A310 STRUCTURAL TESTING FOR CERTIFICATION PHILOSOPHY AND APPLICATION TO MEET CURRENT DURABILITY AND DAMAGE TOLERANCE REQUIREMENTS

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Abstract

The AIRBUS INDUSTRIE A310 was one of the first aircraft to be certified under the new Federal Airworthiness Requirements (FAR) 25.571 (1) and Joint Airworthiness Requirements (JAR) 25.571(2), which include damage tolerance requirements.

These new damage tolerance requirements for civil aircraft structure will be summarized, and due to the complex nature of these damage tolerance requirements, interpretation and the means of compliance are presented.

The AIRBUS INDUSTRIE philosophy regarding durability and damage tolerance testing as applied to A310 full scale fatigue test articles is given in detail for development and certification purposes. All major aspects from initial planning to the current test status at time of preparation of this paper will be summarized for two primary full scale durability and damage tolerance tests, i.e.

- center fuselage/wing test
- rear fuselage/fin test

The essential findings presented in this paper will include examples of areas with important test results concerning fatigue(crack initiation), crack growth characteristics and residual strength behaviour of damaged structure.

Repercussions of these findings regarding the test articles, in-service fleet and future production aircraft will be given, including development and modification of the structural inspection program.

Details of planned completion of major full scale tests will be presented along with conclusions drawn from current test results at time of paper preparation as they effect future structural testing methods for certification of commercial aircraft.

I. Introduction

The design of aircraft structure has been, in the past, based on static analysis and test, with consideration of lower, repeated loads and their effect on the initiation of cracks within the structure based on fatigue ("safe-life") analysis and dynamic test. However, a change in design criteria during the past several years requires additional consideration of structural integrity assuming certain damage to exist in the structure. This damage can be the result of material processing, manufacturing procedures, environmental effects(corrosion, for example) or accidents.

This new criteria is generally refered to as "damage tolerance",which means that the structure, though damaged to a limited extent, is capable of sustaining the expected dynamic loads plus a predefined static load (usually design limit load) until the damage can be detected and repaired. The detail criteria are given in References (1) and (2) and means of compliance in References (3) and (4).The overall effect of this criteria is to require additional testing to establish not only the locations of likely fatigue crack initiations and the associated times to initiation, but also the growth rates of such cracks under spectrum loading and their resulting critical crack length within the complex full scale structure,considering all load redistributions within the cracked structure.
This can only be satisfactorily accomplished with full scale test articles. Therefore, for the A310 program, extensive fatigue and damage tolerance tests were planned and successfully performed with agreement of the French and German airworthiness authorities. The purpose of this paper is to review these tests and present conclusions and recommendations.

II. Application of Current Durability and Damage Tolerance Requirements

The current durability(fatigue)requirements (1)(2) are essentially unchanged from previous requirements, that is, to provide a "safe" structure that can demonstrate a maximum operational life with a minimum of maintenance cost and down time.This is the economic life until the time that cost of repair or replacement of major structural elements becomes prohibitive due to extensive fatigue cracking.This requirement was met in the past for the AIRBUS INDUSTRIE A300 by several full scale fatigue tests,each representing a portion of the primary structure,to a minimum of two life times.

Since experience has shown that damages due to material processing,manufacturing procedures, environmental effects, or accidents, as well as fatigue loading, do occur in aircraft structures, provision must be made to insure that such damages do not grow to an unexceptable size before they can be detected and repaired. To provide for this, the new damage tolerance requirements were developed. To meet these new requirements significant new parameters must be determined:

a) The type and initial size of damage to be assumed for each major structural element;

b) The extent of damage growth in the structural element (and adjacent elements for multiple load path design);

c) The detectability and repairability of damage before it reaches critical proportions;

d) Inspection access, acceptable inspection techniques, and safe periodic inspection intervals.

Application of these requirements for analysis and test leads to establishment of:

a) The inspection threshold based on time to crack initiation (detectable) or on assumed initial quality flaw size;

b) Detectable flaw size assumptions based on the inspectability of the structure and the inspection capability;

c) Minimum residual strength of the damaged structure;

d) Inspection interval based on the time to propagate detectable flaws to critical size under fatigue and fail-safe loading.

