OCCASION

This publication has been made available to the public on the occasion of the 50th anniversary of the United Nations Industrial Development Organisation.

DISCLAIMER

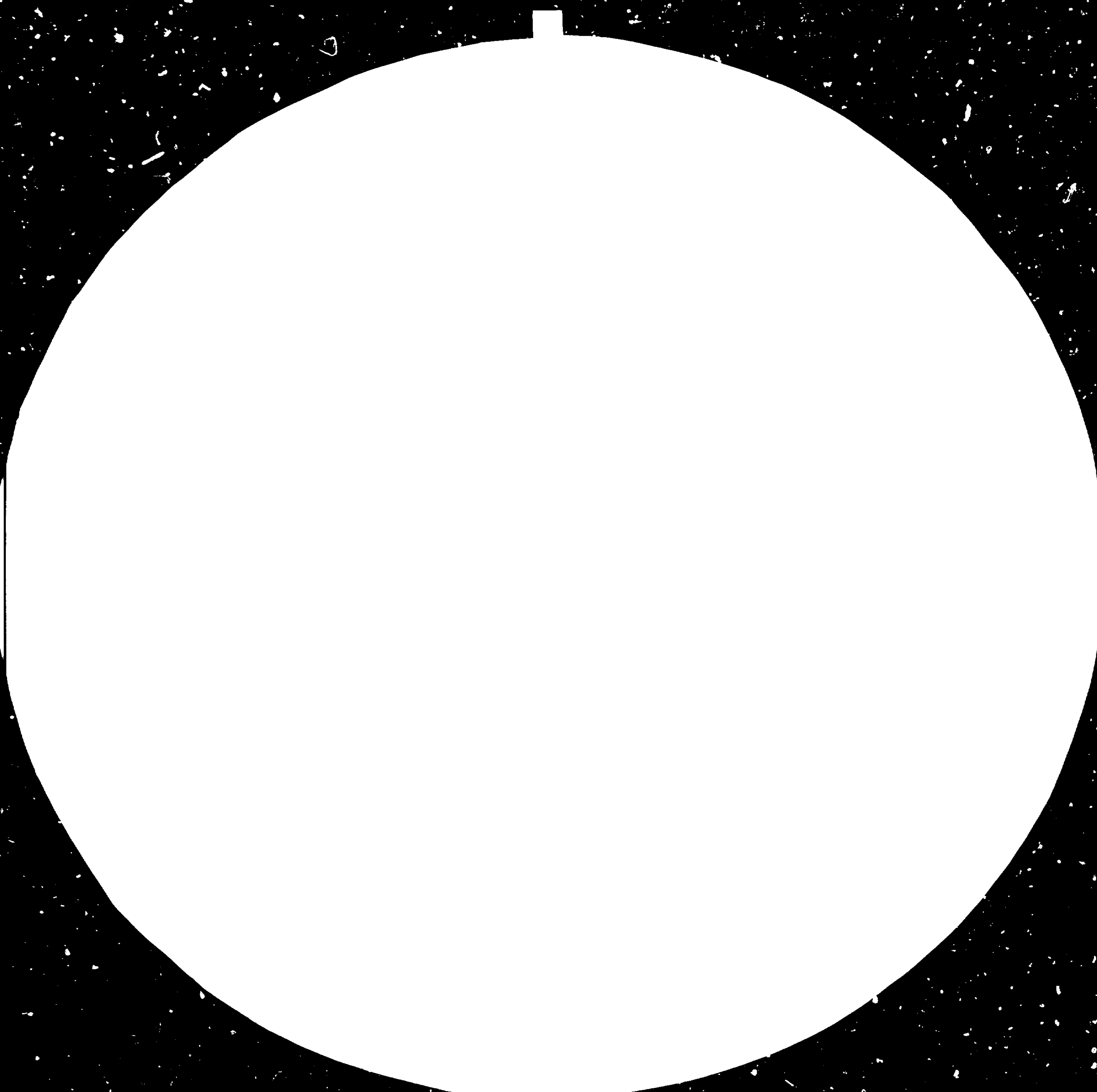
FAIR USE POLICY

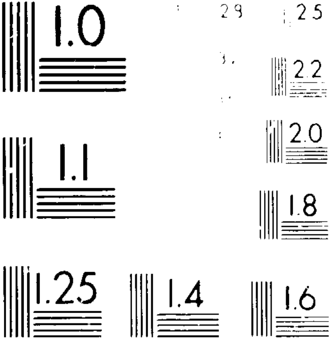
CONTACT

Please contact publications@unido.org for further information concerning UNIDO publications.

For more information about UNIDO, please visit us at www.unido.org

UNITED NATIONS INDUSTRIAL DEVELOPMENT ORGANIZATION
Vienna International Centre, P.O. Box 300, 1400 Vienna, Austria
Tel: (+43-1) 26026-0 · www.unido.org · unido@unido.org




1.0
2.5
2.2
2.0
1.1
1.8
1.25
1.4
1.6

13313

United Nations Industrial Development Organization

Distr.
LIMITED

ID/WG.416/5
2 February 1984
ENGLISH

International Conference on
Carbon Fibre Applications

São José dos Campos, Salvador, Brazil,
5-9 December 1983

SURVEY LECTURE AND SPECIAL EXPERIENCES IN F.R.G.*

(carbon fibres).

by

Klaus Brunsch**

* The views expressed in this paper are those of the author and do not necessarily reflect the views of the secretariat of UNIDO. This document has been reproduced without formal editing.

** Messerschmitt-Bölkow-Blohm GmbH, Helicopter Division, 8000 Munich, F.R.G.

V.84-81158

INTRODUCTION

Although carbonfibres are wellknown technical products since T.A. Edison, it took them a long while to become a candidate for structural application. In the time before CFC glassfibre components had demonstrated quite a lot of favourable properties, but there was no broad application in aerospace due to the limitations in stiffeness.

In the time after the Royal-Aircraft Establishment patent for carbonfibre production had be granted (after 1964) carbonfibres more and more became a strong competitor with the metals used in aerospace. Meanwhile the increasing use of CFC has spured the development of new light alloys to such an extend, that with components made of such alloys weight savings of in excess of 10% can be expected.
This report tries to give an overview of the experience gathered with CFC structures for aerospace application mainly in Germany.

MATERIALS

CFC always comprises two to three constituents:
The fibres, the matrix and the fibre-matrix interface. Therefore experience with CFC material will be reported with reference to the above constituents. Fibre related experience could be gathered with a considerable high volume application of the high strength type fibres for aircraft components. The experiences with high- and ultrahigh modulus fibres mainly result from space activities. Table I gives an overview of the elastic properties of the standard ud-laminates made out of those fibres.

With all fibre types a consistant high quality could be observed. Due to the higher quantity used of T 300 and XAS-fibres excessive information is available. Within one decade these fibres demonstrated a steady, rather moderate increase in strength and decrease of scatter, table II. Only two unfavourable events must be reported, both related more to the interface. Some while ago the surface treatment was executed too intensively, improving ILS considerable but also dramatically increasing notch sensitivity. This is overcome today.

Recently a not notified change in sizing/finish seems to have been made by a fibre supplier. This influenced significantly the results of DSC testing. Shelf live might be reduced.
Investigation of this incident is going on. Like the fibres the resin systems also have developed to a high quality level.

With the 120°C curing systems no problems have been reported for many years. With the 170°C curing systems the adverse influence of flow controlling additives on wet glass transition temperature causes some problems.

There is a good chance for improved systems to be on the market soon. Again in the early times it had to be learned that influence of additive especially on solvent (MEK) resistance must be considered.

With some resin systems to be used for wet processes like filament winding legal problems have been imposed recently. Due to much more stringent health- and/or environmental requirements some space qualified systems will not be available any longer. Expensive qualification of new systems will be necessary.

For such qualifications a lot of material testing has to be performed. There are two categories of material testing: Acceptance testing and testing for material engineering data evaluation.

Engineering data evaluation must be done with rather expensive test-specimen. For tensile properties evaluation the so called "BAC" specimen proofed to be the best. For compressive strength testing a modified "Celanese" test is recommended. Test-specimen see figure 1. Some of the data evaluated using aside of others the test-specimen mentioned before are contained in table III.

EXPERIENCE WITH SPACE APPLICATION

Any material successfully to be used must be selected with respect both to mechanical properties and "design to cost". Carbonfibres, especially the high modulus types are expensive compared to metal. Therefore space applications, which yield the maximum profit of high strength and modulus to density ratio are their domain.

Todays most powerfull civil satellites in orbit are those of the INTELSAT V series. Approximately 25% of the dry satellite mass is made out of Carbon-Fibre-Composites. The structural components are solar arrays, antennas, the feedersystem and the antenna support-structure.
The material used for the solar arrays is Thornel 75S fibre and Ciba CY 209 / HT 972 resin. The ultrahigh modulus fibre was selected to provide sufficiently high natural frequencies of the panels at minimum weight and low coefficient of thermal expansion, similar to that of the solar cells. Design for proper frequencies is required because of the load spectras at start and launch (table IV). The resin CY 209 system was selected first because it is qualified to meet the less than 1% TWL (Total Weight Loss) and the less than 0,1% VCM (Volatile Condensible Materials) requirement and second for filament winding applicability.

By use of these materials the mass per usable m^2 of solar array structure could be reduced from 5 kg for the aluminum design to 1,3 kg for that with CFC.
The solar array consists of 3 panels and the central joke. It is the deployable type. The joke consists of rectangular beams, the panels are CFC skin-Al-honeycomb design.

For all laminate production filament winding was used. Experience gathered is

- the required fibreorientation can be maintained within less than $\pm 3°$. This is required, because of the maximum of flexural and shear stiffeness being very narrow for 0/90° and $\pm 45°$ high modulus fibre laminates
- ultrahigh modulus fabrics are not available, fabrics are woven to standards. Filament winding technology provides flexibility, the theoretically predicted mass of fibres per unit of area can be applied exactly. The face sheet layer can be cut of the mandrel without problems. The large radius in the dome areas doesn't cause problems
- impregnation is without influence on allignment
- laminates are "quasi symmetrical", minimum thickness is as low as 0,1 mm
- no logistic problems like with prepregs exist - storage time, laminate thickness and so on

The rectangular beams shown in figure 3 also completely are produced by filament winding. The ud-chords are wound on narrow drums and cut off.

With the INTELSAT V panels the thickness of the face sheets is 0,16mm nominally. The cell width of the honeycombs being 10mm, the support is local only.

The flexural loading of the panels results in compression in the skins which might fail with instability i.e. with wrinkling. For calculation of minimum wrinkling load stands

$$\sigma_{wr} \leqq 0.816 \cdot \sqrt{\frac{E_c t_f E_f}{t_c}}$$

f = face

c = core

Due to the large ratio of cell width to face thickness local instability - intercellular buckling was expected. For design verification about 150 coupons have been tested.
With these the principal validity of the above equation was found, the factor had to be modified from 0,816 to about 0,30 (figure 4).

The second area of unique CFC application for space are antennas. Here the mass saving potential is gratefully acknowledged, but match winning is dimensional stability independent of temperature. To demonstrate this property an antenna of 90 cm Ø was tested from -140°C to +120°C for 70 cycles.
The maximum deformation normally to the antenna dish was measured to be 0,3 mm at one point only. Average dislocation was 0,1 to 0,2 mm.

