A320 FULL SCALE STRUCTURAL TESTING FOR FATIGUE AND DAMAGE TOLERANCE CERTIFICATION OF METALLIC AND COMPOSITE STRUCTURE

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Abstract

The AIRBUS INDUSTRIE A320 commercial transport aircraft has been certified in February 1988 with consideration of the damage tolerance requirements as specified by FAR 25 Amendment 45 (1) and JAR 25.571 (2).

Details of full scale structural testing, performed for fatigue and damage tolerance certification, will be presented.

Part 1 of this paper deals with the two major full scale fatigue and damage tolerance tests for metallic structures, i.e.

- center fuselage/wing test
- rear fuselage test

The essential aspects of the test programs are given with special regard to the evolution of fatigue and damage tolerance testing of primary metallic structures of AIRBUS INDUSTRIE aircraft within the last 15 years.
The installation of composite horizontal and vertical tailplanes causes significant thermal loads and has therefore an important impact on the fatigue test program, which is presented in detail.

Results of fatigue testing are given and the repercussions regarding test specimen, in-service fleet and future production aircraft are demonstrated for an example.
Details of damage tolerance tests, performed to verify the planned structural inspection program, are presented.

Part 2 of the paper deals with the full scale structural testing of the composite vertical tail regarding static, fatigue and damage tolerance testing.
The major details of the A320 vertical tail design are given.

The certification philosophy according FAR and JAR regulations is presented with special respect to the damage tolerance requirements and the artificial damages considered in the test.

A description of the vertical tail test specimen is given together with the major test objectives as well as details of the test load program.
The test sequence of the A320 vertical tail full scale component test is described including the various tests along with the essential results.

I. Introduction

The AIRBUS INDUSTRIE A320 aircraft has been designed and certified under consideration of the current requirements. An essential part of the certification procedure is the execution of full scale structural tests to demonstrate sufficient static, fatigue and damage tolerance behaviour of the structure under maximum ultimate load and repeated operational loads, respectively.
The AIRBUS INDUSTRIE decision to design a composite vertical tail for the A320 aircraft required to justify the composite vertical tail by a separate test. Using the AIRBUS INDUSTRIE philosophy which had been successfully applied to the A300 and A310 metallic structures and to the A310 composite vertical tail, the A320 tests have been carried out according to the principles developed for the previous aircraft models. The static test was performed with a complete aircraft specimen in contrast to the fatigue and damage tolerance multi-section testing using three separate specimens for the metallic structure. The static, fatigue and damage tolerance testing of the composite fin has been done on one specimen.

This ICAS-constribution comprises two parts: Part 1 is dealing with the full scale fatigue and damage tolerance testing of the metallic structures, Part 2 with the full scale testing of the vertical tail composite structure.

Part 1 - Full Scale Structural Testing for Fatigue and Damage Tolerance Certification of Metallic Structures

II. Full Scale Fatigue Testing under Consideration of Current Requirements

The AIRBUS INDUSTRIE A320 commercial transport aircraft has been certified in February 1988 with regard to the damage tolerance requirements as specified by FAR 25.571 incl. Amendment 45 and JAR 25.571 and their interpretations by the four national airworthiness authorities involved in type certification (STPA-France, CAA-Great Britain, RLD-Netherlands, LBA-Federal Republic of Germany).
The current damage tolerance requirements, (1) and (2), and means of compliance, (3) and (4), prescribe, that the structure which is damaged to a limited extent is capable of sustaining the expected dynamic loads plus a predefined maximum static load (usually limit load) until the damage can be detected and repaired.

These requirements have already been applied to certify the AIRBUS A310 in March 1983 and A300-600 in March 1984. According to the AIRBUS INDUSTRIE philosophy which has been applied to all AIRBUS aircraft types, these requirements are fulfilled by theoretical justifications for all structural significant items (SSI's) which are supported by extensive fatigue and damage tolerance testing performed at full scale specimens.

The aim of A320 fatigue and damage tolerance testing, is to justify the design goals which are

- 24 000 flight crack free life
- 48 000 flight economic repair life

with consideration of a safety factor of 2.5 for metallic structure, i.e. 120 000 flights will be simulated in the full scale tests.

According to the content of the requirements and the AIRBUS philosophy, successfully applied during A300 and A310 development, the A320 testing is split into two major phases:

- phase 1 (60 000 flights) fatigue testing including initial flaws

- phase 2 (60 000 flights) damage tolerance testing with applied artificial damages

The full scale fatigue testing of the AIRBUS A320 metallic structure is performed on three separate specimens which are: -

- front fuselage (EF1) (CEAT/Toulouse)
- center fuselage/wing (EF2) (IABG/Munich)
- rear fuselage (EF3) (MBB/Hamburg)

so that all three portions of the structure could be tested simultaneously, but independently. This procedure allows the simulation of an optimized test load spectrum for each individual test specimen by maintaining a common basic spectrum. Additionally this multi-section testing leads to reduction in running and inspection time compared with single-specimen testing. This paper deals with the major details of fatigue testing of the two latter tests.

The overall test planning is shown at the example of the EF2 (center fuselage/wing), s. Figure 1.

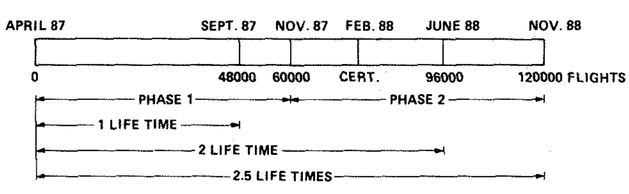


Figure 1: A320 Test Schedule
Center Fuselage/Wing Specimen

The execution of the test will take approximately one and a half year and at certification time (February 88) 1.5 life times have been simulated.

The actual status at time of preparation of this paper (End of April 88) is the following:

center fuselage/wing: 88 400 flights
rear fuselage : 73 100 flights

III. Description of A320 Full Scale Fatigue Test Specimens

A. Center Fuselage/Wing Specimen (EF2)

The EF2 test specimen is a part of the A320 airframe, with typical production configurations and comprises of the following primary structures:

- fuselage center sections incl. emergency exit doors
- wing box complete incl. fixed leading/trailing edge structure, engine/pylon support fittings, main landing gear support structure
- dummy elements (slat/flap tracks, pylon/engine assemblies, main landing gear)

An overview of the test specimen is given in figure 2.

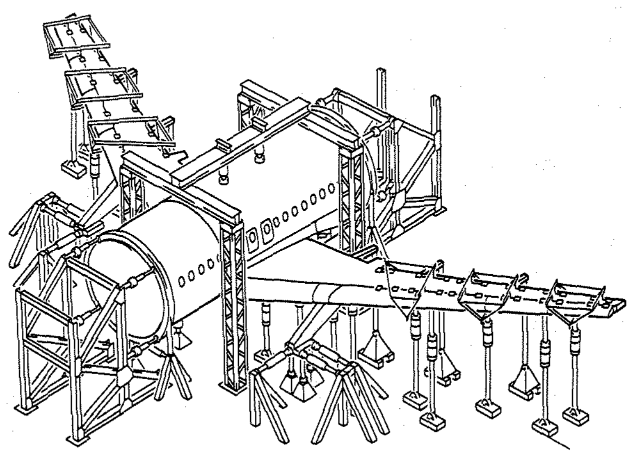


Figure 2: Center Fuselage/Wing Specimen
and Loading System

The simulated flight and ground loads are applied to the test specimen by means of 55 dual action hydraulic jacks (see also Figure 2), internal pressure is simulated additionally.
The test loads are applied through 172 bonded pad blocks to the wing structure and through 80 fittings attached to the passenger/cargo floor besides load introduction at dummy elements and steel bulkheads.
The specimen is statically determined supported by six rods which are fixed to steel bulkheads at forward and aft end of the specimen.
Both specimens, EF2 and EF3, are tested in a laboratory environment. For economic reasons the air volumes in the fuselage are reduced by easily transportable Styrofoam blocks which are supported on a track system.