In general, an evaluation of the structure under typical load and environmental spectra must show that catastrophic failure due to fatigue,corrosion or accidental damage will be avoided throughout the operational life of the aircraft.This requirement applies to all structural elements whose failure, if remained undetected, would lead to loss of the aircraft. Due to the vast complexity of modern aircraft structures, variations in material processes, manufacturing procedures, environmental conditions,etc.,and insufficiencies in current analytical prediction methods,such a damage tolerance evaluation must include extensive testing of full scale structure. In order to satisfy these new requirements, several full scale fatigue and damage tolerance test programs for the A310 were planned and are currently in progress.

This paper will briefly describe:

- Test planning
- Significant test findings which result from the original design
- Details of damage tolerance tests and preliminary results
- Repercussions of test findings
- Conclusions and recommendations

III. A310 Major Full Scale Fatigue and Damage Tolerance Tests

A. Planned Tests-

Four major full scale fatigue and damage tolerance tests were planned for the A310 as shown in Figure 1.

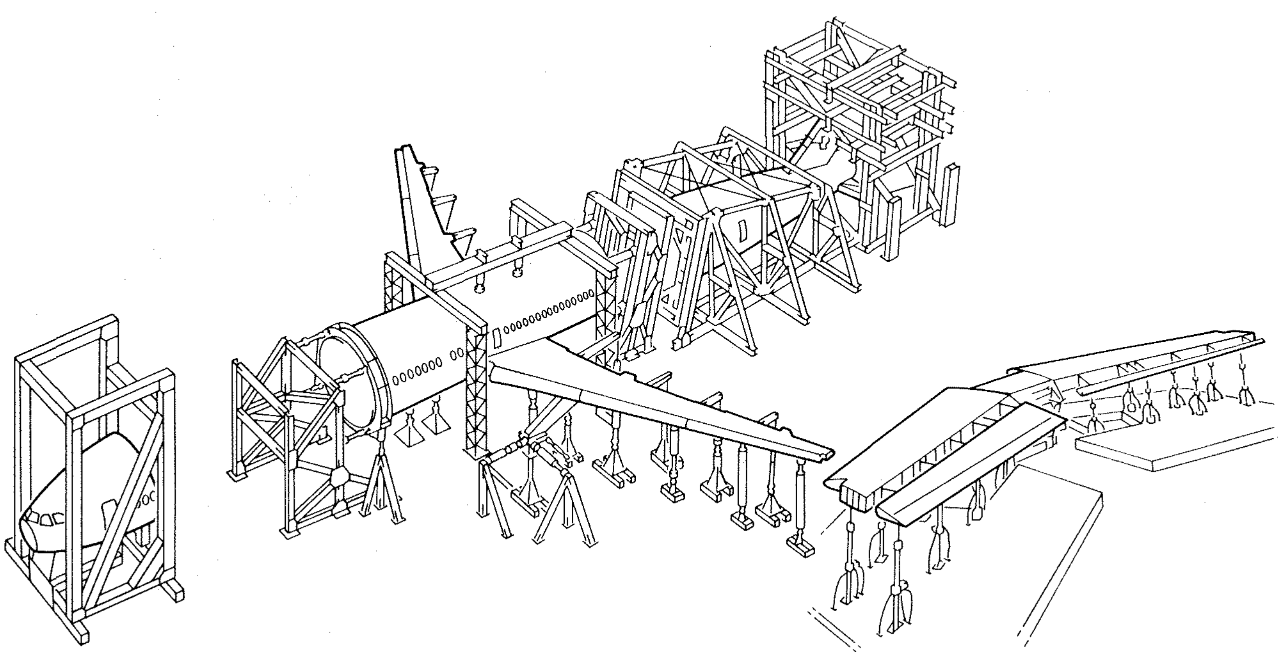

Figure 1 A310 Full Scale Fatigue and Damage Tolerance Tests

They are:

1) Center fuselage/wing test (IABG-Munich)
2) Aft fuselage/fin test (MBB-Hamburg)
3) Forward fuselage test
(Continuation of A300 test, MBB-Lemwerder)
4) Horizontal tailplane test (MBB-Lemwerder)

Due to space limitations this paper will be limited to the major tests 1) and 2) above.

