ICAS-88-5.10.2

COMPOSITE SECONDARY AND PRIMARY STRUCTURES FOR PILATUS AIRCRAFT

Experience from the development and considerations for future applications

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ABSTRACT

Within an advanced composite technology programme, several design and manufacturing methods have been evaluated for a number of composite components on existing PILATUS aircraft, covering both secondary and primary structures. Cost effectiveness, weight reduction, certification requirements and in service implications are taken into consideration. Series production as well as prototype components are described covering technical and economical aspects of design and fabrication and compared to the respective metal version components, followed by a conclusive summary showing the potential of the outlined design concepts and manufacturing methods for future applications on the specific aircraft described.

1. INTRODUCTION

PILATUS is an independant company within the Oerlikon-Bührle-Group, manufacturing mainly three types of aircraft, namely the PC-6 light transport aircraft and the PC-7 and PC-9 turbo trainer types with a total average production rate of about 50 units per year.

An advanced composite technology R+D-programme was started four years ago in order to:

* Develop and define composite design and manufacturing technologies which are more advantageous than the existing metal manufacturing techniques regarding technical, economical as well as in service aspects for the application on existing and for future projects.

* Get experience in the certification of composite components.

* Introduce such composite techniques within the PILATUS design and manufacturing departments.

The programme was subdivided in two parts:

1) Composite manufacturing techniques utilizing the existing infrastructure at PILATUS have been developed and subsequently applied in the design and series production of composite fairings and other secondary structures for the existing aircraft types.

2) In order to evaluate the potential of composites for primary structures, a PC-9 composite horizontal tail prototype is developed.

Selected results of both programme parts will be presented in this paper.

2. COMPOSITE FAIRINGS AND OTHER SECONDARY STRUCTURES

2.1 Parts overview

Composite fairings have been used for over a decade on PC-6 and PC-7 aircraft. Being manually fabricated out of woven glass fibre materials and room temperature curing resins, they offered comperatively low manufacturing costs for parts with complex geometry, but exhibited no weight advantage over the aluminium structures and quite a high weight scatter.

In an attempt to reduce both weight and the scatter inherent in the traditional manufacturing methods, the following parts have been selected for advanced composite manufacturing evaluation and subsequent series production:

* PC-6 Extended wing tip of "H4" version
* PC-9 Main Undercarriage doors
  Wing tips
  Inertial separator air duct panel
  Cockpit interior lining panels

under the following objectives:

* Weight and / or cost reduction
* Manufacturing techniques adapted to existing infrastructure
* No restrictive operational/in service implications
* Hand laminating repair techniques must be applicable.

FIG.1 and 2 give an overview over all the series production composite parts on the PC-6 and PC-9.

