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# DEVELOPMENT OF THE A300 FIN IN MODERN COMPOSITE FIBRE CONSTRUCTION

D. SCHULZ
Messerschmitt-Bölkow-Blohm
Unternehmensbereich Hamburg

## Abstract

On the base of a research program sponsored by the German Government the companies MBB and VFW started developing the Airbus - fin box in fibre reinforced plastics in 1978. In 1984 one fin shall be certificated for airworthiness and be tested in airline service. In this paper, program and design aims are set up. Main results achieved during the first development phases are reported. Environmental conditions to be considered permit the use of 120°C - resin systems. As the result of analytical and experimental investigations with respect to weight, production costs, maintenance and reliability, a structure was chosen which is primarily reinforced by open - section stringers. By using a low cost production concept the increased composite material cost can be offset.

## I. Introduction

Since 1976 there have been strong efforts within the partners of Airbus Industry for applying advanced fibre materials to the secondary structure of the Airbus A300 / A310. Essentially, this refers to spoilers, landing gear doors, floor struts, rudder, leading edges and fairings. The fin leading- and trailing edges and rudder (Fig. 1; 2; 3) will be brought into commercial airline service in 1983. While for the existing leading - and trailing edge sandwich structure a simple material exchange of glass to carbon fibre reinforced plastic (GFRP/CFRP) could be realized, the rudder had to be totally redesigned. The rib - stiffened metal rudder will be replaced by a CFRP - sandwich structure without inner ribs. The simple configuration of the new rudder allows an additional 18% weight saving and a reduction of the total production costs. On the basis of this preparatory work which generates special experience with fibre constructions in the fields of design, production, quality assurance and later in actual airline service, the companies MBB and VFW began to develop the Airbus - fin box in CFRP in 1978. The research program is sponsored by the German Ministry of Research and Technology (BMFT). The fin will be the first piece of composite primary structure for the Airbus. This structural part is especially suitable for demonstrating the applicability of advanced fibre materials in civil aircraft constructions. It distinguishes itself by being removable and therefore easily interchangeable. The large area structure ($A = 45m^2$) is typical for wings and tail units. The development program is divided into six stages (Fig. 4) Preparation, conception and definition stages ended in March 1980 with the completion of the preliminary design work. At the end of the following developmental and testing stages, five fin boxes will be produced. The program will be completed in 1984/86 with flight service.

## II. Objectives and Design Principles

The design principles are mainly induced by the research program objectives and the requirements derivable from the Airbus program.

### Research Program Objectives

- Demonstration that the application of composite material to primary structures is a suitable step to reduce the direct operating costs of civil aircraft without any loss of safety.
  - reduction of the structural weight by approximately 20%.
  - comparable production cost
  - acceptable maintenance costs in relation to the existing metal structure.
  - preparation of the basis for a medium-term economic series introduction of the CFRP-fin.

### Airbus Requirements

- The outer geometry of the fin is clearly defined.
- The fin box must be interchangeable.
- The fin structure must meet the airworthiness requirements FAR 25, amendment 45 and special conditions.
- Economic repair life:
  48 000 flights for A300
  40 000 flights for A310
  as a design aim.

### Conventional Fin Structure

The fin can be divided into the following main components (Fig. 1)

- fin box
- fin leading edge
- rudder
- fin fairings

Its main dimensions are:

| | |
|---|---|
| height | - 8.3m |
| chord depth at the root | - 7.8m |
| chord depth at the tip | - 3.1m |

The fin box is the main structural part of the fin. It takes all airloads acting on the box area itself and the joining loads coming from the leading edge, rudder and trailing edge. The resulting loads are transfered to the fuselage structure by 3 transverse load-carrying fittings and 6 fittings acting mainly in the vertical/longitudinal plane. These 9 fittings guarantee full fail-safety. The interchangeability of the fin is achieved by detachable bolts. The box is designed with 3 spars in the lower and 2 spars in the upper part. Normal rib pitches vary between 420mm and 500mm. The ribs are of a trusswork and shear web design. The skin is made from sheet material mechanically and chemically milled to varying thicknesses due to loading requirements and is stiffened by Z-shaped stringers with a pitch of 115mm. At the rear spar 7 aluminium fittings are attached to pick up the rudder hinge fittings and 3 actuators for performing rudder deflection. The general assembly is joined by riveting and bolting.

### Design Principles

The before-mentioned aims and requirements lead to the following design principles for the CFRP-fin box

- The fin box shall be applicable to A300 and A310 as well.
- The outer geometry has to be identical to the metal one at room temperature; the same lofting has to be used.
- All interface points to the fuselage, leading edge and rudder must be identically positioned and shaped.
- The structural design must meet the requirement that interface forces must not increase more than 5%, which is within tolerance with respect to the

strength of the adjoining parts.

- o The structural design must have fail-safe features and guarantee an economic repair life of 48 000 flights.
- o Environmental conditions have to be considered.
- o The structure must allow the application of simple inspection methods.
- o Number of components has to be reduced.
- o The design must allow the application of cost competitive manufacturing processes and materials.

## III. Production Conception

Considering the program objectives it becomes obvious that the success of the program depends strongly on the possibility of using cost competitive manufacturing processes. Medium - term economic series introduction means later than 1986. By then several hundred metal fins will have been built and the production costs will have been reduced to a major extent. It must be shown that the production costs of the CFRP - fin box shortly after the beginning of series production will decrease 80% of the metal fin costs (Fig. 5). Since the material costs at the end of the 1980's will be higher for CFRP than aluminium, in order to achieve this considerable cost reduction, advanced structural concepts and manufacturing processes had to be considered. An analysis of the production costs for the existing metal fin box is shown below:

| Work Package | % of Total Production Costs |
|---|---|
| Panels & Fittings | 63% |
| Spar Webs & Fittings | 4% |
| Ribs & Fittings | 19% |
| Assembly | 14% |

Table 1

This shows that special efforts on the panels, ribs and assembly could be profitable. During the concept stage a configuration study showed that an orthogonal stiffened panel concept was the best compromise based on the criteria:

- production cost
- weight saving
- reliability
- maintenance
- inspection

It was decided that such panels have to be designed and manufactured as integral parts. Skin, stringers and rib clips will be "one - shot cured".

The cost reduction for the rib production can be mainly achieved by reducing the number of ribs. Considering the necessity of having ribs at the rudder hinge fitting positions and studies concerning rib pitch optimization, the number of ribs for the composite fin box could be reduced as follows:

metal fin box - 26 ribs
composite fin box - 18 ribs

Both afore mentioned measures, the integral panel concept and the reduction of the number of ribs, bring down the assembly costs accordingly.

For realizing the cost effective "one-shot curing" production of the panels, two concepts were developed:

a) mat production process (Fig. 6)
b) module production process (Fig. 7)

The Mat - Concept is preferable in the production of closed sectioned stringers. Shaped rubber mats reaching from rib to rib are laid on a compact substructure that is the female mold for the ribs and stringers. Rib laminates are brought between the shaping blocks and A - section - stringer laminates are laid directly in the shaping mat. The inner volume of the stringers is filled with a rubber pipe which is formed according to the inner surface of the stringers. This grid - work is than covered by the skin laminates. During the curing process within the autoclave, the pressure is achieved for all parts of the skin, ribs and stringers by the flexible mat and rubber pipes. After the curing process has been finished, the mats can be dismantled easily. Implosion techniques can be used in the case of the rubber pipes.