The primary goals of these tests are:

1) Demonstration of the A310 design durability goals which include:

. 20 000 flights crack free life,
. 27 000 flights minor repair life,
. 40 000 flights economic repair life,
by simulation of 90 000 test flights;

2) Determination of local stress distributions and concentrations by the extensive use of strain gauges;

3) Determination of the crack propagation behaviour of the complex structure by installation of artificial damages at selected areas earliest at 52 000 flights and by monitoring the crack propagation of selected existing natural cracks;

4) Validation of inspection methods, including probable detectable crack sizes, proposed for A310 Structural Significant Items (SSI);

5) Justification of repairs or modifications that result from test findings after initial design of the aircraft;

6) Determination of the residual strength of the cracked structure subjected to cabin pressure.

Details of the significant test results to date will be given in the following paragraphs.

The test status as of end of April 1984 was:

Center fuselage/wing test = 75 000 flights
Aft fuselage/fin test = 52 000 flights

The test schedule of the major test 1)as it relates to important A310 milestones is given in Figure 2.

	1982	1983	1984
Start of test	▽		
Certification		▽ 40 000F	
Start of crack propagation tests		▽	56 000 F
Current status			▽ 75 000 F

Figure 2 A310 Test Schedule
Center Fuselage/Wing Specimen

B. Test Set-Up -

Figures 3 and 4 show details of the test set-ups. Significant characteristics of these tests are shown in Table 1.

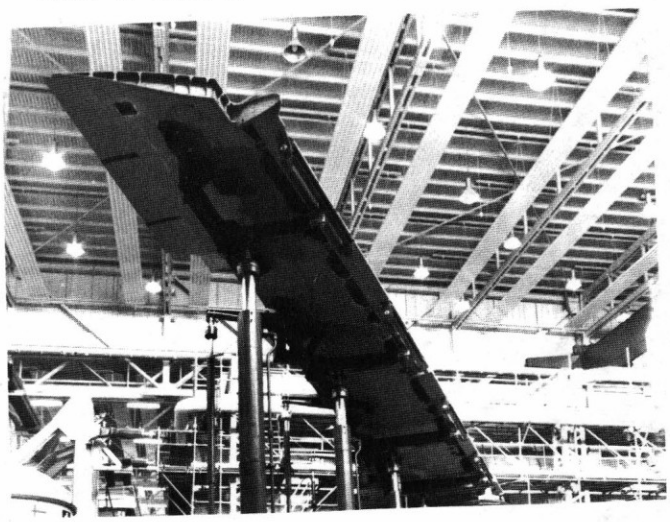

Figure 3 Center Fuselage/Wing Test Details

Test Specimen	Structure Included	Dummy Structure	Loading Elements	Measurements
Center fuselage/ wing	Fus.frames 22-61, complete wing box, fixed leading and trailing edges, wing pylon (LH side)	Main ldg.gear, wing pylon(RH), slat and flap tracks	Hyd.jacks: 51 Control links: 6 Internal pressure	Strain gauges:1196 Deflection transducers : 35
Aft fuselage/ fin	Fus. frames 65-91, complete fin box to rib 16	Horizontal tailplane, fin box above rib 16	Hyd.jacks: 13 Internal pressure	Strain Gauges: 860 Deflection transducers : 10

Table 1 A310 Full Scale Fatigue and Damage Tolerance Test Characteristics

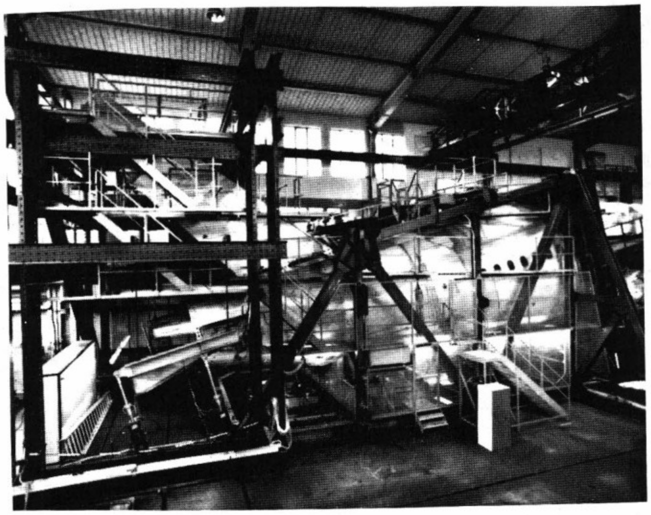

Figure 4 Aft Fuselage/Fin Test Details

Figure 5 shows the detail loading system for the center fuselage/wing test.

