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DESIGN AND ANALYSIS OF COMPOSITE HELICOPTER FUSELAGE STRUCTURES

D.MÜLLER, R.MÜLLER, R.PFALLER
MESSERSCHMITT-BÖLKOW-BLOHM GMBH, MUNICH, GERMANY

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DESIGN AND ANALYSIS OF COMPOSITE HELICOPTER FUSELAGE STRUCTURES

D.Müller MBB
R.Müller MBB
R.Pfaller MBB

ABSTRACT

Since composite application is no longer limited to secondary structures and rotor blades, wider use of design and analysis methods as typical for fuselage structures has come into consideration.

Using the dynamic system of an existing light transport helicopter, essential primary structure groups of conventional Al-construction were substituted by composite structures.

The aim was, to demonstrate the possible weight-saving at competitive manufacturing costs.

This paper describes the design and analysis approach and quotes first measured weights and strength test data.

1. INTRODUCTION

The objective of the presented activities is, to investigate various aspects of this method of construction, especially when applied to helicopter structures:

- weight reduction
- reduction of number of parts
- reduction of manufacturing costs
- crash behaviour
- manufacturing and quality
- joining of composite components
- reliability and damage tolerance behaviour

Special problems concerning these fields should be identified, solved or a possible approach should be demonstrated.

The hardware-outcome will be certain fuselage components for various testing purposes, a complete fuselage structure for conducting static strength tests and a fully equipped composite fuselage structure for dynamic testing and in-flight tests.

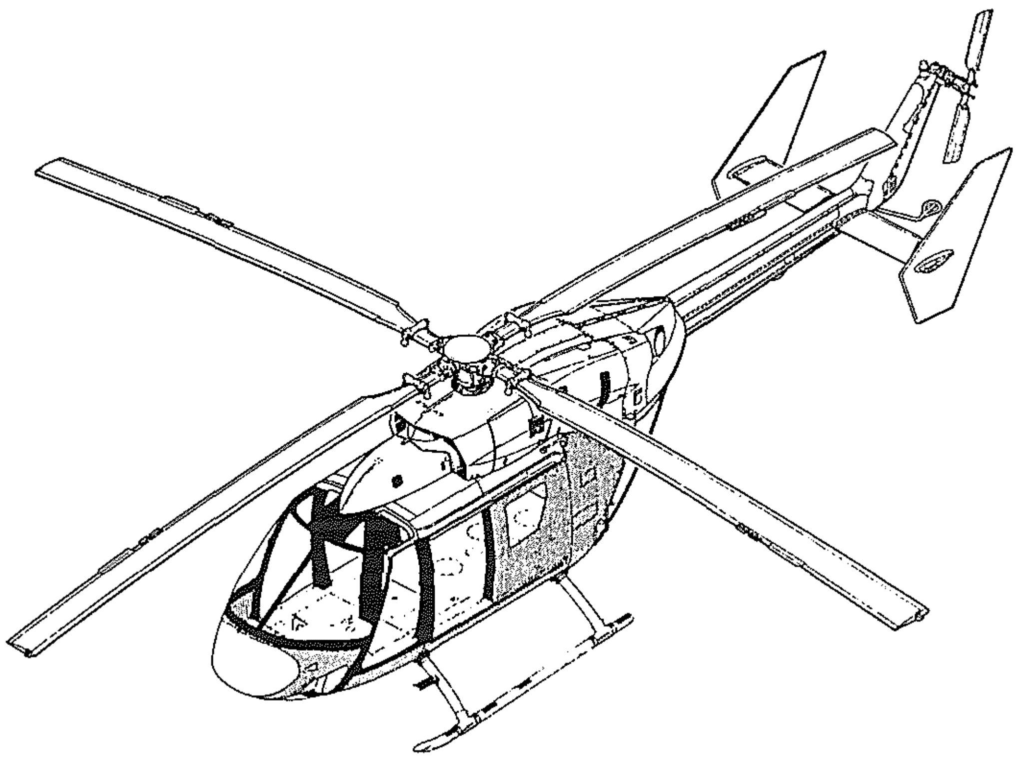


Fig. 1-1: Composite Light Helicopter Fuselage

2. DESIGN

2.1 Arrangement of Major Components

The fuselage of most existing helicopters is a conventional aluminium structure with skin, frames and stringers (Fig. 2-1). Only a few sections, are built in aluminium sandwich. Typically for in-production helicopters cowlings, sliding doors, nose trap doors, aft doors, vertical fin fairings, horizontal stabilizer and end plates are already made of composites.

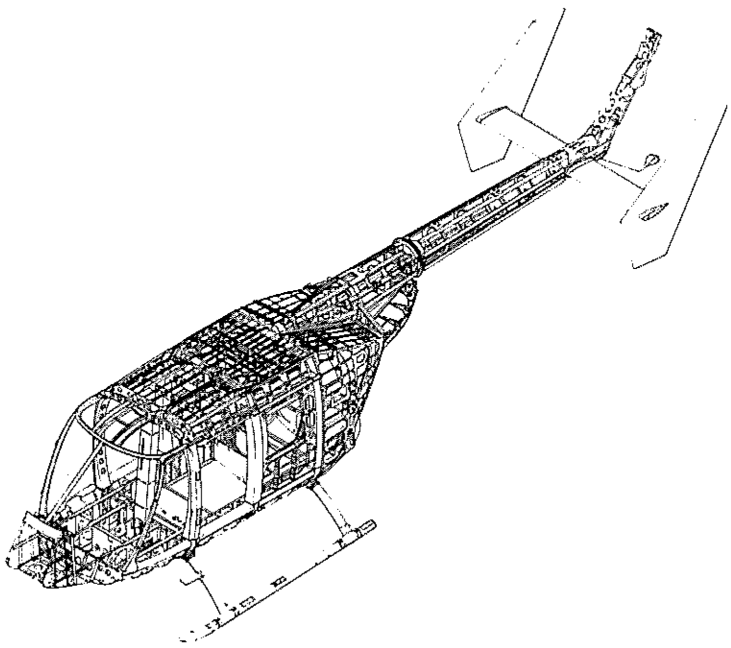


Fig. 2-1: BK 117 metallic airframe

Excluding the transmission deck all parts of the primary structure of an conventional airframe are substituted by composite materials (Fig. 2-2).

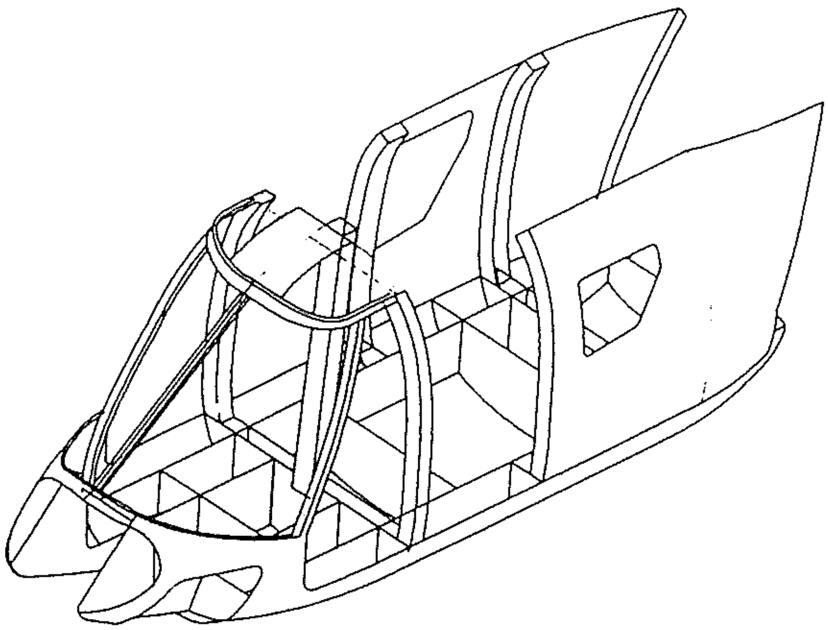


Fig. 2-2: Composite part of the experimental airframe

The sectioning was made under the following assumptions:

- realize large components, built if possible by co-curing frames
- good accessibility
- good mould release
- simple and cost-effective production
- clear load pathes
- simple joints
- extensive preintegration

This leads, as shown in an exploded view fig. 2-3, to the following major components:

a) Cockpit left
b) Cockpit right
c) Subfloor structure } Lower fuselage
d) Bottom shell } assembly
e) Side panel left
f) Side panel right
g) Transmission deck