B. Rear Fuselage Specimen (EF3)

The EF3 specimen consists of the primary structure of the rear fuselage including cargo and passenger doors, see Figure 3. The original horizontal tail structure made from CFRP, which is subject of a separate fatigue test with predefined environmental conditions, is replaced by a dummy element used for load introduction purposes. The vertical tail and rudder are also made from CFRP, and are tested separately due to the following resons:

- Contraction and expansion loads at the attachments due to the different expansion rates of the metallic fuselage and the CFRP vertical tail at different temperatures, require loads in opposite directions at both structures. This can only be provided in separate fatigue tests, because the vertical tail is statically indeterminately supported, unlike to the CFRP horizontal tail.
- The certification tests for the CFRP stabilizers have to be performed with due consideration to environmental effects which would greatly extend the test time at EF3.

In addition the tailcone structure is replaced by a dummy construction to provide realistic load introduction into the main support frame of the horizontal tail rear spar.
The simulated flight, ground and temperature loads are applied by 26 dual action hydraulic jacks, 18 of which located at the vertical tail attachments. Furthermore internal pressure loading is simulated. The specimen is supported and attached to the test rig, by a steel bulkhead at the front end. The steel bulkhead is removable for inspection purposes, and for filling and removal of the Styrofoam blocks used to reduce the air volume in the fuselage.

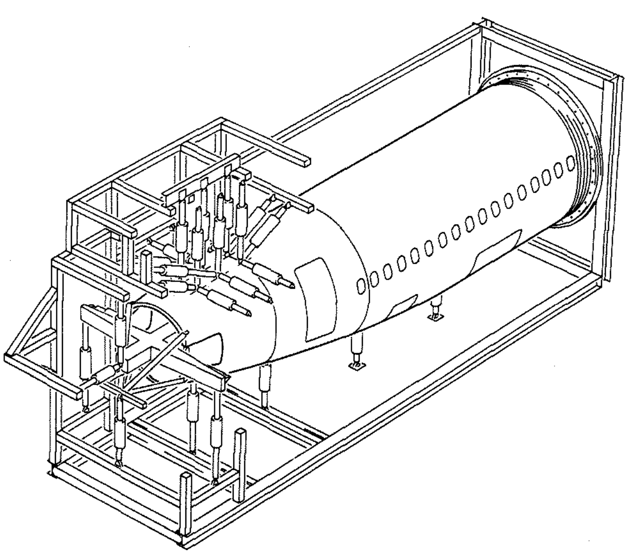


Figure 3: Rear Fuselage Specimen and Loading System

IV. A320 Fatigue Test Program and the Evolution of the Full Scale Fatigue and Damage Tolerance Testing of AIRBUS Aircraft

In this chapter the fatigue test programs for the A320 EF2 and EF3 specimens are presented with special regard to the evolution of the fatigue test procedures of AIRBUS aircraft during the past 15 years.

The first aircraft type produced by AIRBUS INDUSTRIE was the A300 B2 wide body, which had its first flight in 1972. The A300 versions have been followed by

- A310-200 (first flight 1982)
- A300-600 (first flight 1983)
- A310-300 (first flight 1985)
- A320-100 (first flight 1987)

Full scale fatigue tests had been performed for both, A300 B2 and A310-200, as multi-section testing in laboratory environment similar to the A320 as described in chapter III.
However, significant improvements have been introduced in A310 and A320 testing to achieve the most possible realistic testing of the structure, to fulfill additional requirements specified in the latest requirements according to chapter II and to improve the economic aspects of the test itself and resulting repercussions. The following improvements in comparison with the first A300 B2 multi-section testing have been made.

A. Improvements on Load Application

- Installation of three independent hydraulic jack rows, instead of two at slat tracks, wing reference line and flap tracks to achieve a more realistic representation of the chordwise aerodynamic load distribution and of exact loads at all movable surfaces (first applied to the A320).
- Arrangement of the bonded pad blocks at the wing in less fatigue sensitive areas, i.e. lower inner wing skin and upper outer wing skin, to improve the inspectibility for early detection and easy monitoring of natural and artificial damages (first applied to the A320).
- Arrangement of six instead of three hydraulic jacks at each engine/pylon dummy to represent the complex pylon spectrum in order to achieve correct pick-up loads and representative stresses in wing attachment structure (first applied to the A310).
- Arrangement of additional hydraulic jacks to simulate the undercarriage retraction jack loads, aileron loads and tailcone loads and to improve the accuracy of the results in these areas (first applied to the A320).

B. Improvements to the Fatigue Test Load Program

The fatigue and damage tolerance testing of the A300, A310 and A320 full scale specimens was performed by simulation of flight-by-flight loading with consideration of an average flight mission profile and for all the relevant loads for revenue and crew training flights (ratio 50 : 1).

In principle the A320 test load program has been based on the definitions taken from the two earlier aircraft types. However, significant modifications have been introduced to improve the accuracy of the results and to take into account specific A320 features such as the load alleviation function and use of tailplanes made from CFRP.
The major improvements are:

- An increase in the number of different flight types (especially for EF3 specimen), firstly to allow a greater variety of combinations between vertical and lateral loads to simulate round-the-clock-gusts and secondly to obtain the greatest range of combinations for the different flight types and different temperatures profiles.

- An increase in the number of flight segments simulated in test with corresponding gust/maneuver spectra so as to achieve a better accuracy in the simulation of gust/maneuver distributions versus flight.

- Definition of the low trunction level (cut-off of high loads at low frequencies) in order to achieve a crack growth rate which will not be reached by 99% of all aircraft.

- Definition of two different omission levels (cut-off of small loads at high frequencies) for phase 1 and phase 2 testing to achieve the best compromise between economic fatigue testing and accuracy of results.

- Consideration of non-linearities in the vertical gust loads due to introduction of a load alleviation system, i.e. the aircraft cross section loads due to vertical upgusts depend non-linearly on the gust velocities.

- Application of thermal loads resulting from the different thermal expansion and contraction rates of the fuselage and the CFRP vertical tail and the CFRP horizontal tail.