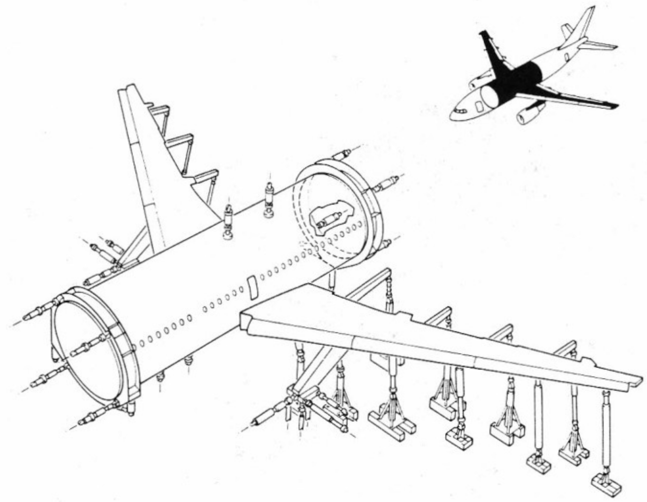

Figure 5 Center Fuselage/Wing Loading System

Each specimen is loaded with flight-by-flight spectra which include all significant load cases resulting from the representative flight profile which is based on a block time of 90 minutes.

The flight and ground loads are represented by 61 test load cases for the center fuselage/wing test and more than 300 test load cases for the aft fuselage/fin test.

These load cases are combined in more than ten different flight types which are applied in a randomized sequence to the test specimen.
Figure 6 shows flight types which are typical for both test specimen.

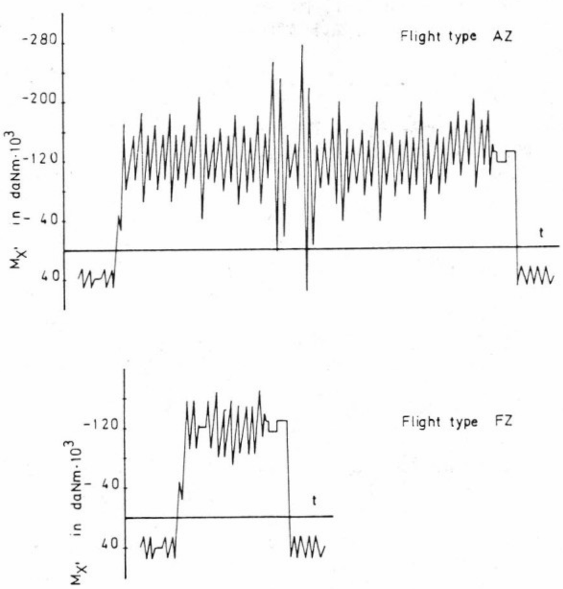

Figure 6 Typical Flight Type Loading

IV Significant Fatigue Test Results and Repercussions To Date

A. Center fuselage/wing test -

Only the results from the fuselage section of this test will be given here. Of the test findings from this specimen, three are of particular interest and are here described in detail.

1) T-Fitting at Frame 50a/Stringer 25

Figure 7 shows this part in detail. It is in the area of the main landing gear/ fuselage load introduction.