Reflectivity also was found to be extremely good. Comparative measurements High-Modulus-CFC versus aluminum showed, that in the 1,3 to 1,7 and in the 2,8 to 3,2 GHz frequency, the CFC was as good as the metal. Up to frequencies ~ 14 GHz pure CFC shows reflectivity factors of up to 98%.

The lessons learned with CFC space applications are:
HM-CFC structural components feature outstanding good behaviour both mechanically and electrically. Application for central structural components is limited by poor heat-transfer capability. Ongoing R+D efforts will increase effectiveness of structures.

EXPERIENCE WITH TRANSPORTATION AIRCRAFT

When the utility of CFC components became evident most of major aircraft companies started programs to build up carriers confidence in the new material. Small structures like the Boeing 737 spoiler were developed and brought into service. These service evaluations showed durability like with metals, ease of repair and did train service personal to handle properly such components.

The programs mentioned above were started when the prognosis for carbonfibre prices were very encouraging. Unfortunately carbonfibre prices are still rather high compared with metals (figure 5) and the economic situation of most of the carriers does not let them pay additional money for weight saving.
At MBB's commercial AC division a CFC component will get the go ahead only if it costs equal or less than a comparable one out of metal. This can be achieved only by carefull evaluation of appropriate technology which must be taken in account at the very begin of the design and by dramatic reduction in number of parts and fasteners. A good example for such a technology/design is the CFC rudder of the DC 10 This was produced in a one shoot technology using Thermal Expansion Moulding. Silicone rubber bags are a standard tool today with the known disadvantage to stand for a limited number of cycles only. With the development of the airbus vertical stabilizer it was aimed to avoid the costly repeated production of rubber tools. The use of the thermal expansion of aluminum only combined with vacuum bag technology is cost effective. Series production for the airbus familiy was decided by Airbus Industry.

To the authors knowledge, the airbus vertical stabilizer is

the largest civil transportation CFC component to go in series production. Some of the most interesting features will be reported.
Cost effectiveness mainly results from the technology used for the integrally stiffened about 3 by 11 m skin panels, which are produced with the so called modular concept. CFC fabric skins are layed onto the mould, which also is made out of CFC. At the same time ±45° fabric tapes are wrapped around the aluminum "moduli", (see figure 6). Once all the moduli are covered they are assembled on a jig. There is considerable compression chordwise, but non spanwise. UD-tapes are layed on that side, which later is layed on the skin. For that purpose the jig is rotated 180° and the mould positioned underneath. For ease of that the mould is air cushion suspended. After the "moduli" have been layed onto the mould and fixed by a frame running all around the mould the inside ud-tapes of the flanges are laminated.

The vacuum bag is then tightened to the mould and the complete lay up cocured. The result of that is shown in figure 7.

The resulting CFC stabilizer box is predicted to be approximately 5% less in cost and 17% less in weight compared to the metal version. The predicted cost reduction is based on the assumption, that the wrapping of the moduli, the assembly of the moduli and also the cleaning of the moduli is performed by robots or robot like machines.

EXPERIENCE WITH MILITARY AIRCRAFT

In civil application like with space-and transportation aircraft-components design to cost normally is decision making. With military equipment superiority consideration rank first. This is the reason for advanced composites being used for military aircrafts since they became available. Application of CFC components started with the history making boron epoxy horizontal stabilizer of the F 14. Todays advanced composite structures concentrate on use of the by far less costly carbonfibres, the share of composites in structural mass break down increases continuously (figure 8). Control surfaces are state of the art, wet-wings (Harrier) and front fuselage components are in series production. To come to that state a lot of work needs to be done. Typical sequence of such a development is described using the CFC-Tornado taileron as an example. Aerodynamic shape and also loads were known from the metal version (figure 9). Preliminary design made use of so called carpet plots. These carpet plots - the tensile RT strength of the 0°/90°/45°-laminate family of T 300/914C is shown as an example (figure 10) - are calculated by computer programs. The input data are to be evaluated by carefully testing the "standard" 60% volume of fibre unidirectional tape. Carpet plots for example of bearing strength must be based of course on tests with laminates.

In the preliminary design two versions of the highly loaded internal structure have been investigated - an all CFC and a mixed CFC-metal design. Although some novel structural CFC components and the related technology had been developed successfully, the CFC-metal design was selected (figure 11) for lower risque.

To finalize the design "Nastran" computation was used. Both skin laminating plan with reference to ply sizes and orientations and honeycomb density have been optimized based on that calculation.
Taileron have been produced without major problems and are in flight and fatigue test now (figure 12).
Out of the total mass of the taileron 36% are CFC only. The weight saving compared to metal version is 17%.

Approximately the same weight saving was calculated for a technology research CFC fighter cockpit. The technology research cockpit is about 3,5 m length by 1,4 m width and 0,9 m height. It weighs 290 kg with the reinforcements to allow assembly with the test fixtures and should be 272 kg without. The low weight low cost design consists of two integrally stiffened shells (figure 13), the cocured stiffeners are ⊥ shaped. For ease of production the skins are laminated on male mandrels (figure 14) and then transferred to female galvano nickel moulds. The stiffeners are preformed on silicon rubber tools and inserted in the mould together with the rubber cores. Skin panels have been produced without problems, one completed cockpit (figure 15) will be tested electrically and structurally.

Lessons learned:

- Access to experience with military Aircraft CFC components is limited. Information was obtained from the AFML, that there was no adverse influence on composite structure, that had be flown to skin temperatures in excess of the wet glass transition temperatures of the composite. It must be noted, that flights of such a speed always are of short duration only.
- To provide battle damage tolerance the strain at ultimate design load shall not exceed 0,35% to 0,4%.
- Complex one shot cured components must be designed with allowance for minor defects. Especially cocuring of skins and I-beams always is associated with the risque of flaws in the direction of the I-beams. Those flaws tend to be at the corners where the shearwebs attach to the skin.

HELICOPTER EXPERIENCE

Since the great success of the BO 105 glassfibre composite rotorblades everybody uses composite rotorblades.
The programs to replace the metal blades of the US-Forces CH 46 and CH 47 helicopters total approximately 200 Mio US $.

In spite of the great number of composite rotorblades being either in service production or under development the CFC experience is limited because of the large GFK share.

With the BK 117 (MBB-KHI) mainrotorblade some experience with carbonfibre-fabric is being made. Initially these blades were built with glassfibre only. Testing inflight the outboard region of the blade was found to be to low in torsicnal stiffeness. About two meters of glassfabric of the skin were replaced by carbonfibre fabric (figure 16) and since that the helicopter is improved dramatically in static and dynamic stability. In the "wet" production process of these blades the mixed fabric skin-glass- and carbonfibre fabrics - does make no problems. MBB does expect to have the excellent long time serviceability for the carbon-skin-blades which is known for its glassfibre-blades. Adverse influence of the electric conductivity of the carbon skin is of low probability, simulated lightning testing with the unprotected CFC skin did not cause catastrophic damage (figure 17).

Whilst MBB did use CFC to increase torsional stiffeness, Sikorsky did use it for the opposite objective. The "flex beam" of the bearingless S 76 tailrotor (figure 18) is entirely made out of high strength CFC. The design of a rotorblade always must avoid coincidence of natural bending frequencies of the rotating beam rotorblade with exciting frequencies. The natural frequencies both in flapwise and chordwise direction must be relatively high, thus defining the required flexural stiffeness (EJ) distribution over R.

With the low flexural stiffeness of GFC the Js and with them the areas have to be considerably larger than with CFC. As the shear modulus of GFC and CFC-UD is about the same, the glassfibre version would be much higher in torsional stiffeness, the structure would not withstand the repeated deformation for pitching.

The Sikorsky S 76 flex beam CFC tailrotor is in service now since 1978. No problems with the CFC flex beam have been reported within many hundredthousands of flight hours. A different area for profitable use of CFC are drive shafts.

This has been demonstrated by the development of an experimental tailrotor drive system for the BO 105. The metal version - stainless steel tube, titanium fitting and stainless steel leave springs (figure 19) was replaced by a filament wound HM and HT CFC tube and a glassfibre composite coupling. The composite version (figure 20) was successfull in static and dynamic test, but most important is that it was superior weight and costwise (table V).

With reference to cost must be added, that for logistic reason the number of bearings of the metal version was maintained although the increased flexural frequency of the CFC version would have allowed reduction.

Despite the advantages described above the composite drive shaft system is not in series production. This is because of the miss match of coefficient of thermal expansion of the aluminum tail boom in series production and the CFC tubes. For future MBB helicopters the use of advanced composites both for tailboom and -drive system is foreseen.

The experience with CFC applied for fuselage components is limited only. With respect to the exterior surface, the share of composites is rather large - about 40% for BO105, 50% for BK 117 and 70% for the Agusta A 129. Most components with these percentages are secondary structures,made out of glass- and aramidefibres, nearby no carbonfibre. A major break through is to be expected once the All Composite Airframe Programs have proven the promised weight and cost reductions. Both Bell and Sikorsky will have an "ACAP" fuselage completed in 1984. Weight saving is predicted to be 22%, cost saving should be 17% compared to the comparable metal version. A typical fuselage section made out of both CFC and AFC is shown in figure 21. This section designed for BO 105 size was used to investigate technologies and resulting weight and cost saving potential. Results of these investigation indicate the objectives of the ACAP to be realistic.

TECHNOLOGY

For many years the nearby only available technology - aside of filament winding - was hand-lay-up. This did include cutting of the plies, removal of release and supporting film and laying of the plies. Today a lot of machinery is offered and great care is recommended to avoid unprofitable investments. Based on MBB production requirements experience some examples are reported, which are more generally composite than CFC only releated.

Materials used mainly are glass- and aramide-fibre-prepreg, i.d. broad good, which is used in secondary structures with one or two layers only. For this application a "Gerber-Cutter" (reprocicating knife)is used to cut the plies. Most of the remaining work is done by hand.

The rather expensive "Gerber" will not be used to cut the plies to be wrapped on the "moduli". The reason is, that the "Gerber" fixes the broad good to the cutting table by vacuum and therefore does not allow to remove supporting and release film in advance of cutting. For series production the plies for the "moduli" will be cut by an NC-controlled waterjet. Waterjet technology allows the automated removal of all auxiliar materials prior to cutting.

In the R+D phase MBB's helicopter division uses steel rule dies to cut plies for the CFC plates of a rotor hub. The plies are identic in geometry but vary in fibre orientation. The steel rule dies work effectively only with the foils on the prepreg. For mass production the cutting technology must be reconsidered due to the lack of a reliable automated technology to remove foils off the plies.

The application of a laser cutting device for serial CFC cutting is - at least now - not planned at MBB.

With laying techniques the situation is similar. Although an "Ingersoll" tape laying machine is installed at the civil AC division it will not be used for the airbus vertical stabilizer. Handlaying of the C-fabric-prepreg is more cost-effective than tape laying of 4 layers to get the same "symmetric" 0°/90°-laminate. The tape laying machine is used for components with great amounts of UD-tapes required.

The military AC division is installing a tape laying machine coupled with a cutting device. As the cutting device operates much faster than the laying machine, ply-families will be laid to balance machine capabilities. With respect to the small military AC market in Germany, ply transfer by hand to the mould appears to be superior in cost effectiveness.

If cutting is performed by means of steel rule dies the die can be used for ply transportation also. This is considered for the fabrication of the rotorhub plates, but first the problem of removing the foils must be solved. Laying of the rotorhub plies is no problem because of the press curing.