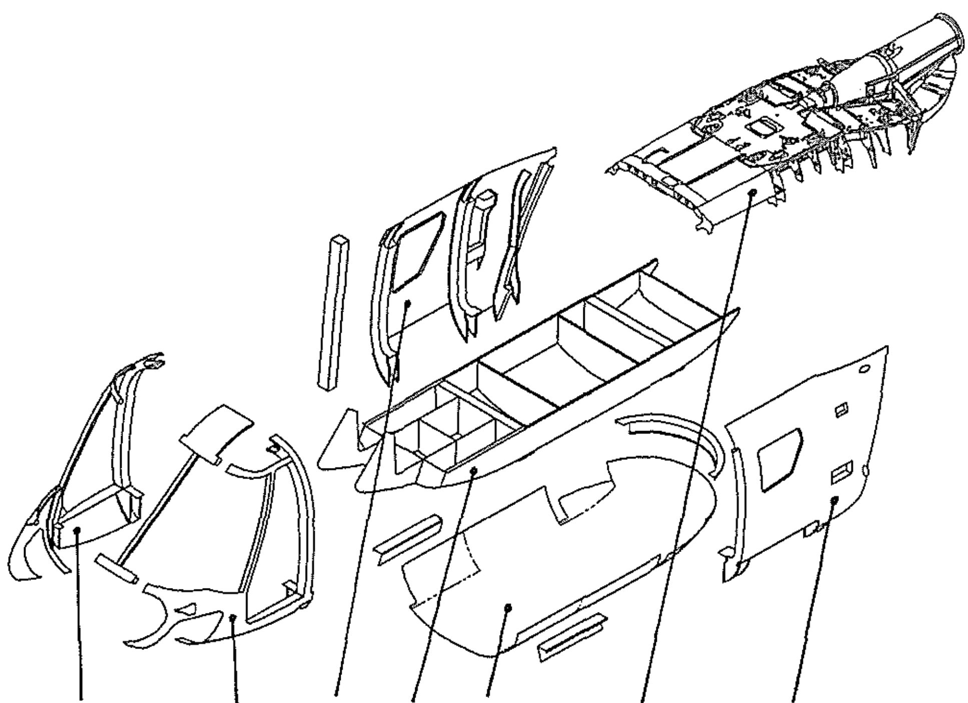


Fig. 2-3: Derivation of major components

Comparing the aluminium airframe to the composite one, the number and the shape of the main ribs and spars are almost the same. The reason of this is the taking over of the original transmission deck, all secondary structures as doors, windows etc., the dynamic system and all other subsystem-components from an existing helicopter. This limited the freedom of designing. A significant difference to a metal airframe is however that nearly all skin-stringer elements are substituted by aramid-NOMEX-core sandwich, or in some cases carbon-NOMEX-core.

Aramid is used where thin skins are necessary, because they promise better impact strength during handling the part than carbon does.

Reinforcement spars and frames are made out of monolithic carbon fibre to obtain high strength and stiffness.

Fig. 2-4 shows the distribution of the airframe's substituted material.

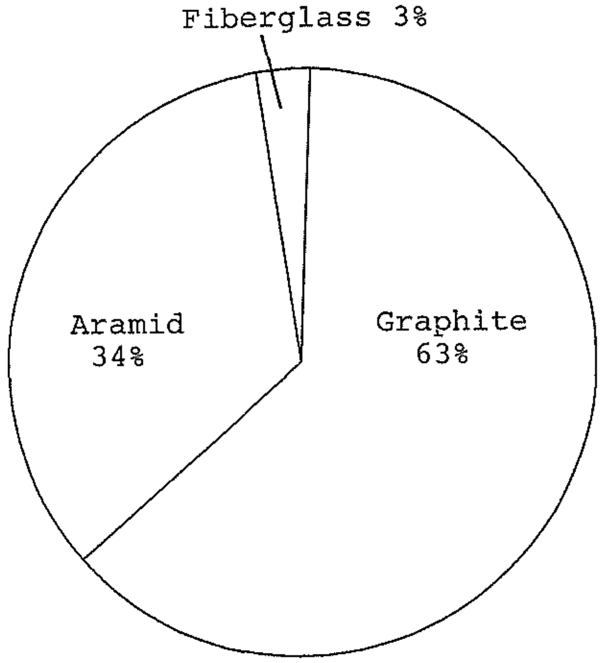


Fig. 2-4: Distribution of composite materials

2.2 Structural Joints

Within the main components many joints are made unnecessary by co-curing components. For example the hat-type spars are integrated in the side panel and in the cockpit by co-curing. The advantages are: fewer parts to be handled, lower manufacturing costs than achievable by bonding or riveting. All sandwiches are cured in one step too. Those parts, which cannot be cured in one step, are bonded and rivetted. The primary task of the adhesive is shimming, because unlike metal structures, composite materials cannot be deformed mechanically to provide tight fitting. Main components are joined by rivetting. Fig. 2-5 shows the principal joining areas and indicates three kinds of fasteners: blind rivets, HI-LOKS and blind bolts.

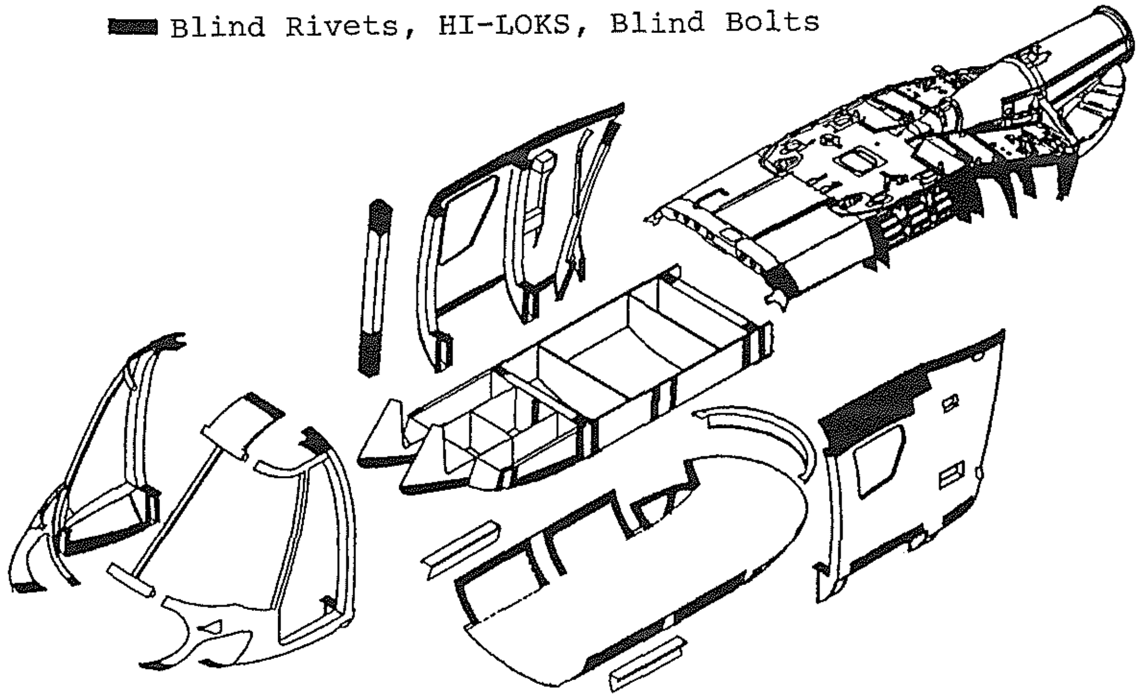


Fig. 2-5: Fasteners for the airframe

2.3 Electrical Effects

The following important aspects had to be regarded:

- earth connection for return conductors and for housings of electrical equipment
- earth for antennae
- screen against HF-fields
- protection against lightning
- protection against electrostatic charge

In order to solve these problems and to provide an appropriate electromagnetic character of the airframe, the side panels and the center post are covered with an alu-mesh (Fig. 2-6).

Additionally ten aluminium wires run from the transmission deck to the floor board and connect these two metallic components. Three of them have contact with the alu mesh layers.

