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Full scale glare fuselage panel tests

R.W.A. Vercammen and H.H. Ottens

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ABSTRACT
A GLARE fuselage panel, representative of the crown section of the Fokker 100 fuselage just in front of the wing, has been tested in the curved fuselage panel test facility that was recently commissioned at NLR. Panels are loaded by internal air pressure resulting in tangential stresses in the panel and by axial loading representative of both the cabin pressure and the fuselage bending due to taxiing and gust loading. A fatigue test was performed in which 180,000 flights (two lifetimes) were simulated. After the fatigue test no damage was observed. The fatigue test was followed by static test to Limit Load and to Ultimate Load. Finally the panel was laded to failure at 1.32 Ultimate Load. This paper will describe the test set-up in some detail, demonstrate the obtained uniform strain distribution in the panel, show the fatigue loads applied at high test frequency and present the results of the GLARE fuselage panel tests which proof that the use of GLARE leads to a substantial weight reduction without affecting the fatigue or static strength.

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FULL SCALE GLARE FUSELAGE PANEL TESTS*

Roland W.A. Vercammen, Harold H. Ottens
National Aerospace Laboratory (NLR)
Amsterdam, The Netherlands

SUMMARY

A GLARE fuselage panel, representative of the crown section of the Fokker 100 fuselage just in front of the wing, has been tested in the curved fuselage panel test facility that was recently commissioned at NLR. Panels are loaded by internal air pressure resulting in tangential stresses in the panel and by axial loading representative of both the cabin pressure and the fuselage bending due to taxiing and gust loading. A fatigue test was performed in which 180,000 flights (two lifetimes) were simulated. After the fatigue test no damage was observed. The fatigue test was followed by static tests to Limit Load and to Ultimate Load. Finally the panel was loaded to failure at 1.32 Ultimate Load. This paper will describe the test set-up in some detail, demonstrate the obtained uniform strain distribution in the panel, show the fatigue loads applied at high test frequency and present the results of the GLARE fuselage panel tests which proof that the use of GLARE leads to a substantial weight reduction without affecting the fatigue or static strength.

INTRODUCTION

In fuselage design studies there will always be the necessity to test components in a realistic way. The fuselage panel test facility developed and built at NLR (fig. 1) offers the possibility to test fuselage skin sections, with curvatures ranging from panels of relatively small aircraft like the Fokker 50 to panels from relatively large aircraft like the Airbus A300, at a high fatigue testing speed (ref. 1). The fatigue test loads simulate flight simulation loading conditions by loads in circumferential direction caused by cabin pressurization and axial loads representative of both the cabin pressure and the fuselage bending due to taxiing and gust loading. The new fuselage panel test facility has also the possibility to perform static strength and residual strength tests.

In order to verify the applicability of GLARE A** as a fuselage skin material Fokker designed and built a Fokker 100 fuselage panel with a GLARE A skin and GLARE N stringers, representative of the crown section just in front of the wing. GLARE A as skin material is

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* This investigation has been carried out under a contract awarded by Fokker Aircraft B.V. according to the commitments made by Fokker in the Brite Euram IMT 2040 project "Fibre reinforced metal laminates and CFRP fuselage concepts".

** GLARE A=GLARE 3-2/1-0.3: 2*(0.3mm 2024 sheet) + (0.25mm cross-ply glass prepreg)
GLARE N=GLARE 1-3/2-0.3: 3*(0.3mm 7475 sheet) + 2*(0.25mm UD glass prepreg)
GLARE C=GLARE 3-3/2-0.2: 3*(0.2mm 2024 sheet) + 2*(0.25mm cross-ply glass prepreg)

weight favourable compared to GLARE C when the amount of necessary doublers for countersunk riveting is restricted to only doublers in the axial lap-joints. In the frame-skin attachments no doublers were required as the frames were connected to the stringers by means of cleats in stead of to the skin by means of castellations as is normally done by Fokker. Applying cleats led to a panel design with a threefold test objective:

- Verification of the applicability of GLARE A as a fuselage skin material for loading conditions which are representative of the crown section of the Fokker 100 fuselage.
- Generation of test evidence on the static strength and fatigue behaviour of rigid stringer-frame attachments in the GLARE fuselage.
- Determining the deterioration of the static strength of the GLARE skin after two times the design life (2*90,000 flights).

