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THE DEVELOPMENT OF A COMPOSITE HELICOPTER FUSELAGE AS EXEMPLIFIED ON THE BK 117

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THE DEVELOPMENT OF A COMPOSITE HELICOPTER FUSELAGE
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ABSTRACT

The aim of this paper is to present the development of a composite fuselage.

This technology-program, founded by the german ministery of defence, should shwo, that fibre reinforced materials can also be used in primary structures, such as airframe structures for a new helicopter generation.

The excellent properties of composites are well known and they were utilized in many parts of existing helicopters like the BO 105 or BK 117. But it is the first time that a (nearly) complete fuselage was designed, manufactured and tested in Europe.

This paper will give an overview about the main areas of activities:

- Design philosophy (model-tools-parts)
- Manufacturing aspects
- Test and certification of this flying demonstrtor, based on a BK 117
- Weight/parts saving capacity in comparison to an equivalent metal version.

1. INTRODUCTION

MBB has been one of the pioneers in use of composite materils for helicopters. The development at these areas had started in the early 60th with rotorblades (GFRP) and secondary structures such as engine cowlings. More and more parts of the helicopter changed from metal to composites because of the well known good properties. FRP (fibre reinforced plastic) applications today are: rotorblades, cowlings, sliding doors, nose trap doors, aft doors, vertical fin fairnigs, horizontal stabilizer and end plates.

Based on this substantial knowledge of composite materials and the great experience at MBB, a composite airframe program, sponsered by the german ministry of defence, was startet in 1985.

The aim was, to develop a demonstrator which could show whether composites are profitable compared to metal airframes, including a full scale fest with one fuselage and a flight test phase with a second fuselage.

MBB has selected a BK117 as the experimental rotorcraft, since this version resembles new transport helicopters most closely (Fig1.-1). So, the results of this studie are transferable to other systems.

The major activities of the investigations were:

- demonstration of weight advantages versus metal airframe
- demonstration of parts saving capacity
- preparation of guidelines for the material-oriented design of new airframes
- preparation of computing methods and design characteristics
- development of a manufacturing method suitable for "large-scale components"
- development of suitable joining methods for FRP/metal and FRP/FRP
- effect of FRP on electrical sytems and EMC.

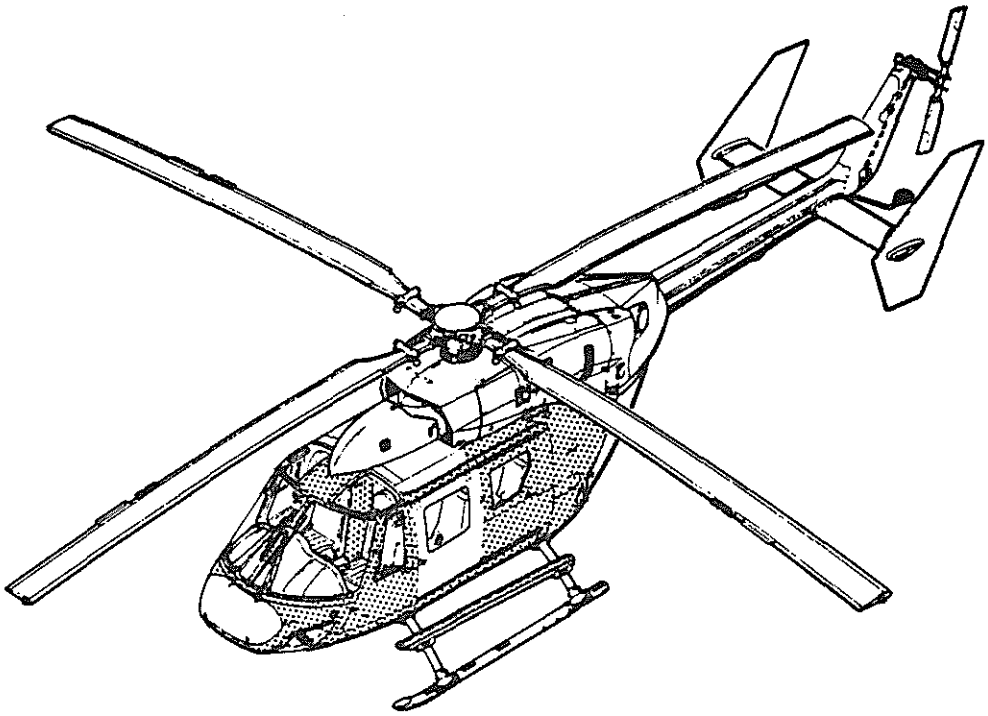
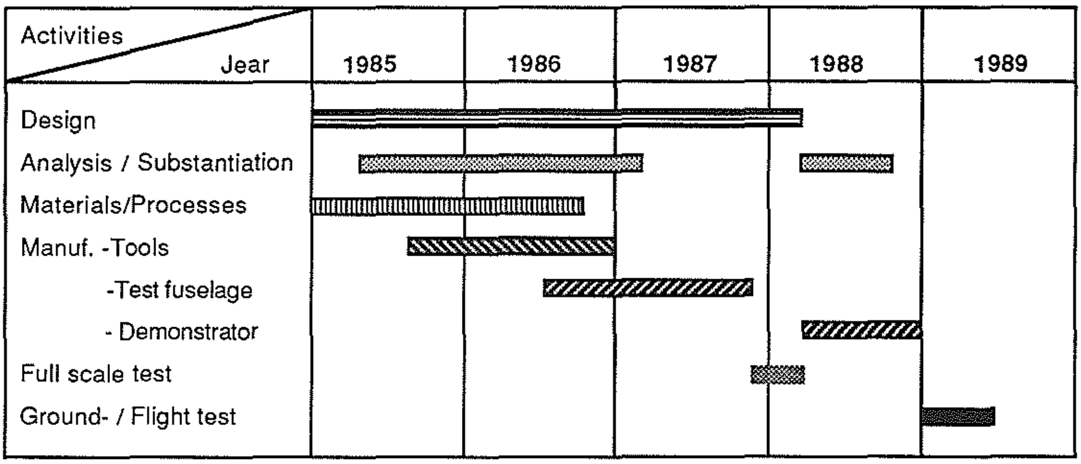


Fig. 1-1: BK 117 with the fibre composite fuselage

Time shedule



2. DESIGN AND ANALYSIS

The example of a BK117 fuselage is inteded to show how FC (Fibre Composite) designs can be crated for major helicopter components which are to be weight efficient and cost-effective. The problems encountered due to the use of FRP in helicopter fuselage design are to be identified, and the effect to the manufacturing process on component quality is to be investigated. The resultes of this project should show whetehr airframe of FC or metal are more suibale for new helicopters. Futhermore, the two kinds of design are to be compared taking the substuted fuselage parts as an example.

Arframe-component development represents a compromise between the technological requirements and what is technically feasible, rspectively the available experimental rotorcraft.

2.1 Arrangement of Major Components

The fuselage of most existing helicopters is a conventional aluminium structure with skin, frames and stringers (Fig. 2.1-1). Some lower fuselage sections are built in aluminium sandwich.

Based on this design, for the composite fuselage, the complete airframe below the transmission check was selcted to be substituted. (Fig. 2.1-2). The joining area from the composite fuselage to the metallic transmission check was good for studing the connection of FRP and metal.