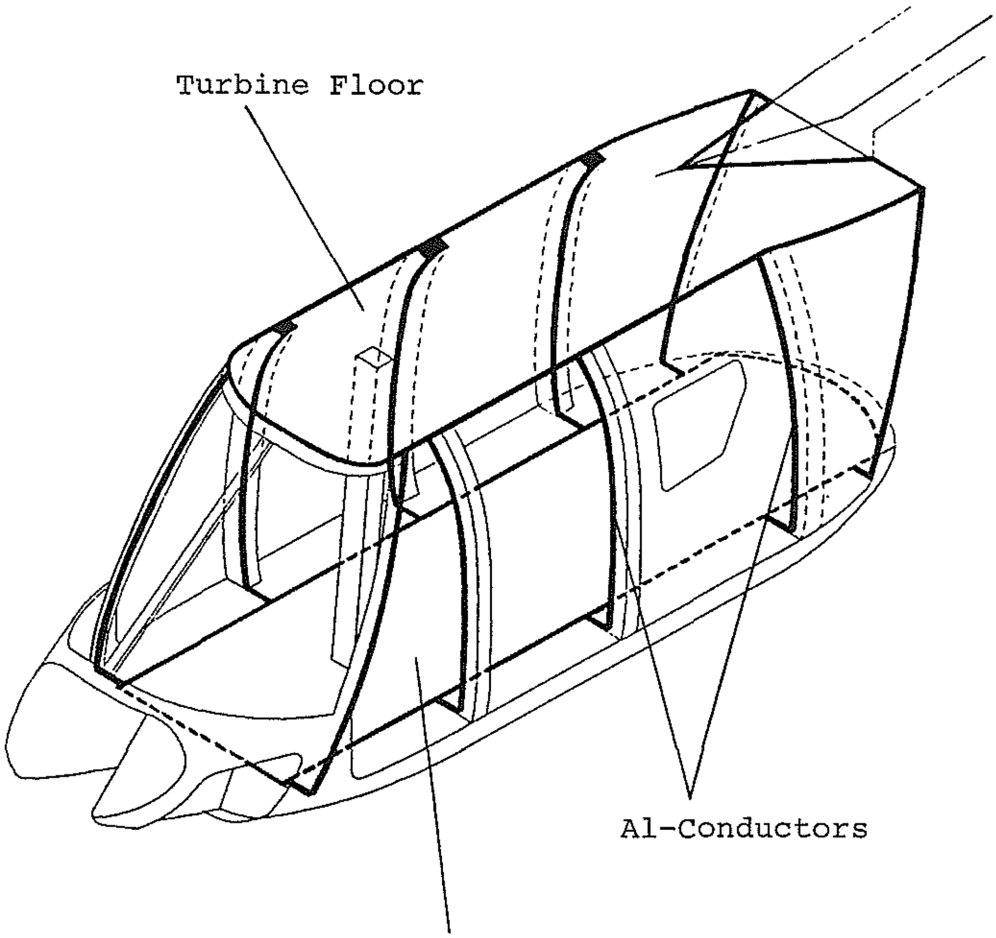


Fig. 2-6: Basic electrical concept

The cabling is arranged in two cable runs.

3. STRESS ANALYSIS

3.1 Finite Element Model

To perform a complete anlysis of the flux of force and the stress distribution within the composite fuselage structure, it was necessary to create a finite element model. Only by this means it appeared possible to successfully obtain the goal of designing an advanced composite fuselage.

The base for a finite element model was already at hand. During the development and testing of the BK 117 aluminium fuselage a FE-model was generated and verified by structural tests. In that period several loadcases were applied to the model and calculations were performed. With respect to basic stiffness and strength requirements and taking care for already existing subsystems which had to be integrated, the preliminary design was developed based on computer runs mentioned above.

The second loop of the design and stress analysis process consisted in generating the composite finite element model. Since the overall geometry did not change, the node coordinates could be adopted for the composite model, likewise the major part of the element mesh. The mesh had to be altered according to new material properties, new material distribution and requirements concerning the accuracy of later calculations.

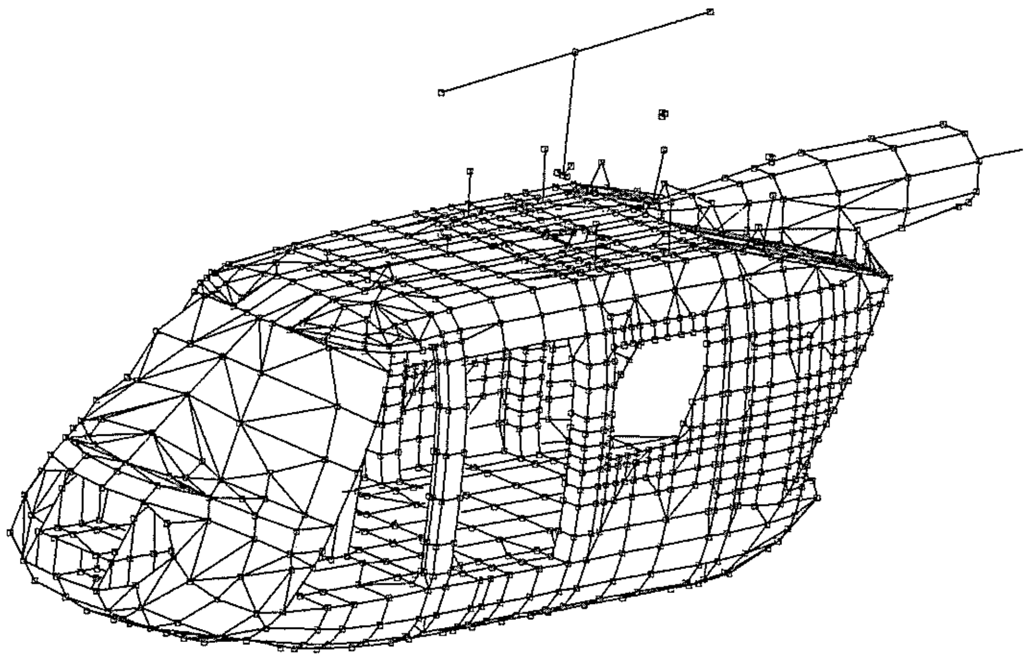


Fig. 3-1: Finite Element Model

To fullfill these requirements within the given schedule and progress of the project, the fuselage-idealization had to be broken down to submodels. These submodels were seperated analogous to the sectioning of the principal components during the design and manufacture process.