The most challenging task in mechanisation of CFC structures production is laying of the plies, especially on double curved surfaces. This area is investigated intensively and with high cost investments, mainly in the USA.

The results achieved up to this day are such, that MBB in the moment considers hand-lay-up of the precut plies to be cheaper than automated laying. This consideration is based on 200 kg or less CFC prepreg to be consumed per day.

Lessons learned with CFC technology are:

- The small volume production of aerospace CFC structures requires a well balanced mix of automated and manual processing to be costeffective

- waterjet cutting of prepreg does not reduce material mechanical properties
- progresses in auxiliary materials can decide on technology and/or design to be used.

QUALITY CONTROL EXPERIENCE

Composite Components have been developed and produced already before prepregs were on the market. Wet processes still are used, the quality assurance "tools" both for wet and prepreg technology are available.
For the "wet" technologies like filament winding resin and hardener are tested both physically and chemically at income inspection. Typical tests are

for resin:	-density
	-epoxid equivalent
	-viscosity
	-colouring
for hardener:	-amin/anhydrid value
	-composition (HPLC)
	-melting point (TMA)
	-latent heat

As to the testing of the resin of a prepreg there is some controversity in the moment. The requirements range from mechanical testing only to complete physico chemical analysis.
An overview of the physico-chemical methods is given in table VI .The standard tests are to assure material quality at minimum cost. With that in mind a very costeffective test-specimen for prepreg-ud-tape has been developed. A single layer of prepreg-tape is covered with a thin poliamid foil and then cured. The cured tape is cut to strips of 10 mm width and tension tested to failure. Tensile modulus and ultimate strength are measured.

This simple test combined with the information of fibremass/unit of area provides excellent global information on fibre tensile properties and/or fibre distribution and/or fibre alignment.

With the mass production of glassfibre composite rotorblades it had to be learned, that the ILS does provide limited information on surface adhesion only. In glassfibre acceptance testing a peel test was introduced. A similar test for carbonfibres is being developed.

The single layer test-specimen is used for carbonfibre-fabrics also. For test result interpretation, the specimen is treated as a 0°, 90°/s-laminate.

The fibre itself, which is dominating the composite properties is tested in laminated form only. For wet processes ud-laminates are produced for evaluation of tensile strength and modulus, ILS and may be peel strength also.

With in process testing a well balanced compromise must be found to assure quality to maximum reliability at reasonable cost. But prior to that, all the equipment intended for use must be controlled starting with the environment of shops, tool-dimensions, heatflow and temperature distribution of tools and so on. All parameters easily to be reported shall be. For example any scale used is to be linked to a printer. If there are major differences in thickness of components and/or moulds, the readings of many probes for temperature are to be documented.
Also the materials have to be selected with respect to in process quality assurance or inspectability.
Again for example: To reduce the risque of poor fibreorientation with C-fabrics, a "Kevlar" tracer is to be used.
In addition to that so called critical processes have to be indentified and special /additional tests/test-specimen have to be specified.

For the time after the component is cured and controlled for dimensions NDT is to be applied. Acoustic transmission testing is in series application with most of the CFC-components produced today.
Thermography, Holography and X-ray technology are well-known also. Computerized processing of the acoustic or X-ray signals improves detecting capabilities.
In the development phase computer tomography can be very usefull too. It helps to produce very reliable information eg. of flaw size which then can be compared with acoustic test reading intended to be used in series production.

The"NDT"to be used depends on the individual structure and on the defects looked for. The holographic image of a great flaw between CFC skin and PVC-foam core of a rotorblade is shown in figure 22.

For space structures proof test loading appears to come back again. This can be accepted only with respect to the relative small number of components and the relative high prices payed for space structures.

Conclusion

CFC is a well adapted aerospace material today. It has spured metal technology to considerable improvements (Al-Li alloys) and is being improved itself to the maintain lead among the aerospace materials. Ability to handle CFC is an earnest requirement to be competitive in the international aerospace market.

60 Vol.%		M40A		75S	T300/914C	XAS/914C
		RT	77K	RT	RT	RT
$E_{\|\|}$	kN/mm²	225,5	240	298	133	133
$E_{\perp}$	kN/mm²	7,4	11,45	3,68	9,3	9,3
$G_{\#}$	kN/mm²	4,3	5,77	3,06	4,6	4,6
$\nu_{\|\|\perp}$		0,26	0,26	0,36	0,28	0,28

TABLE I: ELASTIC PROPERTIES OF THE UD-LAYER

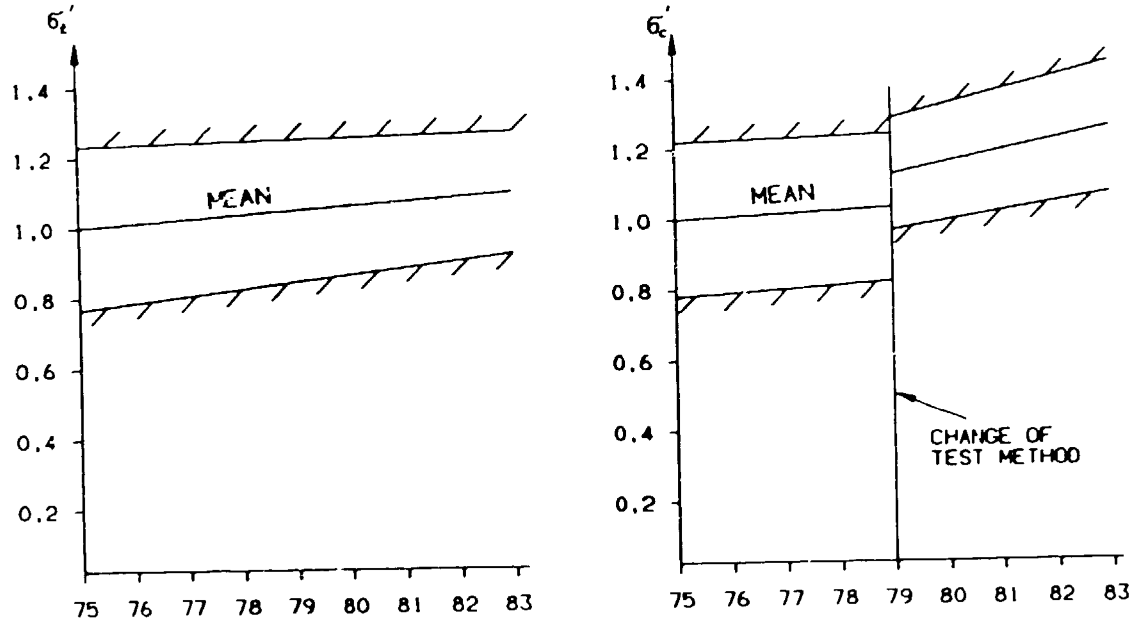


TABLE II: IMPROVEMENT OF CFC UD-TAPES, HT FIBRES

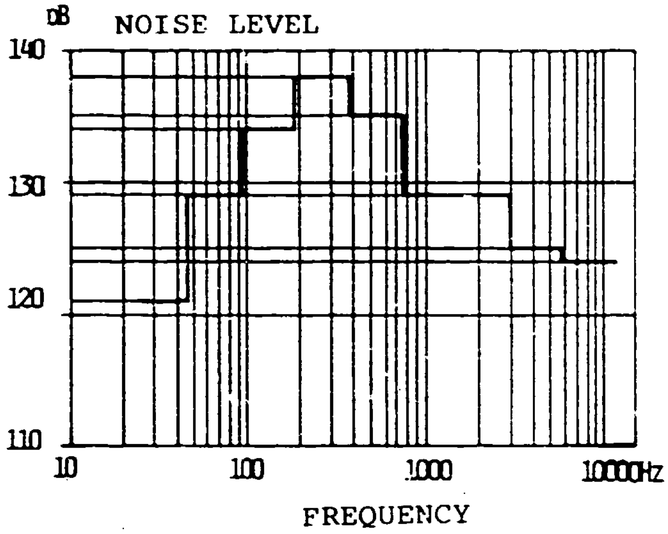
		T300/914C	XAS/914C	M40A/CY209	75S/CY209
Tension					
N/mm^2	$\sigma_{\|\|}$	1613	1580	1422	1572
	s	46		3,8	35
	V	2,9	5,6	0,3	2,2
N/mm^2	$\sigma_{\perp}$	36	36	30	
	s	5,5			1,4
	V	15,3	8	2,4	10
Compr.					
N/mm^2	$\sigma_{\|\|}$	1812	1714	760	526
	s	72,9		51	78
	V	4	5,5	6,7	15
N/mm^2	$\sigma_{\perp}$	218	206	94	72
	s	12,8			2,6
	V	5,9	6,2	4,0	3,6
τ_{ILSS} N/mm^2		109	98	30	38

TABLE III: MATERIAL STRENGTH PROPERTIES 60% FIBRE VOLUME,RT

START AND LAUNCH PHASE

SURVIVING OF LOADS FROM

- SINUSOIDAL VIBRATION (LONGITUDINAL DIRECTION)
- ACOUSTIC VIBRATION
- STATIC LOADS FROM START ACCELER ATIONS
- 1ST RESONANT FREQUENCY ≥ 30 Hz



DEPLOYMENT EVENT (GENERATOR)

LATCH-UP SHOCK FROM DEPLOYMENT FINAL ANGULAR VELOCITY 60°/SEC.:

100 NM BENDING MOMENT AT 130 N SHEAR FORCE AT PANEL CORNER PART

IN ORBIT

- LOADS FROM 400 N BOOSTER THRUST
- LOADS FROM ATTITUDE CONTROL MANEUVERS
- LOADS FROM EXTREME TEMPERATURES $-180°C \leq T \leq 100°C$
- NO SIGNIFICANT CHANGES OF GEOMETRY AT WAVE GUIDES AND ANTENNAS
- PRECAUTIONS FOR VENTING AND ELECTROSTATIC DISCHARGE

OUTGASSING RESISTANCE: TWL ≤ 1% (TOTAL WEIGHT LOSS)
VCM ≤ 0,1% (VOLATILE CONDENSIBLE MATERIAL)

TABLE IV: REQUIREMENTS OF SPACE STRUCTURES

Part	Mass /g/			Number of Parts		
	Metal	Composite	Mass-reduct.	Metal	Composite	Parts-reduct.
Long tailrotor drive shaft (l=3940) with a coupling on one side and a flange on the other and additional 4 bearings	7780	3730	4050	89	54	36
Long tailrotor drive shaft(l=3904) with a coupling on one side and a flange on the other without bearings	6980	2930	4050	72	46	26
Short tailrotor drive shaft (l=731) with a coupling on each end of the rod	1930	860	1070	101	37	64

	Basis: Parts for 200 helicopters					
Component	Total costs /%/		Material costs /%/		Manufacturing costs /%/	
	Metal	Compos.	Metal	Compos.	Metal	Compos.
1 drive shaft incl.2 flanges without bearings	46	69,3	33,8	29,9	12,2	39,4
2 couplings	52,2	20,3	30,0	8,0	22,2	12,3
assembly	1,8	0,4	0,5	0,4	1,3	--
	100,0	90,0	54,3	38,3	35,7	51,7

TABLE V: DRIVE SYSTEM MASS- AND COST COMPARISON

Method	PCI	chem. charact.	specifi- cation	acceptance control
HPLC (HPTLC)	- (fingerpr.)	+	- + +	+
GPC		+	-	
GC		+	- ?	
IR	+ (fingerpr.)	+	- +	+
NMR		+	-	
AAS		+	-	
DSC	+	+	- +	+
DMA		+	-	
TMA		+	-	
TGA		+	-	
Viscosity	0	+	- +	+
Vanhograph	0	+	-	
Element. analysis		+	- +	
Torsional pendulum DMA (lamin.)		+	- +	+
Standard tests		+	- +	+

Use of methods if available:

+ must

0 can

- possible

Methods for specification are specific for each individual system. They can be selected on the basis of chemical characterization in accordance with the supplier.