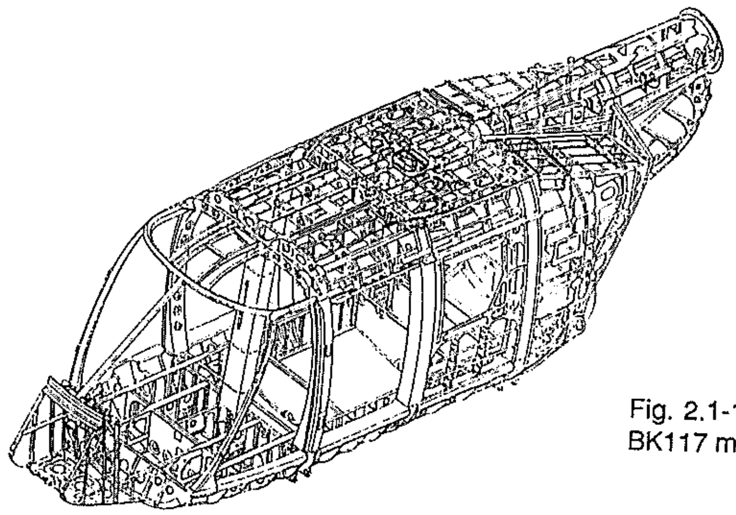


Fig. 2.1-1:
BK117 metallic airframe

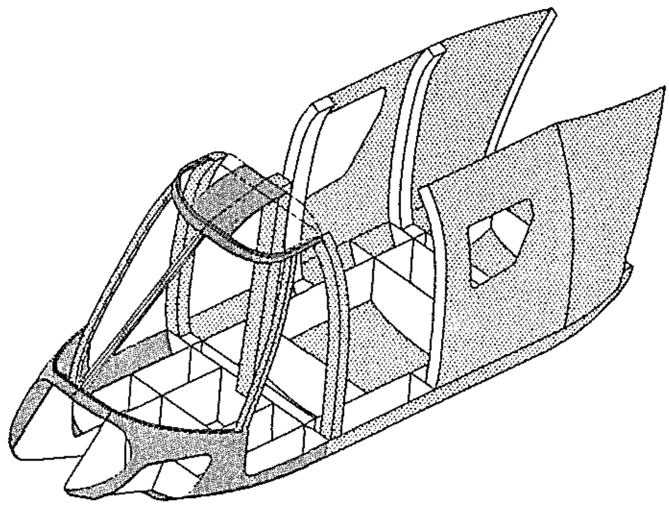


Fig. 2.1-2:
Composite part of the airframe

The sectioning was made under the following assumptions:

- realize as large components as possible
- good accessibility
- good mold release
- simple and cost-effective production
- clear load pathes
- simple joints
- extensive preintegration (one shot)

These considerations led to the following major components, shown in the exploded view Fig. 2.1-3.

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
h) Centre post
i) Cockpit middle

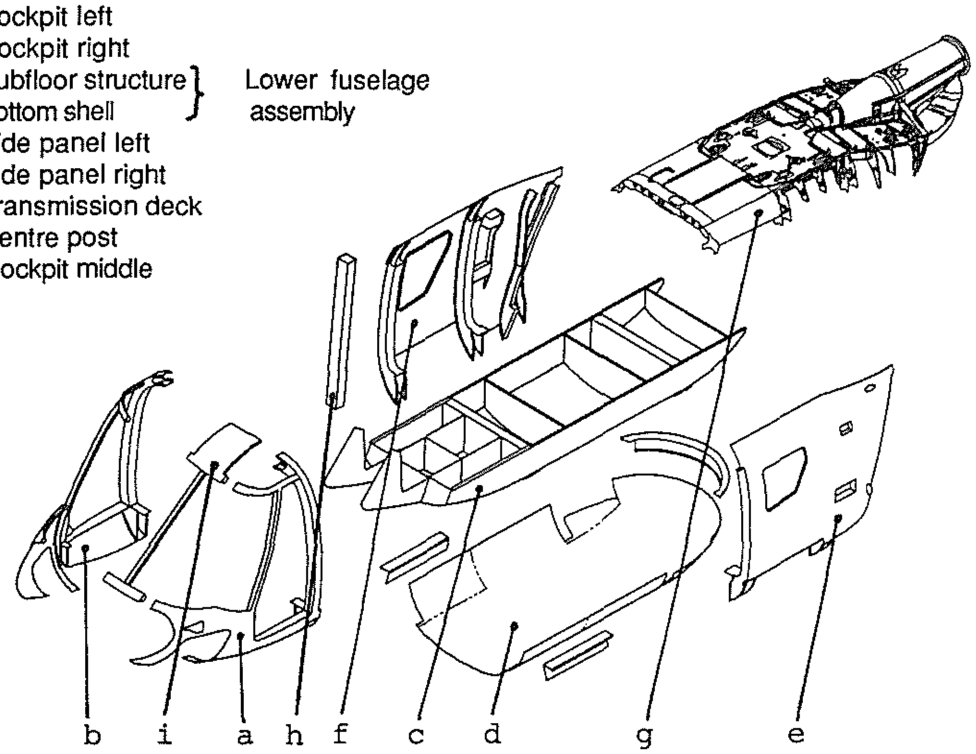


Fig. 2.1-3 Derivation of major components

Comparing the aluminium airframe to the composite one, the bumber and the shape of the main frames and stringer 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 the existing helicopter. This limited the freedom of designing. A significant difference to a metal airframe is however that nearly all skin-stringer elements are subsituted by aramid-NOMEX-core sandwich, or in carbon-NOMEX-core sandwich.

Aramid is used in lower loaded sandwich skins because they promise better impact behavior and due to the weight advantages.

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

The distribution of composite materials within the substituted airframe is: 75% Carbon, 22% Aramid and 3% Glass.

2.2 Structural Joints

Within the main components many joints are no longer necessary by using co-cured components. For example the hat-shaped frames 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 riveting.

2.3 Electrical Effects

The following important aspects have to be regarded:

- ground connection for return conductors and for housings of electrical equipment
- ground 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 were covered with a thin aluminium layer by metalspraying. Also 10 aluminium wires were connected between the transmission deck and the floor board (Al-sandwich) to get a Faraday-cage. Three of them have contact with the Al-layer. Additionally some of the ground connections for the electrical equipment within the side panel had to be transfered to a common metallic point.

2.4 Stress analysis

To perform a complete analysis of the force flow 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 adapted 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 reqirements concerning the accuracy of later calculations.

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

3. MANUFACTURE AND INTEGRATION

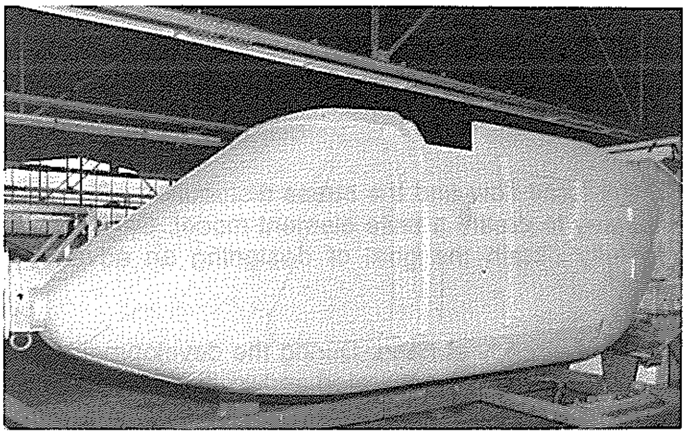
3.1 Manufacturing Concept

Within the scope of this study, various manufacturing methods were selected in order to determine whether differences exist in:

a) Component manufacture applying co-curing (side panels)
b) Component manufacture by bonding cured components (lower fuselage assembly)
c) Combination of co-curing and bonding techniques (cockpit assembly)

In order to be able to test all the methods under the same conditions, CFRP molds were selected to make the FRP parts, which were all formed from a common master model. (Fig. 3.1-1). This affords the following advatages for prototype manufacture:

- Since the coefficients of expansion are about the same for mold and component, there is no danger that the component will be damaged while cooling down.
- The components fit accurately in the molds after cooling down, i.e. further bonding steps can be carried out without additional rigs.
- Markings on the edges of the molds, which were copied from the master model when laminating the CFRP shells, serve as reference points for measuring the components in the mold.
- Large components with a high degree of integration can be made applying the so-called "co-curing technique".