C. Major Details of the A320 Fatigue Test Program

For A320 flight-by-flight fatigue and damage tolerance testing the load spectra have been developed for all relevant ground and flight loads and are applied in different flight types with consideration of the following variations:

- 3 variations in ground load (taxi bumps) including 3 variations in landing impact loads (EF2 and EF3)

- 8 gust flight types representing smooth to severe flights in different weather conditions (EF2)

- 8 flight types combining gusts and maneuvers in normal operation (revenue) flights (EF3)

- 2 flight types combining gusts and maneuvers in crew training flights (EF3)

- 4 variations in each of the a.m. 10 flight types due to different combination of gusts and maneuvers (EF3)

- 7 temperature profiles to represent the thermal loads between the metallic fuselage and CFRP tailplanes (EF3)

These variations result in a total of 21 different flight types for the EF2 specimen and 324 different types for EF3 specimen which are an optimized realization of the complex loading of in-service aircraft (revenue and crew training flights) to achieve full representative crack initiation and crack growth results.
The sequence of loads in the flight segments is randomized. Load time histories for typical flight types (EF2) are shown in Figure 4 for a center/rear fuselage station in rear spar attachment area of wing:

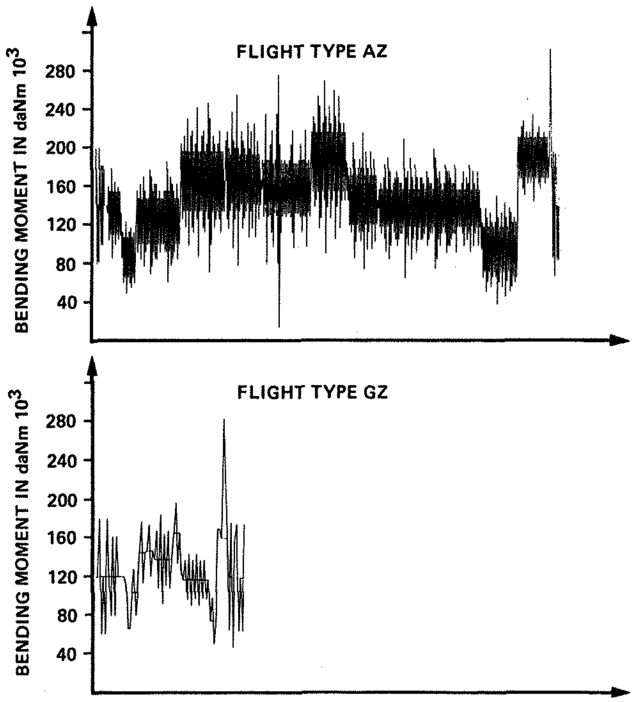


Figure 4: Typical Flight Type Loading

The number of test load points varies from 63 in a smooth flight to 2288 for a rough flight resulting in a test time of 1:42 minutes and 46 minutes, respectively.
The different flight types are applied in blocks of 4800 flights (1/10 of the design aim) in which the rough flight types (A to D) are positioned on an "equally" spaced distribution and the smooth flights (E to H) are positioned on the results of a random number generation program.

The thermal loads which have to be applied to consider the different expansion and contraction of the metallic fuselage and the CFRP tailplanes vary with the differences between the actual temperature and the temperature at time of assembly (room temperature, assumed to 20^o C). Seven different temperature profiles had been defined for the testing of the A310 CFRP vertical tail to consider operation in different climatic zones. The same temperature profiles have been used for both, A320 full scale fuselage testing and CFRP vertical tail testing.

Figure 5 contains the defined temperature profiles and examples for the resulting loads at the fuselage attachment lugs to vertical and horizontal tailplanes.

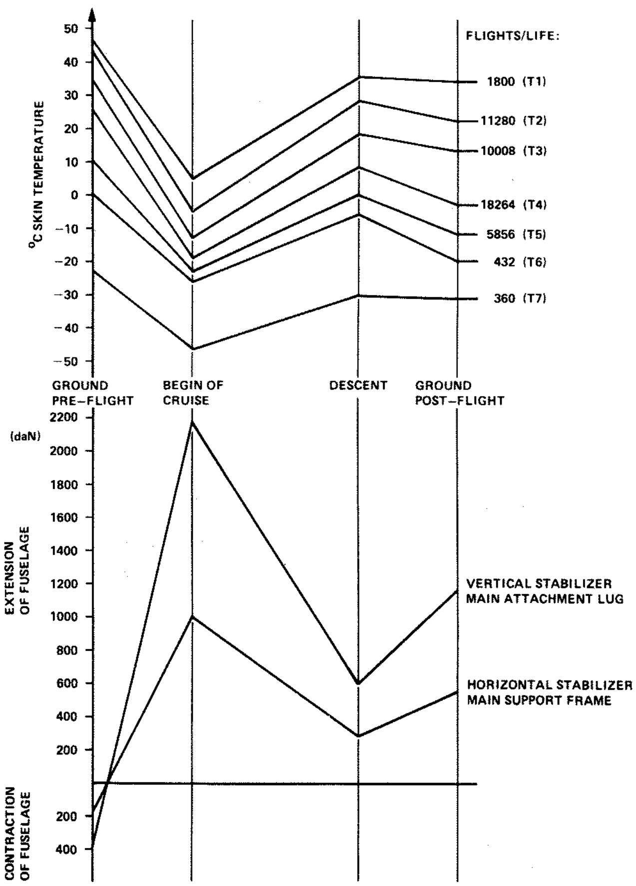


Figure 5: Temperature Profiles and Resulting Loads for Average Profile T4

The resulting action loads at the fuselage attachments, are applied during fatigue testing directly by hydraulic jacks.

Since the A320 is the first AIRBUS INDUSTRIE Aircraft equipped with a load alleviation system special attention has been given to achieve an accurate test representation. The load alleviation system is installed to reduce the vertical gust loads in order to achieve reduced load levels for static and fatigue design and to improve the comfort for the passengers. The installed system reduces incremental gust loads for C.G. accelerations exceeding 0.3g by automatic activation of the spoilers. The load alleviation system leads to a significant reduction of the upgust spectrum in area of lower frequencies as shown in Figure 6.

The overall effect of the reduced spectrum loads is the increase of the crack initiation life especially for the wing. Since the load alleviation leads to truncation of the higher loads which results in a reduced retardation effect (reduced zone of plasticity) and increased crack propagation rates which have been calculated in the pre-development phase and were verified by extensive coupon testing (5).

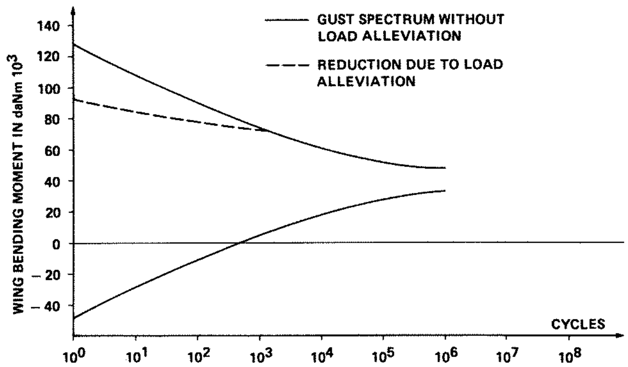


Figure 6: Vertical Gust Spectrum during Cruise at Wing

For consideration of the effect of load alleviation system additional gust load cases (8 for EF2 and 17 for EF3), independent from the linear calculation, have been introduced for C.G. acceleration greater than 0.3g to take into account the non-linearity in higher loads.

The definition of the omission level (lower spectrum truncation) has a significant effect on the crack growth rates as investigated by many authors, e.g. (6) and (7). Since these results are not fully transferable to A320 aircraft because of different spectrum shapes special investigations at CCT-specimens have been carried out for the original A320 spectrum (5) with different omission levels (13, 25 and 50 cycles per flight) for the three materials mainly used for the wing box structure (2024T351, 7010T7651 and 7150T651). The investigation revealed minimum crack propagation life for an average of 25 cycles per flight, see Figure 7.