TABLE VI : PHYSICO-CHEMICAL METHODS FOR QUALITY ASSURANCE OF EPOXY -CF-PREPREG SYSTEMS

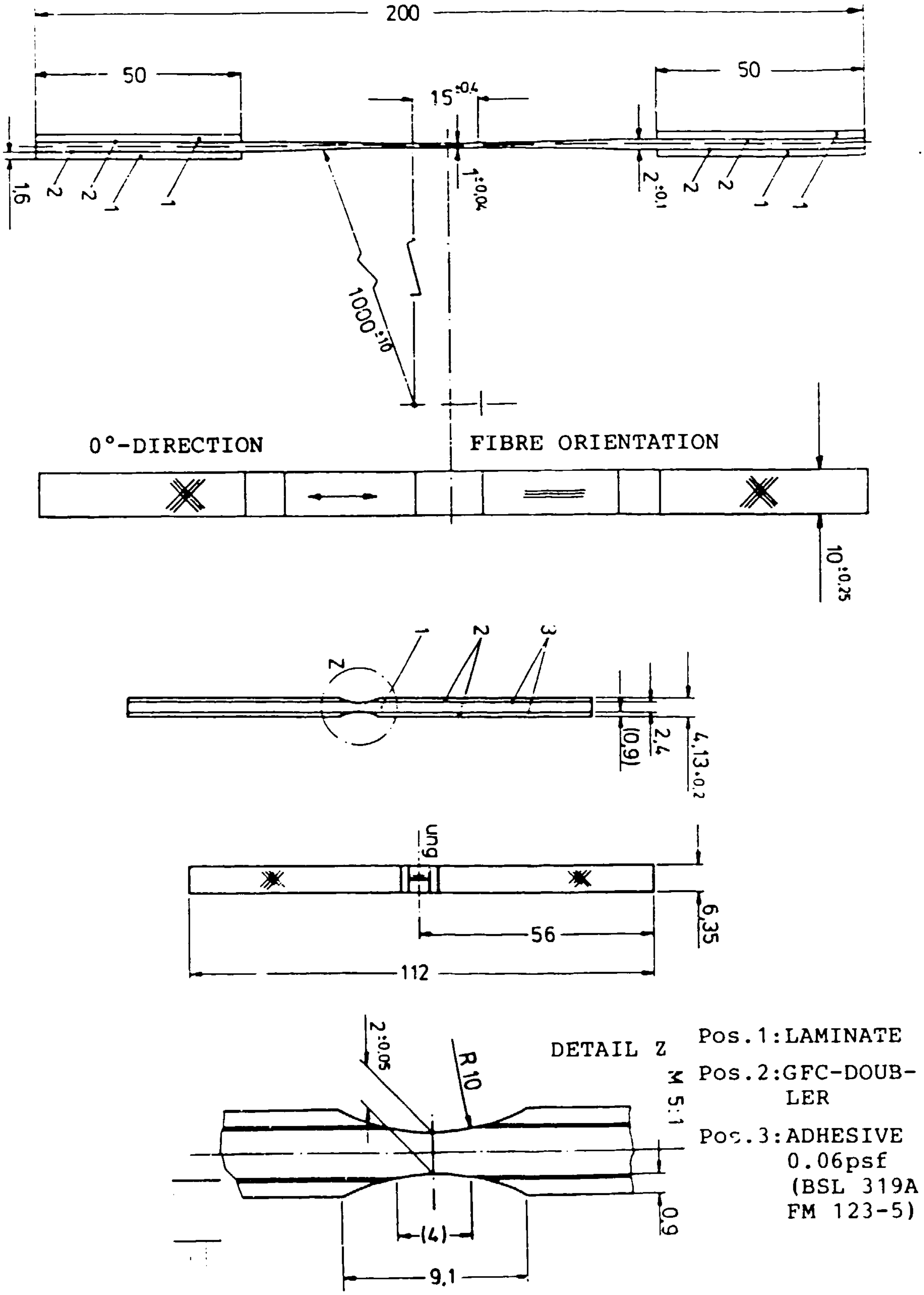


FIGURE 1: TEST SPECIMEN FOR ENGINEERING DATA EVALUATION

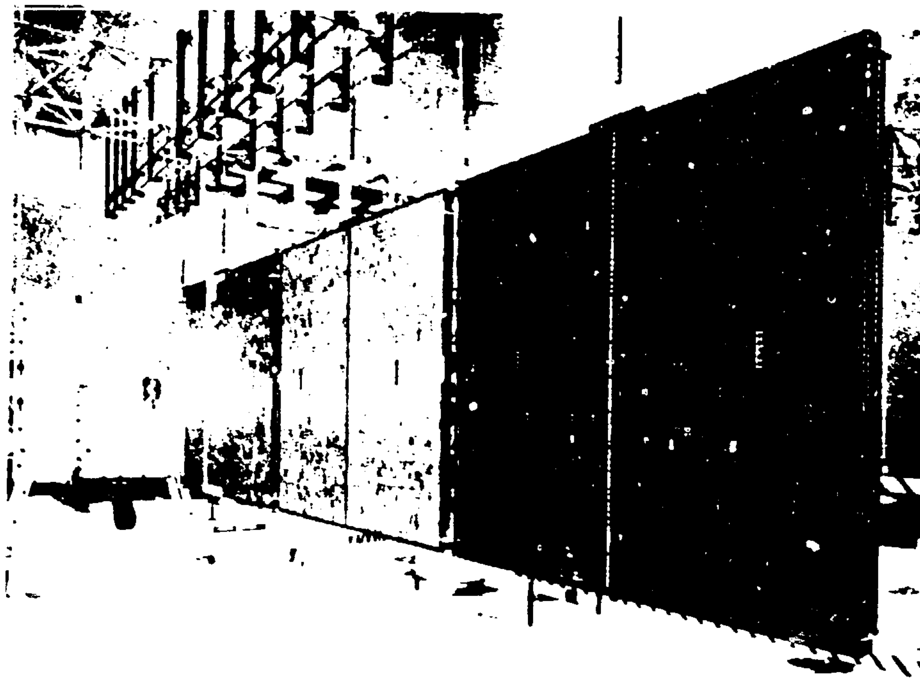


FIGURE 2: SOLAR ARRAY

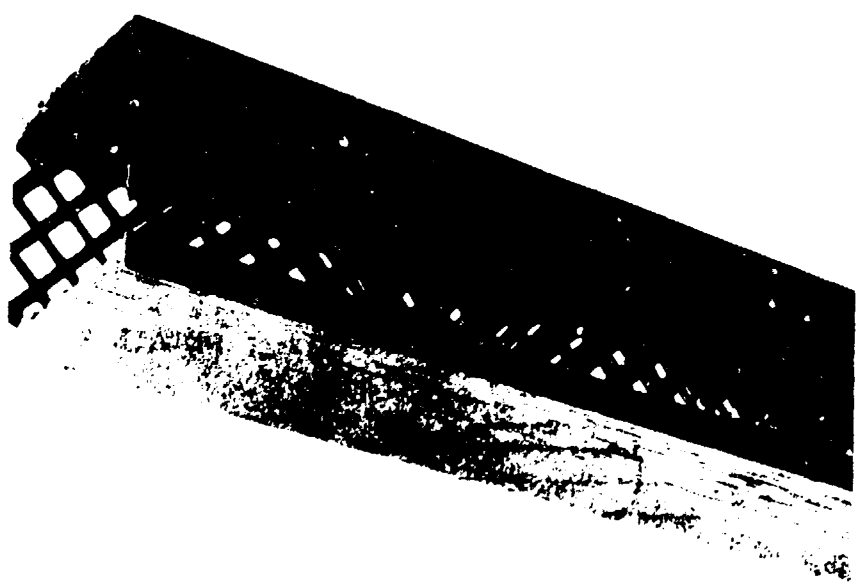


FIGURE 3: FILAMENT WOUND RECTANGULAR BEAM

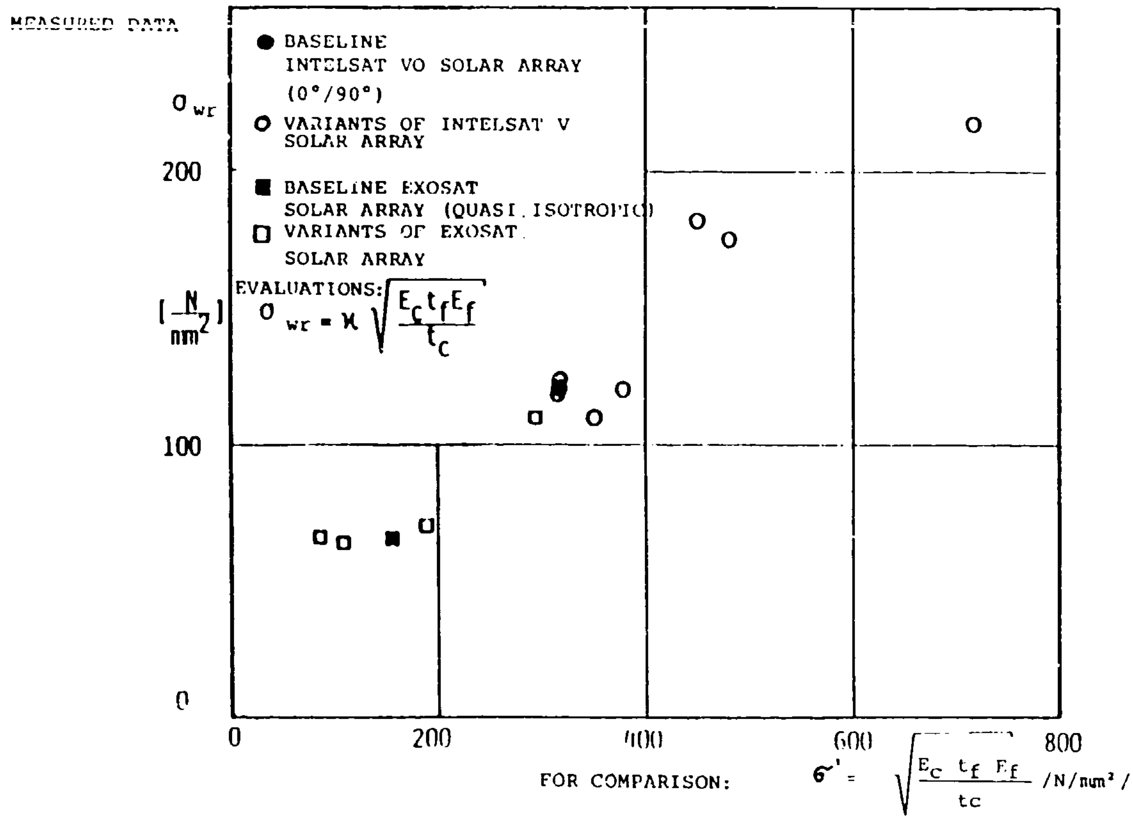


FIGURE 4: WRINKLING TEST RESULTS (FROM COMPRESSION LOADS)