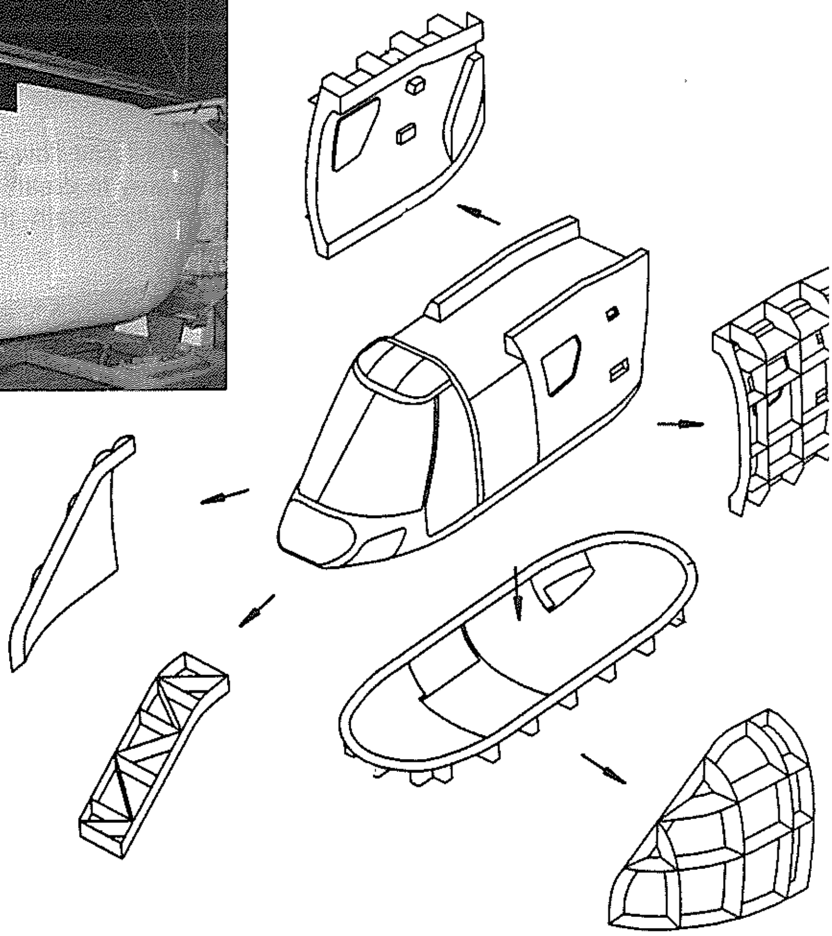


Fig. 3.1-1:

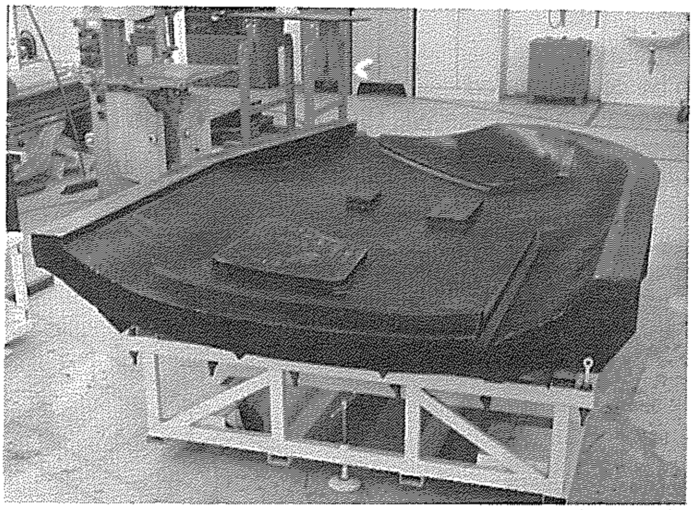
CFRP mold concept

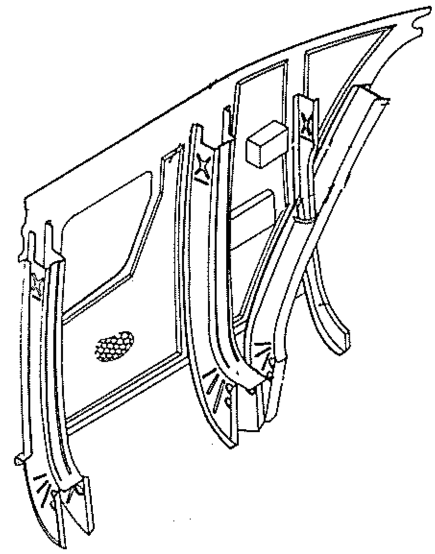
3.2 Component Manufacture

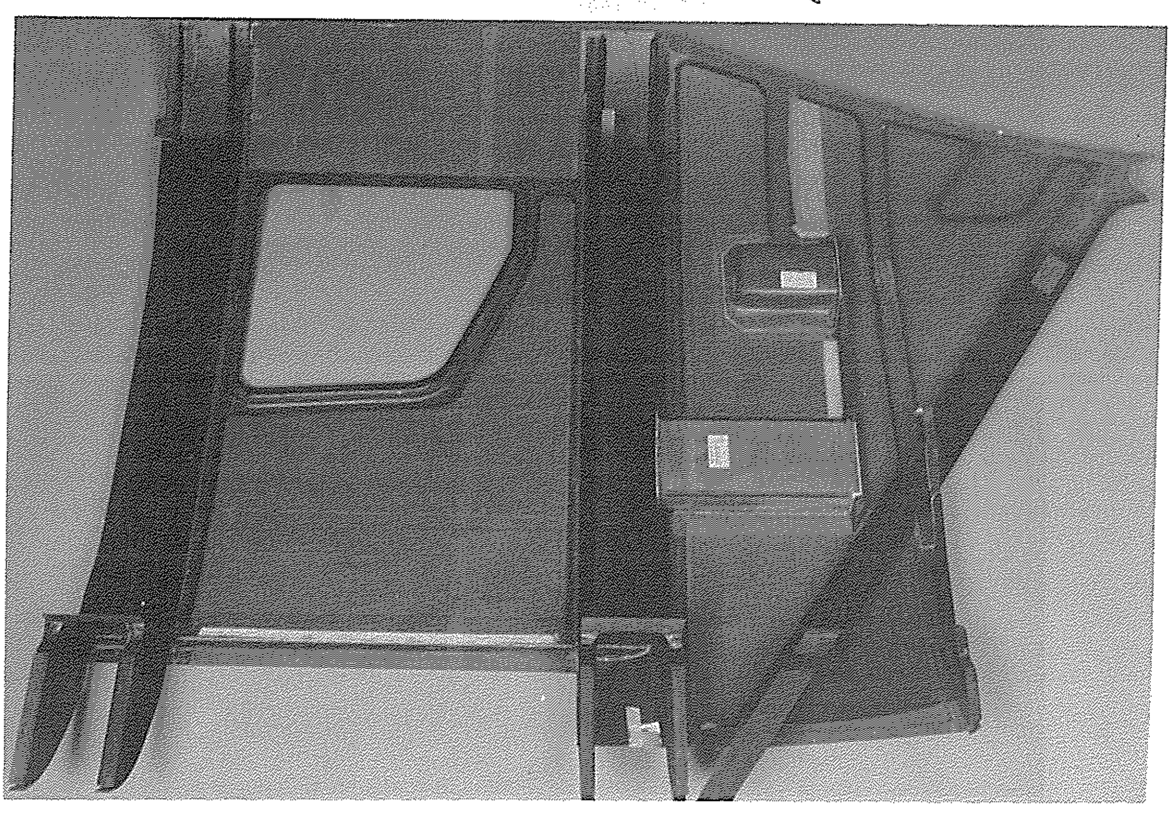
3.2.1 Manufacture of Side Panels

From the manufacturing point of view, this assembly represented the most extensive process-engineering technology. It was manufactured with a high degree of integration (co-curing technique). The precut honeycombs were inserted and positioned after putting the outside skin and the lower engine frame into place. The next step was to insert the inside skin. The hat-shaped frames, which had already been laminaated previously, were now positioned with its CFRP-Mold. Mounting the canted-frame is an exception; it was inserted in the panel already cured. This step was selected in order to test which result can be achieved for highly stressed parts applying this "co-bonding" procedure. After setting up a vacuum bag, curing was carried out in the autoclave at125°C. This technology indeed requires sophisticated tool construction, i.e. accurate adjustment of mold and pressure stamp, but the advantages are:

- no problems as regards piece-part adjustment
- no "adhesive gap".







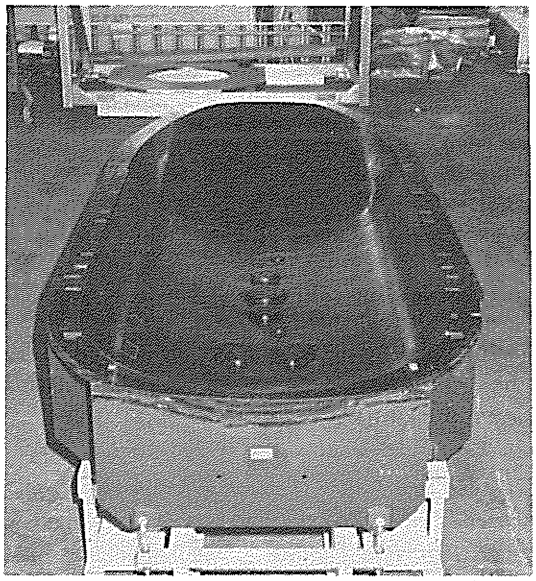
3.2.2 Manufacture of Lower Fuselage Assembly

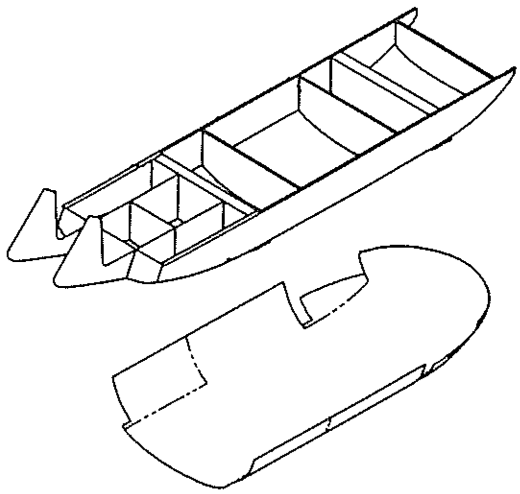
For this assembly all individual components were cured before.These parts such as keelbeams, fuel-cell frames and landing-gear frames were produced in different molds. For the fuel-cell frames and the landing-gear frames, steel-molds were used for studiing the quality of such parts. The keelbeams were manufactured in a modified CFRP-mold.

The face sheets of all sandwich parts were broght together at the joining areas to other parts. The components were designed with monolithic angles at the edges for bonding together.

In order to bond the lower fuselage assembly, the bottom shell was again placed in the CFRP laminating facility. An adjustable metal frame mounted on the edges of the mold served as a positioning and bonding device. Bonding was implemented in two stages:
1. Bonding of subfloor structure
2. Bonding subfloor structure to bottom shell.

The advantages of this procedure are:
- component tolerances can be compensated
- it is ensured that reference planes (in this case WL 1500 for the floor) can be adjusted accurately with relatively simple tooling.
- the scrap-risk is low because of manufacturing individual parts.