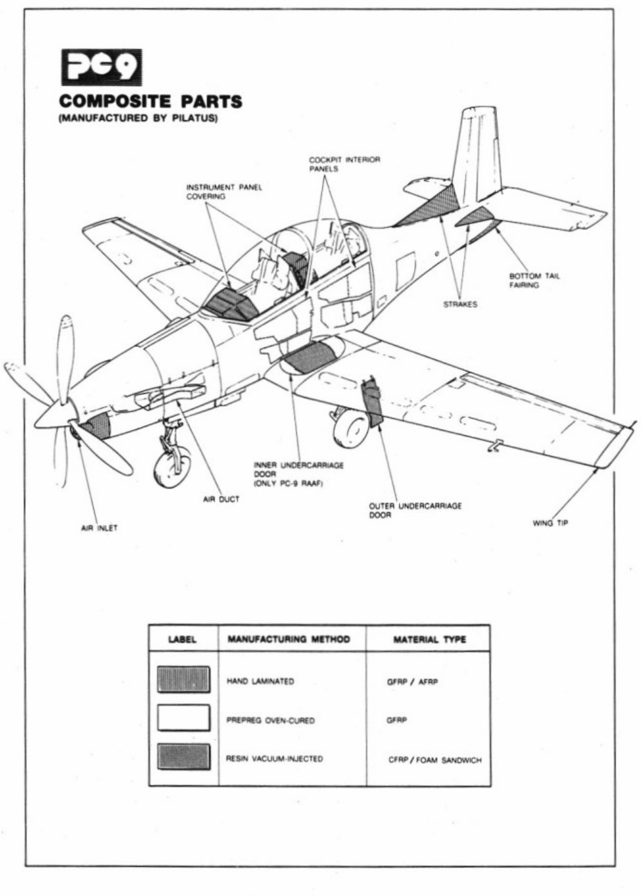


FIG.1 PC-9 COMPOSITE PARTS

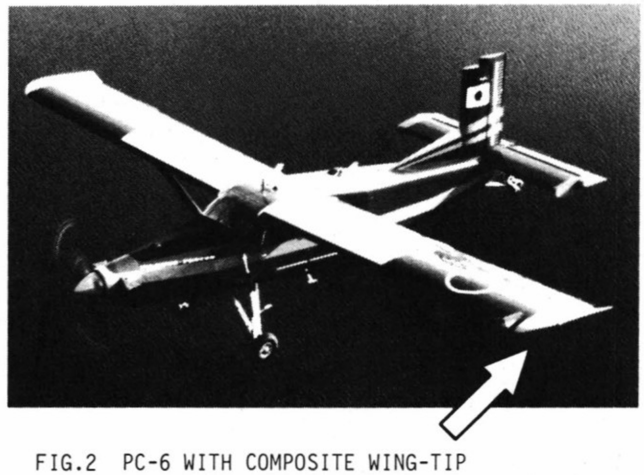


FIG.2 PC-6 WITH COMPOSITE WING-TIP

2.2 Manufacturing techniques

Based on the available infrastructure, the following fabrication techniques have been introduced:

* Prepreg low pressure (vacuum) - oven curing
* Resin vacuum injection using a heated mould

The tools used are mainly hand-laminated composite moulds, cured at room temperature and subsequently postcured to about 20 deg C above the respective component curing temperature.

Quality assurance requirements are met mainly by process control and co-cured traveller specimen testing.

For comparison purposes, prototypes of the same components also have been designed and manufactured using:

* Handlaminating
* Prepreg autoclave curing
* Prepreg heated press curing

In TABLE 1 a comparison is given, showing the results achieved for typical sandwich and monolithic skin/stringer type structures with the above manufacturing techniques applied.(1)

FIG.3 gives a comparison of the relative costs. The main cost drivers are the recurring costs, namely the costs for the curing (based on the differences in machine cost rates) and the lay-up manhours.

MATERIAL		PREPREG		WET RESIN / DRY FABRIC	
FABRICATION METHOD	AUTOCLAVE	OVEN	HEATED PRESS	RESIN VACUUM INJECTION	HANDLAMINATING
EQUIPMENT	AUTOCLAVE	OVEN	HEATED PRESS	OVEN / OIL HEATER	-
LAY-UP	BASE	SIMILAR MORE PERSONEL SKILL AND CARE MAY BE REQUIRED	SIMILAR	POSSIBLE ONLY FOR PARTS OF MODERATE CURVATURE AND SIMPLE LAY-UP. EXACT POSITIONING OF LAYERS MAY BE DIFFICULT	
CURING PROCESS	BASE	SIMILAR	SIMILAR	RESIN PROPAGATION DURING INJECTION DEPENDANT ON LAY-UP AND MATERIAL	LONGER CURING TIMES
LAMINATE THICKNESS	BASE	ONLY THIN LAMINATES	SIMILAR, GOOD DIMENSIONAL CONTROL	SIMILAR	SIMILAR
POROSITY	BASE	HIGH	SIMILAR TO HIGH	LOW	HIGH
FINISH	BASE	POOR, POWDER GEL-COAT APPLICATION MAY BE REQUIRED	SIMILAR TO POOR	GOOD	GOOD
SANDWICH CO-CURING	BASE FILM ADHESIVE MAY BE REQUIRED	SIMILAR	DIFFICULT CRUSHED CORE APPLICATION MAINLY	ONLY WITH CLOSED CELL FOAM CORE GOOD BONDING INHERENT TO PROCESS	POSSIBLE
MONOLITHIC SKIN/STRINGER	BASE	SIMILAR	DIFFICULT COMPLEX TOOLING REQUIRED	POSITIONING MAY BE EXTREMELY DIFFICULT	SIMILAR