The Module - Concept is the result of applying a high degree of automation in production to the specific features of stringer- and rib - stiffened light - weight constructions. In principle, light - weight fibre structures can be realized by designing the different elements for internal forces as follows:

skin / for shear loads
stringer and rib-flanges / for end loads
stringer and rib-webs / for shear loads

While all flanges should be continuous, the stringer webs can be interupted at the grid points, but a shear load transfer to the adjoining webs must be available. Such a web design, shown schematically in Fig. 8 is applied in the module concept. The fin box panel is devided into a substantial number of small boxes shaped by stringer webs and rib webs. The design suggests that stringer- and rib webs of such an unit can be produced in a one step process by simply draping prepregs around rotating module cores, the geometrie of which is defined by the volume between two adjacent stringers and rib clips (Fig. 9). In another mechanized production step, all uncured module parts must be brought together with the layed-up skin and flange prepregs for co-curing. It is obvious that this procedure will save considerable assembly work later for joining rib- and stringer webs, since the stringer webs and rib clips are already joined in the module unit.

## Production Development

During the definition stage of both previously discussed configurations, a number of panels were manufactured in order to establish process parameters. These investigations will be continued in the development stage, but preliminary results from compression tests demonstrate that both methods will lead to an acceptable standard of quality. Special attention was given to module core design to achieve a controlled manufacturing process with respect to pressure and temperature combined with a long core life. Test panels produced with aluminium cores showed good geometry and strength but for the production of more complex structures more elastic cores are preferable. For the mat processing, special attention was given to reproducibility of the general geometry and flatness of all parts with the use of soft tooling and the accomplishment of long life behavior of the rubber mats. Different rubber materials are under investigation.

## Selection of Final Concept

As the result of paper studies [1] during the concept phase, a decision was made to start preliminary design of an I - section stringer configuration having in mind the application of module technology. It was hoped that hardware built in the definition stage would verify this assessment. At the end of the definition phase a review was made on this subject. An evaluation was carried out according to a method given in "Luftfahrttechnisches Handbuch für Konstruktion" [2] for both stringer configurations, considering six criteria of varying importance (Table 2). As it can be seen from Table 2, the profitable values for both conceptions are almost equal, which led to the decision not to change the design work but to continue with I - section stringers.

| Criteria | Factor of Importance | I-Stiffened Structure | | ⎍-Stiffened Structure | |
|---|---|---|---|---|---|
| | | Relative Value | Profitable Value | Relative Value | Profitable Value |
| Production Cost | 1.0 | 1.0 | 1.0 | 0.8 | 0.8 |
| Risk for Program | 0.7 | 0.95 | 0.665 | 1.0 | 0.7 |
| Weight | 0.63 | 0.94 | 0.59 | 1.0 | 0.63 |
| Transferability | 0.5 | 0.85 | 0.425 | 1.0 | 0.5 |
| Maintenance | 0.45 | 1.0 | 0.45 | 1.0 | 0.45 |
| Potential Improvement | 0.4 | 0.95 | 0.38 | 1.0 | 0.4 |
| Profitable Value | - | - | 3.51 | - | 3.48 |

(Profitable Value) = (Factor of Importance) x (Relative Value)

Table 2. Valuation of I- and ⎍-stiffened Structures

Series Production Concept

In Fig. 10 a possible series production line on the basis of module conception is pictured. As outlined before, such a concept must be able to compete on a cost basis at the end of the eighties. The main principle for this production line is minimizing man power and applying automation to the utmost justifiable degree. Almost all production steps are numerically controlled ① .

The production starts by water jet cutting ② the module prepregs, which will be draped on module cores on a bandage machine ④ with the aid of a manipulator ③ . The transport of naked module cores from storage ⑥ to the bandage machine ④ and the draped cores from the bandage machine ④ to the module control box ⑦ will be carried out by a second manipulator ⑤ .

After being controlled, the draped module cores will be transported from the control box ⑦ to the module grating platform ⑧

The grid being complete and pressed together will be positioned on the laminated skin which has been prepared by a layer machine ⑨ . After that, the stringer flanges will be layed upon the bordered stringer webs ⑨ . Autoclave curing will join ⑩ , ⑪. The dismantling ⑫, transporting ⑬ , ⑮ and cleaning of the tools and module cores ⑭ is a highly automated procedure.

## IV. Structural Development

### Material Selection

Environmental conditions, especially humidity and temperature, affect the strength and stiffness properties of the composite structures.

The composite structures must be designed for the worst applicable conditions which are influenced by aircraft mission spectrum, surface coloring and material behaviour. Evaluations of test programs and calculations [1] resulted in the specification statements for extreme structural temperatures being -70°C and +70°C. For the maximum humidity contents a first prediction was made, that the long time value will be 0.9 weight percent. The decrease of the material properties at 0.9% RH and 70°C for the composite T 300 / CIBA 913C is shown in Table 3.

These values were used for preliminary calculations. Since the maximum fin box temperature does not exceed 70°C under high loading conditions, and the fin structure is dimensioned mainly for stiffness requirements, resins with curing temperatures of 120°C can be used. Compared with 180°C - curing systems the following advantages on the production side can be outlined:

- less problems with curing tools due to thermal expansion
- reduced time and energy extent
- reduced thermal stresses
- easier machining

As fibre material KC 20 fibre type according to LN 29694 has been selected. It represents a good combination of stiffness, elongation, costs and workability. The product T300 of TORAY/UCC belongs to this class. For the production of the fin box 8H - satin fabrics and tape prepregs are used, preferable with a surface weight of $400 \mathrm{g/m^2}$. Suitable resin systems are Hexcel F 550, CIBA GEIGY 913C, NARMCO 5209; Fiberite 948 and CODE 95. Hexcel F 550 was designated as the primary resin system for the fin during the definition stage.

| | ambient | 0.9% RH; 70°C |
|---|---|---|
| $E_{\parallel}$ | $1.26 \times 10^{5}$ $\mathrm{N/mm^2}$ | $1.26 \times 10^{5}$ $\mathrm{N/mm^2}$ |
| $E_{\perp}$ | $9.5 \times 10^{3}$ $\mathrm{N/mm^2}$ | $6.7 \times 10^{3}$ $\mathrm{N/mm^2}$ |
| $G$ | $4.8 \times 10^{3}$ $\mathrm{N/mm^2}$ | $3.4 \times 10^{3}$ $\mathrm{N/mm^2}$ |
| $\gamma$ | 0.3 | 0.3 |
| $\sigma_{\parallel t}$ | 1250 $\mathrm{N/mm^2}$ | 1250 $\mathrm{N/mm^2}$ |
| $\sigma_{\parallel c}$ | 1250 $\mathrm{N/mm^2}$ | 750 $\mathrm{N/mm^2}$ |
| $\sigma_{\perp t}$ | 50 $\mathrm{N/mm^2}$ | 35 $\mathrm{N/mm^2}$ |
| $\sigma_{\perp c}$ | 210 $\mathrm{N/mm^2}$ | 100 $\mathrm{N/mm^2}$ |
| $\tau$ | 80 $\mathrm{N/mm^2}$ | 50 $\mathrm{N/mm^2}$ |

| | |
|---|---|
| $E$ / $G$ | Young / shear modulus |
| $\sigma$ / $\tau$ | normal / shear stress |
| $\nu$ | Poisson's ratio |
| t / c | tension / compression |
| $\parallel$ / $\perp$ | in / transvers to fibre direction |