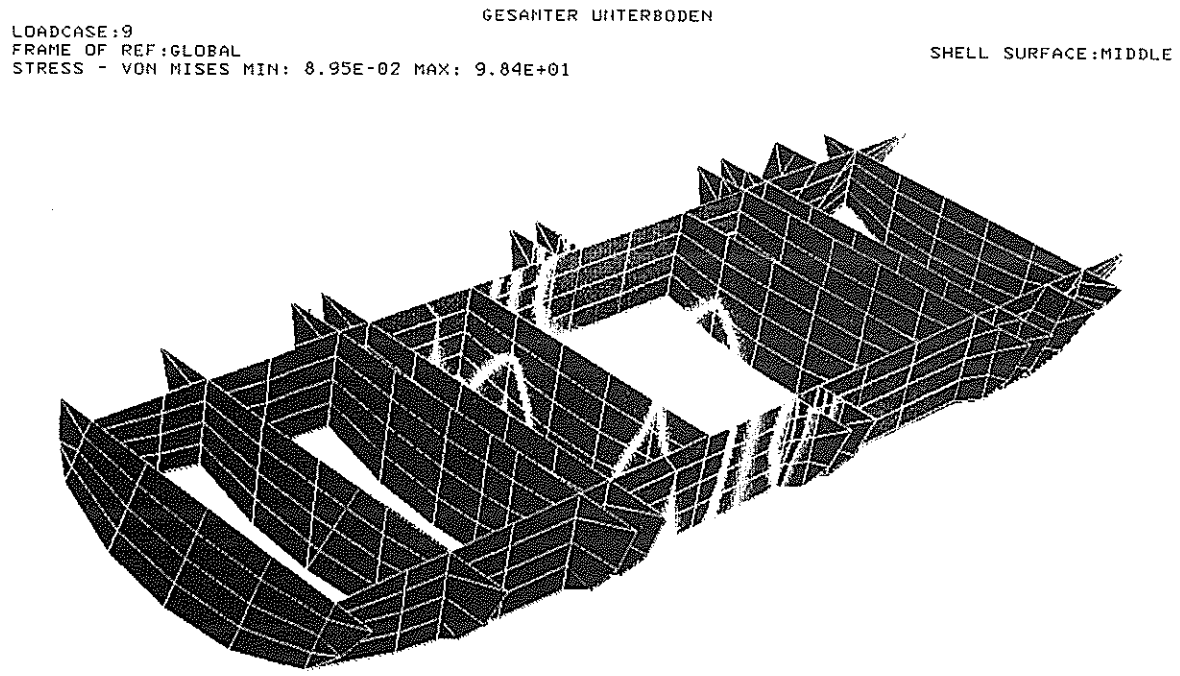


Fig. 3-2: Keel Beams and Spar Elements - Tank loads

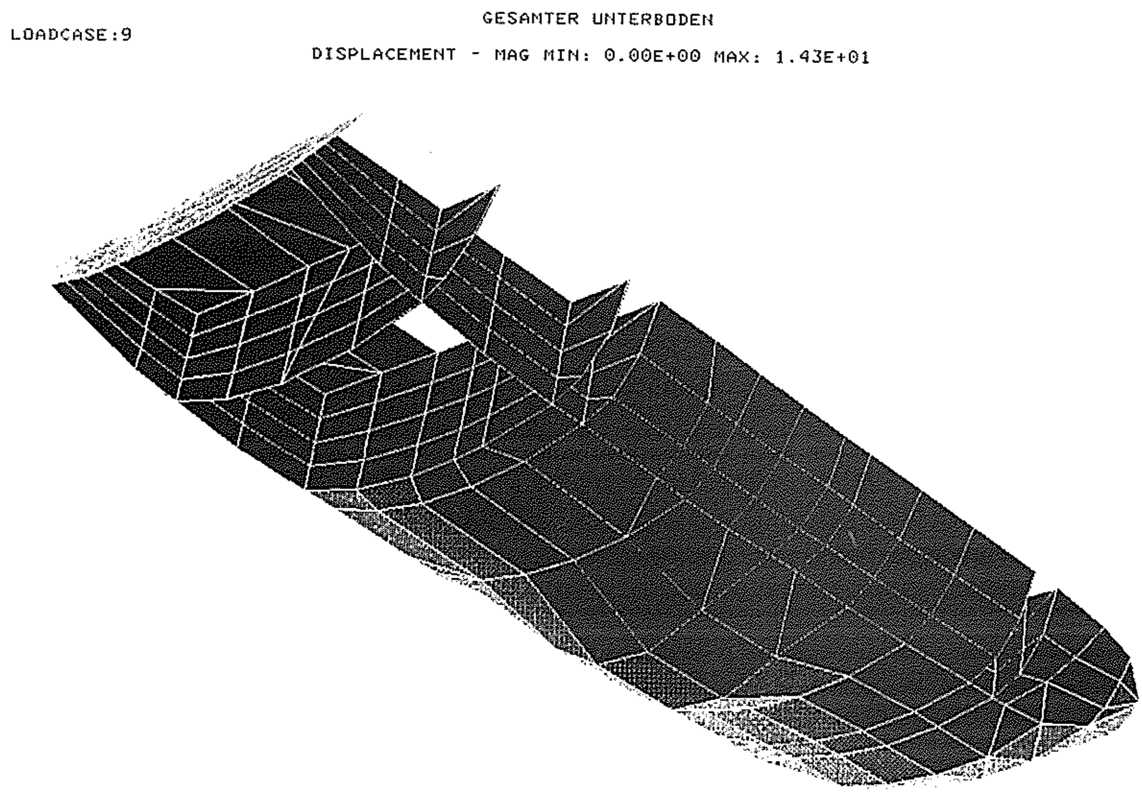


Fig. 3-3: Bottom Panel - Tank loads

Related to the design process and restrictions of the company manufacturing capacity the partial idealization in question had to be modified according to the preliminary design data. The element mesh had to be adjusted, composite properties had to be inserted, FE-computer runs had to be performed. With respect to the resulting stresses and deformations necessary modifications of the design were carried out, the partial idealization was updated. After a final run the design was frozen.

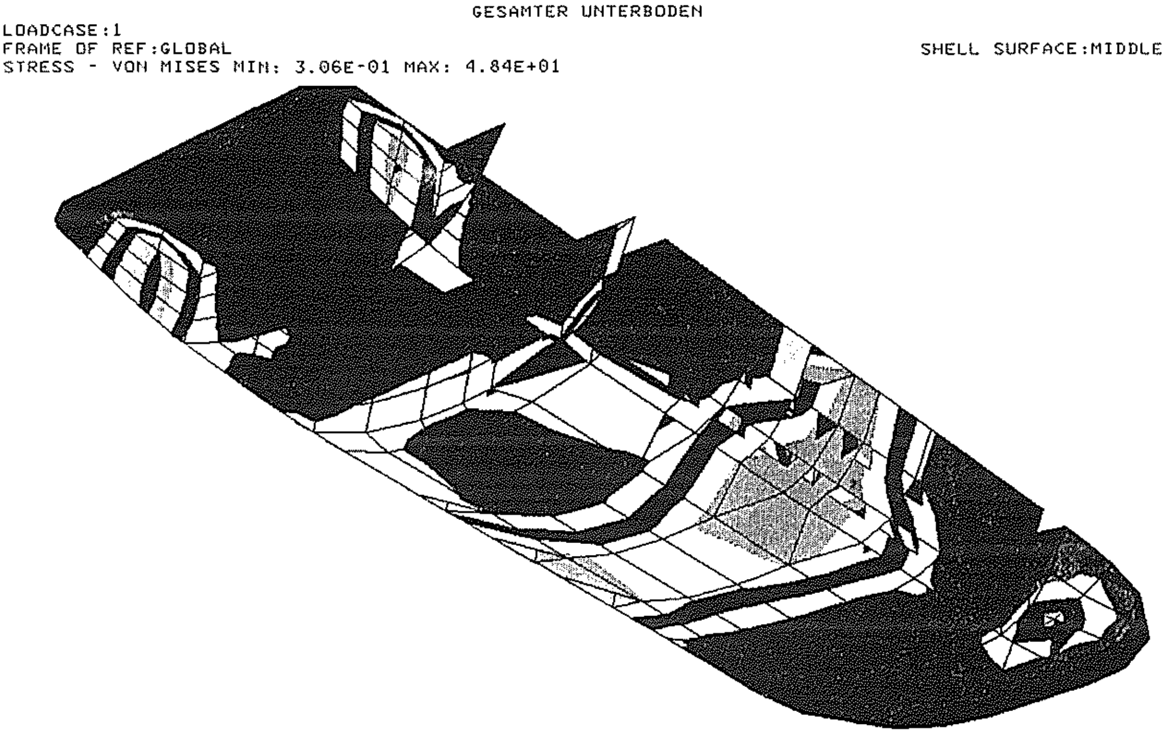


Fig. 3-4: Bottom Panel - Stress Distribution

Certain parts of the model representing the metal components of the fuselage to be included into the composite design could be integrated into the new model. These model-parts were the transmission deck and the cabin-floor. Assembled together with the new composite-submodels resulted in an idealization of the complete composite fuselage structure.

The complete FE-model consists of
- 2004 nodes
- 4122 elements
  - o 1627 beam type elements
  - o 2495 shell type elements

For our purpose linear element types and a linear FE-analysis was sufficient. The finite element code used is NASTRAN release 64 on an IBM. Pre- and Postprocessing was done by SUPERTAB and CAEDS.

For a better presentation of results (stress distribution and displacements) the complete model was broken down to smaller groups, representing interesting structural parts of the fuselage.

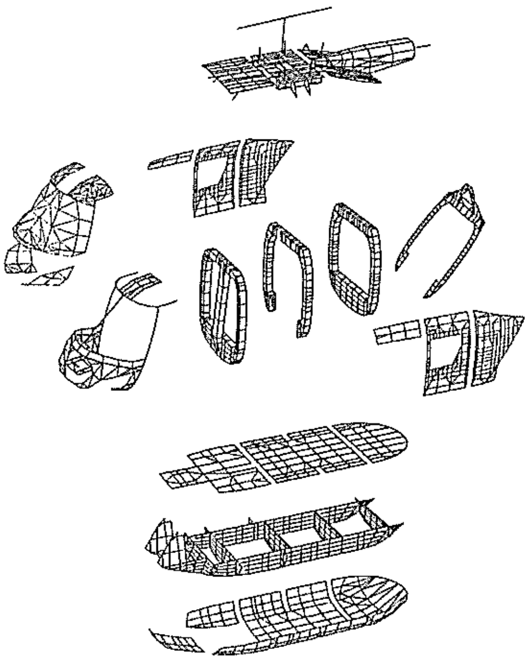


Fig. 3-5: Break down of Finite Element Model

Out of several loadcases according to FAR Part 29 the following load conditions with critical stress distrubutions were selected and applied to the composite-model:

- level landing with drag
- pull up to 3,5g

The external forces, resulting from these loadcases, had already been known out of the BK 117 development and were applied to the composite FE-model.

3.2 Structural Strength and Stability

During the development process of the composite fuselage structure several design versions were run as FE-models. Based on these results the strength and stability analysis was performed.

During the preliminary design phase a rough checking of the strength and stability criteria was done. During the actual design phase, when the number of plies, orientation, type of fabric and the fibre material was varied, the decisive failure criteria were applied.

In general the analysis had to be done for anisotropic, especially orthotropic composites, some of them nonsymmetric. Certain sandwich-panels were not critical regarding strength or stability, but had to be designed with respect to handling resistance.