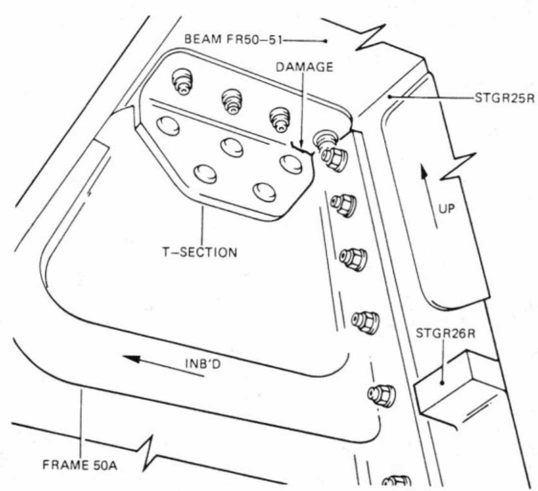

Figure 7 Fatigue Damage at Frame 50a/ Stringer 25

A 10 mm crack was found in the radius of the T-fitting which ties frame 50a to stringer (longeron) 25 at approximately 85 % of the design life (34 000 flights). The repercussions of this result are the following:

a) Production aircraft - Reinforcement of T-fitting by increasing net area and shot peening with high intensity

b) In-service aircraft - (3 Alternative solutions)
   - Installation of production solution before 12 000 flights
   - Shot peening of existing T-fitting before 12 000 flights
   - Periodic inspection every 6000 flights as determined from crack propagation test measurements, beginning at 12 000 flights.

2) Frame 51 at stringer 25 -

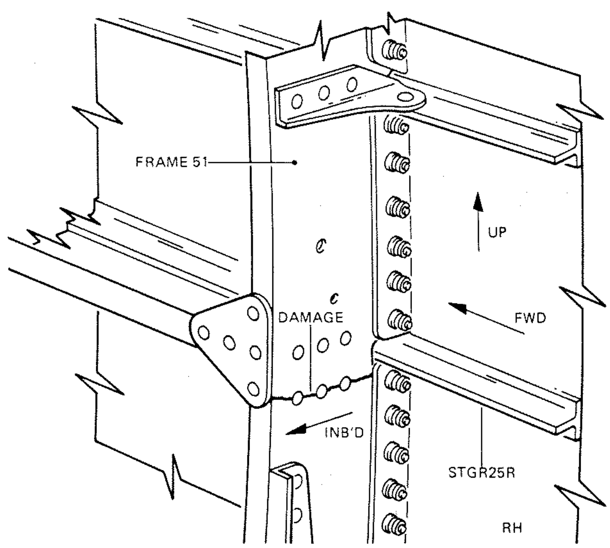

Figure 8 Fatigue Damage at Frame 51/Stringer 25

Figure 8 shows this part in detail.At 93 % of the design life (37 000 flights) frame 51 at stringer 25 was found to be completely failed. The neighboring structure, including the external skin, was closely inspected and found not to be damaged. This part is in the same main landing gear/fuselage load introduction area as the previously mentioned damage 1).
The repercussions of this result are:

a) Production aircraft
   - Reinforcement of the frame by increasing the net area of the inner flange (approximately 100 % increase)

b) In-service aircraft (2 Alternative solutions)
   - Installation of reinforcing profile between stringers 22 - 27 before 12 000 flights
   - Inspection every 1000 flights beginning at 12 000 flights.

3) Frame 46 at stringer 22

Fig.9 shows this part in detail.

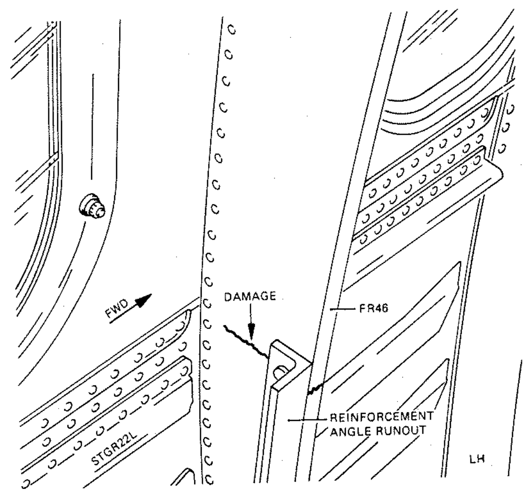

Figure 9 Fatigue Damage at Frame 46/Stringer 22

An 80 mm crack was found at approximately 140 % of the design life (56 000 flights) in the inner flange and web of the frame at the runout of a reinforcing angle. The repercussions of this result are:

a) Production aircraft
   - Redesign of the runout of the reinforcing angle by tapering the width and thickness to reduce the load transfer at the crack location

b) In-service aircraft (2 Alternative solutions)
   - Improvement of the reinforcing angle runout by milling the flange. The modification is to be carried out before 12 000 flights.
   - Periodic inspection every 3000 flights beginning at 12 000 flights.