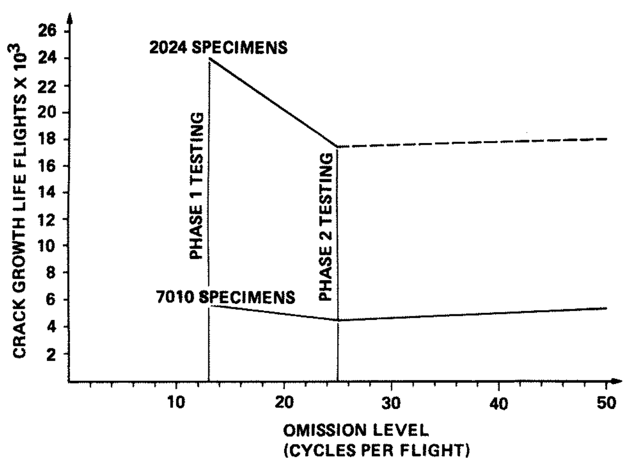


Figure 7: A320 Coupon Tests Results Regarding Omission Level

According to the a.m. results an average number of 25 gust cycles per flight has been introduced at 72 000 simulated flights to achieve most conservative crack propagation results during phase 2 testing. Due to the negligible influence of the omission level on the crack initiation life and due to economical reasons a reduced number of 13 gust cycles were chosen during phase 1 testing, which is in line with earlier decisions made for the A300 and A310 EF2 testing.

V. Results of the A320 Fatigue and Damage Tolerance Testing

A. Fatigue Test Results

The following chapter contains the results of the EF2 and EF3 fatigue testing up until the time of writing. The detailed results are for the fuselage parts under MBB responsibility.
The fatigue tests up to now have fully confirmed the design of the fuselage structure under fatigue and damage tolerance aspects. Since during the development phase, the allowable stress levels had been defined for the design goal of 48 000 flights for all fatigue sensitive areas, no significant damages occured in the areas where stresses can be determined by linear calculations. A few damages have been detected, in areas which are mainly loaded due to constraint deformation, as experienced by many aircraft types.
According to the nature of the loading the crack propagation rate of these damages was very slow in the most cases.
An example of one damage caused by constraint deformation is shown in Figure 8.

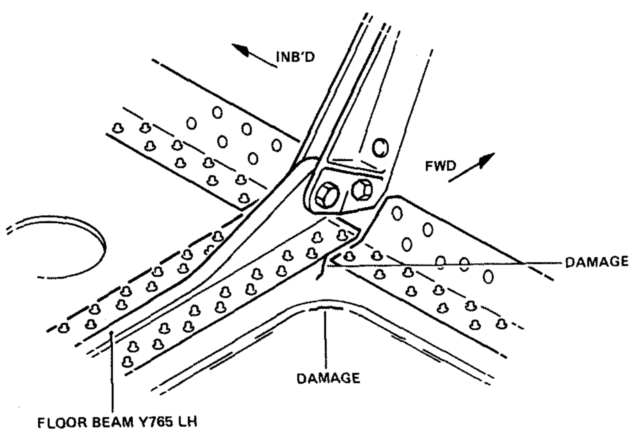


Figure 8: Fatigue Damage in Pressure Deck Membrane at Frame 42

A corner crack had been found in the diaphragm initiating at a rivet hole due to local buckling caused, by constraint deformation. This damage, found at 16 765 flights, led to a modification consisting of reinforcing the diaphragm itself by increase of thickness, installation of an additional tapered support in the concerned corner and cold expansion of rivet holes. This modification was introduced on all A320 aircraft before delivery and also to the EF2 test specimen showing no damages once more.

B. Damage Tolerance Tests

Damage tolerance testing at full scale specimens is carried out to meet the current requirements and to verify the theoretical investigations in the field of fracture mechanics. Therefore several detail artificial damages have been installed in the center fuselage/wing (56 damages) and the rear fuselage (34 damages) test specimens, the majority of them after simulation of 60 000 flights. The primary purpose of these damage tolerance tests is to verify the stress distributions, assumed failure modes, the damage tolerance analysis results and finally the defined inspection intervals.
These investigations are partly a repetition of earlier verification crack propagation tests performed during A300 and A310 full scale fatigue testing, (8) and (9). Comparisons of crack growth calculation and test results have shown, that the MBB calculation method and also the new proposed retardation model lead to conservative results for fuselage structure. A good correlation between test and calculation results can be achieved by consideration of the reduced stress intensity due to biaxial stresses acc. to new MBB crack growth tests under biaxial loading (10), (see results for a circumferential crack in the stiffened upper fuselage panel under flight-by-flight loading).
A summary of the artificial damages introduced in the A320 fuselage parts under MBB responsibility is shown in Figures 9 and 10. These areas represent structural significant items (SSI) which were included in the A320 certification analysis and for which inspections have been defined in the frame of the structural inspection program. The following major items are covered by damage tolerance testing (MBB design part):

- circumferential joints
- longitudinal joints
- cut-outs for doors, windows, antennas, fittings etc.
- pressure floor and keel beam
- attachment lugs for horizontal and vertical tailplanes

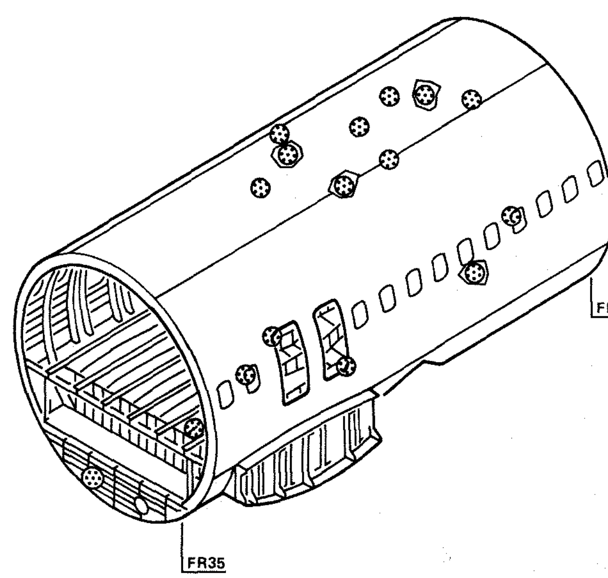


Figure 9: Artificial Damage Locations Center Fuselage

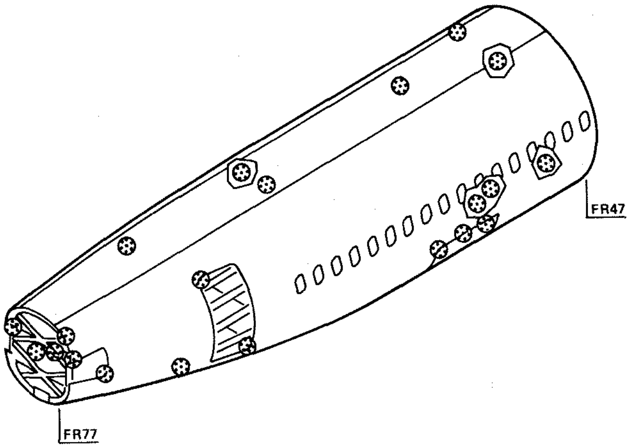


Figure 10: Artificial Damage Locations
Rear Fuselage

Examples of the first three areas are presented in the following.