Table 3 Material B-Values for T 300 / 913 C

Material Testing

A principle question for material testing is how to get realistic humidities for the different materials. The problem can be solved by calculation for specified flight mission if the diffusion coefficients are known. For selected materials the diffusion coefficients were found by tests at the DFVLR in Braunschweig.
On the other hand, the service humidity can be evaluated by comparable tests if for one composite material experience on aircraft with a comparable mission is available. At spoilers of Boeing 737 made of carbon fibre NARMCO 5209 composite, a relative humidity of 0.65% after long time Lufthansa service was found. Recalculations for extreme climates indicated that a humidity content of 0.85% could have been expected.
In order to find a laboratory service climate, specimens made of T 300/NARMCO 5209 - composite were exposed to different climates. It was found that laboratory climate of 70°C and 70% RH led to the desired stabilized humidity of 0.9%. Other composite materials were exposed to the same climate up to a stabilized humidity content. First results are given in Table 4, which due to the comparable flight missions also are valid for the Airbus.

| Graphite Composite | Service Humidity |
|---|---|
| Hexcel W3T - 584 - F 550 | 0.74% |
| Hexcel T6T - 262 - F 550 | 0.75% |
| Fiberite HY - E 1048 A 1 E | 0.85% |
| Fiberite HMF - 133 | 0.86% |
| NARMCO 5209 | 0.90% |
| Fibredux 913C - TS - 5 | 0.98% |

Table 4. Relative Humidity in Composites at the Airbus

Strength tests with "service - climate" - conditioned specimens are under preparation.

## Dimensioning Criteria

The main dimensioning criteria are derived from the requirements that the composite fin box must be applicable to the unchanged A300 and A310 aircrafts. As the weight of the total composite fin is reduced by approximately 20%, the stiffness of the fin box can be reduced by 20% in bending, shear and torsion accordingly. By this a comparable fin box -Eigenvalue and a similar level of safety for flutter onset can be achieved as for the certified aircrafts with metal fin boxes. In addition, all interface loads with fuselage and rudder must remain the same as for the metal fin. For composite structures, the effects of buckling on strength, especially on fatigue, is not completely understood. Therefore, it was decided for the fin box design that local buckling may not occur below limit loading. The same limit was set up for resin matrix cracking. For all strength and stiffness calculations ambient and conditioned material properties must be used.

## Preliminary Strength Calculations

On the basis of the dimensioning criteria and preliminary material data, first calculations for optimizing I - and ⊓ - stringer stiffened structures were carried out. The rib pitches were found to be identical for both stiffener configurations, 700mm at the root and 1000mm at the tip of the fin. The overall dimensions of the stringers and the stringer pitches shown schematically in Fig. 11 are constant over the span. On the other hand, the skin and stringer thicknesses decrease from root to tip. For the preferred I - stringer configuration thicknesses are given at three rib stations in Fig. 12.

## Panel Tests

In the course of the definition stage, compression tests were carried out with 1000mm long panels stiffened by different shaped stringers. All panel configurations were designed for the same stiffness per unit width. The ends of the panels were cast in resin and rib structures were simulated by supporting plates. In Fig. 13 a failed test specimen is shown in the test rig. The test results are depicted in Fig. 14. The ultimate stress $\sigma_u$ is here understood as the ultimate load divided by the total cross - sectional area. In relation to I - section stringer panels it is shown that the ultimate strength is 13% lower for [ - and Z - section stringers and 6% higher for ⊓ - section stringers.

Further it is shown that by the use of ±45° tapes instead of fabrics an increased strength of 6% can be achieved.

A comparison of the weight related stresses $\sigma_u \cdot \rho_{CFRP} \cdot \rho^{-1}$ for CFRP I - stringer panels and aluminium Z - stringer panels indicates a weight saving of 26% for the composite structure.

At the DFVLR for short I - stringer panel sections comparative tests are under preparation to define the influence of moisture and temperature on compression strength.

As the next step, panels with a size of 2300mm x 700mm are to be tested in combined shear and compression loading. Fig. 15 shows a module panel prepared for testing in a test rig described in Fig. 16. A preliminary shear test with such a panel showed that the shear strength will be higher than 120 N/mm² at all stations of the fin box.

## Preliminary Design

As outlined before in the definition stage, the preliminary design for the I - section stringer configuration was accomplished. It resulted in drawings for all main components of the composite fin box [3] .

Selected points shall be discussed in this paper.

- Structural System
  According to the design principles all interface points to the fuselage, leading edge and rudder must be identically positioned and shaped. That means identical positioning of the 3 spars and the 7 main ribs supporting the rudder hinge fittings in relation to the metal structure (Fig. 17). The length of the middle spar could be changed.
- Panels
  As outlined before the panels consist of skin, stringers, spar flanges and rib clips made of carbon fibre fabrics and tapes (Fig. 15). Because these parts are co - cured no further assembly work is necessary. The approximately 40mm high I - stringers (height varies with flange thickness) run parallel to the rear spar with a pitch of 100mm. Special features for assembling the panels to the ribs and fittings can be seen at Fig. 17.2; 17.3.
- Spars
  The spar webs consist of flat fabric laminates stiffened by co - cured composite stiffeners. In the upper part of the front and rear spar, holes are incorporated to make the fin box accessible for assembly and maintenance reasons. At the lower end the shear fittings are co - cured with the spar webs. The spar webs are riveted to the panels by titanium rivets. To the rear spar, aluminium fittings are bolted for supporting rudder hinge fittings and actuators (Fig. 17.3). The principle load transfer is shown on Fig. 18.
- Ribs
  While the upper and lower end ribs and the 7 main ribs supporting the rudder hinge fittings are designed as flat stiffened webs, mainly made of fabric composite (version I on Fig. 17.3), the 9 ribs supporting the panels for stability reasons can be designed as composite truss work, which leads to a lighter structure (Fig. 17.2). It is under investigation if the inner rib flange can be co - cured with the panel.
- CFRP - Fittings
  In Fig. 17.1 the mid spar/fuselage attachment fitting for load transfer in the vertical/longitudinal plane is shown assembled to the fin box. The fitting consists of 2 separate cured parts laminated from fabrics. During the panel curing the 2 fitting parts will be bonded to the fin box panel, achieving a multiple shear connection. Additional bolting is foreseen for safety reasons.
- Weights
  At the end of the preliminary design stage a weight calculation was carried out. The results are presented in Table 5.

| Components | CFRP (kg) | METAL (kg) |
|---|---|---|
| Panels & Fittings | 341 | 404 |
| Spar Webs & Fittings | 75 | 112 |
| Ribs & Fittings | 84 | 136 |
| | 500 | 652 |

Table 5. Weights of the Fin Box

## Finite Element Analysis

In order to meet the requirements of the design and dimensioning criteria with special respect to the interface forces and fin box stiffness, a detail structural analysis has to be carried out to obtain a reliable comparison of the structural aspects of the metal and composite fin. On the basis of the preliminary design work and stress calculations, a structural analysis with the finite element program NASTRAN is prepared. As shown on Fig. 19 the structural idealisation incorporates all stringers, ribs and spars. The rudder is idealized as a flexible beam. The elasticity of the fuselage structure is taken into account by a special super-element. Effort is made for getting a suitable idealisation of the module stringers by splitting the stringer area into 3 elements: the module webs, the inner flanges and the outer flanges. By this means the strength analysis can be carried out separately for the different laminates of the stringers.