Stress distribution roughly depends on
- number of plies
- fibre type
- ply thickness
- ply orientation
- per cent by volume of fibre.

Strength was checked for by applying failure criteria for unidirectional composites, which include the possibility of fibre or matrix failure.

The ultimate strength values are based on experimentally found data of unidirectional and fabric coupon tests, which were performed for the principally used carbon- and kevlar fibre composites:
- longitudinal tensile strength
- longitudinal compressive strength
- transvere tensile strength
- transvere compressive strength
- longitudinal sheer strength.

The measured unidirectional strength data are modified by various factors, taking into account the following influences:
- fabric type
  The max. strength and E-modulus of fabrics differ from these of unidirectional plies. This is considered by applying a fabric specific factor.

- environmental conditions
  The experimentally found decrease of stiffness and strength is taken into account by climate dependant factors for moisture and temperature conditions.

Local stability problems mainly occur on sandwich panels. Stress concentrations were checked with respect to local face-sheet stiffness and core quality according to local stability criteria:
- shear crimping
- face wrinkling
- intracellular buckling

In most cases the face wrinkling criteria was the decisive one.

Global stability criteria were checked for all subcomponents of the structure. General plate buckling criteria, due to compression or/and shear, applied to large sandwich-panels as well as to smaller monolithic panels. All frame elements were checked for buckling, too.

The required safety factors were applied to all analysed structural loads.

4. QUALITY ASSURANCE

The use of composites requires some new methods of quality assurance to verify the manufacture of a reliable, reproducible product.

This begins with the row material acceptance and life cycle control of the prepreg. Also laminating operations and curing had to be controlled.

All moulds for main components are measured exactly after each cure cycle.

To examine "spring-back" the components are measured additionally.

To ensure static behaviour the primary non-destructive inspection method is ultrasonic inspection, especially puls-echo techniques.

During the development of parts experimental types are made first and cut into pieces for computer tomography. This is a good method to determine defect joints and cracks without destruction. Fig. 4-1 shows a part examined with computer tomography. In Fig. 4-2 the more bright lines show the aluminium-filled adhesive. In this lines the dark point is a defect joint wich is scaled-up in the right half.

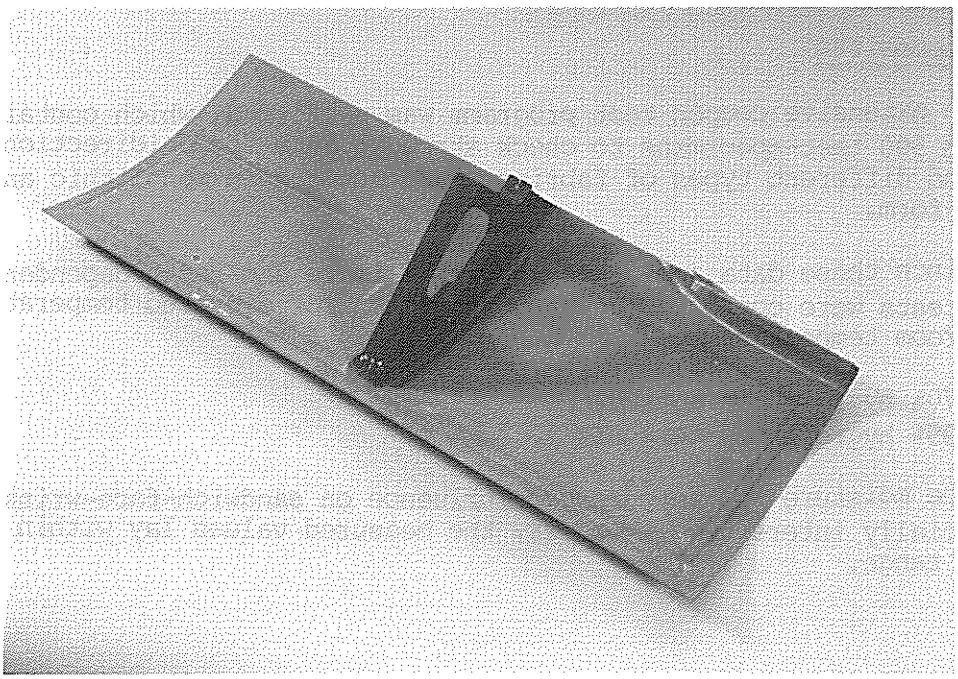


Fig. 4-1: Sandwichpanel bonded to CFC frame

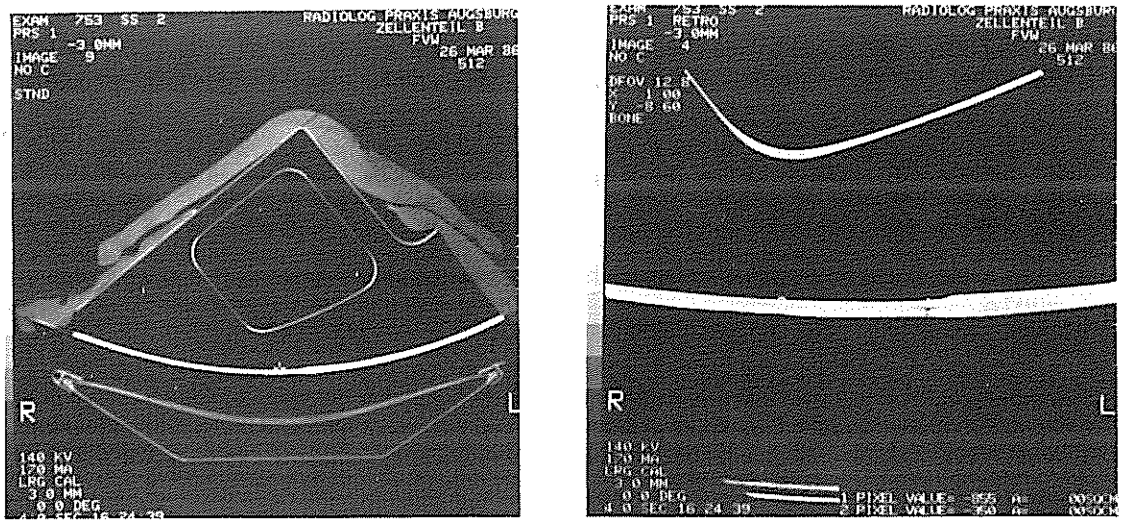


Fig. 4-2: Bonded joint frame-sandwich

5. TESTING

5.1 Coupon Tests

5.1.1 Composites

An extensive coupon test programm was performed, which contained more than 500 test-specimens to check strength data for unidirectional layers and different types of fabrics, manufactured in monolithic and sandwich form.

Since these data appear to be very climate and resin dependent, certain tests were conducted under moisture and/or elevated temperature:

- 20°C (room temperature)
- 70°C
- 70°C, 70% hygroscopic moisture
- 70°C, 95% hygroscopic moisture

The moisture and temperature influence on sandwich-face-wrinkling was especially interesting. However, the measured values lay within the expected range.