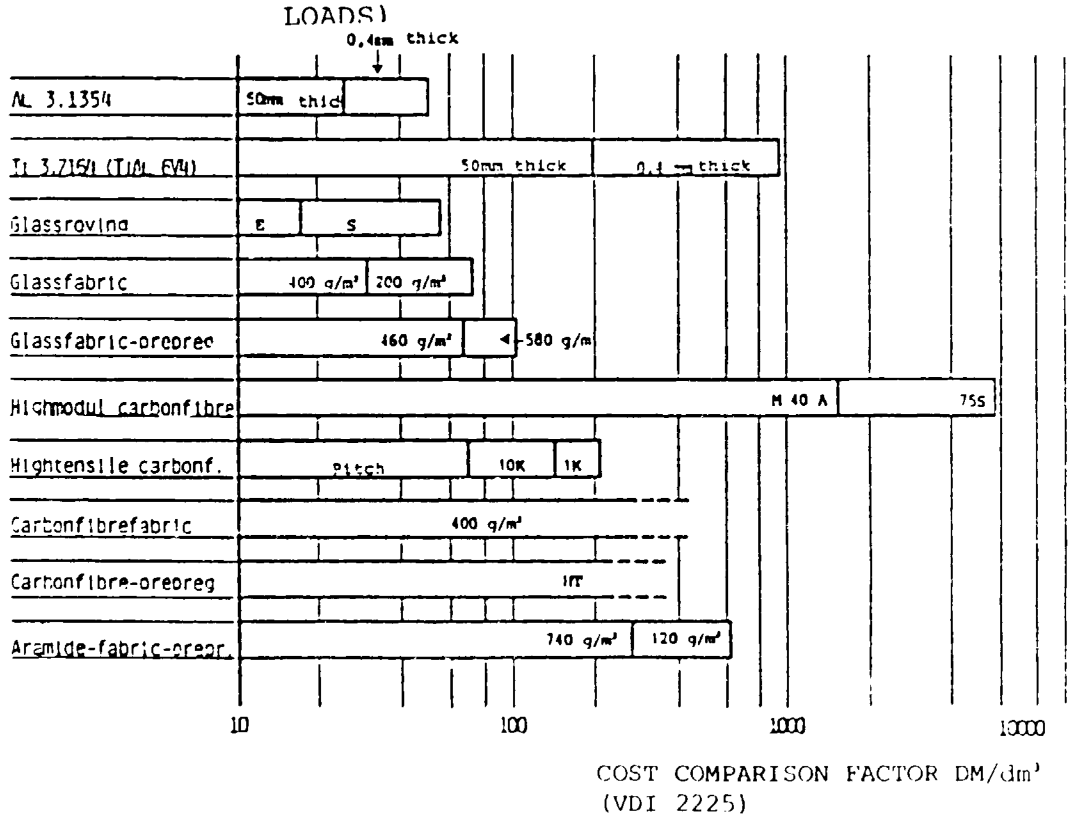


FIGURE 5: MATERIAL COST COMPAPISON

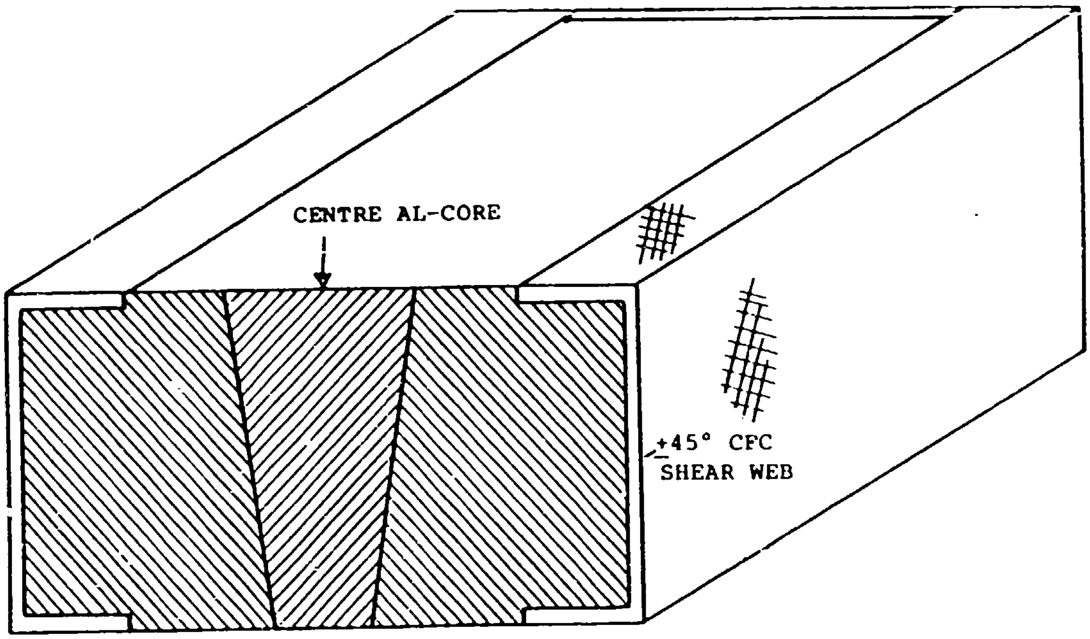


FIGURE 6: CROSSECTION OF MODUL WITH SHEAR WEB

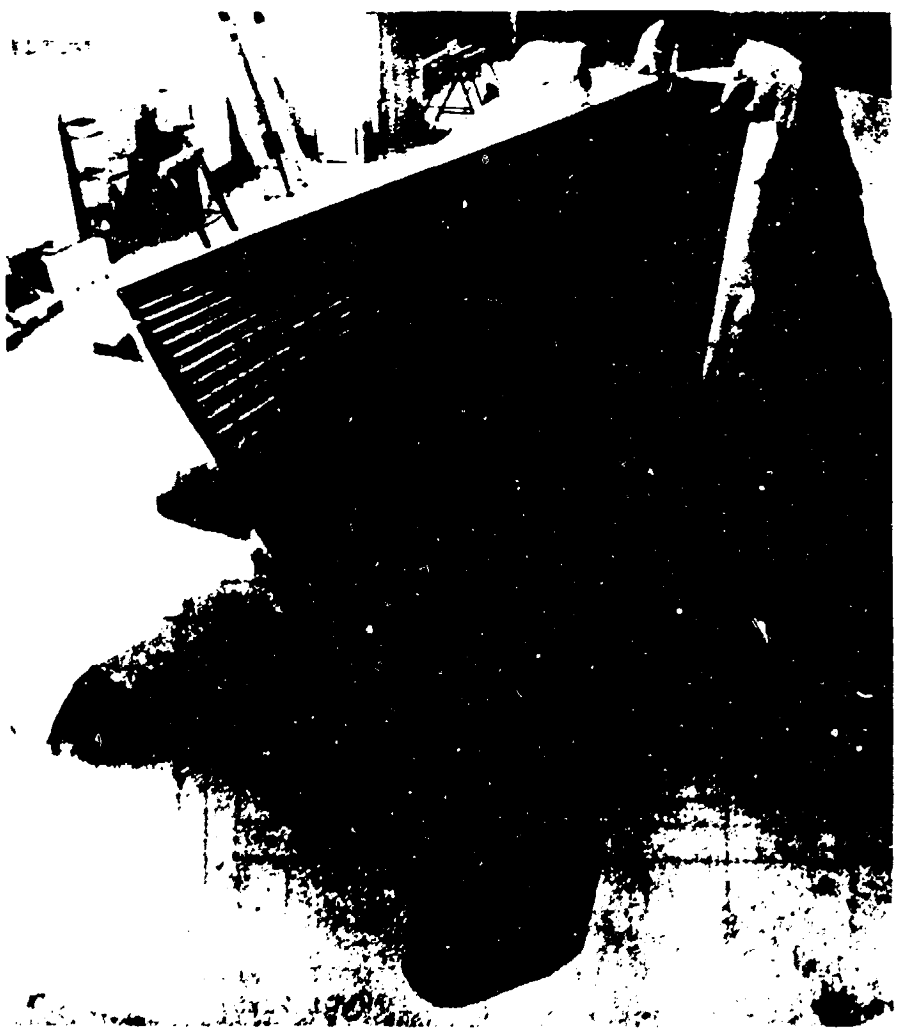


FIGURE 7: INTEGRALLY STIFFENED PANEL

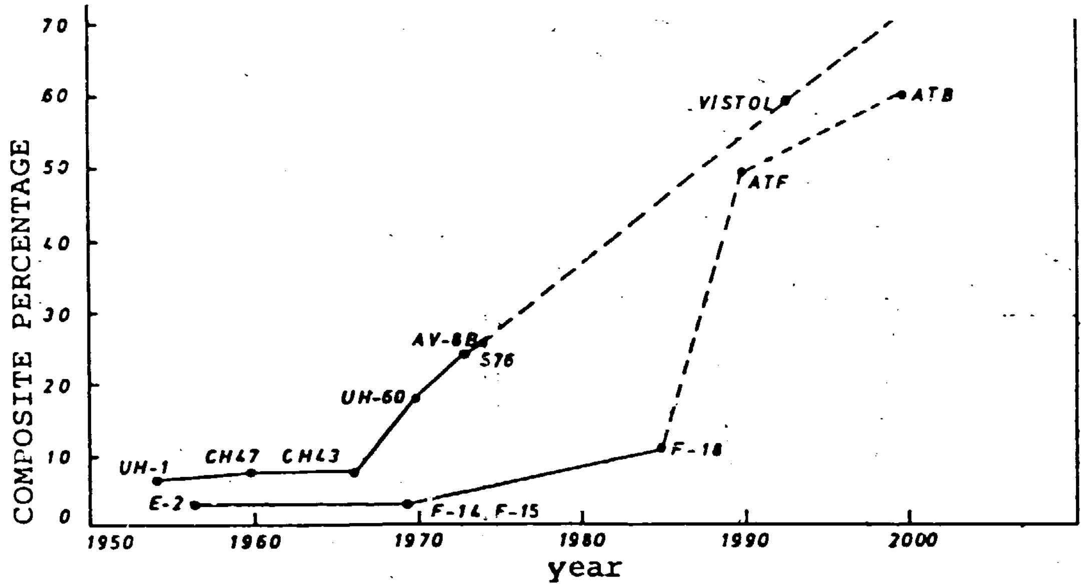
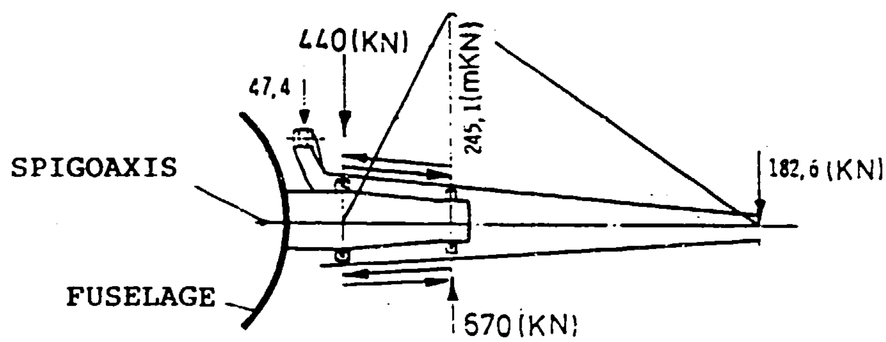