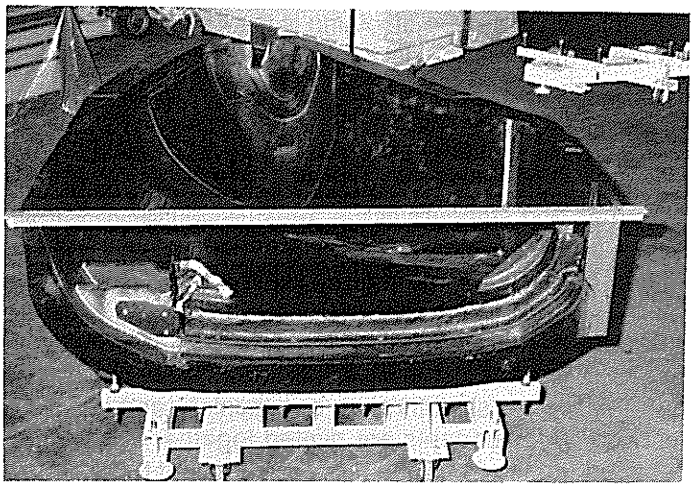


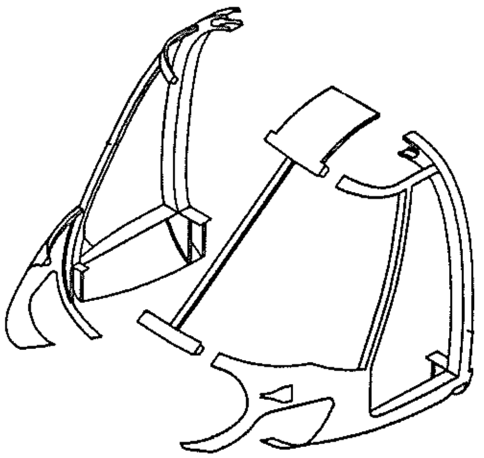
3.2.3 Manufacture of Cockpit Assembly

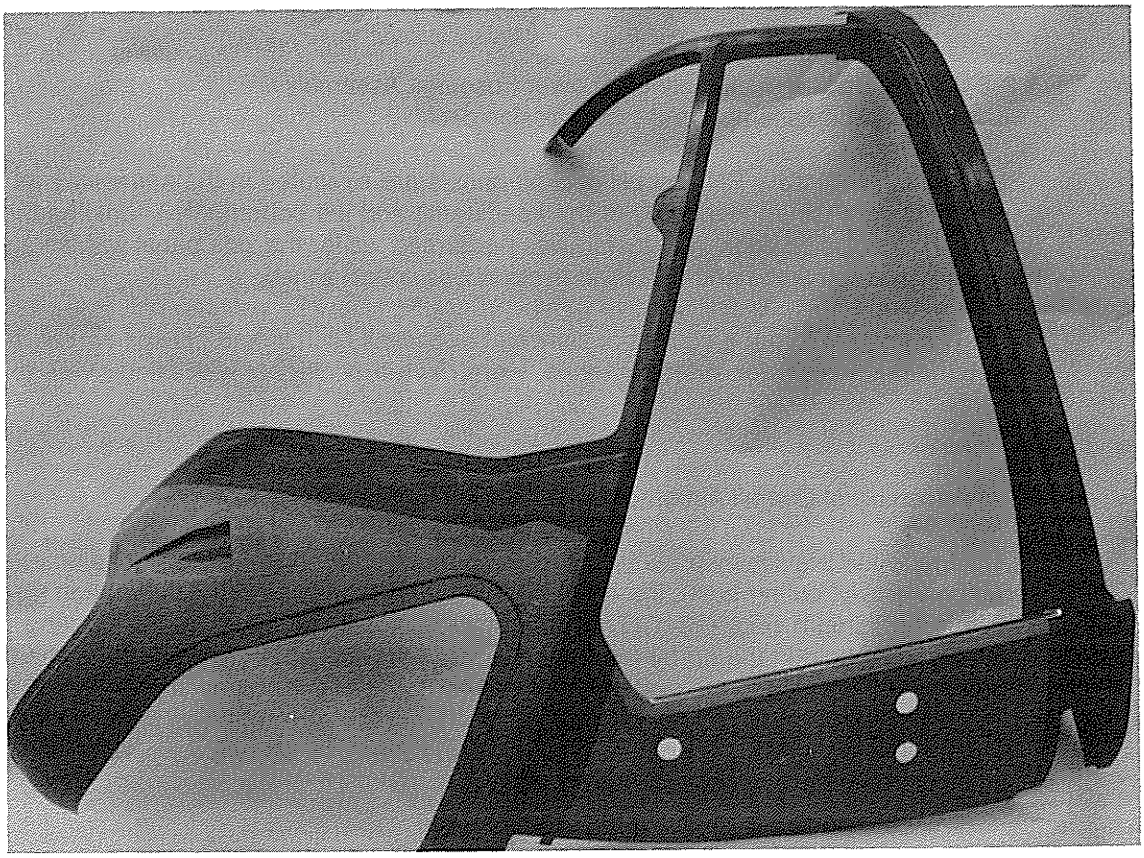
The wet-in-wet and hard-in-hard procedures were combined for the cockpit group. The hat-shaped door frames to the left and right of the cockpit panels were co-bonded, whilst all other monolithic sections, such as cabin frames, door frames and intercostals were bonded in subsequently. The center post, a monolithic CFRP part was laminated in two halfs and co-cured in one shot to a closed box profile.

This manufacturing-method breakdown was the result on the one hand of component requirements and on the other hand of the endeavor to build as cheaply as possible. As mentioned above, co-curing can only be realized with more sophisticated tooling.

Advantages of this procedure are:
- Co-curing is only selected when high loads must be transmitted
- Cost-effective manufacturing method, i.e. tooling optimized to the component.







3.3 Integration

The following was defined for assembly of the composite airframe:
a) Assembly components are bonded
b) Assemblies are riveted together .

The reason was that the composite assemblies are small enough to permit good bonding when suitable facilities are used. Once system pre-integration has commenced, adhesives should no longer be used in an assembly. This was also a reason for saying that assemblies should be riveted together. Further, in this way it is easier to exchange assemblies later in the case of appropriate damage.

3.3.1 Joining Side Panels and Transmission deck

This project is also intended to be an advance development for the method of joining composites and metal. Due to the system, this type of joining is the case on the BK 117 at the transition of the side panel to the metal transmission deck.

For joining the hat-shaped frames, the corner straps are extended upwards so that tongues are formed which serve for load transmission after riveting. The canted frame is obtuse joined to the metal part using doubling sheets. The monolithic sector of the side panel on top was riveted like "sheet metal" to the frames and stringers of the transmission deck. Vrious sectors, which for system reasons had to be taken over from the engine deck, are naturally not "fiber-suitable".

3.3.2 Joining Side Panel and Lowqer Fuselage Assembly

The side panel is joined to the bottom shell through overlapping, and the side panel is then located on the inside. The two FC components are riveted together using blind rivets at the overlapping point. The hat-shaped frames (rear and center) are joined by means of contact angles at the keelbeam and in the floor panel by screw bolts.

3.3.3 Joining Cockpit Assembly and Lower Fuselage Assembly and Cockpit Assembly and Transmission deck

In this sector, the RH and LH cockpit panels are executed in the same way as the side panels and were joined identically to the keelbeams and the bottom shell. Joining the cockpit and the Transmission deck (FC/metal) was performed as for the side panel with screw bolts in the highly stressed areas and with blind rivets in less critical zones.

The joints within the cockpit assembly, i.e. cockpit RH/LH and cockpit center, are large-area bondings which, for safety reasons, were riveted additional with blind rivets to prevent peeling.

3.3.4 Assembly of Overall Airframe

Due to the previously pre-integrated components (side panels, lower fuselage assembly, cockpit assembly, center post), the effort for fixing the assemblies in the integration rig could be kept to a minimum (Fig. 3.3-1) The Transmission deck, required as the starting point for assembly, was held in the rig by the transmission mounting points as well as by the tail-boom connection and leveled to the system dimensions. The reference plane was WL 1500 (= upper edges of floor). Further fixed points are just the four landing-gear fittings in the lower fuselage assembly. In this way it was possible to keep the areas to be riveted perfectly free and to assemble the airframe without removing rig parts.

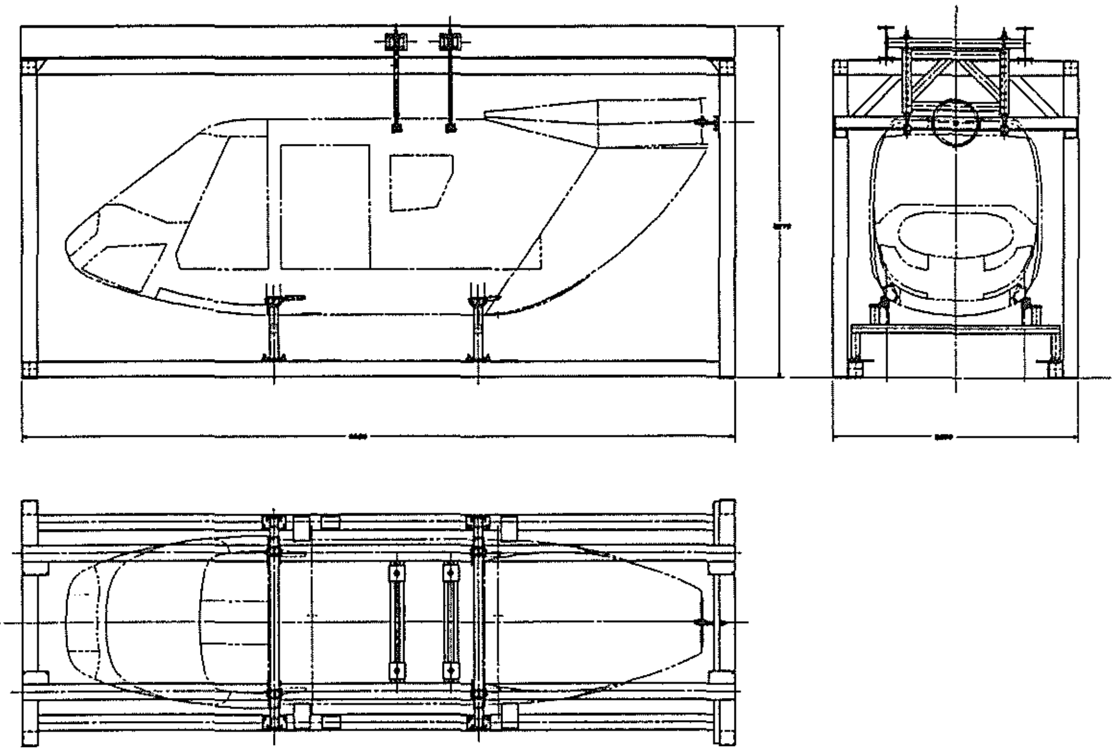


Fig. 3.3 -1: Integration rig

4. STRENGTH SUBSTANTIATION

4.1 General

The strength substantiation was performed to fulfill, among others, the reqirements of FAR § 29 incl. AC 20-107A so far to get a provisional permit to fly for one experimental rotorcraft.