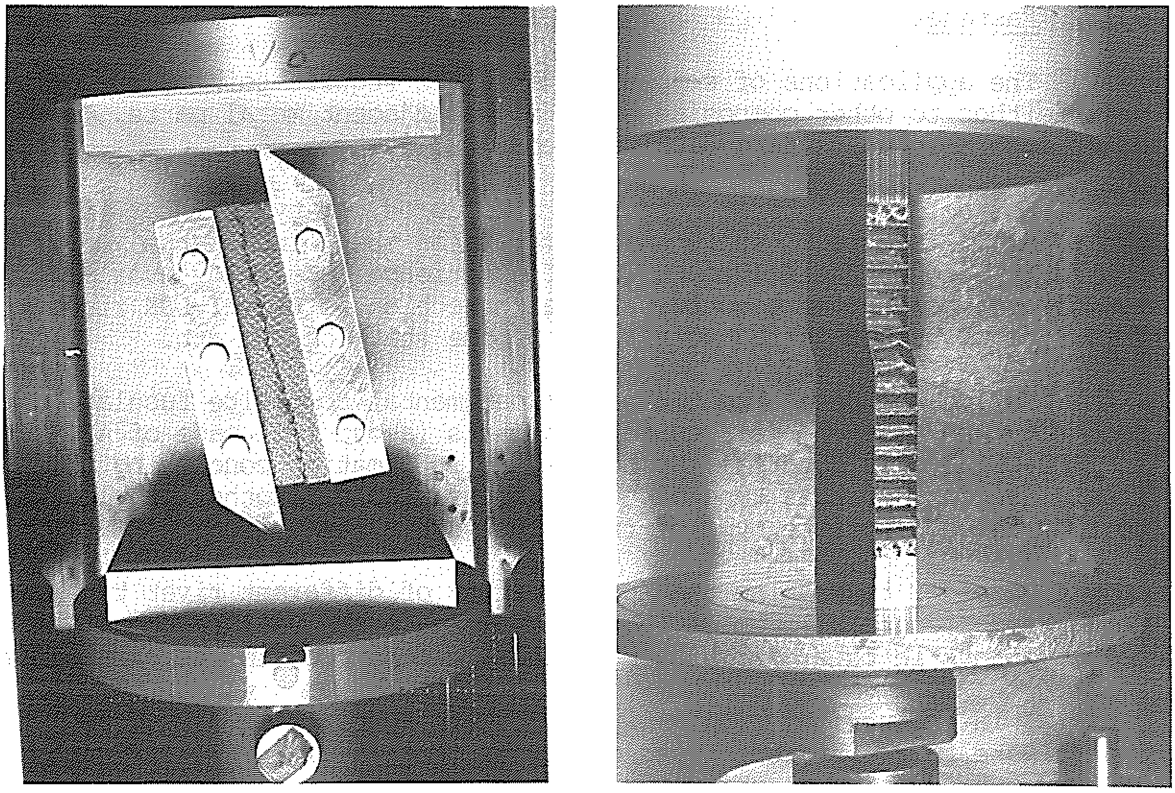


Fig. 5-1: Composite sandwich samples: in-plane shear and compression

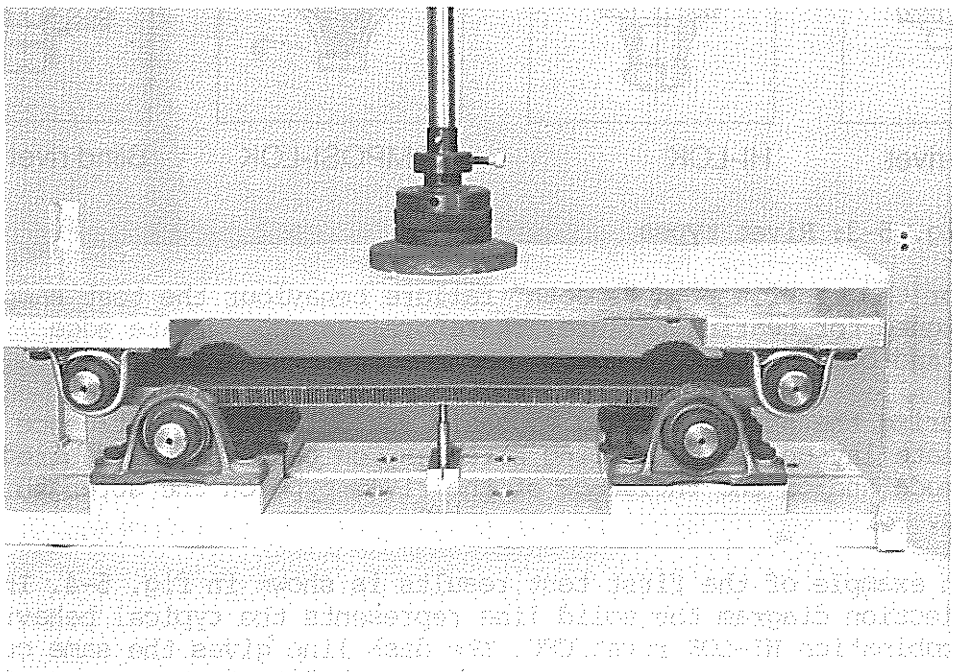


Fig. 5-2: Composite sandwich samples: Bending

5.1.2 Fasteners

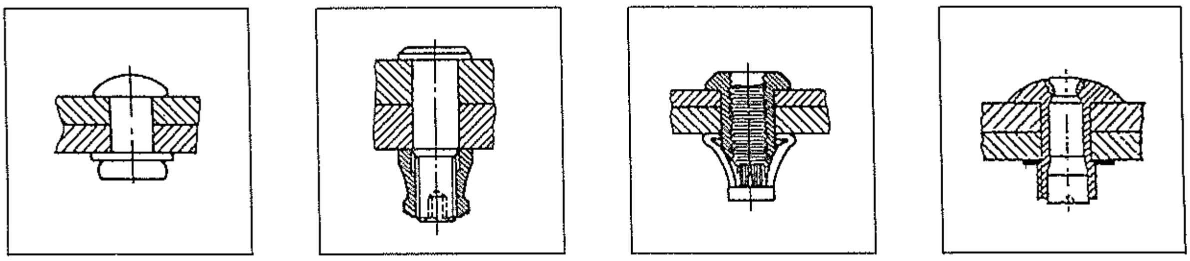
The applications of composite materials require joining composite parts either to other composite parts or to adjacent metal parts. In many cases such joints are formed using mechanical fasteners.

For bolted joints in composite structures, as typical for fighter and transport aircraft, where the thickness of the joined parts varies typically between 2 and 10mm, a lot of experience is already available. A typical standard rivet used for such a connection is the HI-LOK rivet. This type of bolted joint has already become state of the art.

During the development of the experimental composite fuselage the necessity has risen to join quite thin parts (about 1mm) made of AFC, CFC and AFC-CFC hybrid material. Up to now, no information concerning the special aspects of this type of joint was available. Another problem was, to find the appropriate type of rivet for such a connection.

In order to examine the applicability of different rivet types, to determine strength allowables and to investigate the failure behaviour of bolted joints in this composite materials, a test program was carried out.

Thin (about 1mm) single-lap joint test specimen were used to investigate four different types of rivets (Fig. 5-3) in AFC, CFC and AFC-CFC hybrid material under various environmental conditions.



Universal rivet | HI-LOK | COMPOSI-LOK | Blind rivet

Fig. 5-3: Rivet types

For the benefit of a "smooth" failure behaviour the test samples were designed for bearing failure. Therefore, an appropriate edge distance and bolt pitch were used.

During the static testing emphasis was laid on the influence of the laminate lay-up on the bearing strength, on checking the desired failure mode and on the influence of the rivet type on the strength of the joint.

An example of the first test results is shown in fig. 5-4. In the load-deflection diagram the solid line represents the typical behaviour of the combination HI-LOK rivet/CFC. The dash line gives the same curve for a sample were the test was stopped at about 90% of the average final failure deflection. The photograph in fig. 5-4 shows a cross section of this joint.