The loading is applied at the grid points of the idealized structure. In Fig. 20 typical loadings are shown for gust and maneuvre cases.

At the end of the definition phase the calculations were not completed so that results cannot be presented in this report.

## V. Component and Full Scale Tests

### Aging of Test Specimens

Tests for the verification of static, residual and fail-safe strength will be carried out under the most unfavorable applicable climate conditions, which normally means maximum service moisture content combined with maximum structural temperature. As outlined in chapter IV, for the Airbus the maximum structural temperature is +70°C and the maximum service moisture content can be achieved by conditioning the composites at 70% RH and +70°C until weight equilibrium.

When achieving the service moisture content for larger test specimens the following problems can result:

- There is natural aging during the time between production and test. This initial moisture absorption must be taken into account when obtaining the desired moisture value for testing.
- The test specimen might have different thicknesses. Each structural thickness should contain the same moisture content when tested.

The solution to these problems is the following:

Reference specimens for determining the moisture absorption with different thickness will be produced at the same time as the test specimen, weighed, and travel with it until the static test begins. These reference specimens will be weighed periodically before and during the artificial aging.

In order to accelerate the aging process it can be split into two steps (Fig. 21):

1st step:

Temperature: 60°C; RH: 96% - 98%

Duration: Until the thickest reference specimen reaches the desired service moisture content (0.9% for NARMCO 5209). After this the environment is changed to laboratory service climate.

2nd step: Service Climate

Temperature: 70°C; RH: 70%

Duration: Until the moisture content of the thinnest reference specimen has fallen to the service moisture content. The strength test can then be started.

### Component Tests

In the development stage for critical and complex components such as spars, composite fittings, panels and their interfaces, test specimens will be designed, built and tested in order to get an early verification of the applied design features. For the production and quality control of the specimens, those techniques are applied which have been developed or checked by previous tests with respect to applicability to the full size fin structure.

### Full Scale Static Test

For demonstrating stiffness, fail-safety and ultimate strength, a full scale test specimen consisting of the total fin box and dummy structure for the leading edge and rudder will be tested for gust and manoeuvre cases (Fig. 22). As outlined before the test will be carried out at service moisture and 70°C temperature. The aging can easily be realized by using the fin box itself as an environmental chamber.

For this reason the openings of the fin box will be sealed and the aging will be done from within.

## Full Scale Fatigue Test

The fatigue test specimen consists of the lower part of the fin box (Fig. 23). Loads acting at the tip will be applied by a dummy structure. The test will be carried out to demonstrate that 48 000 service flights can be achieved without major failures. Therefore 96 000 test flights of the fin load spectrum must be applied. After 96 000 test flights, artificial cracks shall be saw-cut in the structure and an additional 24 000 flights applied in order to demonstrate that the crack propagation will not become critical with respect to proposed inspection intervals and methods. Over the test life the stiffness will be checked periodically. It is expected that the fatigue life of the fin box is not influenced significantly by the effects of temperature and moisture. Therefore the fatigue test shall be carried out at ambient conditions. This procedure shall be justified by fatigue tests on smaller components. The fatigue test will be completed with residual strength tests after 120 000 test flights. These tests will be carried out at service moisture and a temperature of +70°C.

## Acknowledgement

The author would like to thank all members of the CFRP-fin-team for their work which is the basis of this paper.

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[2] Arbeitskreis Konstruktion
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[3] CFK-SLW-Team MBB / VFW
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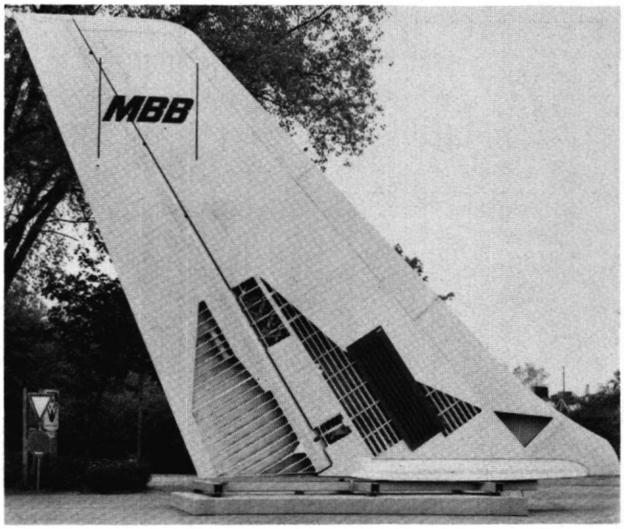