The overall panel dimensions are 1210 mm * 3030 mm containing five aluminium frames with a pitch of 500 mm, seven stringers with a pitch of 147 mm, seven stringers couplings and a longitudinal riveted lap-joint in the panel centre (fig.2). One of the frames is a Z-shaped frame, the others are C-shaped frames. A complete top section made of GLARE A with GLARE N stringers and rigid stringer-frame attachments has a weight that is 63% of the current Fokker 100 design in aluminium (ref.2).

TEST FACILITY

Fuselage skins of most aircrafts are subjected to biaxial loads, owing to bending and pressurization of the fuselage. In evaluating damage tolerance properties of candidate fuselage structures and materials, it is highly desirable that curved structures are tested under biaxial loading conditions. For this purpose one generally uses a barrel test set-up. This is a full-size cylindrical pressure vessel consisting of several interconnected fuselage panels. A barrel test set-up, however, has some features which make it less flexible and therefore less attractive for studies not directly related to a particular aircraft design. The radius of curvature is fixed, a large number of panels has to be tested simultaneously and the test frequency is rather low. In addition, barrel tests are expensive due to the large number of panels and the long testing time. The panel test facility at NLR was developed to avoid the forementioned disadvantages. In the curved fuselage panel test facility, which has flexibility with regard to panel diameter, panel width and panel length, a single fuselage panel can be tested at a relatively high testing speed.

The major components of the test facility are the main frame, the pressure chamber and the load introduction systems (fig. 1). The main frame is a very stiff steel structure. It consists of heavy bottom and top beams and two vertical main columns. A pyramidal shaped frame, which houses the hydraulic actuator, is mounted above the top beam and two auxiliary vertical columns. The panel is mounted in the frame such that the centre of gravity of its cross-section is in the working line of the actuator. The height of the test facility is about 7.2m, the width is about 4.5m. The pressure chamber is formed by a seal and base structure, connected to a transport system. The base structure of the pressure chamber is formed by a stiffened base plate and two support beams which are bolted to the vertical columns of the main structure. The base plate has a large central hole for air supply. At the front side the base plate has curved wooden blocks around the edges which form the side walls of the pressure chamber. An inflatable inner seal is mounted on the wooden blocks and the pressure chamber is closed by the panel. In order to accomplish an air-tight seal without net radial force acting on the panel edges, an inflatable outer seal is mounted at the outside of the panel just opposite the inner seal. The outer seal is bonded on the reaction frame, which consists of an open rectangular steel frame with curved wooden blocks. With the transport system the pressure chamber can easily be shifted aside during the test, which significantly improves the inspectability of the test panel. The chamber pressurization, axial loads and seals pressurization are regulated by a control system.

Pressurization of the fuselage panel is reacted to the test frame leading to tangential stresses in the skin and normal stresses in the frames. The ratio of the stresses in the skin and frames is determined and can be adjusted by the stiffness ratio of the skin-to-testframe and frame-to-testframe connections. The tangential stresses in the skin are taken out by bonded glass fibres. The loads in the frames are transferred to steel rods. Therefore the ends of the panel frames are locally reinforced. The wooden blocks have several holes through which the panel frame tensile rods are guided. The openings between the panel frame tensile rods and the hole edges are air-tight sealed with silicone rubber collars. In axial direction the panel-ends are loaded by rods which are connected to the panel-ends by steel brackets. At these axial panel-ends the stringers are ended and the stringers loads are taken by a gradually increased skin thickness (fig.2). This makes it possible to seal directly on the skin of the panel.