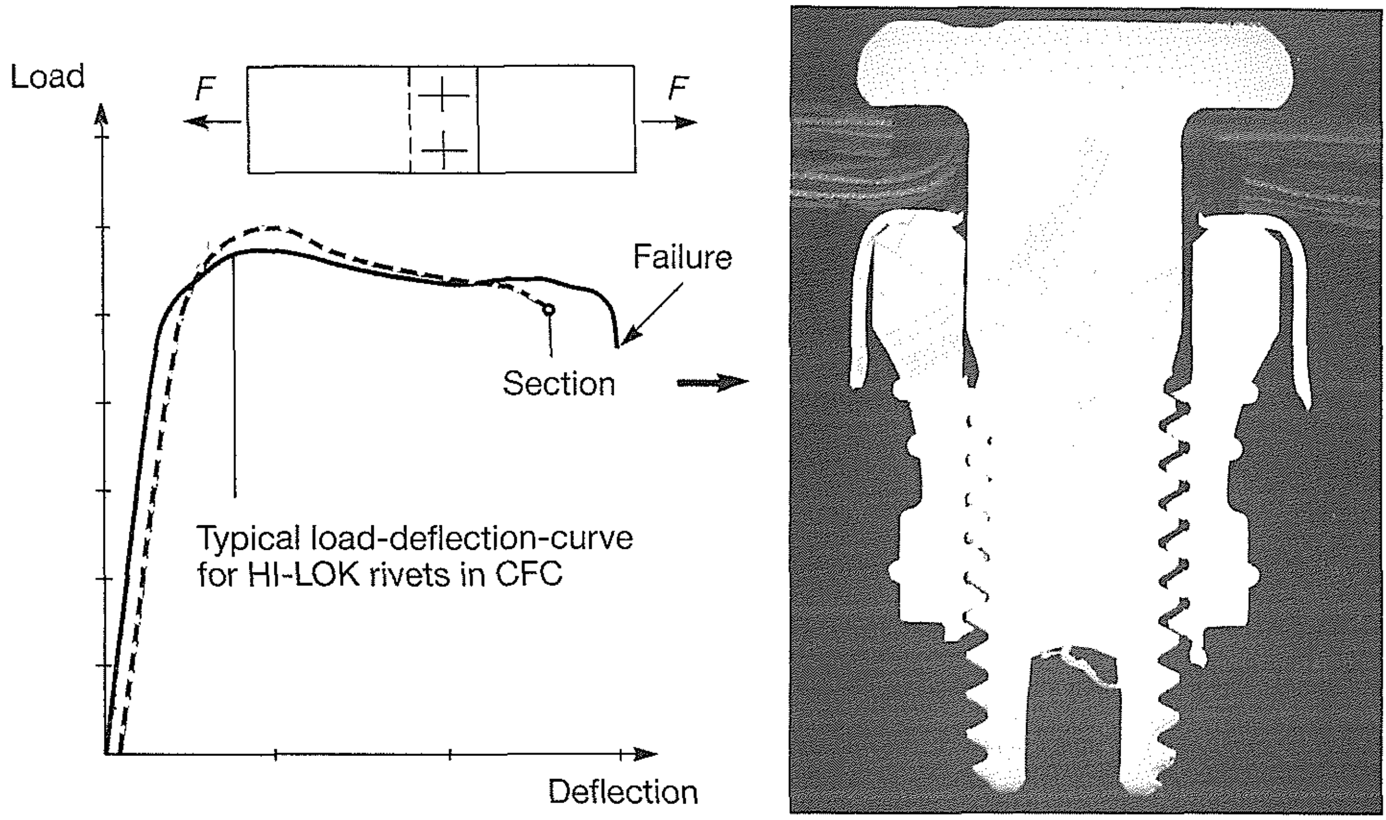


Fig. 5-4: Bearing failure (HI-LOK rivet in CFC)

First test results indicate, that the static strength of a riveted joint in fibre reinforced plastics (AFC and CFC) is approximately at the same level as for a similar joint in an aluminium structure, even at elevated temperature and moisture.

5.2 Component Tests

For the verification of the theoretical work carried out a component test was performed (in cooperation with IABG in Ottobrunn).

For this test a side panel was selected because this has proven to be the best choice with regard to manufacturing aspects, the test procedure and the costs. The whole test piece (see fig. 5-5) consisted of the composite side panel, a part of a metal transmission deck and a piece of subfloor structure.

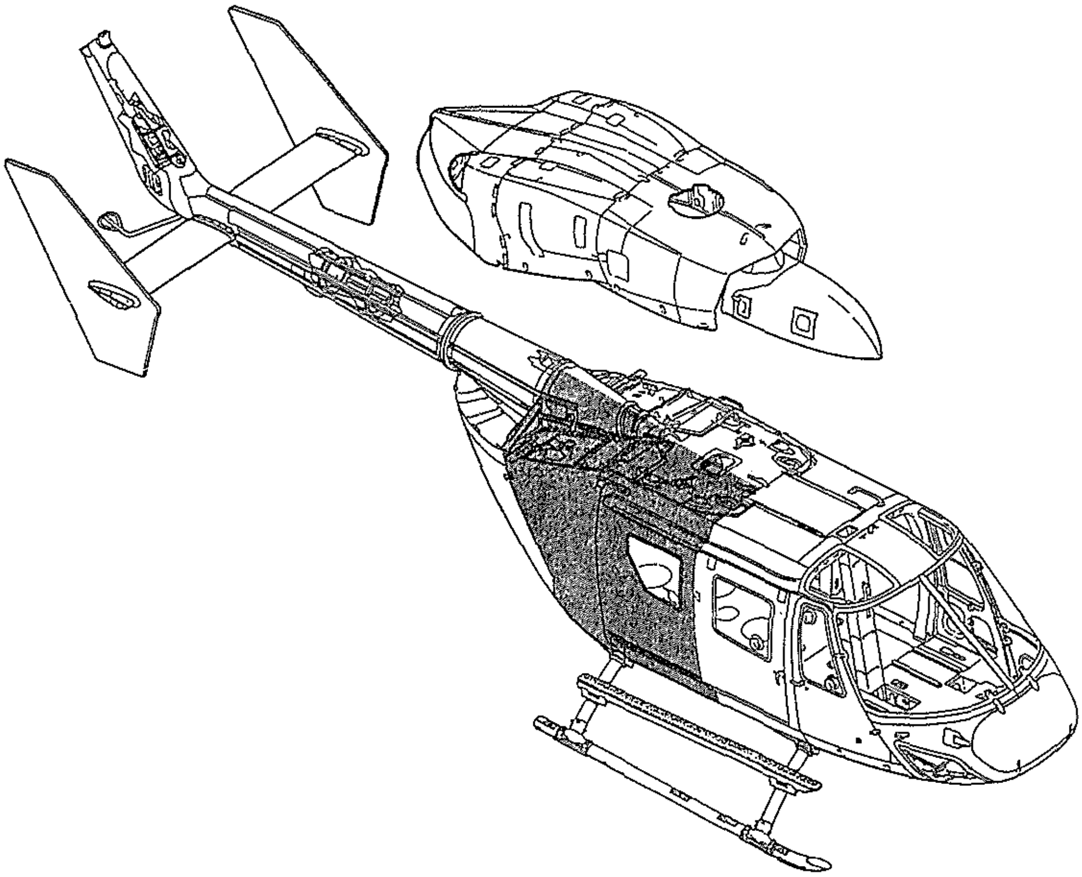


Fig. 5-5: Test assembly for component test

The transmission deck and the subfloor structure (dummy structure) were used to provide realistic boundary conditions for the side panel during the load tests. Both of them ended at the symmetrical axis of the fuselage (Buttock line 0). Only symmetric load cases should be investigated in the component tests. So the appropriate supporting conditions were required at the symmetrical axis of the helicopter fuselage. The boundary conditions for the test assembly are shown in fig. 5-6.

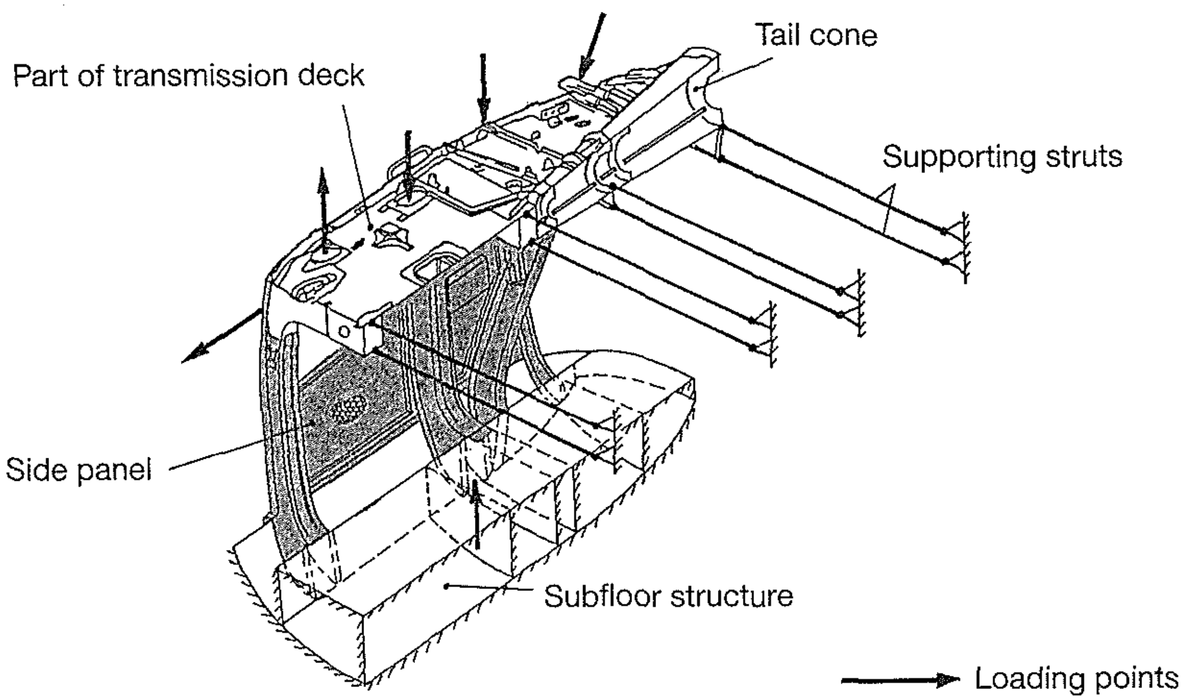


Fig. 5-6: Schematic view of the component test set-up

The subfloor dummy structure was attached to a test fixture. The transmission deck was supported at each frame with a pair of supporting struts. This kind of support for the transmission deck provides only a reaction of in-plane moments and end loads.

With the load tests the ultimate strength and the stiffness of the composite side panel incl. the load introduction points of the transmission, engine, tailboom and landing gear were tested.