Fig. 1 Airbus Fin Structure and CFRP Test Panel

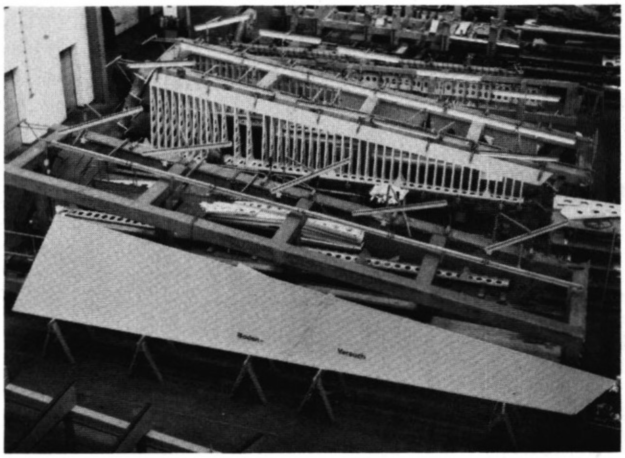


Fig. 2 Airbus Rudders in Metal and CFRP

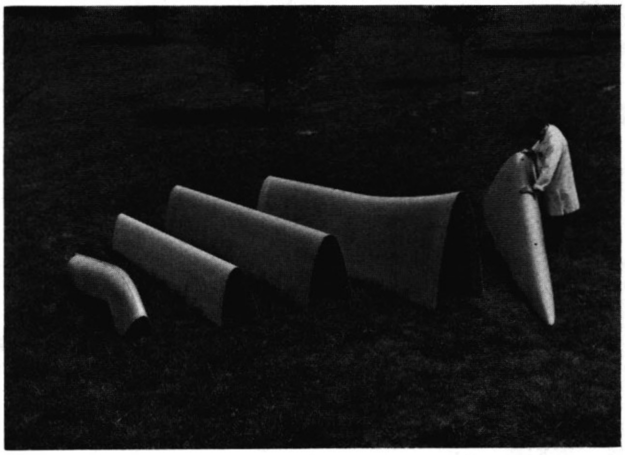


Fig. 3 Airbus Fin Leading Edge Sections in GFRP

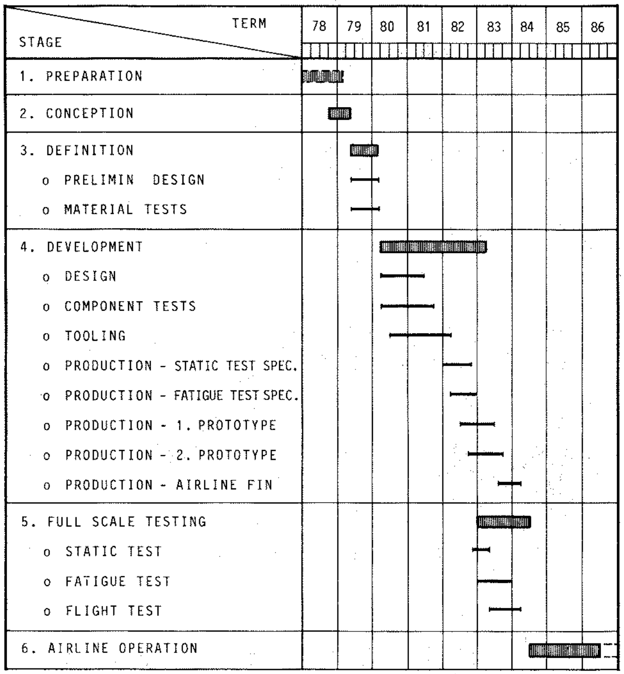


Fig. 4 Schedule of Development
- Airbus Fin in CFRP -

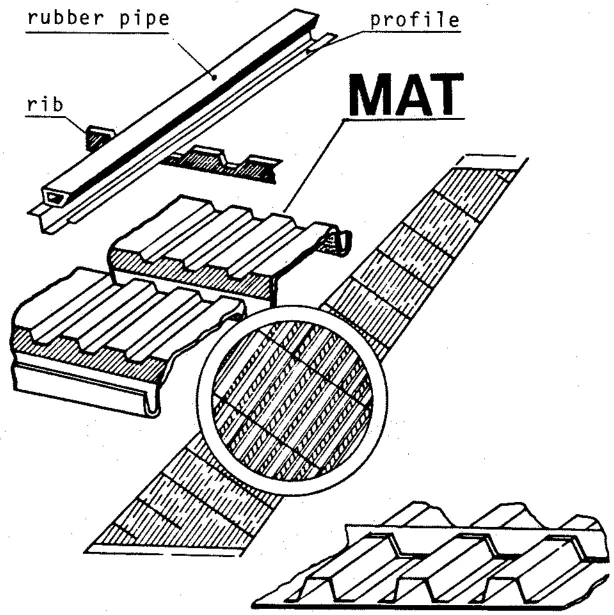


Fig. 6 Mat Production Process

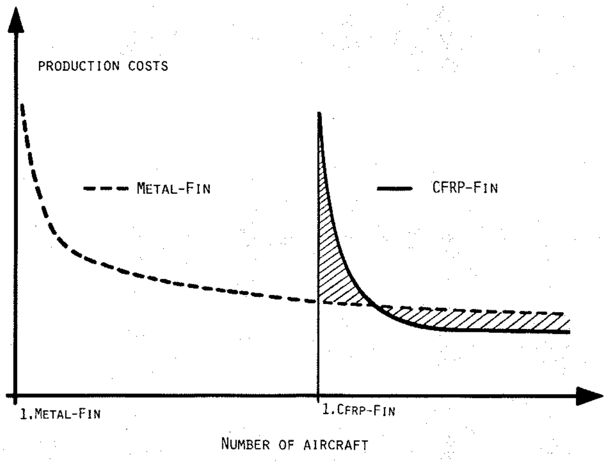


Fig. 5 Development of Series Production Costs for Metal and CFRP Fin in Principle

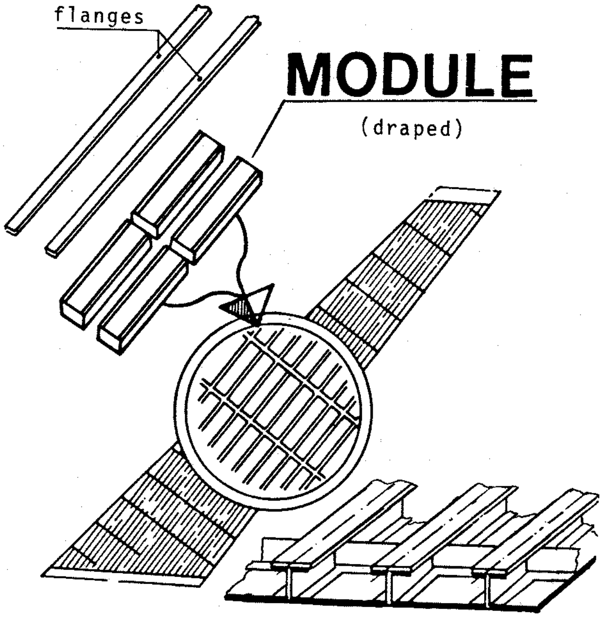


Fig. 7 Module Production Process

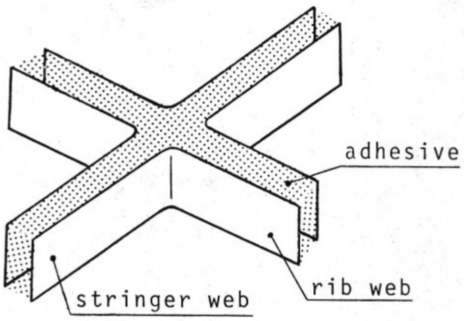


Fig. 8 Scheme of Web Assembly "Module Structure"