FIGURE 8: COMPOSITES USED WITH EXISTENT AND PLANNED FOR FUTURE AIRCRAFT



BENDING MOMENT	$M_{b\ max}$	=	245 (mkN)
SHEAR LOAD ON INTERNAL STRUCTURE	q_{max}	=	2000 N/mm
LARGEST LOAD ON BEARING	Q_{max}	=	670 (kN)

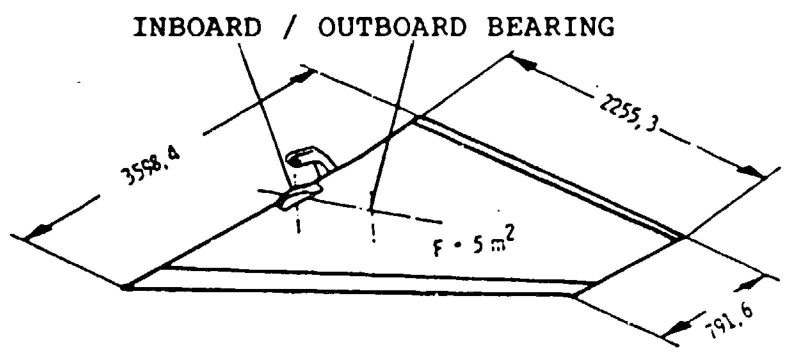


FIGURE 9: TORNADO TAILERON LOADS AND DIMENSIONS

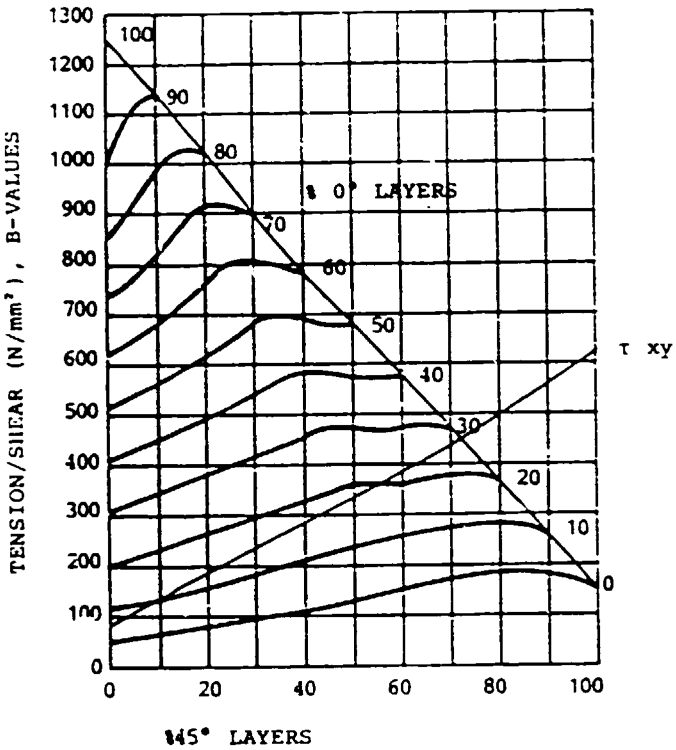


FIGURE 10: TENSILE STRENGTH 0° / 90° / 45° LAMINATE FAMILY, RT

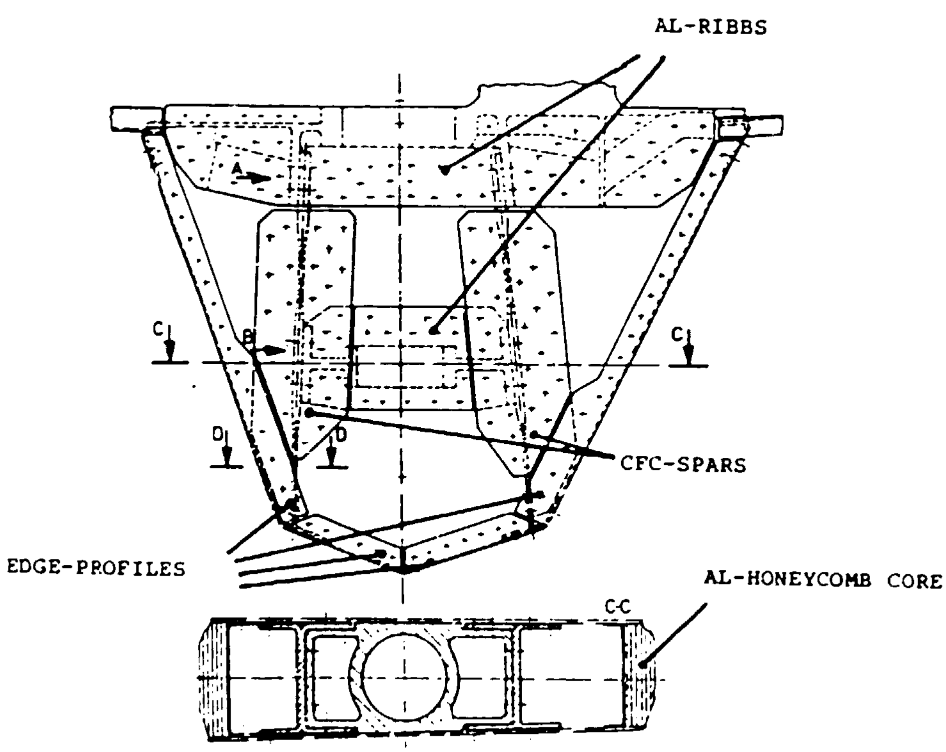


FIGURE 11: TAILERON INTERNAL STRUCTURES