Circumferential Joint. Figure 11 shows details of the structure of a typical circumferential joint and the damage installed.

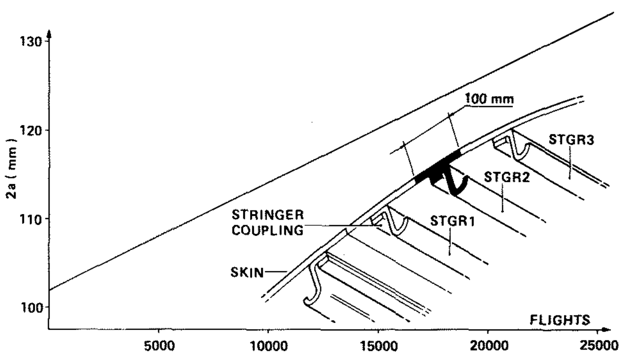


Figure 11: Artificial Damage at Circumferential
Joint, Upper Shell

A 100 mm circumferential skin crack was installed in the stringer coupling area at Frame 47 near crown center line above a cut-through stringer and coupling. The goal of this test is to show the efficiency of external detailed visual inspections as defined in the structural inspection program. The crack propagation, observed over a period of approx. 25 000 flights, was slower than predicted using uni-axial stresses in calculation. The difference is small in case of consideration of bi-axial stresses.

Longitudinal Joint. Figure 12 shows the design of a typical A320 longitudinal lap joint and the location of the crack retarders installed at each frame intersection.
The goal of this test was to verify predicted crack growth behaviour in the inner panel lower rivet row and the intended crack turning effect in crack retarder area which allows external visual inspections of a previously hidden crack. To date the crack turning is verified and the crack propagation is slightly slower than predicted using earlier experience from similar design features and A300 and A310 aircraft. The small differences can be explained by the different stiffness rate at A300/A310 and A320 aircraft.

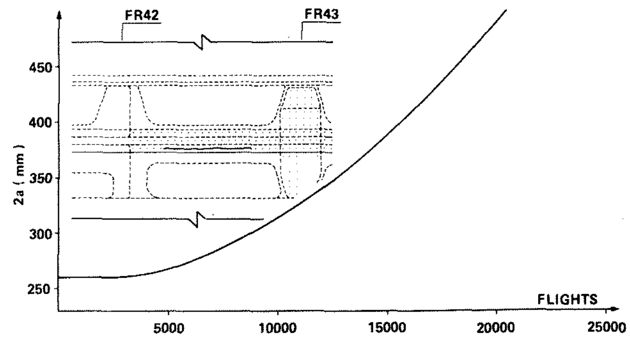


Figure 12: Artificial Damage at Longitudinal
Joint, Inner Panel Lower Rivet Row

Skin Panel at Rear Emergency Exit Door Cut-Out.

Figure 13 gives design details of the highly stressed aft lower cut-out corner.

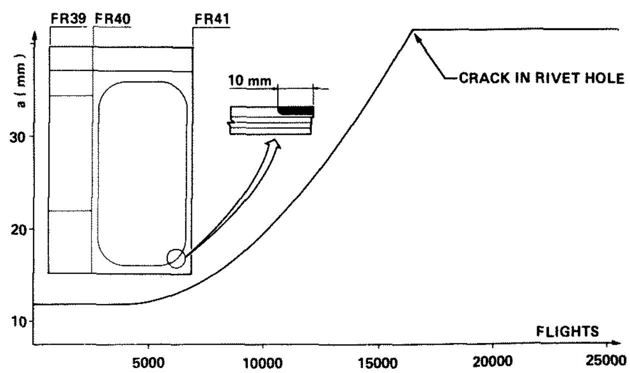


Figure 13: Artificial Damage at Emergency Exit
Cut-Out, Skin Panel

The crack is installed in the skin panel which is above three corner doublers.
This crack is from special interest since the stress intensity at the tip end is highly influenced due to stress redistribution caused by the undamaged doublers. To date an acceptable correlation between test and calculation is noticed. However, further comparisons are necessary up to the end of the test because of the increased influence of the intact doublers on the stress intensity.

Part 2 - Strucural Testing of the Airbus A320 Full Scale Composite Vertical Tail

VI. Description of A320 Vertical Tail

When the A320 project was launched in 1984 the A310 carbon fibre fin box, which is the first carbon fibre primary structure in civil aviation series application, was just in the component test phase with the full scale test starting at the end of 1984.

Nevertheless, there was great confidence in the performance and safety of carbon fibre structures and especially in the design of the A310 carbon fibre fin box, so that within AIRBUS INDUSTRIE the decision was made, to develope, manufacture and certify a full composite vertical tail for the A320 without a metal back-up solution which in case of the A310 was available.
To reduce development and manufacturing risk and costs, the main design features of the A310 composite vertical tail were also applied to the A320 composite fin layout. Consequently, the A320 fin is fixed by six main attachment fittings and three transverse load fittings at the metallic fuselage. With a total height of 5.87m the vertical tail consists of glass fibre sandwich leading edges, a monolithic carbon fibre torsion box, a rudder with glass/carbon fibre side panels and monolithic carbon spar web, glas/carbon fibre trailing edge panels and fairings. The materials in use are: Ciba 913C, 916G, Hexcel F550 and AIK-EHG.

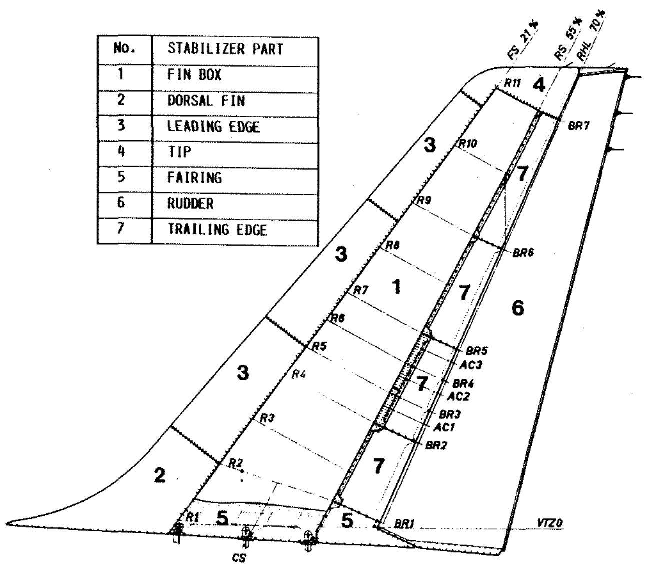


Figure 14: A320 Composite Vertical Fin

The torsion box is assembled by two side shells stiffened with I - stringers, front and rear spar web with beaded maintenance holes, a center spar web stub in the root area, 11 web ribs, several fittings for the rudder control system and seven rudder hinge arms.
Most of the parts are manufactured in the modular technique, which was developed by MBB for the A310 torsion box and is explained in (11).
The lightning protection system of the vertical tail consists of non-conductive glass fibre leading edges, the glass fibre tip cap with metallic diverters, the metallic rudder tip cap, the glass fibre portion of the upper rudder skin, internal metal straps riveted to the composite side shells at front and rear spar, metallic rudder trailing edge straps and several bonding jumpers to ensure electrical continuity at mechanical joints.