The advantages of using unidirectional glass fibres is that the loads are very evenly introduced over the length of the panel (ref. 3). Therefore hardly any distance is required for stress redistribution, i.e. the stress distribution is uniform at a very small distance from the panel edge. In addition, the use of unidirectional fibres does not result in local stiffening of the panel edges in axial direction. The length of the glass fibre sheets was chosen sufficiently large to limit the rotation of the fibres at the upper side of the panel owing to axial elongation of the panel. The small rotation that occur during the test alter not significantly the load transfer through the panel. The glass fibres are bonded to steel tangential plates. Because of their large width the tangential plates act more or less as hinges. The outward movement of the panel due to pressurization is therefore nearly radially and will not result in a significant change in the shape of the panel from circular to oval. The tangential clamping system was designed such that the angle between tangential plates and the vertical column was adjustable to allow for a large range of panel diameters to be tested.

PANEL LOADS

The GLARE panel is designed for the Fokker 100 loads in the crown section at Fuselage Station 14911. The axial loading sequences of the fatigue spectrum are derived from the spectrum applied in the Fokker 100 full scale test (ref.4). The axial load is written as:

$F_{axial} = a_1 * M_y + a_2 * \Delta p$ with

F_{axial} = axial load in fuselage panel M_y = bending moment at Fuselage Station 14911

Δp = cabin pressure $a_{1,2}$ = Fuselage Station dependent constants

The spectrum consists of 36 repeating testblocks of 5000 flights. Each testblock of 5000 flights is subdivided in four subblocks of 1250 flights. Three subblocks are exactly equal, the fourth block is equal but for one severest flight. Within this spectrum eight flight types with different gust loading severity have been defined. Figure 3 shows the axial loading and frequency per 5000 flights for these typical eight flight types. Each flight has five segments: ground, initial climb, climb/descent, cruise and approach. During the ground segments the cabin pressure Δp=zero, during the cruise segment $\Delta p=\Delta p_{max}$. In case of climb/descent the cabin pressure variates between zero and Δp_{max}.

The fatigue test is followed by static tests. The GLARE panel is subjected to one Limit Load case, two Ultimate Load cases and a failure strength test. These static load cases are intended to demonstrate that after two times the design life (2*90,000 flights) and possible undetectable cracks in the GLARE skin the residual strength is still sufficient to carry Limit and Ultimate Load. The Limit Load case equals cabin pressure $\Delta p=\Delta p_{max}$ plus Limit Load bending moment. The first Ultimate Load case is the cabin pressure Ultimate Load case: $\Delta p=2*\Delta p_{max}$. The second Ultimate Load case equals 1.5*[cabin pressure $\Delta p=\Delta p_{max}$ plus Limit Load bending moment]. The second Ultimate Load case is followed by the failure strength test at $\Delta p=\Delta p_{max}$ by increasing the axial load until failure of the panel.

STRAIN DISTRIBUTION

At fastening of a test panel the length of each axial tensile rod and frame loading rod can be adjusted with nut-keys such that over the panel width and length a uniform strain or stress distribution is achieved when the panel is loaded. The amount of load that is taken out of the panel frames also depends on the stiffness of the rods. The rods are therefore panel dependent parts and have load cells incorporated. The nut-keys and the output of the load cell in each panel frame can be used for proper adjustments.

The strain distribution in the middle of the panel, obtained by using the glass fibres, frame loading rods in tangential direction and tensile rods at the reinforced, non-stiffened panel-ends in axial direction, is shown in figure 4 and 5. These graphs visualize a smooth distribution of the axial and tangential strain levels and an uniform tangential strain distribution within 100 mm of the panel edges loaded with glass fibres. Strain level variations at the gauges R13a, R14c and R16a are due to the lap-joint, radius differences of panel halfs and stringer couplings. The tangential strain distribution over the panel length is given in figure 6. Over the panel width the influence of the stringers run-outs and skin reinforcements is negligible after the first frame (fig. 7).