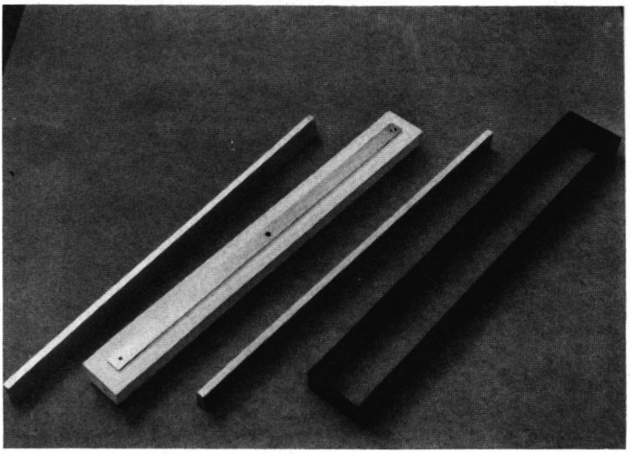


Fig. 9 Module Core made of 3 Sections and CFRP Module

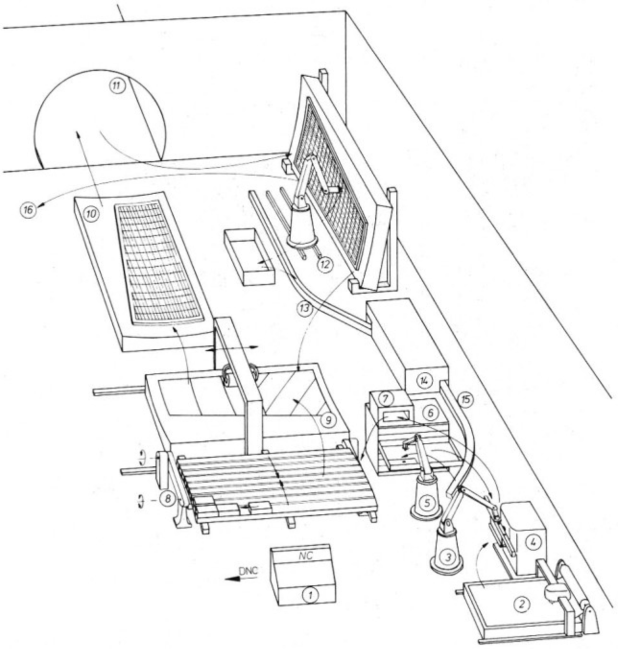


Fig. 10 Series Production Line for CFRP Fin Panels

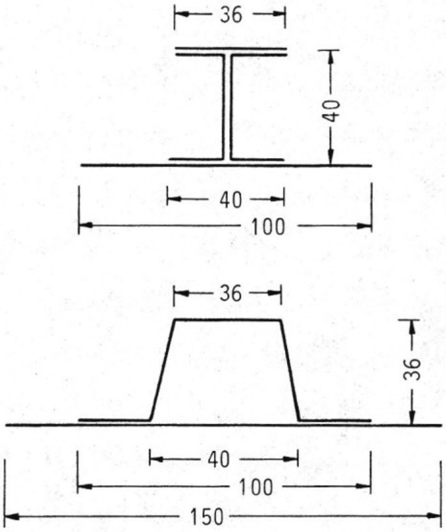


Fig. 11 Overall Dimensions of Stringers and Stringer Pitches for CFRP Fin Panels

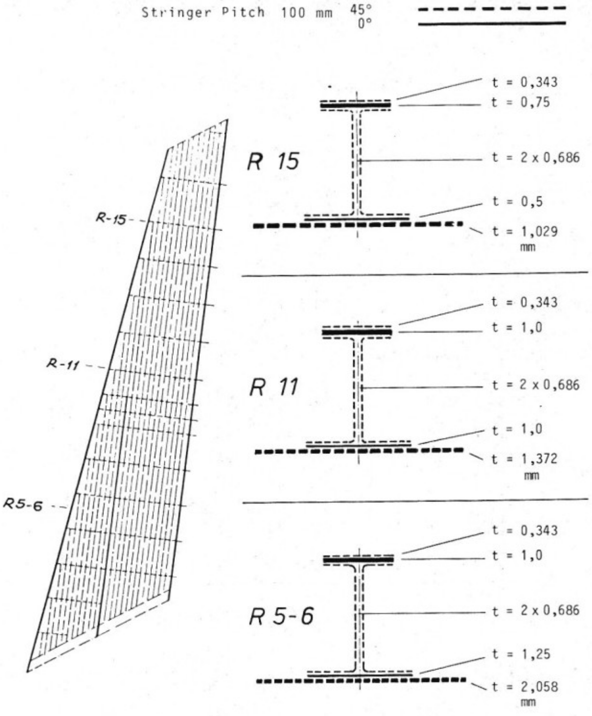


Fig. 12 Thicknesses of Stringers and Skin of CFRP Fin Panels

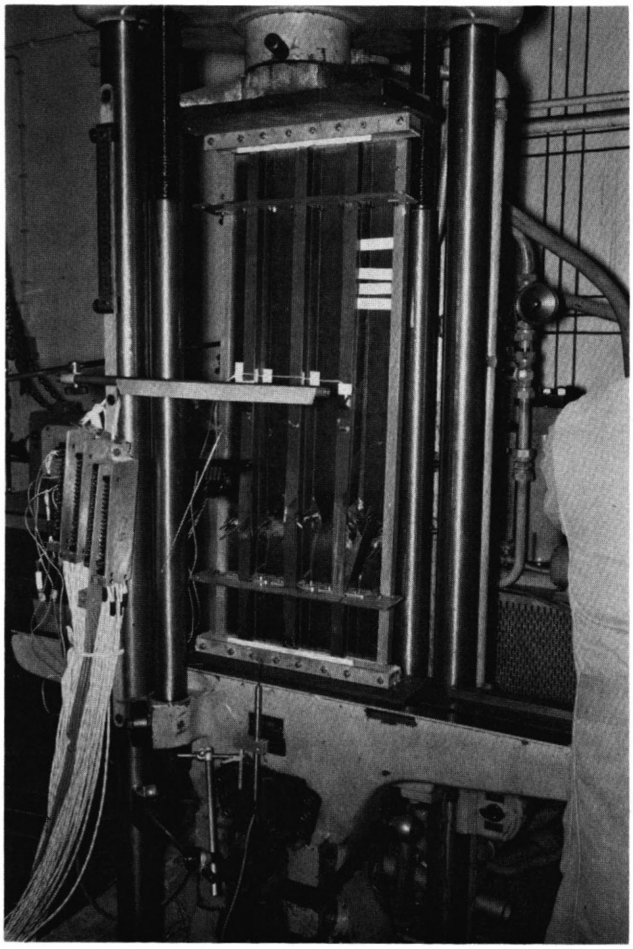


Fig. 13 Compression Test with
I-Stringer Panel 1000 x 350 mm

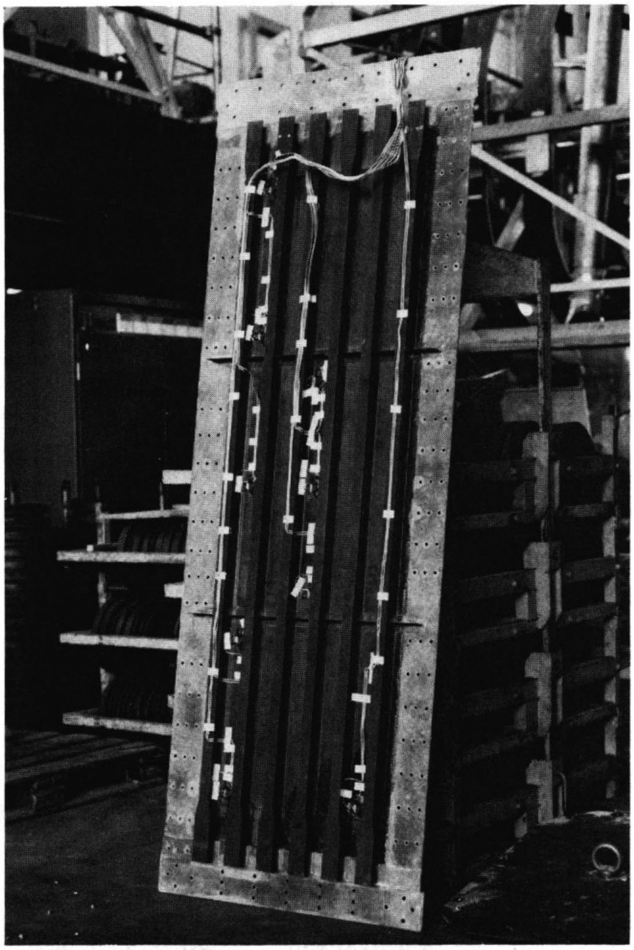


Fig. 15 "Module" Test Panel
2300 x 700 mm

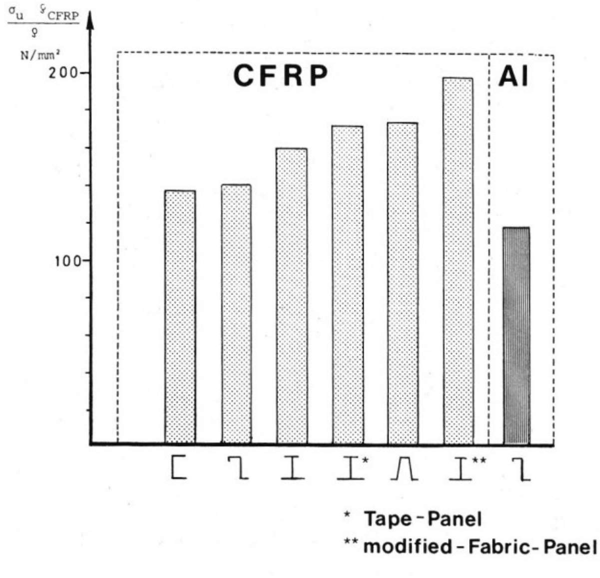


Fig. 14 Weight related Ultimate
Stresses achieved by
Compression Tests for
Rib Station 6/7 of
CFRP Fin Box