B. Aft fuselage/fin test -

Of the test results from this specimen, only two significant damages resulted from original design problems which required redesign. Both of these items relate to the torsion box runouts above the aft passenger/crew doors at frame 72, stringer 9, left and right sides as shown in Figure 10.

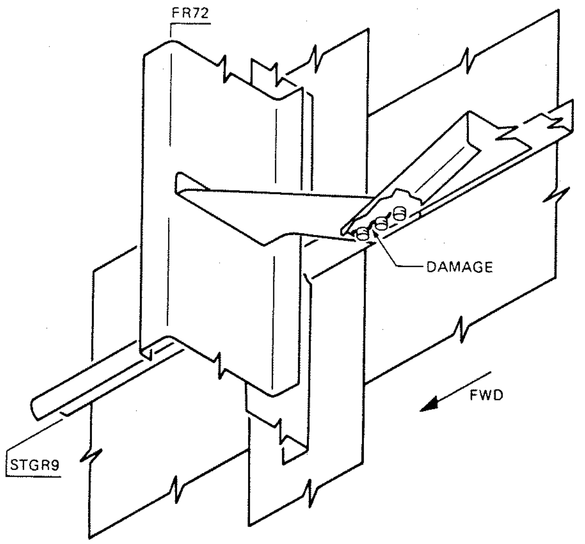

Figure 10 Fatigue Damage at Frame 72/Stringer 9

At 48 % of the design life (19 327 flights), a 14 mm and a 41 mm crack was found at the end of the torsion box runouts, left and right hand sides, respectively. The repercussion of this test result was to redesign the gusset plate which attachs the torsion box runout to Frame 72 in order to reduce the torsional deflection at the runout angle. Since these damages were found early in the test program, no retrofit of in-service aircraft was necessary.

The goal of all of the previously described solutions is to restore full structural integrity and service life to the structure while at the same time giving maximum flexibility to the operator.

This was accomplished by:

- Measurement of crack growth under spectrum loading or microfractographic analysis of crack surface
- Development of production and retrofit modifications
- Installation of modifications on test article
- Comparative stress measurements with modification
- Justification of modification by fatigue analysis using measured stresses
- Further testing of modification

V. Damage Tolerance Tests

As a result of the previously discussed new damage tolerance requirements,several detail artificial damages have been installed in the mid fuselage/wing and aft fuselage/fin test articles. These damages are summarized in Table 2. Most of the damages were installed at 56 000 flights in the center fuselage/wing specimen and at 52 000 flights in the aft fuselage/fin specimen.

Specimen	Structure	No. Damages
Center Fuselage/Wing	Wing	13
	Center Wing	21
	Pylon	9
	Mid Fuselage,Forward	13
	Mid Fuselage,Center/Aft	16
Aft Fuselage/Fin	Pressurized Fuselage	15
	Non Pressurized Fuselage	3