To verify and establish material allowables as well as design allowables at RT and different environmental conditions, a coupon test program was conducted which comprised approx. 800 coupons.

Both, the analytical and the experimental strength substantiation were performed considering static loads mainly. An experimental strength substantiation of the fatigue behaviour was not regarded to be necessary, because the provisional permit to fly is only intended for approx. 50 flight hours and critical stress concentrations in composites can better be recognized at static loading than at dynamic loading (see Fig. 4.1-1).

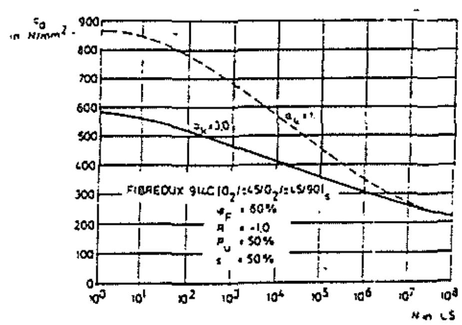


Fig. 4.1-1: S/N curves for notched and unnotched Gr/Ep specimen

Fig. 4.1-2 gives an overview about the procedure of the strength substantiation.

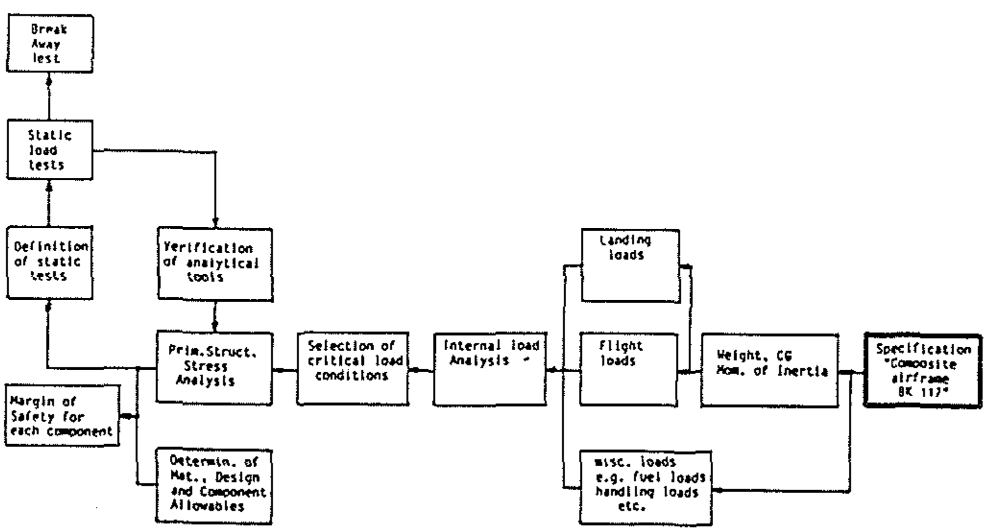


Fig. 4.1-2: Flow chart strength substantiation

4.2 Analytical Strength Substantiation

Besides classical methods a Finite Element model (FE-model, see Fig. 4.2-1) has been used for the analytical strength substantiation. It based on an already existing FE-model of the metal version, however the different design concept (sandwich design vs sheet/stringer) and the different material behaviour (anisotropic vs isotropic) had to be taken into consideration for the modeling.

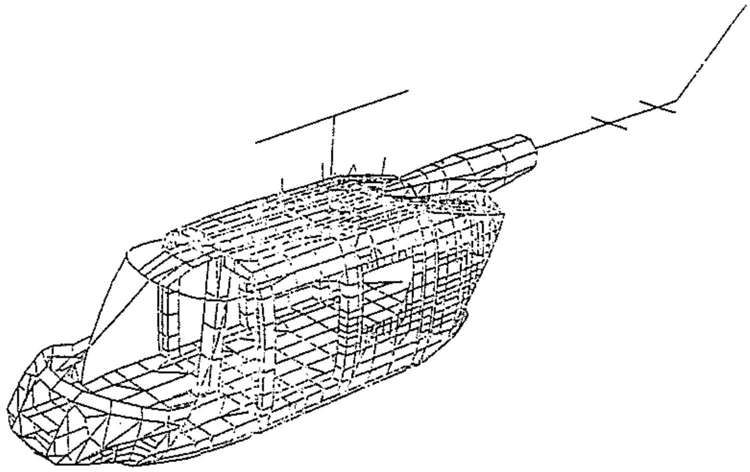


Fig. 4.2-1: Finite Element Model of composite fuselage BK117

Various flight- and groud load conditions were calculated with this FE-model.. For each structural component the critical loads (axial forces, shear forces etc.) were identified to do the stresswork regarding strength and stability criteria for composites.
As far as the strength became the critical failure mode for a component, first a single ply analysis acc. to the so called classical lamination theory was performed and the first ply failure hypothesis (fibre and matrix failure) was applied. Second for damage tolerance reasons allowable laminate strains were defined, which needed not to be exceeded at ult. load. Both measures were regarded to be indispensable to ensure an optimum light weight design.
However, for most components stability criteria became the critical failure modes. Basically two different stabilty modes had to be considered; local instabilities like crippling of monilithic flanges or e.g. wrinkling of sandwich facings and global instabilities like elastic buckling of thin monolithic panels or overall buckling of sandwich panels. None of the local instabilities were allowed up to ult. load because they were supposed to limit the load carrying capability of the structure i.e. a possible loss of integrality of a structural component. Global instabilities like the elastic buckling of thin monolithic panels were allowed below limit load, in case the load surplus could be redistributed to adjacent structural members up to ult. load. However the overall buckling of sandwich panels was treated the same way like the local instabilities.

4.3 Experimental Strength Substantiation

4.31 General

The experimental strength substantiation was accomplished on the one hand to proof and certify by test the airframe's airworthiness and on the other hand to verify the analytical tools. The philosophy which was behind that, was the qualification of the analytical tools by comparing calculated vs measured results of two significant loading cases, in order to be able to do accordingly the strenght substantiation for all other critical loading conditions by analysis only. Tests have been conducted as well on component - as on complete airframe level.

4.3.2 Tests on structural component

Several tests were conducted on a structural component at different evironmental conditions in order to get at an early stage a feedback about the validity of the methods taken for the dimensioning and therefore to minimize the risk for the certificational tests on the complete airframe.

The structural component which was selcted represents almost one half of the fuselage middle and aft portion (see Fig. 4.3.2-1 and 4.3.2-2). This component was supported and loaded in such a manner that it had to behave like the whole structure would do under realistic flight and ground load conditions. Such a part was chosen because most of the load introduction points from x-transmission engines, tailboom and landing gear were inclusive and the interfaces e.g. between the composite structure and the metallic roof structure could be tested too.