VII. Certification Philosophy

In Feb. 1988 the vertical tail had been certified as a primary composite structure of the A320 according to FAR Part 25 Amendment 45 (1) and JAR 25 (2), published by the American and European Authorities respectively. In addition Airworthiness Discussion Points from the European Authorities and the following FAA Advisory Circular were considered: (3), (12).

A. Damage Tolerance Requirements

Special care was taken to show compliance with the damage tolerance requirement, to ensure that should serious fatigue or accidental damage occur within the operational life of the aircraft, the remaining structure can withstand reasonable loads without failure or excessive deformation until the damage is detected (3) and repaired. For composite primary structures, like the A320 vertical tail, this means in detail:

- It has to be shown by test, that structures with non visible damages are able to carry ultimate design load (150% of limit design load) at worst environmental conditions for the whole operational life of the aircraft.
- Visible damages, easily detectable by scheduled inspections have to be validated by static test between 100% and 150% of limit design load, under worst environmental conditions and by fatigue test with multiple times of the desired inspection interval.
- Obviously visible accidental damages easily detectable by walkaround inspection, have to be validated by static test applying 'Get-Home ' loads between 70% and 100% limit design loads at ambient conditions.
- Inspection methods should be developed and inspection intervals should be established in such a way, that damages will be detected before the damage reaches the limits for required residual strength capability. In the case of the 'No-Growth' damage design concept, which is applied for the A320 vertical tail, the inspection intervals should be established by taking into consideration the residual strength associated with the assumed damage and the probability of occurence of this damage per flight.
- When repairs are documented in approved aircraft manuals, it should be demonstrated by test, that these repairs will restore the structure to an airworthy condition.

B. Damages Considered for Damage Tolerance Test

For the damage tolerance substantiation tests of the A320 vertical tail the following discrete source damages were considered:

Intrinsic damages introduced into the structure during the fabrication process due to:	Service damages introduced into the structure during the operational life due to:
. porosity . delaminations . weak bonds leading to delaminations in service . impacts . misaligned bore-holes	. impacts . lightning strike . bird impact . environmental effects . system failures leading to debondings

The above listed instrinsic damages were introduced in coupon and subcomponent specimens, to determine the max. allowable intrinsic damage sizes, and substantiated in the full scale component test under static and repeated load for the whole test period.

Damages of the vertical tail, which may occur during the operational life are mainly inflicted by impacts, and were considered on the subcomponent and full scale component level.
Lightning strike tests demonstrated the high efficiency of the lightning protection system, with only limited damages at the upper fin box and rudder side panels under worst lightning conditions, which were taken into account at the full scale test component.
Bird strike test conducted with 8lbs. birds and striking velocity of 250 kn didn't show any penetration of the leading edge sections or damage of the fin box. Therefore no accidental damage due to bird strike needed to be considered for the full scale damage tolerance test program.

Further, it could have been shown by testing, that in the case of APU rotor failure no damage of the vertical tail is to be expected due to the APU containment.
System failures due to jamming of one rudder servo control control valve, and due to misadjustment of the rudder controls were simulated within the full scale component test.
The worst environmental conditions (70°C, wet condition) were taken into account for the whole full scale component test program.

Relative to visibility the above mentioned damages can be classified as follows:

- non visible damages (from a distance
- visible damages of two meters)
- obviously visible (by walkaround) damages

Due to the difficulties of making a clear classification in non visible and visible damages, in case of the EF5 test specimen the visible damages were validated like the non visible damages for the whole test period.

No.	DAMAGE / DEFECT TYPES	DETECTED BEFORE DELIVERY	DETECTED IN SERVICE	JUSTIFICATION TEST FATIGUE	JUSTIFICATION TEST STATIC
1.	MANUFACTURING DEFECTS (porosity, delaminat. etc.)				
1.1	ALLOWABLE SIZE (not to be repaired)	SQUIRTER	NO	1.5 LIVES	ULTIMATE LOAD HOT / WET
1.2	NONALLOWABLE SIZE (Repair)	SQUIRTER (visual)	1C 4C		
2.	LARGE MANUFACTURING DEFECTS				
2.1	LARGE AREA DELAMINATIONS / DEBONDING (caused by manufacturing errors)	NOT DETECTED	8C	1/2 LIFE	LIMIT LOAD HOT / WET
2.2	REMOVAL OF BOLTS AND RIVETS IN RUDDER CONNECTION AREAS		1C 4C		
2.3	CUT OF 1/3 OF THE RUDDER HINGE ARM LUG AT BR 1		4C		
2.3	SCRATCHES ON FIN BOX AND RUDDER		1C		
2.5	DELAMINATION OF RUDDER SPAR WEB STIFFNER (repaired)		4C		
2.6	DELAMINATION OF STIFFNER RIB 1 (repaired)				
2.7	DELAMINATION OF STIFFNER REAR SPAR (repaired)				
3.	IMPACTS				
3.1	SMALL IMPACT not visible	NO	NO	1.5 LIVES	ULTIMATE LOAD HOT / WET
3.2	IMPACT DAMAGE barely visible/not repaired	SQUIRTER VISUAL	1C		
3.3	IMPACT-DAMAGE visible from a distance of 2m, to be detected at next inspection, repaired	VISUAL			
4.	ACCIDENTAL DAMAGES				
4.1	LIGHTNING STRIKE		WALK-AROUND	one A-flight	LIMIT LOAD Ambient Temper.
4.2	SEVERE IMPACT, obviously visible				

Table 1: EF5 Damage Tolerance Concept

All types of damages were considered in the EF5 test specimen and the associated static, fatigue and environmental test conditions are listed in Table 1.

VIII. Description of the Vertical Tail Full Scale Component Test (EF5)

The static and fatigue tests were performed on a complete vertical tail, inclusive rudder control units inside of a styroprene climate chamber.