TABLE 1 COMPARISON OF MANUFACTURING ASPECTS FOR DIFFERENT FABRICATION METHODS

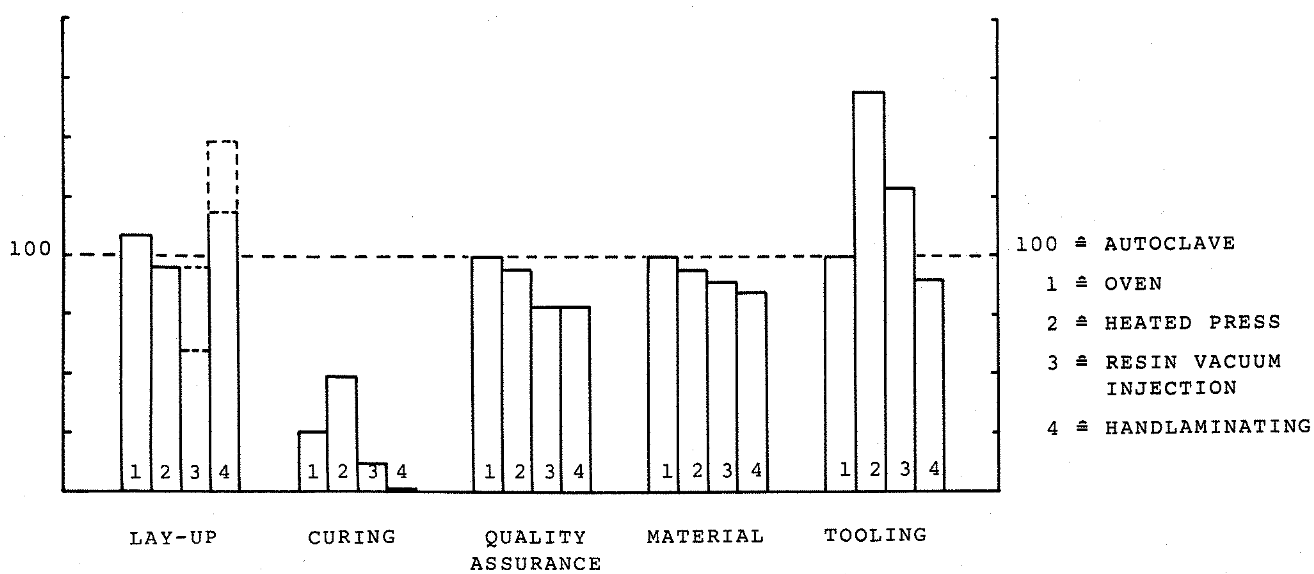


FIG.3 COMPARISON OF RELATIVE COST ITEMS FOR DIFFERENT FABRICATION METHODS

2.3 Parts description

Four of the parts developed are subsequently being discussed, illustrating the different design and manufacturing methods applied and investigated:

PC-6 H4 Wing Tip (See FIG.4):

The main design and manufacturing characteristics are:

Skin and ribs are mainly AFRP (Aramid Fibre Reinforced Plastic)-laminate using prepreg, low pressure / oven- cured at 120 deg C.

The skin is protected with one layer of GFRP (Glass Fibre Reinforced Plastic) and the edges are thickened with additional GFRP-plies for countersunk fasteners.

A co-cured epoxy powder gel coat is applied to the skin for surface finish improvement.

U-type ribs were used in the first prototype design. For the series production version the flanges for the rib attachment have been integrated in the skin laminate, eliminating a wing tip thickness tolerance problem for the rib to skin bonding.

The ribs are bonded to the skin using a room temperature curing paste adhesive and riveted for positioning.

FIG.4 PC-6 COMPOSITE WING-TIP

PC-9 Main undercarriage doors:

For the outer and inner undercarriage doors the following fabrication techniques have been investigated, including manufacture of prototype articles:

* Prepreg/vacuum pressure oven-curing
* Prepreg/autoclave curing
* Resin vacuum injection into heated mould

For the series production versions, the resin vacuum injection method has been selected.