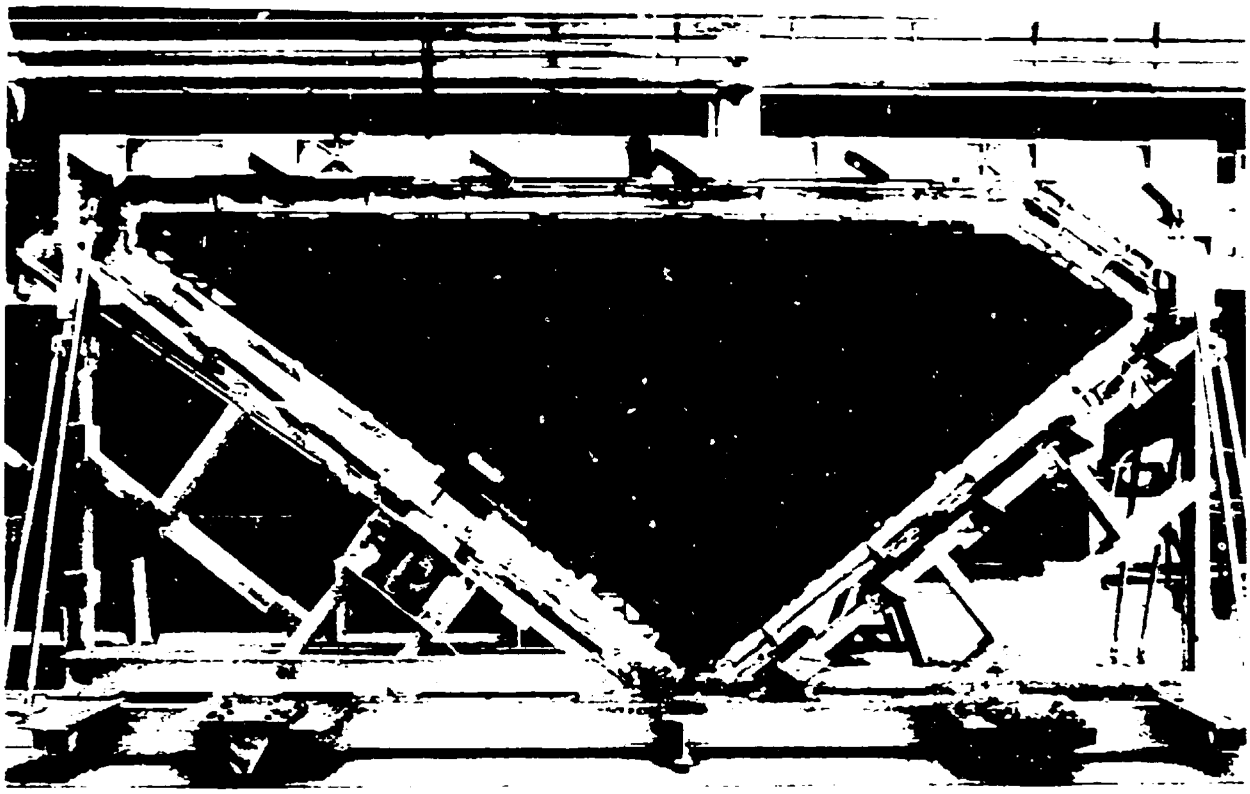


FIGURE 12: TAILERON IN BONDING FIXTURES

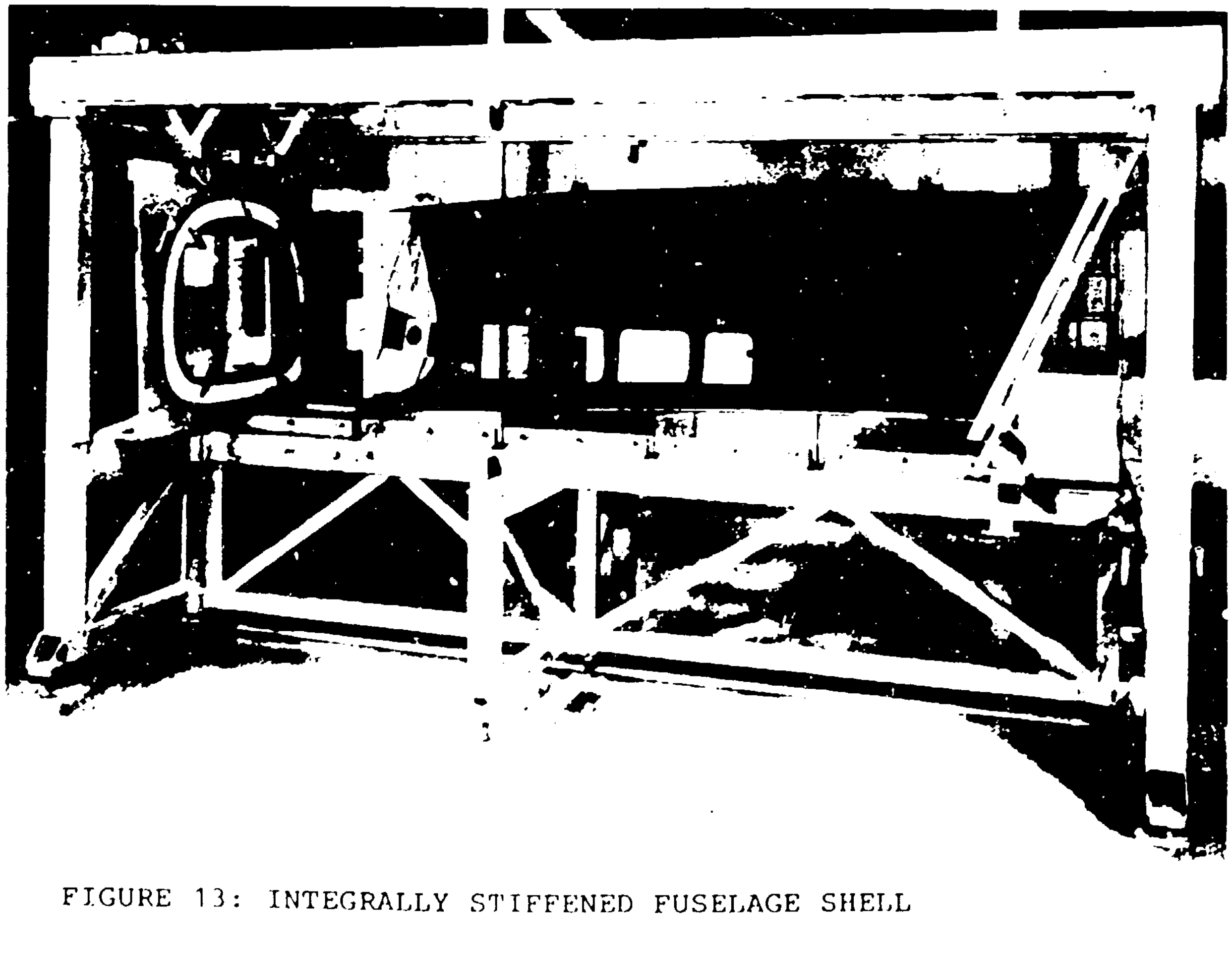


FIGURE 13: INTEGRALLY STIFFENED FUSELAGE SHELL

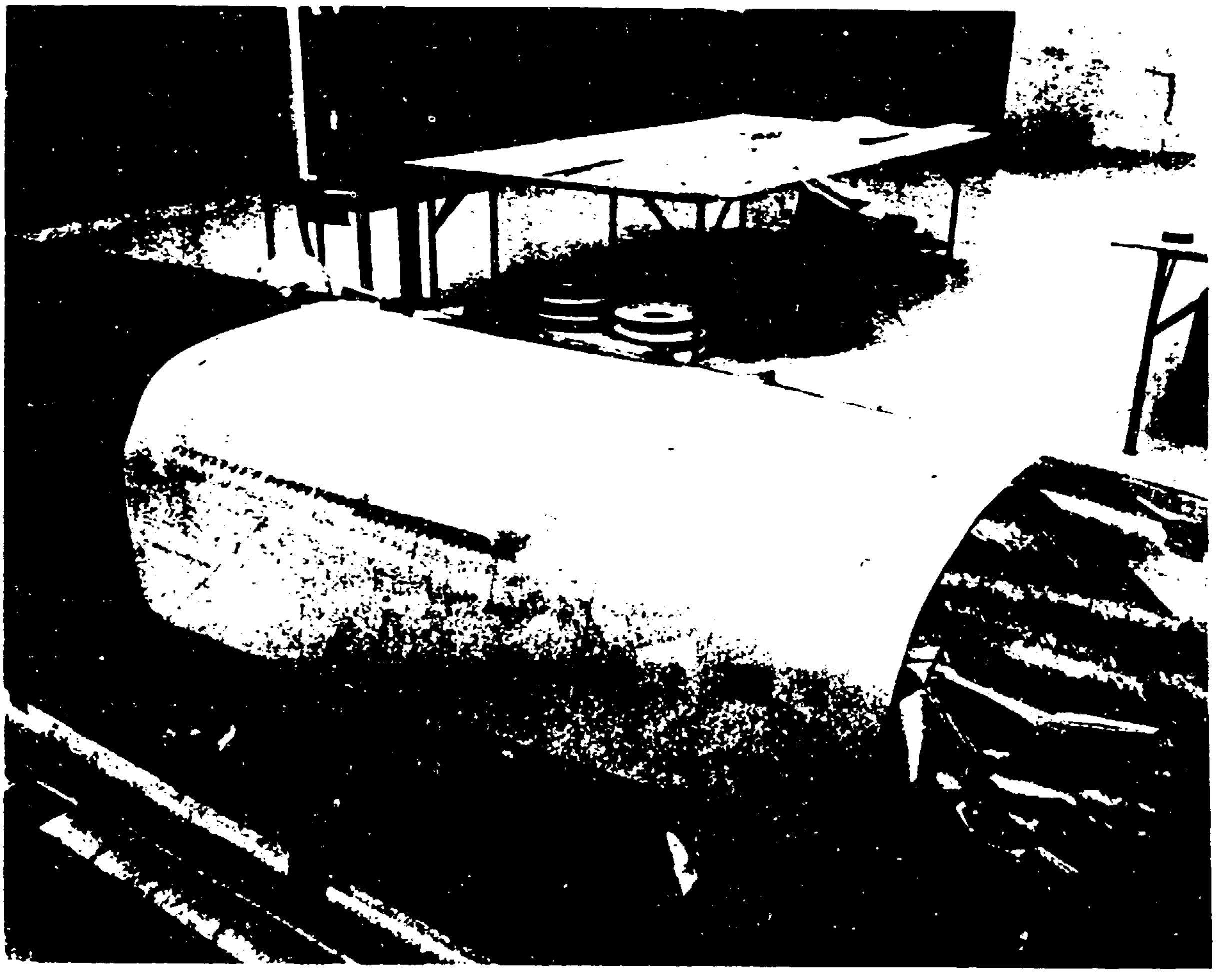


FIGURE 14: MALE LAYING TOOL

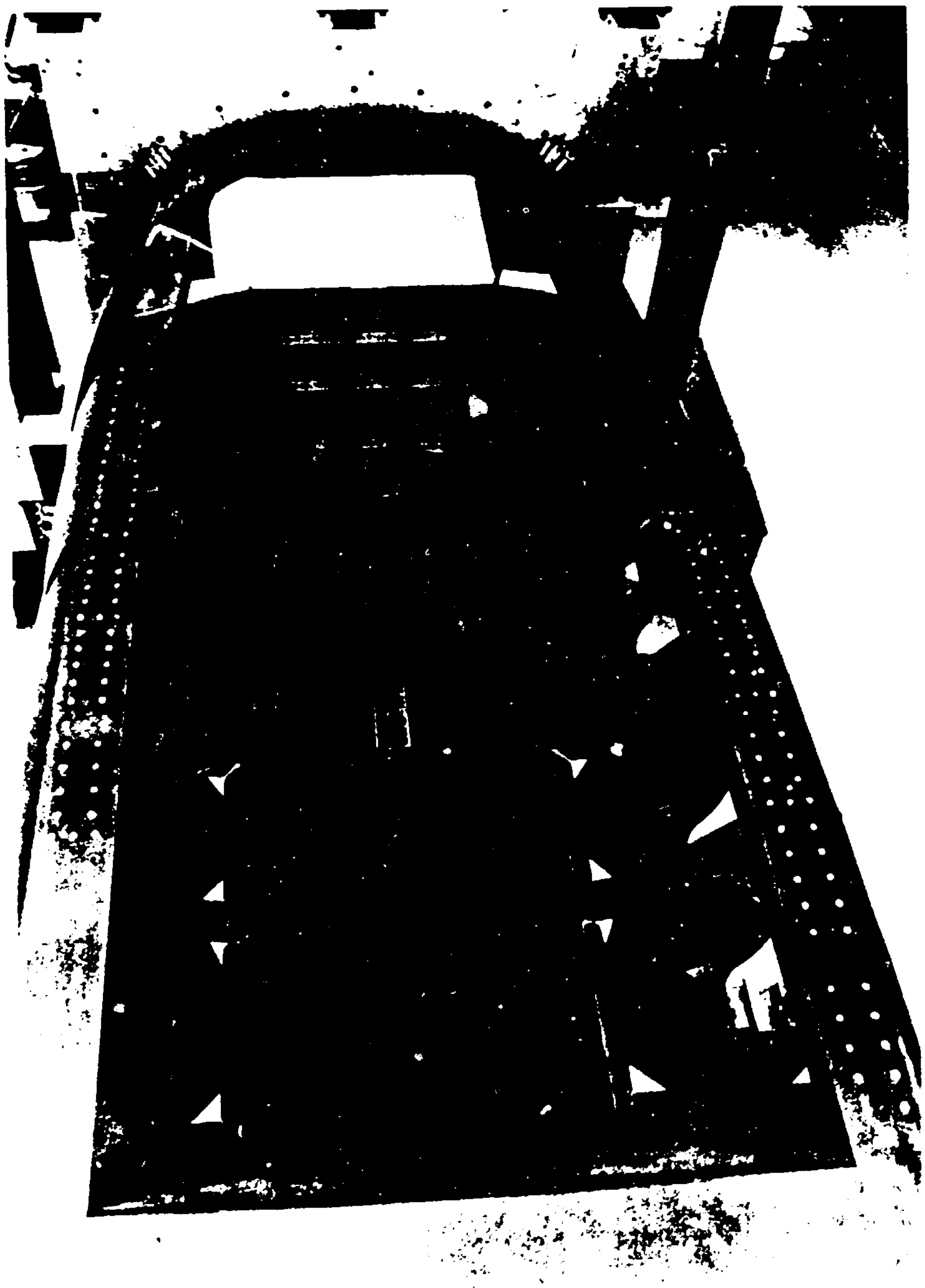


FIGURE 15: COCKPIT CFC STRUCTURE ASSEMBLY

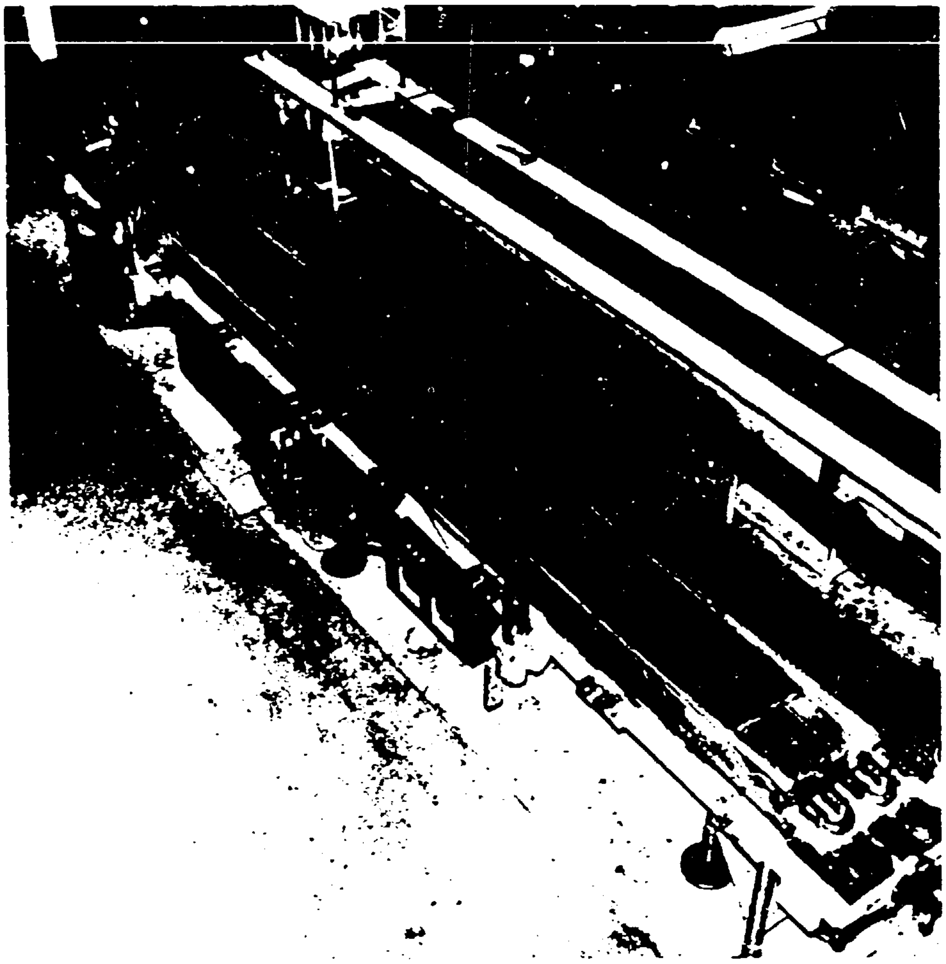


FIGURE 16: ROTORBLADE WITH CFC-TIP IN MOULD

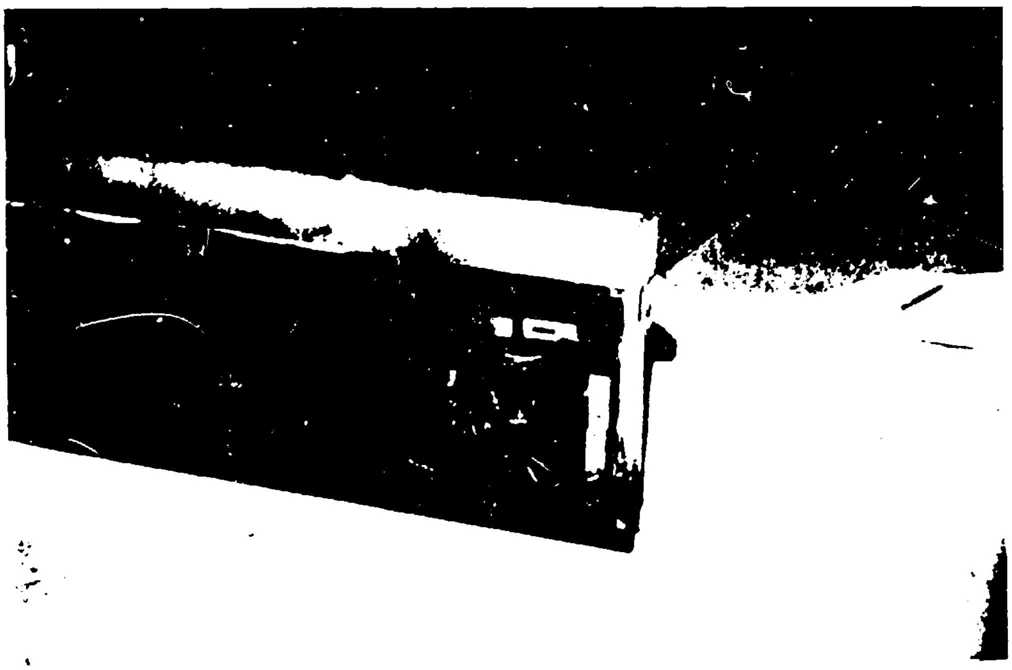