Table 2 Damage Tolerance Tests

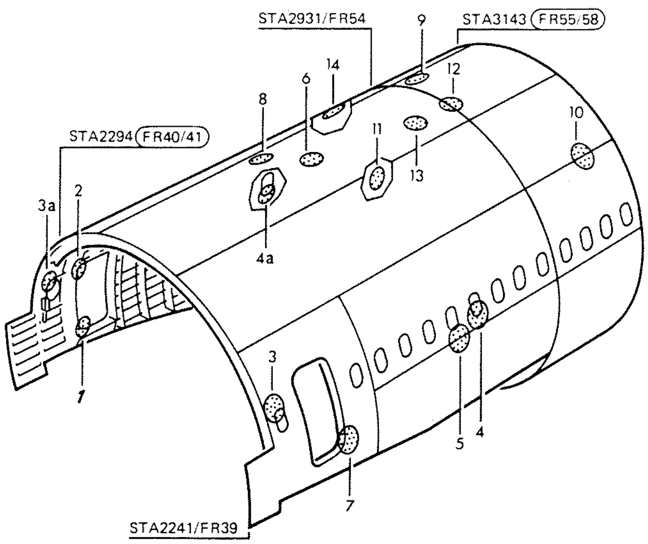

Figure 11 Artificial Damage Locations
Mid Fuselage, Center/Aft Portion

Locations of the artificial damages in the center/aft portion of the mid fuselage and in the aft fuselage are given in Figures 11 and 12, respectively.
These areas represent structural significant items (SSI) which were included in the A310 certification analysis and which led to certification in March 1983. The locations were considered in developing the MBB structural inspection program. The primary purpose of these damage tolerance tests is to verify the methods, material data, stress distributions and assumed failure modes used in the crack propagation analysis required by the certification authorities. Six areas will be discussed in more detail:

- Longitudinal lap joints
- Cutouts for doors and windows
- Upper shell skin
- Frames
- Aft pressure bulkhead
- Major attachment lugs