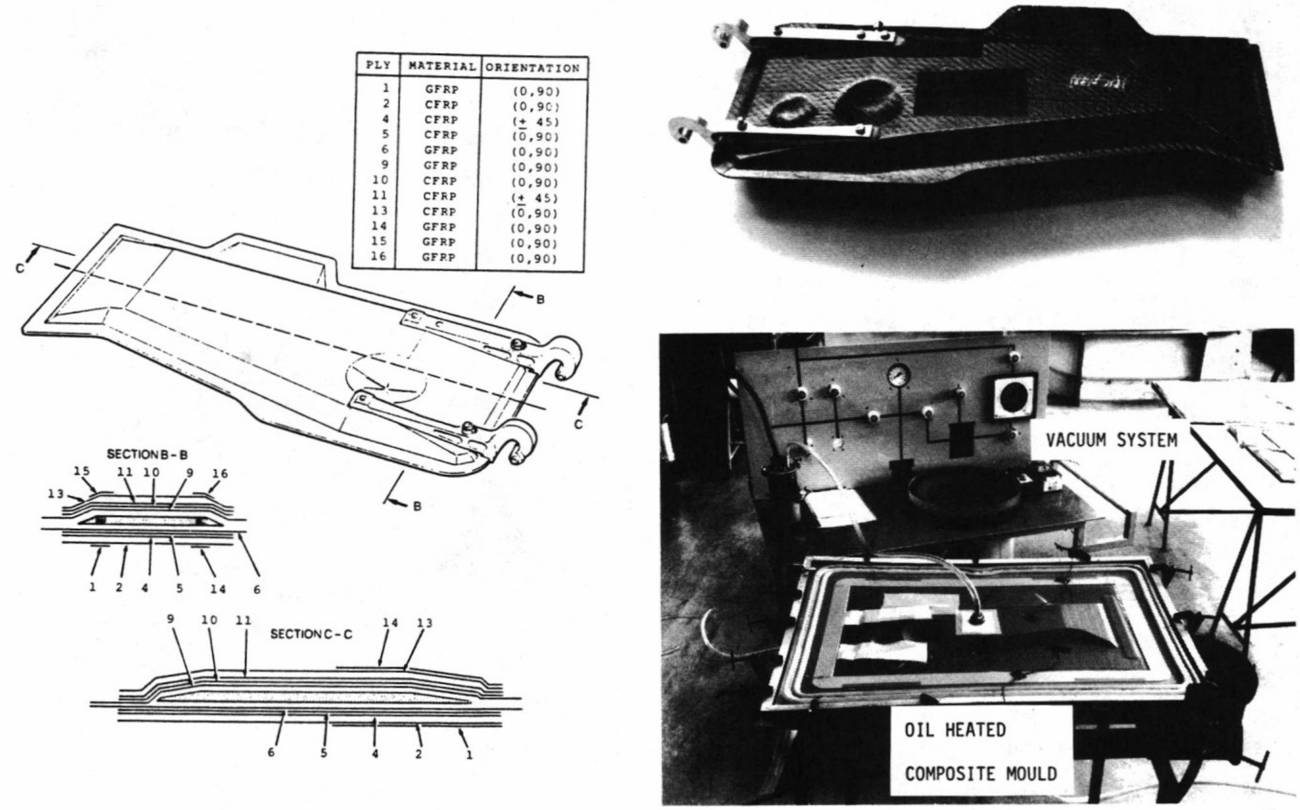
The doors are stiffness critical parts, a CFRP (Carbon Fibre Reinforced Plastic)-sandwich is therefore used.
A GFRP-layer is added on the laminate surface at the fitting positions.

For the vacuum injection version, the lay-up has to meet strength, stiffness as well as resin flow control requirements for the injection. The viscosity and curing characteristics of the resin system have to be balanced.
The selected system is Ciba Geigy LY 564/HY 2962.
The core material must be a closed cell foam.
The selected material is Divinycell (Diab Barracuda).
For the inner and outer door prepreg versions, Nomex-honeycomb as well as PMI-foam (polymethacrylimid) have been used as core material.

Details of the construction as well as a photograph of the injection moulding using oil-heated tooling are given in FIG.5.

PC-9 Wing Tip:

Using a two part negative tool, the wing tip prepreg-GFRP-laminate is oven cured as a single piece, applying an epoxy powder gel coat for surface finish. A prototype manufactured by resin vacuum injection moulding technique showed difficulties with fabric layer positioning and resulted in a large amount of waste material caused by the geometry of the part.