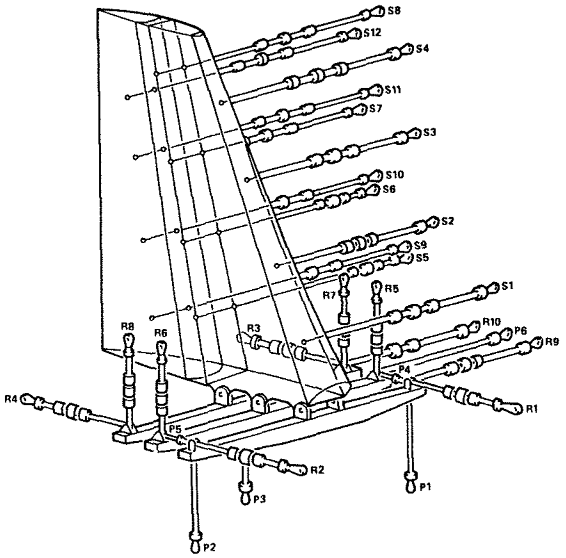


Figure 15: EF5 Test Set Up

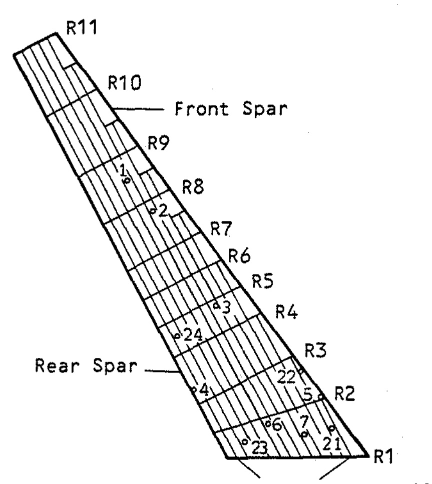
Due to load introduction the EF5 was equipped with reinforced leading edge sections, which were not subject of the demonstration tests. The flight loads from the critical gust and maneuver load cases were applied by means of 12 actio hydraulic jacks and distributed on 46 loading pads acting in compression only. The fuselage compliance was simulated by means of 10 reaction hydraulic jacks and 6 reaction struts, which served additionally for statically determinate support of the complete test set up.
Forces due to the difference in temperature expansion rates of the vertical tail (CFRP) and fuselage (aluminium) in the course of flight ground flight load cycles, were additional considered for the fatigue tests as well as for the static tests, and were applied by the X-cylinders (R1-2-3-4) and superimposed with the X-cylinder reaction forces of the flight loads.
The moisture absorption of the structure was controlled by component and pilot specimens added to the test component in some cases immediately after manufacture.
The temperature of the structure was measured by 7 temperature sensors fixed to the outside and inside of the structure. For comparison of the calculated strain distribution, done by FEM-calculation for the decisive load cases, with the actual strain distribution of the EF5 test specimen, the EF5 was equipped with a total of 121 strain gauges and 118 rosettes mainly bonded back to back. With 37 lateral displacement pick ups the displacement of the fin in x-y-z direction could be measured.

A. Description of the Specimen

Before the test specimen was mounted into the test rig several artificial damages were introduced into the structure, some of them already during manufacturing to simulate intrinsic damages.
Most of the damages were introduced by impacts. The requirement for impacts was to produce visible damages or, at locations with thick laminates, to impact with at least 50 Joules impact energy in case of non visible damages.
Figure 16 shows the position, energy level and damage sizes of the 24 impacts on the right fin box skin panel.
The left rudder side panel, the spar webs and some ribs were impacted with a total of 24 impacts.
In addition the test specimen was equipped with eight repairs, 6 introduced into the torsion box and 2 rudder repairs.
All these repairs have been tested for the total test period and are published in the Structure Repair Manual of the A320.
Riveted and bonded repair solutions were considered, but bonded repairs are approved only for limited sizes.
One typical skin - stringer rivet repair solution is shown in Figure 17.

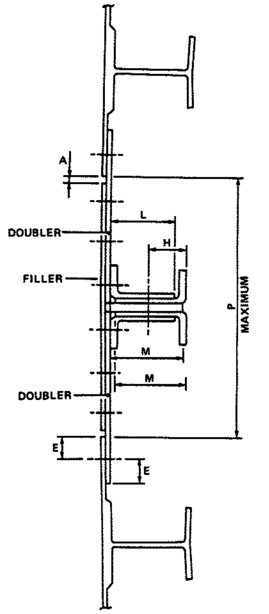
B. Test Objectives and Test Loads

The aim of the EF5 static, fatigue and damage tolerance testing was to demonstrate that the A320 fin is able to carry all relevant loads for the whole operational life of the aircraft even with a predamaged structure unter worst environmental conditions. In addition, the test should validate the 'NO-Growth' design philosophy for damages. The operational life of 48 000 flights should be justified with a life factor of 1.5 and the critical static load cases with a safety factor of 1.5.



Impact No.	Diam. (mm)	Laminat (mm)	Engergy (J)	Damage visible	(mm^2) NDT
1	12	1.5	13	230	635
2	50	1.5	12.1	250	1257
3	12	4.1	13	44	868
4	12	5.6	20.2	no	1360
5	50	6.5	17.8	no	720
6	12	7.0	19	no	1400
7	12	3.7	13	no	665
21	12	6.7	40	20	2700
22	12	5.7	40	40	2045
23	12	7.5	40	20	2070
24	12	6.5	40	13	1815

Figure 16: Impacts on right Skin Panel of Torsion Box



Dimension code (mm)	
A	1.
E	10.
H	19.
L	32.
M	36.
P	152.

Figure 17: Skin Stringer Rivet Repair with Precured CFRP Sections

Using a life factor of 1.5 a load enhancement factor of 1.15 was applied to the fatigue loads to account for the possible high scatter in life direction due to the flat S-N curves of composite structures. This philosophy was accepted by the authorities because the EF5 did have initial defects, like impacts and delaminations, at the start of the fatigue test and thus represents the minimum acceptable production standard.
Three critical load cases were considered:

- lateral gust with the highest bending moment
- lateral maneuver with the highest torsion moment
- rudder maneuver with the highest rudder hinge moment

The 'flight by flight' fatigue test program loads were simulated for 10 different flight types.
The load cycles of each mission segment and the flight types were simulated in a randomised form.
The gust and maneuver load steps were related to the static limit load cases reaching 72 % of gust limit load and 81% of maneuver limit load. The omission level was 18% of gust limit load and 26% of maneuver limit load. One lifetime includes 183440 lateral gust cycles, 167520 maneuver cycles during normal operation and 35900 maneuver cycles during crew training.
The temperature loads at the main attachment lugs due to different expansion coefficients of the CFRP fin and aluminium fuselage, were simulated by 7 temperature profiles taking into account the geographical location, solar radiation and aerodynamic heating.
The system failure cases 'Force Fighting' due to desynchronisations between the rudder servo controls and 'Spool Valve Jamming' due to jamming of one servo control control valve were additionally validated in the course of the full scale test.

IX. Test Sequence of the A320 Vertical Tail Full Scale Component Test

A. Rudder Function Test with Loaded Structure

The full scale test program commenced with the rudder function tests, with the structure loaded. In the course of one of these tests the rudder was deflected by $\pm 30^{\circ}$ with the fin box loaded at limit load level of the maneuver load case with max. bending moment.

B. Static Tests up to Limit Load

The tests were performed under following conditions:

- 70°C and 0.72% moisture weight content
- simulation of 'Force Fighting'
- validation of the design gust and maneuver load cases with superposition of 70°C fuselage loads

C. Fatigue Test up to 48400 Flights

The test was performed under following conditions:

- room temperature and 0.72 moisture weight content
- simulation of 'Force Fighting'
- enhancement factor of 1.15 for fatigue loads

After accumulation of 37496 test flights the test was stopped, due to a damage at the left main attachment fitting of the front spar. The damage was caused by delamination of the root rib connecting angle (see Fig. 18) from the main fitting. Subsequently, this delamination caused several delaminations in the fitting laminate until the load introduction section of the fitting failed.

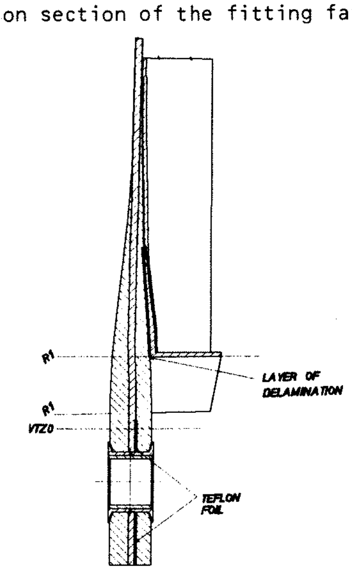


Figure 18: Delamination of the Rib 1 Connecting Angle

The investigation of the damage revealed that the bonding of the root rib connecting angle and the fitting was overloaded by the transverse reaction force of the main fitting. Consequently an additional rivet line, connecting the root rib angle with the main fitting and skin panel was introduced as serial solution and retrofitted in the full scale test component. Additionally the damaged section of the EF5 test component was cut out

and replaced by a new one. The new section was riveted by means of inner CFRP doublers and countersunk rivets to the parent structure like a flush rivet repair.
Due to the replacement of the damaged section the test program was extended by 37496 flights to validate the new main attachment section for 1.5 lives. After the repair a limit load test with the decisive load cases was conducted to prove the rivet repair, and finally the fatigue test was continued up to 48400 flights.