Also the riveted joints between the composite side panel and the transmission deck as well as the subfloor structure were checked. For these tests the side panel was equipped with about 170 strain gages and 6 hydraulic cylinders were used for applying the loads. Fig. 5-7 shows the side panel prepared for the tests.

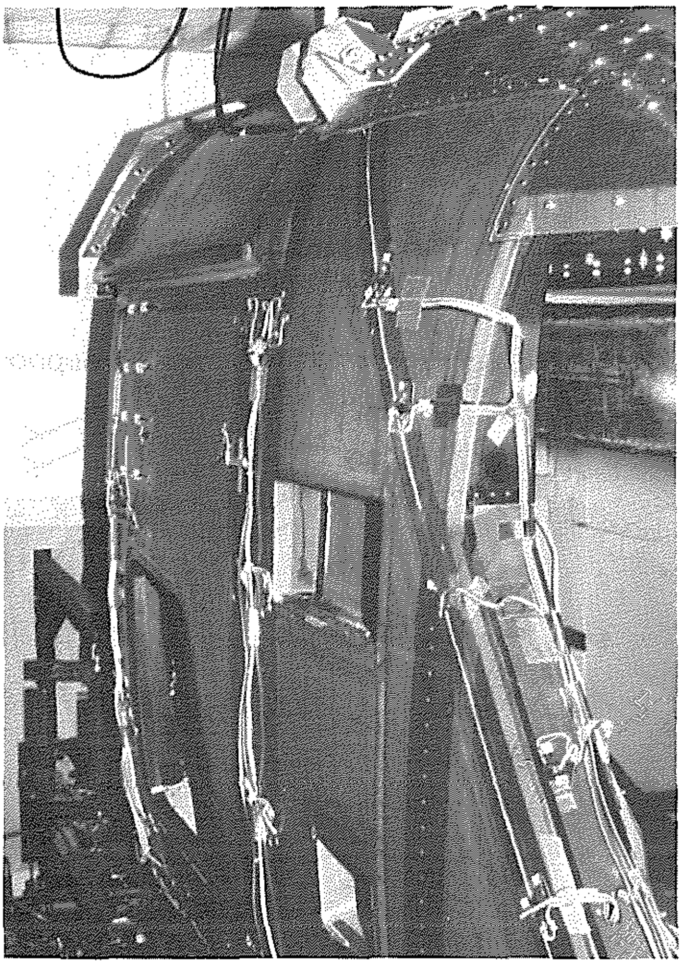


Fig. 5-7: Side panel prepared for load tests

In the side panel tests the component was loaded according to various critical loading conditions for this area, e.g. a flight load case ("Pull-up to 3.5g" - FAR 29.337) and a landing load case ("Level landing with drag" - FAR 29.501(c)). First the test piece was loaded up to limit load for all loading conditions and then up to failure at 70°C. For simplification of the test procedure the influence of moisture was neglected in these tests. Failure occured always beyond ultimate load (Limit load x 1.5).

The tested side panel is able to carry all the required loads. Furthermore, an examination of the results has shown, that the failure loads and failure modes had been predicted correctly by the stress analysis.

6. CRASH BEHAVIOUR

For new helicopters design aspects concerning the surviveability of the crew during accidents become more and more important. One generally regarded basis for the crashworthy design of helicopters are the requirements according to MIL-STD-1290.

During the development some basic work for designing crashworthy helicopter subfloor structures was performed (in cooperation with the DFVLR in Stuttgart).

The design of the composite subfloor structure was limited because all the equipment (e.g. the tanks), the landing gear and the flight control system had to be taken from an existing helicopter. During the development of the composite fuselage the crash-critical components of the subfloor structure were identified (Fig. 6-1).

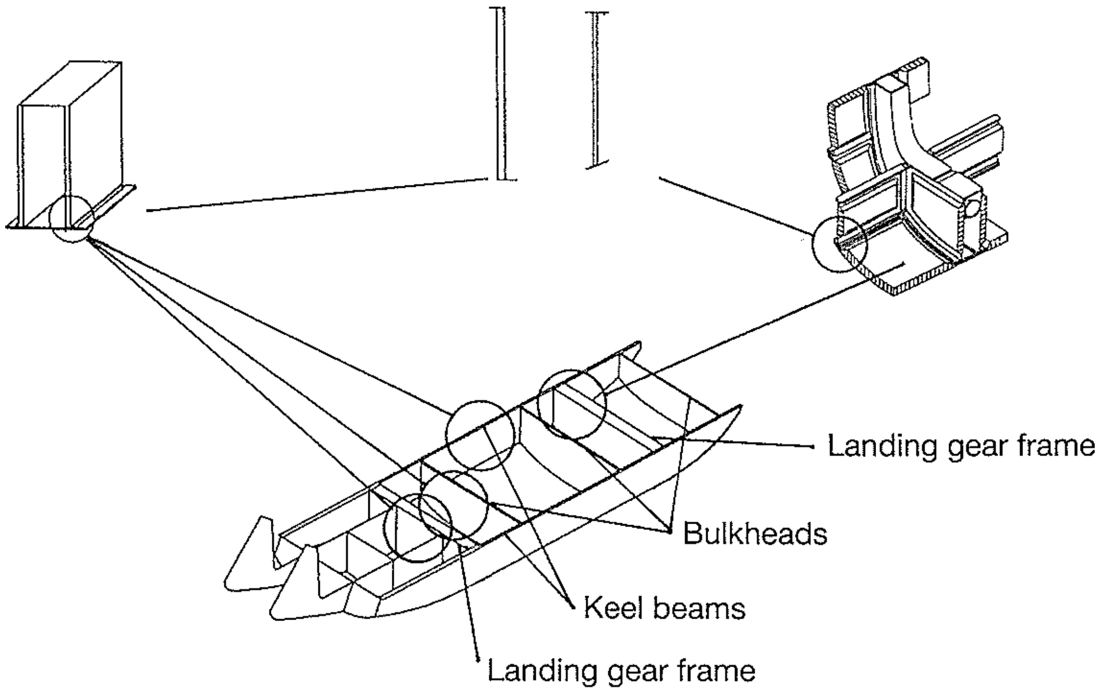


Fig. 6-1: Crash-critical areas of the subfloor structure

Quasi-static and dynamic crushing tests with components from the crash-critical areas of the composite subfloor structure were performed in order to compare the energy absorption capability of the composite structure with the one of the basic version with regard to future projects.

First test results indicate, that the energy absorption capability of a sandwich design (FRP-facing or Al-facing, NOMEX-core), as used in the subfloor structure of experimental fuselages, is generally not very high. To improve this behaviour, the sandwich parts need to be modified, e.g. with additional monolithic stringers (which means also additional weight) and special trigger mechanisms for initiating a controlled crushing.

To meet the requirements of specifications like the MIL-STD-1290 would require a different design than the one used in the experimental composite fuselage.

7. REDUCTION OF WEIGHT AND NUMBER OF PARTS

A great advantage of composites is the reduction of weight and cost.

Up to now there are just components for two airframes built and cost savings cannot be determined exactly. As a reference point for cost savings the reduction of parts is listed. There are only hard parts (for example core blanks and cured components) and no single prepreglayers listed in the table. As fig. 7-1 shows, part numbers are reduced significantly. Rivets, screws etc. are not counted but certainly reduced in the same way.

The weight-saving is only analytically determined but in the case of the left and right side panel the completed parts confirmed the calculation after being weighed.

	Weight			Number of Parts		
	Typ. Al Airframe	Composite Airframe	Saving	Typ. Al Airframe	Composite Airframe	Saving
Cockpit left	9,7kg	6,6kg	32%	125	29	77%
Cockpit right	9,7kg	6,8kg	30%			
Lower fuselage structure	26,4kg	18,5kg	30%	283	29	90%
Lower fuselage skin	18,4kg	12,0kg	35%	79	17	79%
Side panel left	15,1kg	10,2kg	32%	117	10	92%
Side panel right	15,0kg	10,2kg	32%	117	11	91%
	94,3kg	64,3kg	32%	721	96	87%
		66,7kg*	29%			

* with Additional weight to avoid electrical effects (2,4kg).

Fig. 7-1: Comparison of weight and Number of Parts

CONCLUSIONS

This programm allows to compare a conventional aluminium fuselage structure and an advanced composite fuselage structure. Effects concerning development, manufacturing and in-flight operation are studied. Since all other systems of the helicopter are identical to a conventional, existing helicopter, the results will show some typical characteristics for introducing a composite helicopter fuselage.

The experience gained during the research and development done with this program will be transferable to future helicopter projects, like MBB-specific developments as well as international cooperations e.g. PAH2/HAP/HAC3, NH 90 or EUROFAR.