PLY	MATERIAL	ORIENTATION
1	GFRP	(0,90)
2	CFRP	(0,90)
4	CFRP	(± 45)
5	CFRP	(0,90)
6	GFRP	(0,90)
9	GFRP	(0,90)
10	CFRP	(0,90)
11	CFRP	(± 45)
13	CFRP	(0,90)
14	GFRP	(0,90)
15	GFRP	(0,90)
16	GFRP	(0,90)

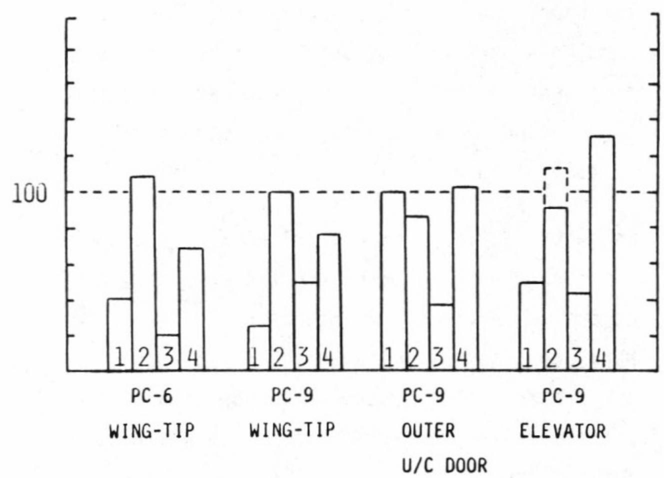
FIG.5 PC-9 OUTER UNDERCARRIAGE DOOR MANUFACTURED BY RESIN VACUUM INJECTION TECHNIQUE

PC-9 Inertial separator air duct:

A GFRP-prepreg component which as well is oven-cured.
Significant lay up time savings have been achieved compared to the original hand laminated parts. Manufacturing using a heated press required more expensive metal tooling.

2.4 Comparison with the respective metal versions

Since for most of the composite parts described above, sheet metal alternative prototypes were also designed and manufactured, a direct comparison with the respective composite parts can be made.
A comparison of the relevant design and cost data is given in FIG.6.



100 = METAL VERSION
1 = DETAIL PARTS RATIO
2 = WEIGHT RATIO
3 = NON RECURRING COST RATIO
4 = RECURRING COST RATIO

FIG.6 COMPARISON OF COMPOSITE AND METAL VERSION

Weight savings by replacing metal with composites could not be achieved due to the higher minimum laminate thickness requirements for composites compared to the quite common 0.5 mm (0.020 in) and 0.4 mm (0.016 in) sheet metal skin thickness.

Non recurring cost savings are achieved mainly by lower tooling costs, but additional costs for the specimen and the component testing must be considered. Despite the higher material costs and quality assurance efforts, the recurring costs can be kept lower than, or comparable to the ones for metal parts mainly due to low machine cost rates of the established "non-autoclave" manufacturing techniques.

2.6. Conclusions

Weight and costs are only two of many parameters that have to be considered when comparing metal and composite components for the application in aircraft structures.

General advantages of composites for example, their resistance against corrosion, their surface smoothness and the possibility of optimizing a component in its material properties according to the loading, significantly contribute to the overall picture obtained.

In general, the experience PILATUS gained up to now from the development and the application of composite secondary structural parts described above corresponds quite well with that of other aircraft manufacturers, showing a big potential for the application of composites for future aircraft. (2)

For the particular situation at PILATUS, having comparatively small production volumes and rates, the contribution of the established "non-autoclave" manufacturing methods related to the cost savings achieved is quite essential.
The established design and fabrication techniques will therefore be applied not only to similar, but also to different kinds of secondary structural parts of new projects.
A typical example of such a new application are engine cowlings.
FIG.7 shows a prototype of a GFRP/AFRP-Nomex-sandwich cowling component for the PC-9.