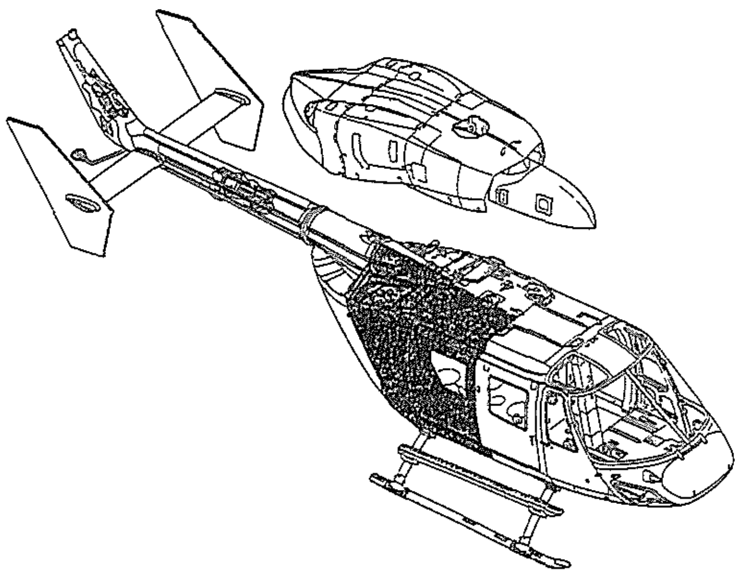


Fig. 4.3.2-1: Location of the component within the fuselage

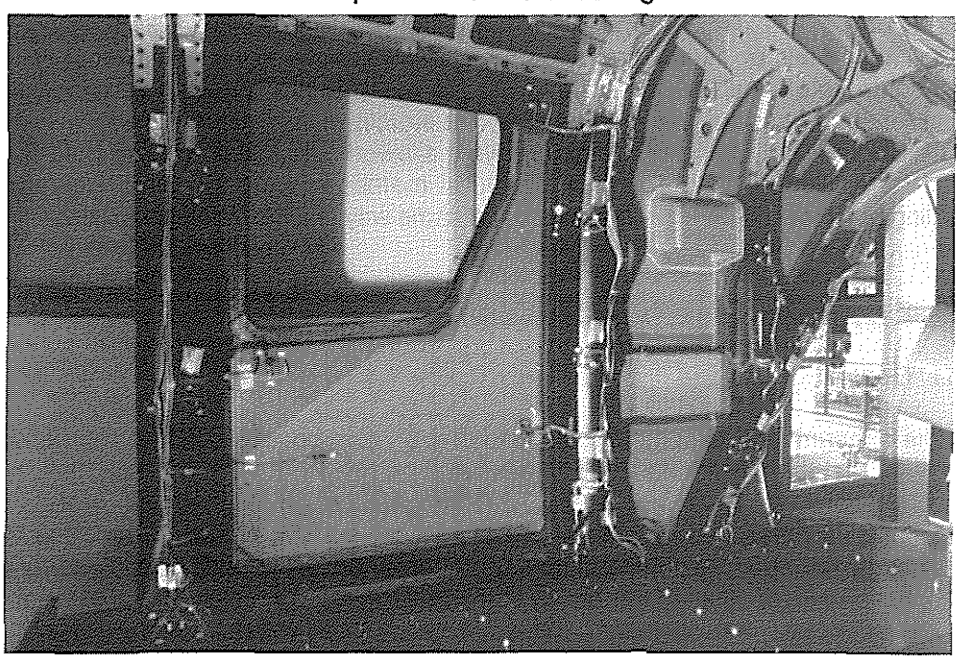


Fig. 4.3.2-2: Test setup for component test

4.3.3 Tests on complete airframe

The main objective of the tests on the complete airframe was to demonstrate that the fuselage could support limit loads without detrimental or permanent deformations and that ultimate loads (limit loads x 1.5) could be applied without causing the structure to collapse. The tests were conducted at RT only because the influence of elevated temperature on the structure had been demonstrated already by the component tests. Out of the several critical loading conditions one significant flight- and ground load case was selected respectively to proof the strength of the airframe.

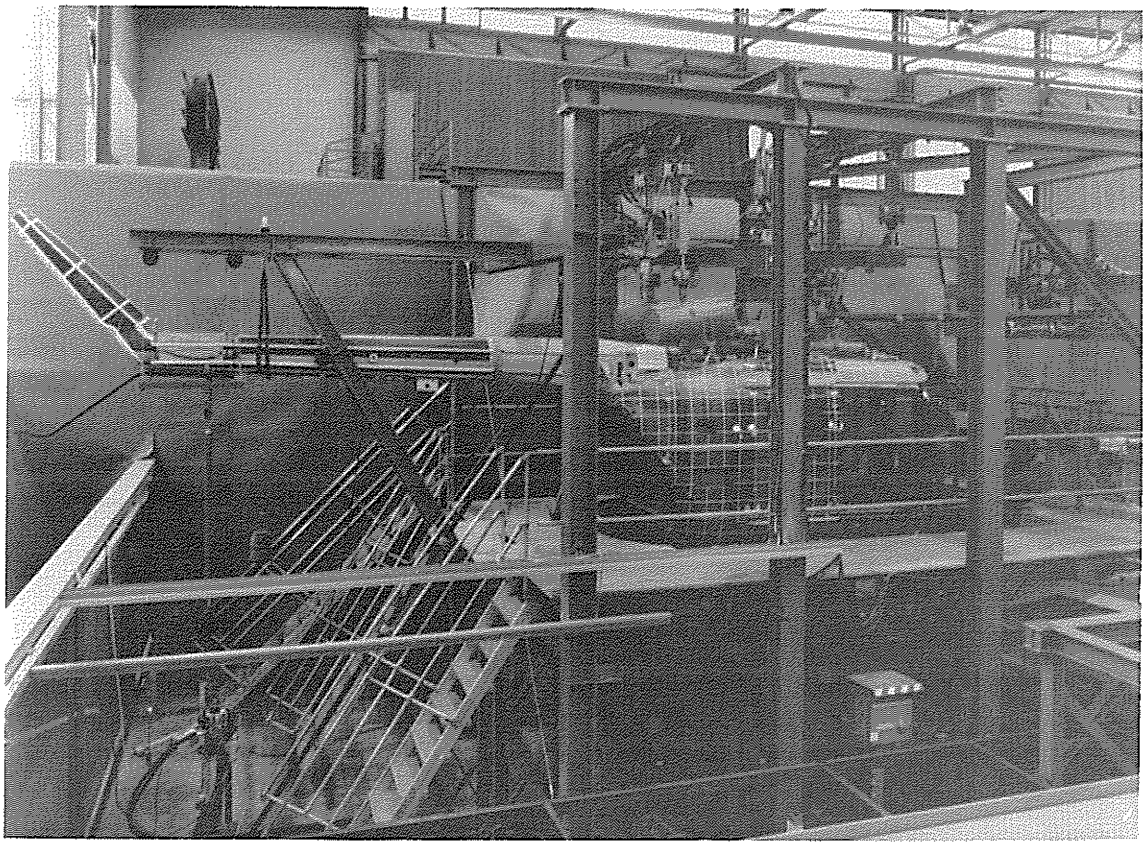


Fig. 4.3.3-1: Test set up for complete airframe tests

The Figs. 4.3.3-2 and 4.3.3-3 show a comparison of measured vs calculated defor- mations at limit loads.