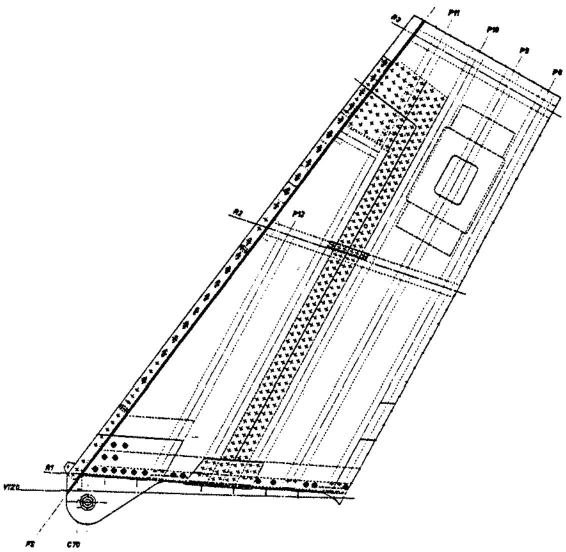


Figure 19: Repair due to the Front Spar Main Attachment Failure

D. Static Test up to 150% Limit Load

The tests were performed under following conditions:

- 70°C and 1.2% moisture weight content
- simulation of 'Force Fighting'
- validation of the design gust and maneuver load cases with superposition of 70°C fuselage temperature loads

E. Fatigue Test with Additional Damages for Damage Tolerance Demonstration of 24000 Flights

For the damage tolerance demonstration the following artificial damages were introduced into the structure:

- large area delamination of one stringer module from skin between two rib stations
- large area delamination of cover layer from core of right rudder sandwich panel
- delamination of several rib and spar web stiffeners
- delamination and cut of 1/3 of one rudder support fitting
- removal of bolts and rivets in rudder connection
- scratches on rudder and torsion box side shells, spar webs and fittings

In parallel to this damage tolerance test, the system failure load case due to a jammed servo control control valve was tested successfully.
The tests were carried out at following conditions:

- room temperature and 1.1% moisture weight content
- enhancement factor of 1.15 for fatigue loads

F. Static Test up to 100% Limit Load with Lightning Strike Damages and Severe Impact Damage

Lightning strike damages derived from lightning strike tests were introduced into the torsion box and rudder upper part. A severe impact, delaminating one stringer from skin caused a total delamination of 4500mm^2.

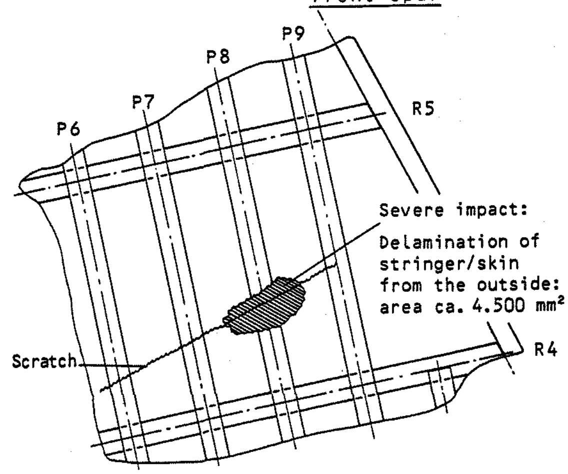


Figure 20: Severe Impact at Stringer P8

With these severe damages the structure was tested at following conditions:

- room temperature and 1.1% moisture weight content

After the successful test the severe damages were repaired as temporary repairs. This implied, that the torsion box severe damages were repaired with aluminium sections and plates simulating a field riveted repair. After repair the EF5 was loaded again up to limit load at following conditions:

- 70°C and 1.1% moisture weight content

G. Fatigue Test up to Test Flight 109496

The test were performed at following conditions:

- room temperature and 1.1% moisture weight content
- simulation of 'Force Fighting'
- enhancement factor of 1.15 on fatigue loads.

The actual status at time of preparation (end of april '88) of this paper is 109496 flights.

H. Residual Strength Test

After completion of the fatigue test program, a residual strength test up to 160% limit load of the maneuver load cases and up to rupture at gust loading will be conducted at following conditions:

- 70°C and 1.2% moisture weight content
- simulation of 'Force Fighting'
- superposition of 70°C fuselage loads

X. Conclusions

The existing fatigue and damage tolerance results for the fuselage and vertical tail can be considered as a good verification of the design which was developed with respect to long crack free and economic repair life and long intervals for structural inspections, respectively.
Acceptable correlation between damage tolerance calculation and test results has been observed for the in-house calculation methods used by MBB.

The whole test philosophy and accomplishment is a sufficient approach to achieve most realistic results with consideration of economical aspects. Therefore the total concept will be the basis for similar tests with the new launched AIRBUS INDUSTRIE projects A330 and A340.

References

(1) Federal Aviation Regulations (FAR) Part 25, Paragraph 25.571, including Amendment 45, 'Damage Tolerance and Fatigue Evalution of Structure', December 1978

(2) Joint Airworthiness Requirements (JAR) Part 25, Section 1, Paragraph 25.571, 'Damage Tolerance and Fatigue Evalution of Structure', Change B

(3) FAA Advisory Circular AC 25.571-1, 'Damage Tolerance and Fatigue Evalution of Structure', September 28, 1978

(4) JAR Part 25, Section 2, Paragraph ACJ 25.571, 'Damage Tolerance and Fatigue Evalution of Structure (Acceptable Means of Compliance)', Amentment 7, November 1980

(5) I.G. Gray, 'Fatigue Crack Propagation Programme for the A320 Wing', Presented to 14th ICAF in Ottawa, Canada; June 1987

(6) T. Swift, 'Verification of Methods for Damage Tolerance Evalution of Aircraft Structures to FAA Requirements', Presented to 12th ICAF in Toulouse, France; May 1983

(7) J. Schijve, 'The significance of flight-simulation fatigue tests', June 1985

(8) H.-J. Schmidt, C.L. Hammond, 'A310 Structural Testing for Certification Philosophy and Application to meet Current Durability and Damage Tolerance Requirements', Presented to 14th ICAS in Toulouse, France; September 1984

(9) H.-J. Schmidt, C.L. Hammond, 'AIRBUS INDUSTRIE A310 and A300-600 Experience with Practical Applications of Damage Tolerance Design in Fuselage Primary Structure', Presented to 13th ICAF in Pisa, Italy; June 1985

(10) N. Ohrloff, 'Comparative Crack Growth Calculations for Flight Simulation Loading', MBB-UT, Federal Republic of Germany, 1988 (Company Proprietary)

(11) D. Schulz, 'Development of the A300 Fin in Modern Composite Fibre Construction' ICAS-80-16.2

(12) FAA Advisory Circular AC No: 20-107A 'Composite Aircraft Structure' 4/25/84