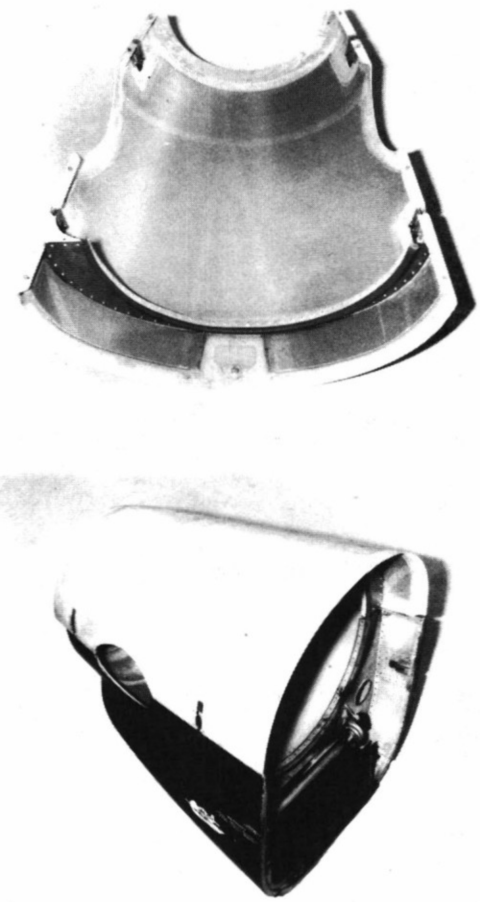


FIG.7 PC-9 COMPOSITE ENGINE COWLING

3. COMPOSITE PRIMARY STRUCTURES

Within the second part of the technology programme already mentioned a prototype horizontal stabilizer and elevator for the PC-9 are being developed in composites.
The major objectives of this part of the

programme are to provide a basis for the evaluation of larger primary structures in future aircraft programmes and to get experience in the certification procedures for primary composite structures, as which both the stabilizer and the elevator are classified.

3.1 Design aspects

According to the general programme objectives given above, the requirements to the design are:

* Weight/cost reductions, mainly achieved also by reduction of the number of detail parts.
* Advantageous operational and in service implications.
* Design and manufacturing methods applicable to future aircraft components.

The structural requirements for the prototype parts described are:

* Interchangeability with metal parts
* Identical load, stiffness and mass balance requirements as metal parts
* No buckling up to ultimate load

Environmental conditions:

The maximum structural temperature due to solar heating and the max. moisture content condition have to be considered as the most critical factors governing the material selection and subsequently the design and fabrication processes.

For a typical turboprop trainer aircraft like the PC-9, only a very short cooling period for the heat soaked structure can be assumed from taxiing to take-off, resulting in significant higher max. temperature values than defined e.g. for subsonic transport aircraft. (3), (4)

FIG.8 gives a temperature vs. time recording typical for a dark surface painted composite elevator for PC-9, showing a max. design temperature of + 93 deg C.

The design reference condition for max. laminate moisture content is 70 deg C/85% relative humidity.

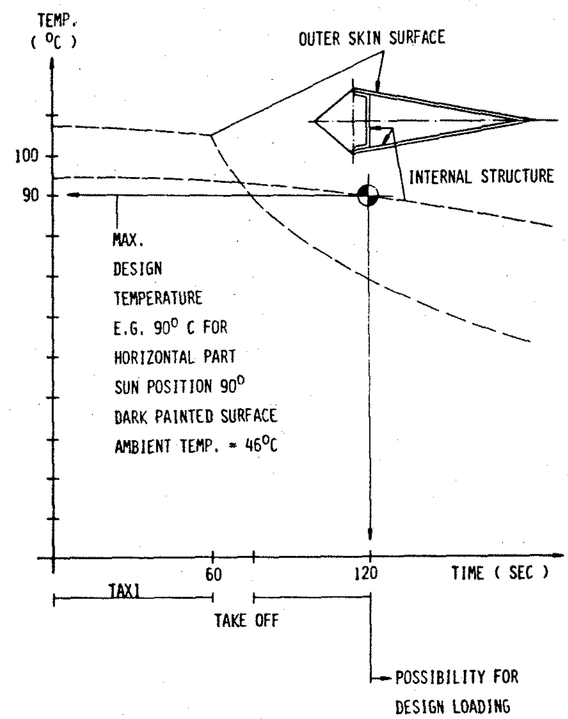


FIG.8 TEMPERATURE / TIME RECORD OF PC-9 ELEVATOR

3.2. Composite elevator PC-9

Design (see FIG.9):

The elevator is a semi-monocoque Nomex-honeycomb sandwich design comprised of U-type spars, a sandwich midrib and two ribs at the tip, all adhesive bonded to the inner face of the skin sandwich, thus allowing for impact damage on the outer sandwich face without weakening the bondline. The high and undisturbed chordwise skin stiffness also gives a better surface smoothness and allows to reduce the number of ribs.