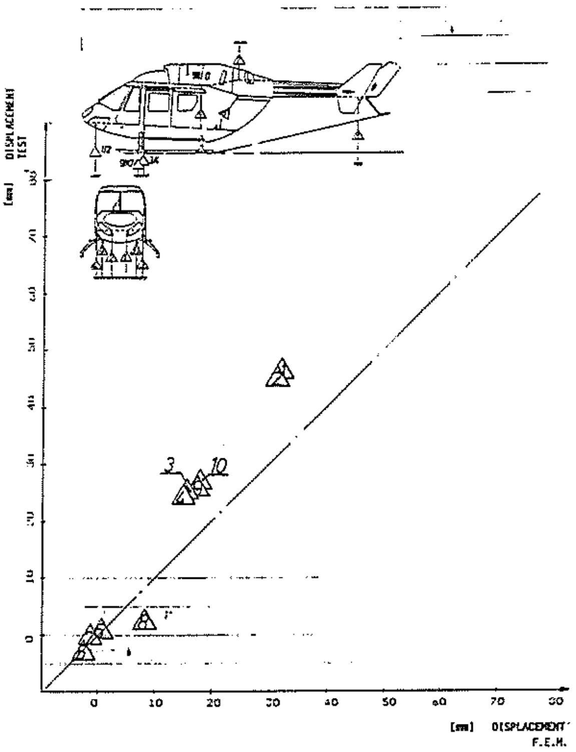


Fig. 4.3.3-2: Comparison of measured vs calculated deformations Loadcase: Pull up to 3.5 g

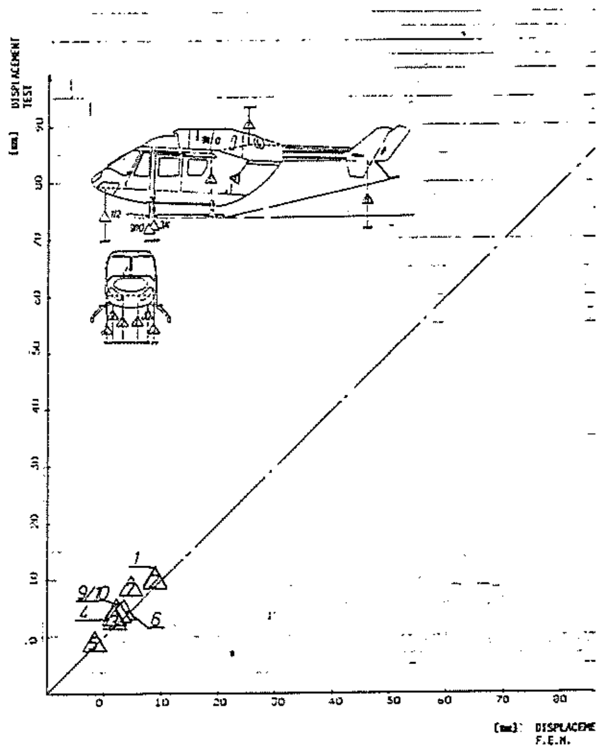


Fig. 4.3.3-3: Comparison of measured vs calculated deformations Loadcase: L/L with drag

After the certificational tests had been completed successfully, the structure was damaged artificially such as could happen typically during service and maintenance. Then the airframe was loaded again but now up to failure, simulating a flight load case. Finally the structure failed at 168% of limit load by separation of one of the mainframes which got typical damages prior to that test.

5. RESULTS

5.1 Design and Analysis

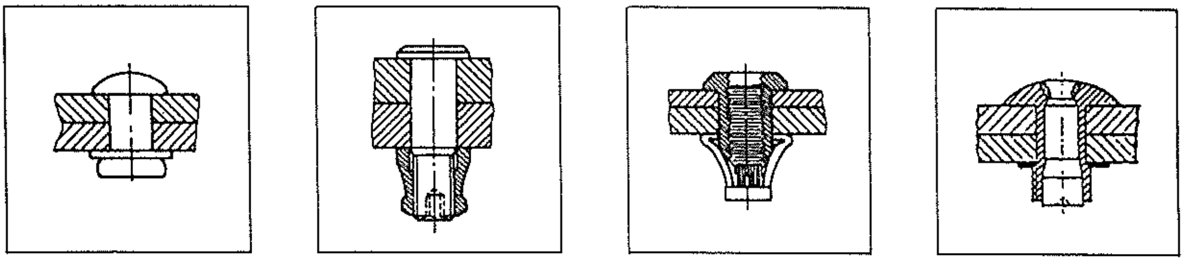
Although there were many restrictions because of the use of all original parts of the BK117, a very good solution was found for the composite fuselage. After calculating the composite parts by a new overall FE-Model the design components and assemblies met all the given requirements from the BK117-A4.

5.2 Materials

An epoxy resin system curing at 120° was selected since the temperature requirements for the airframe did not exceed 80° C. Furthermore, material characteristics for some kinds of CFRP were available from other programs, so that merely and special helicopter-oriented characteristics remained to be determined, which were required for dimensioning purposes. 800 samples for all necessary values. Also a lot of examinations have been done to find the right CFRP-materials for our tooling conceopt. A RT-curing system was selected (because of the not heat-resistant Model) which could be tempered up to 160°.

5.3 Manufacture

Expectations regarding the manufacture of "large-scale components" for helicopter structure could be fully met. All the selected manufacturing methos yielded very good results. The joining methods are obviously feasible. Manufacturing quality is reproducible. In addition, initial investigations have shown that manufacturing FRP structures need not be more expensive than making metal ones.
Using existing fasterners (e.g. HI-LOK)in thin primary helicopter structures would led in our understanding to a overdesign. After testing the combinations of FRP/FRP and FRP/metal by using different types of rivets (Fig. 5.3-1), the results were as good as those of metal/metal.



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

Fig. 5.3-1: Rivet types

5.4 Mass and Parts Balance

Table 5.4-1 shows how the component weights can be reduced by using composites. Furthermore, the calculated weight is compared to the actually achieved (weighed) weight. This demonstrates clearly that the calculated results correspond extremely well with reality.

As is seen from the cmparison of the part quantities, the number of the FC parts is reduced quite considerably if only the cured parts are counted. If the honeycombs are also counted as FC piece parts, the savings in parts are still significant (the prepreg blanks were not counted).

Assembly group	WEIGHT (KG)				NUMBER OF PARTS		
	Al airframe	Composite fuselage calculated	Composite fuselage weighed	saving	Al airframe	comp. fuselage	saving
Lower floor structure	47,2	32,5	32,3	32 %	362	56	84 %
Side panels (LH, RH)	30,1	20,4	19,1	36 %	234	45	81 %
Cockpit	21,8	14,9	12,8	41 %	107	25	77 %
Centre post	4,3	3,0	2,4	44 %	18	4	78 %
Kit to avoid electr. effects	incl.		2,4			20	
Rivets			1,0		(12000)	(1500)	(87 %)
	103,4	70,8	69,0	33 % *	721	150	79 %

*In this value is n ot regarded the weight for painting,aditional cabling and the rivets.

Tab. 5.4-1: Comparison of mass and number of parts

5.5 Electrical Systems/EMV

No final results are yet available for these items. However, it can be stated that a modified ground concept must be prepared for electrical systems and that siutable EMC measures must be implemented. (EMC investigations will only be carrried out on the experimental rotorcraft).

6. CONCLUSIONS

This programm allows to compare the conventional BK 117 metal-fuselage and the advanced composite fuselage for the same system. The development shows the possibility to substitute aluminium structures by saving 25 - 30% of weight. The reduction of number of parts is significant and leads to the expectation that the manufacturing costs are not higher than for metal. The tests of components and the complete fuselage gave the expected results in comparison to the calculations. New manufacturing and joining methods were satisfacory introduced. During a flight test programm in the beginning of 1989 we will point out, that this BK 117 with a composite fuselage fulfills the same flight envelop as the metallic version.

The experience gained during the research and development done in this program will be transferable to future helicopter projects, like MBB-specific developments as well as international cooperations e.g. PAH 2, NH 90 or EUROFAR.