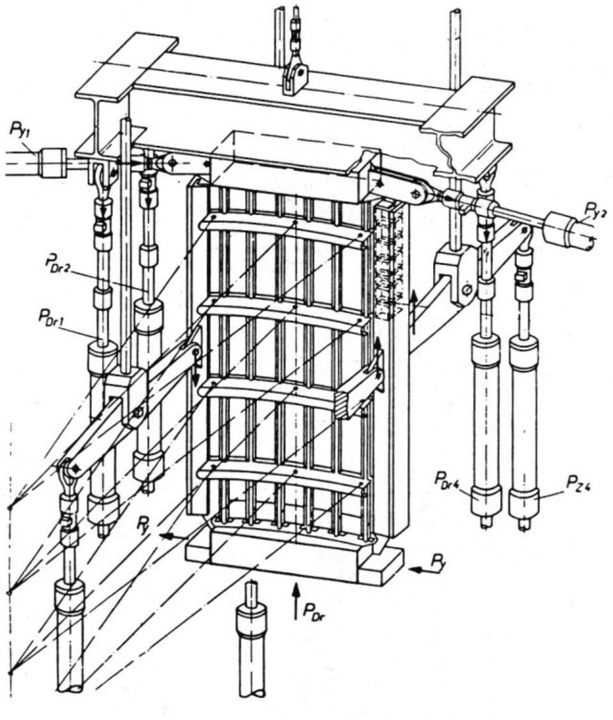


Fig. 16 Rig for Shear/Compression Test

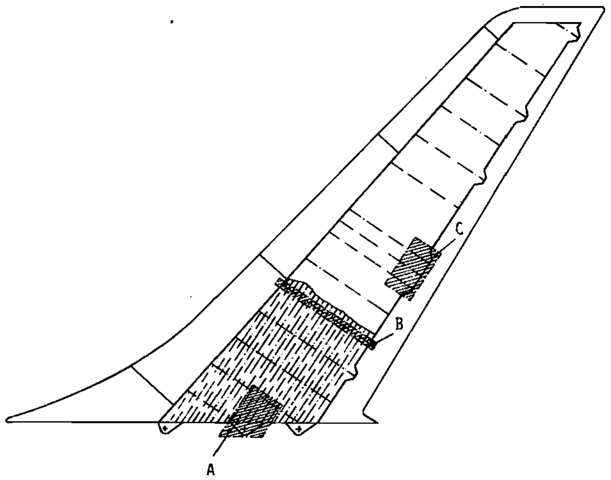


Fig. 17 Airbus CFRP Fin
Preliminary Design Work

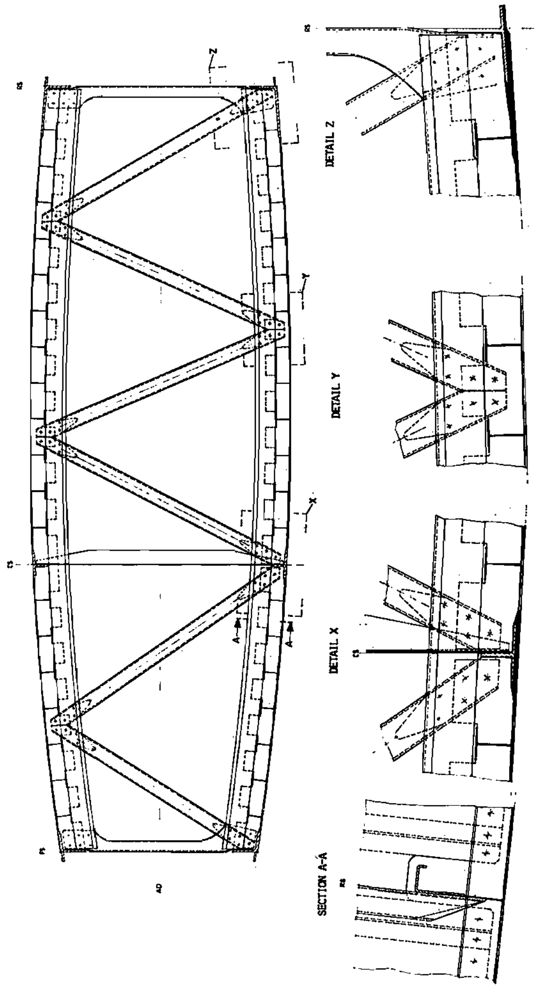


Fig. 17.2 Detail "B" - Rib 6

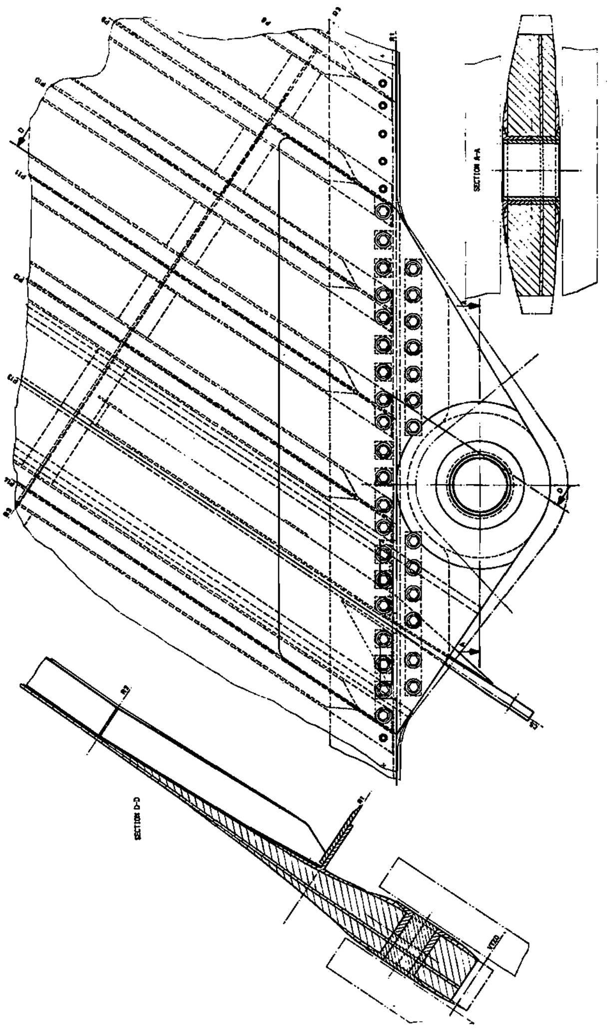


Fig. 17.1 Detail "A"
Fuselage Attachment Fitting

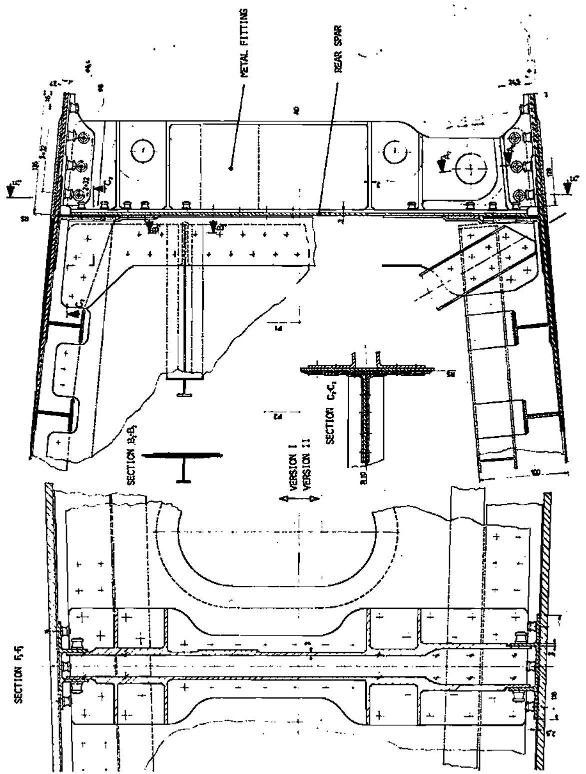


Fig. 17.3 Detail "C" - Rear Spar Area

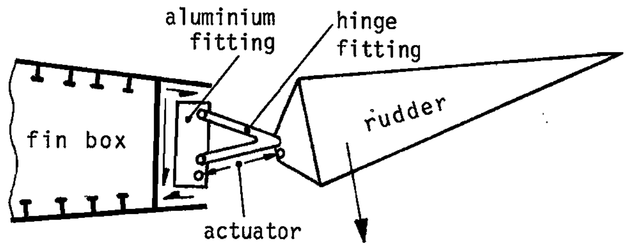


Fig. 18 Interface-Rudder/Fin Box

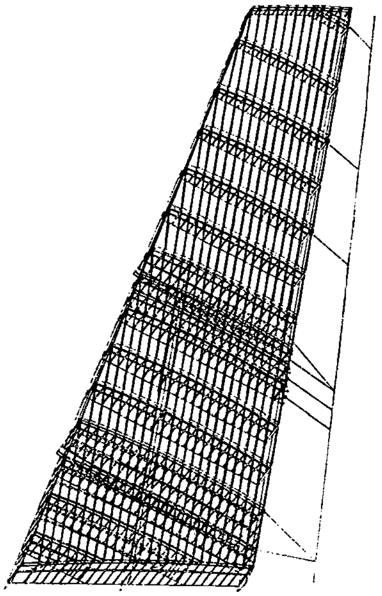


Fig. 19 Model for FEM Calculation

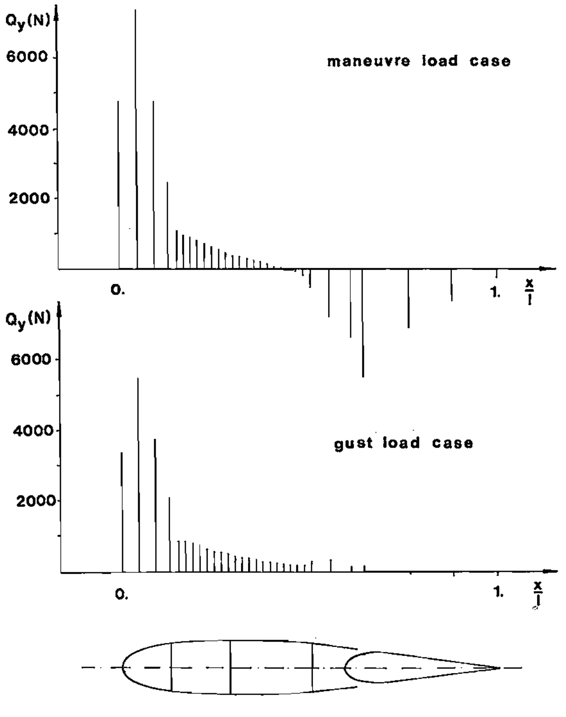


Fig. 20 Ultimate Node Forces at Rib 7

I. Aging Step with 60°C / 95 % RH
(Duration: Until the thickest ref.-probe reaches the service moisture content.)