The materials used are:

* CFRP fabric prepreg: AIK EHKF-251-245

* GFRP fabric prepreg AIK EHG-250-44
  both 150 deg C curing
* Honeycomb core Aramex 3.2-48
* Adhesive FM 73 M (film), AW 139 (paste)

PMI-heat resistant foam has also been evaluated as core material but has been rejected due to its critical impact damage behaviour in combination with the thin sandwich face laminate thickness and the need for additional pre-forming operations.
The lightning protection system is also shown in FIG.9. The sandwich skin of the horn region (lightning strike zone 1A/1B) is fabricated using GFRP only.
The aluminium profile forming the trailing edge is electrically bonded to the metal airplane structure.

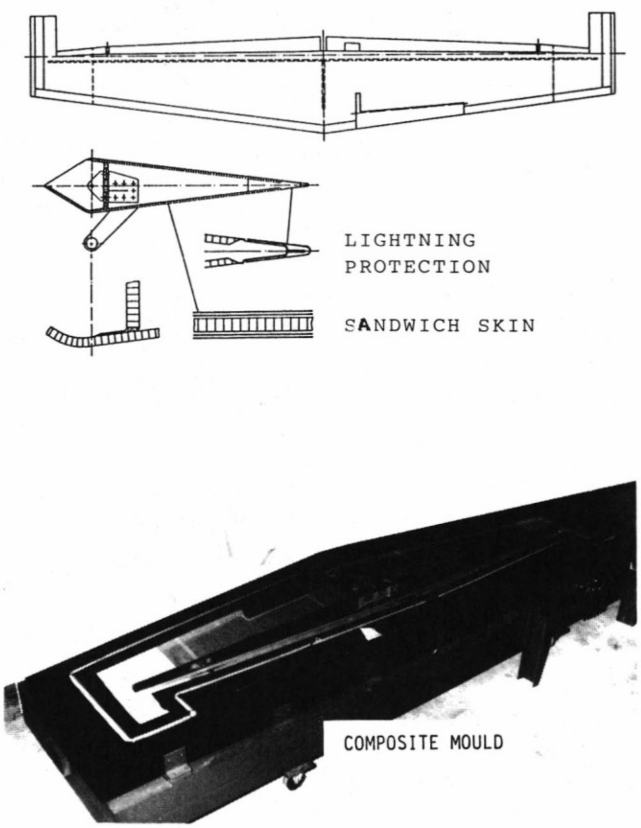


FIG.9 PC-9 ELEVATOR DESIGN AND TOOLING

Fabrication methods:

The skin and rib sandwich parts are separately autoclave cured at 150 deg C and subsequently adhesive bonded at 120 deg C.
Again an epoxy powder gel coat is applied to the sandwich outer face for surface finish improvement and sealing.
The introduction of reliable NDT-methods that are cost-effective under the restriction of a relatively small production volume is a major but still not satisfactorily completed task.
Thermovisual methods supported by Ultrasonic-methods are most promising. For the prototypes, manually performed US-testing was used.

Certification procedure:

Material test results form the basis of a static Finite-Element-analysis, which subsequently has been verified by static ultimate load testing of detail components as well as of the full scale prototype at ambient and elevated temperatures.
Moisture effects are accounted for by additional load factors, which were derived from conditioned specimen tests.
In order to allow for a good damage tolerance behaviour, the design strain allowables have been set conservatively low (limit strain allowable = 0.002).
The flying prototype has been proof-loaded up to limit load at room temperature and the results of this test have been compared to those obtained with the previous test article.
Fatigue and damage tolerance behaviour is demonstrated at specimen and detail component level.

Comparison to the metal version elevator:

FIG.6 gives the relevant data for the composite and the metal version respectively.

The structural weight of the unbalanced composite structure is 5% less, but for the balanced part no weight reduction has been achieved with this prototype compared to the metal version.
This penalty is the result of the described

lightning protection system, the requirement of interchangeability with the metal part, the minimum sandwich face laminate thickness and the conservative strain allowables.
The non-recurring costs again can be kept low due to the low tooling costs, but the costs of the additional testing for certification are quite considerable.
On the other hand the recurring costs of the composite elevator exceed the costs of the corresponding metal part, which is mainly explained by the relatively high operating costs of the autoclave.