At constant bending moment ($Fax_{bending}$=constant), the radial expansion of the panel increased due to raising cabin pressure. When, at constant cabin pressure ($Fax_{\Delta p}$=constant), the axial force was raised the radial expansion of the panel decreased. The panel displaced inwards to the pressure chamber to resist the increasing tangential contraction, due to the Poisson’s ratio effect due to the axial extension. The increase of the tangential contraction and the displacement of the panel inwards led to a decrease of the tangential strains in the skin.

FATIGUE TEST

During the fatigue test the defined 180,000 flights were applied. The testing frequency was about 10,000 flights per 24 hours. The axial loads had an accuracy of 0.5%, the chamber pressure reproduced within 3%. The panel was inspected by eddy current at the axial lap-joint, stringers run-outs and checked visually at the stringers couplings, doublers run-outs, stringers at the connection with the cleats, frames at the connection with the cleats and load introduction points of the frames. In accordance with the lifetime predictions (ref.2) no cracks or damages were found.

STATIC TESTS

The fatigue test was followed by static tests. The first static load case was the Limit Load test. For this load case figure 7 gives the axial strain levels from station zero till the fourth frame. Clearly visible is the pillowing of the skin. The lower axial strains in the second and third bay are caused by the stringers couplings and difference in stiffnesses of the frames. During this Limit Load test the panel extended linear, no permanent deformations took place. Some results of the first Ultimate Load case at 2*Δp, are visualized in figure 5. Figure 5 shows the tangential strain levels in the middle of the panel. The axial strain levels are less than 20% of the tangential levels. Like during the Limit Load test pillowing occurred and the panel expanded linear without permanent deformations. After the 2*Δp Ultimate Load case the second Ultimate Load test was applied. No final failure occurred and the strain distribution was, except for plasticity which occurred above Limit Load, conform the distribution during the Limit Load test. Conclusion: the Ultimate Load tests showed sufficient residual strength of the GLARE panel after two lifetimes fatigue loading.

To determine the deterioration of the panel after two lifetimes fatigue loading, the first few load steps of the Limit Load and Ultimate Load tests are compared with reference tests performed before the fatigue test. The deterioration of strain levels in axial direction was a small fixed value of 50 µstrain, caused by settling of the brackets, doublers and rivets. In tangential direction negligible differences ($\leq$ 1.0%) were to be noticed.

After the second Ultimate Load test, the cabin pressure was fixed at $\Delta p = \Delta p_{max}$ and the panel was axial loaded to failure. The axial strains are visualized in figure 8. Failure occurred at an axial load of 770 kN, i.e. 1.32* axial Ultimate Load. This axial load is 15% higher than the theoretically expected axial failure load (ref.2). The skin failed between the third and fourth frame at the cross-section of the last rivets of the stringers couplings (fig.9). Conclusion: in the crown section of the Fokker 100, GLARE A as fuselage skin material and GLARE N as stringer material are applicable. The skin, lap-joint and rigid stringer frame attachments (cleats) had enough fatigue and static strength.

CONCLUSIONS

The new full-scale fuselage panel test facility has shown to operate correctly. The load introduction in the panel in axial and tangential direction was uniform and realistic fatigue loads due to cabin pressure and fuselage bending could be applied at high frequency (10,000 flights per 24 hours).

The conclusions with regard to the Fokker GLARE panel are:
- No cracks were found in lap-joint, cleats, stringers, skin, frames and stringers couplings after the two lifetimes fatigue test (inspection was done visually and by eddy current).
- The panel as a whole did not deform permanent at Limit Load.
- At both Ultimate Load testcases no failure occurred.
- Final failure occurred at 1.32 * axial Ultimate Load, which is 15% higher than theoretically expected.
- GLARE A can be used as fuselage skin material in the crown section of the Fokker 100. GLARE N can be applied as stringer material.
- The rigid stringer-frame attachments (cleats) are applicable in the crown section of the Fokker 100 fuselage.

Because of the excellent fatigue behaviour with sufficient static strength and a weight reduction of 37% the designed GLARE panel proved the feasibility of GLARE as fuselage material.

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