FIGURE 17: LIGHTNING DAMAGE ON ROTORBLADE

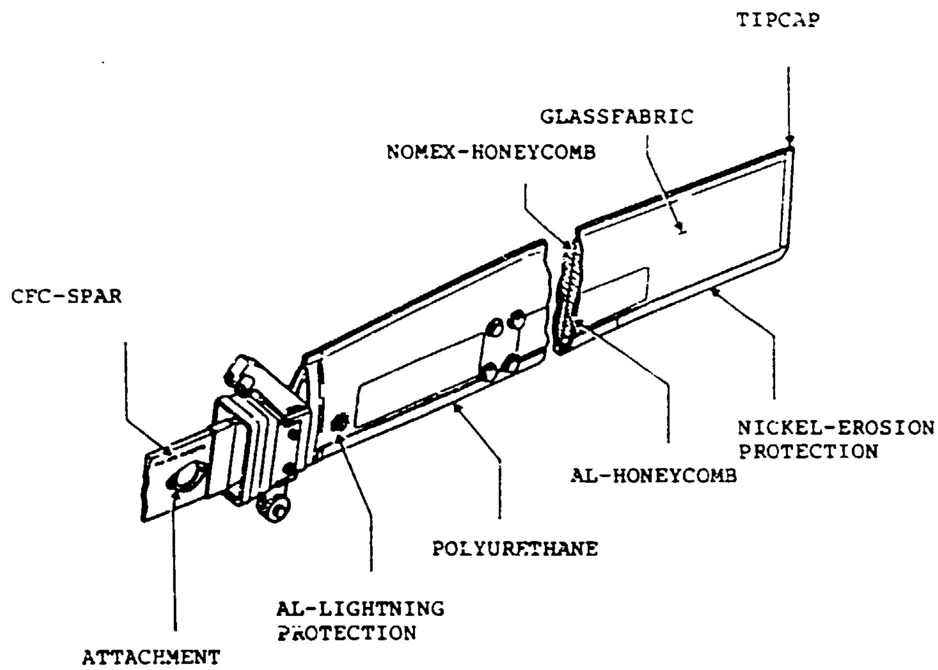


FIGURE 18: BEARINGLESS TAILROTOR

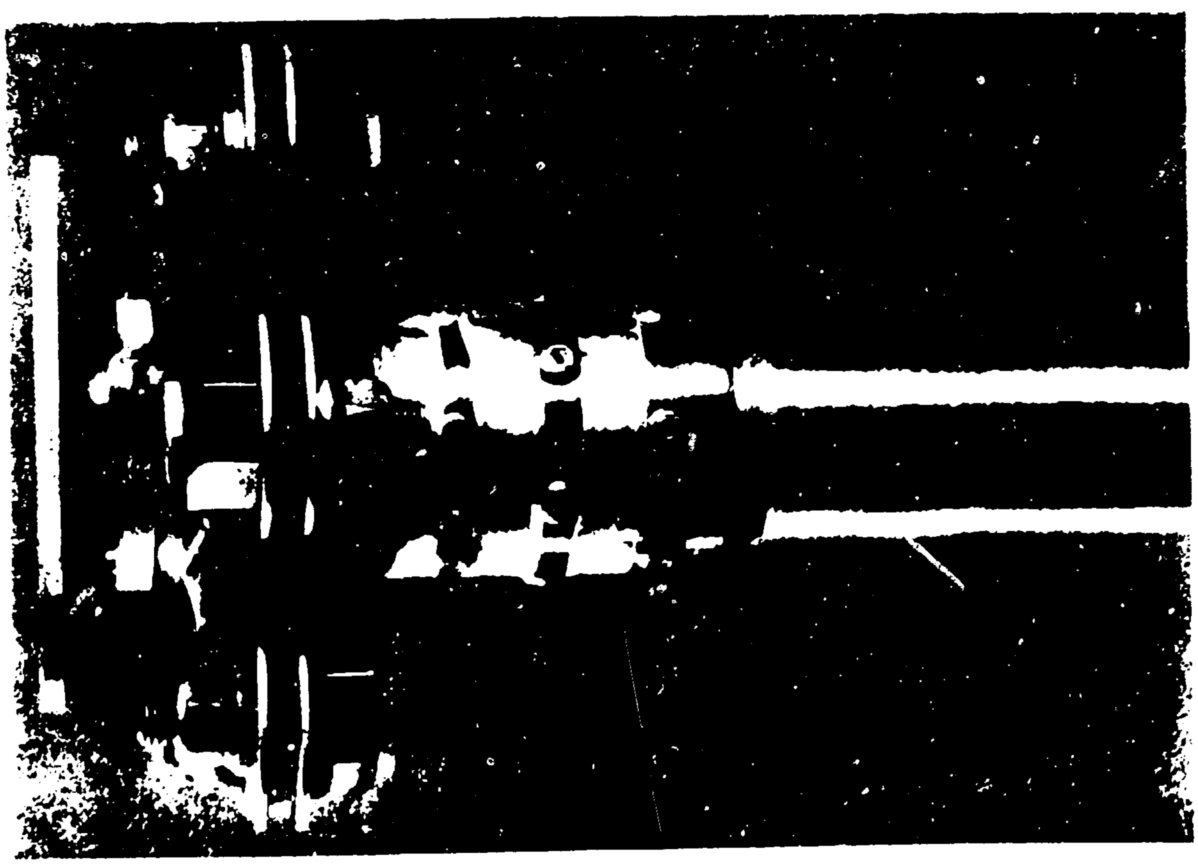


FIGURE 19: METAL VERSION OF DRIVE SHAFT ASSEMBLY

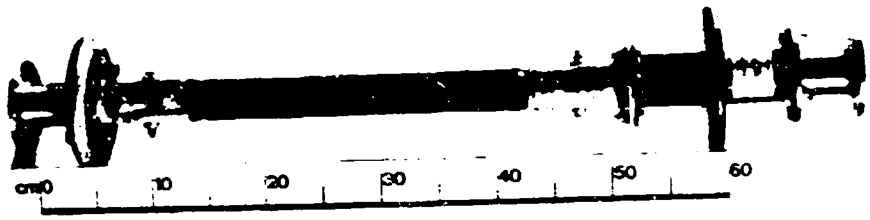


FIGURE 20: COMPOSITE DRIVE SHAFT ASSEMBLY

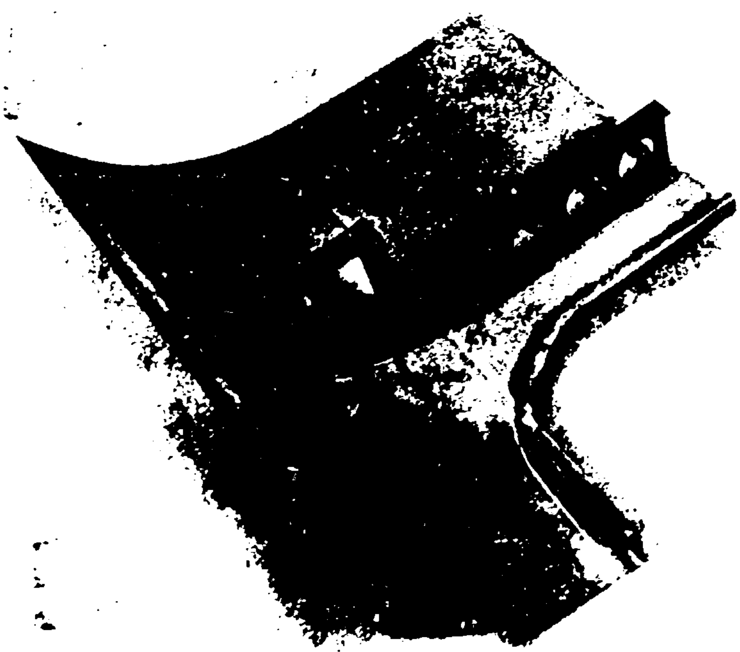


FIGURE 21: COMPOSITE HELICOPTER AIRFRAME SECTION, ONE SHOT CURED

FIGURE 22: HOLOGRAPHIC IMAGE OF CFC ROTORBLADE WITH SKIN DEBOND