3.3. Composite stabilizer

Design aspects:

The design objectives and structural requirements are identical to those set for the elevator.
Instead of a sandwich type construction a monolithic skin with integrated stringer design has been selected. Stringers are intergrated in both spanwise and in chordwise direction in order to minimize the number of ribs.
The materials used are CFRP tape and fabric as well as GFRP fabric prepreg impregnated with either AIK EH-251 or Fiberite 984 epoxy resin system.
Details are shown in FIG.10.

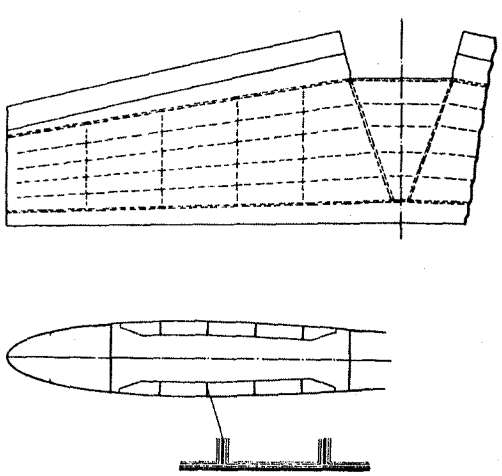


FIG.10 PC-9 STABILIZER DESIGN

For the specific primary structure described, the achievable weight savings compared to the respective metal component are limited by the no buckling criterion as well as by the minimum laminate thickness requirement due to impact energy absorbtion capability, fabrication and sealing criteria.
In addition the requirement of interchangeability with the respective metal part results in restrictions for an optimum composite design.

FIG.11 shows the relative weight vs. specific loading of a wing box typical for the horizontal stabilizer described in sandwich-, monolithic skin/stringer- and metal construction for the three design criteria, strength, buckling and minimum laminate thickness.
The loading at the critical section of the PC-9 stabilizer is taken as reference.
The figure shows that a potential for weight savings is given particularly for components being specifically heavier loaded than the described stabilizer component.

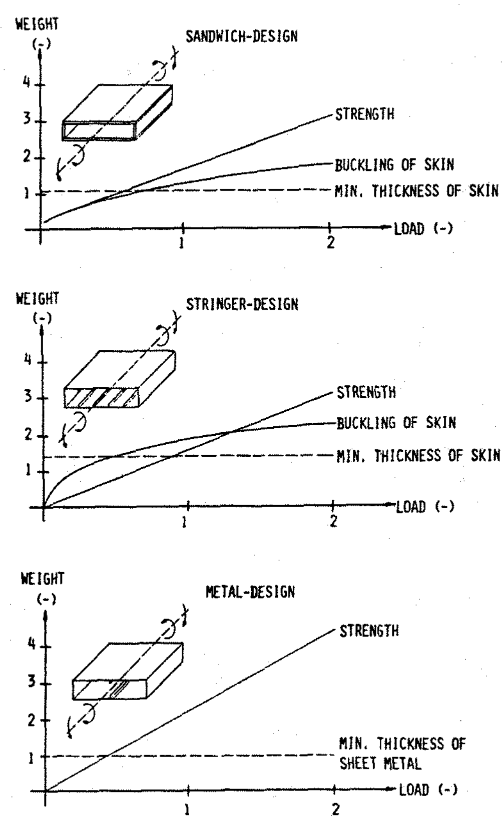


FIG.11 WEIGHT ANALYSIS OF DIFFERENT DESIGNS

Fabrication methods:

The monolithic skin is fabricated using a modular lay-up technique and autoclave curing.
The separatly fabricated spar and ribs are bonded to the skin using film adhesive.
The tooling is mainly composite.

3.4 Conclusions

For the application on primary structures of the category of aircraft described, composites show the biggest potential for weight savings on larger and comparatively heavy loaded components.

Due to the mass balance requirement, the lightning protection system at the trailing edge is crucial to the weight savings achievable for tail control surfaces.

For structural parts having less stringent c.g. position and lightning protection requirements, e.g. stabilizers, flaps or wing box structures, the potential for weight savings is considerably bigger.

Based on durability and damage tolerance specimen tests, the design strain allowables have to be set successively at higher levels in order to take full advantage of the capabilities of CFRP-composites.

Such testing also reduces the certification costs by replacing the full scale component fatigue/damage tolerance tests.

Manufacturing costs can be kept at levels comparable to metal constructions by careful design, minimizing the integration steps and introducing reliable quality assurance procedures.

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