II. Aging Step with 70°C / 70 % RH
(Duration: Until the thinnest ref.-probe reaches the service moisture content.)

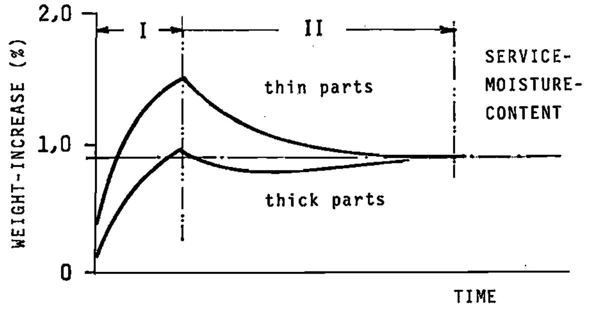
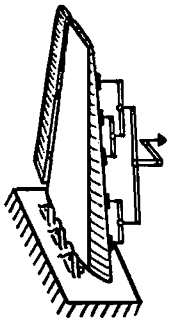
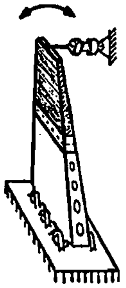


Fig. 21 Aging Process for Test Specimens



- O CHECK FOR STIFFNESS
- O FAIL SAFE-LOADING
- O ULTIMATE LOADING (J = 1.5)
- O $T_{MAX}$ = 70° C;
- O MOISTURE CONTENT = 0.9 WT %

Fig. 22 Full Scale Static Test



- O 120 000 TEST FLIGHTS, AMBIENT HUMIDITY AND TEMP.
  - CHECK OF STIFFNESS AFTER FATIGUE LOADING
  - CHECK OF CRACKPROPAGATION
- O DEMONSTRATION OF FAIL-SAFETY AND RESIDUAL STRENGTH
  $T_{MAX}$ = 70° C; RH = 0.9 %

Fig. 23 Full Scale Fatigue Test