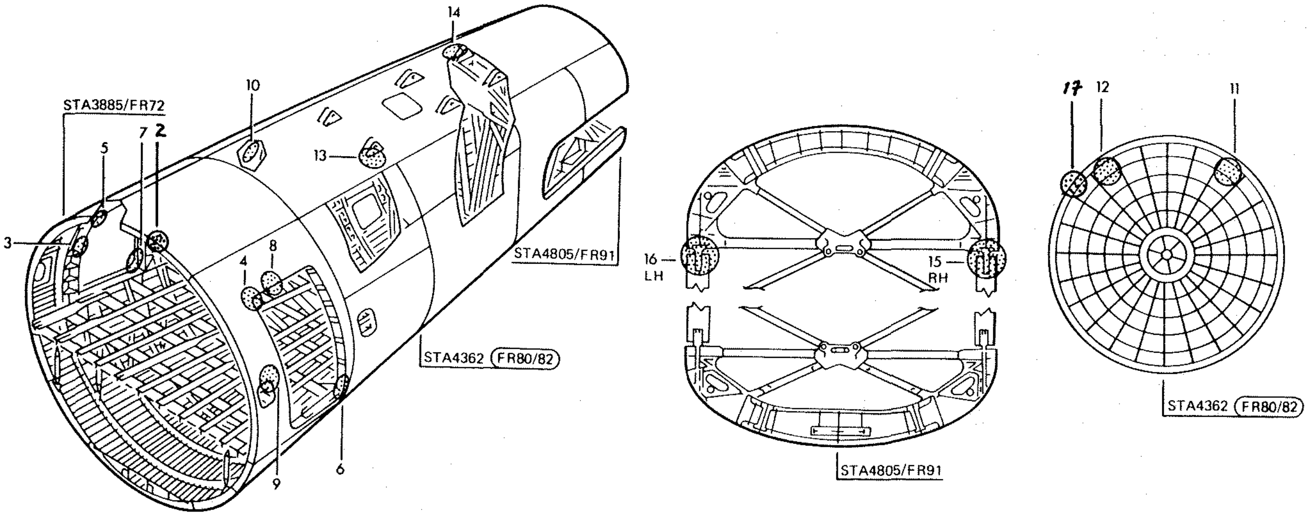

Figure 12 Artificial/Damage Locations
Aft Fuselage

A. Longitudinal Lap Joints -

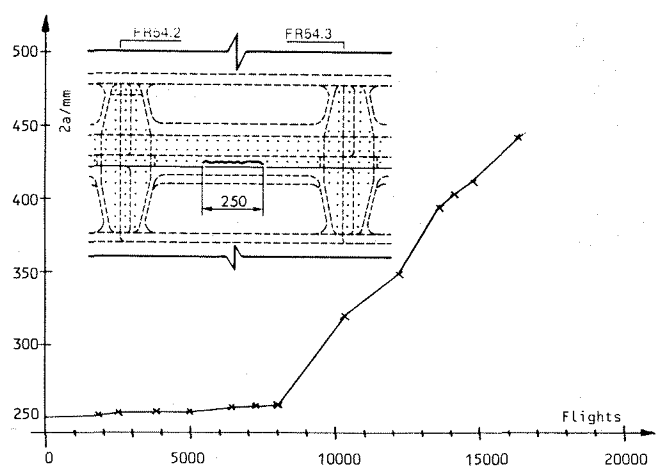

Figure 13 Mid Bay Damage at Longitudinal Lap Joint

Figure 13 shows the details of a mid bay crack installed between Frames 54.2 and 54.3 at stringer 13 (left side). This crack of 250 mm was installed at 56 000 flights, in the inner skin,which is hidden from external and internal inspection. Titanium crack retarders at the adjacent frames were designed to slow the propagation of such cracks and also to make such cracks externally inspectable before they become critical. Results of the crack propagation test up to 72 000 flights are also given in Figure 13.These results show the analysis crack propagation, which was based on A300 test results, to be conservative.This was due to the presence of additional fatigue induced natural damages and a thinner joint doubler in the A300 test. As expected, the crack became visible externally as it approached the crack retarders at the frames.

B. Cutout for Passenger/Crew Doors -

These doors are of particular interest since each one has several possible crack locations and resulting load redistributions. For this reason several tests were planned for the upper and lower corners with separate cracks in the door fail safe ring and the doubler, at the edge of the door cutout and at the first fastener from the edge, where the stress is lower but the stress concentration is higher. Figure 14 shows these locations. Crack propagation tests began at 52 000 flights and will be compared with analysis results.

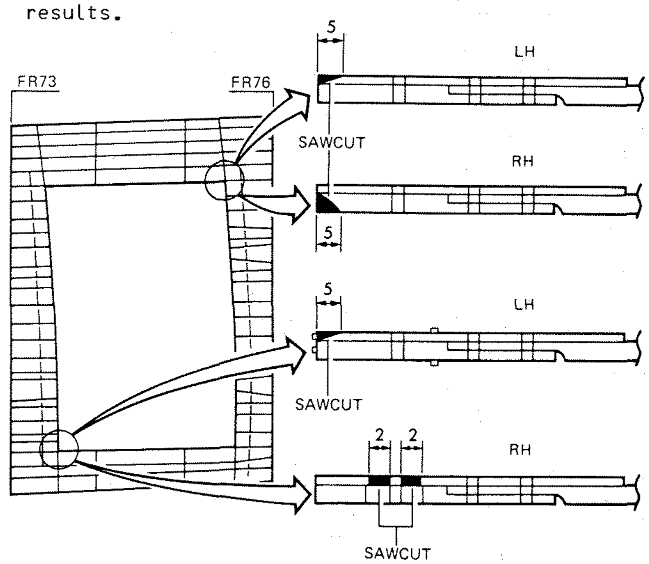

Figure 14 Damage at Passenger/Crew Door

C. Cutouts for Windows -

Due to the high stresses at certain window cutouts,two cracks were installed at 70 000 flights one (8 mm) in the skin at Frame 41 - 42 (LH) and one (5 mm) in the window frame (RH). This is in the highest stressed area of all window cutouts and therefore covers all window locations.

Figure 15 shows the details of these artificial cracks. No crack growth results are currently available.

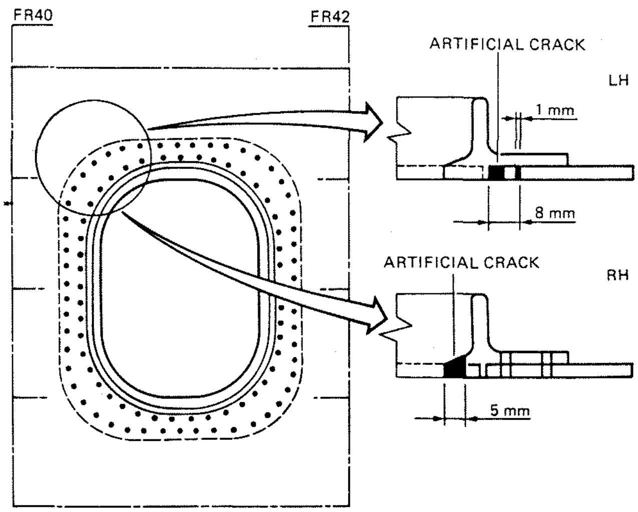

Figure 15 Damages at Window Cutouts

D. Upper Shell Skin -

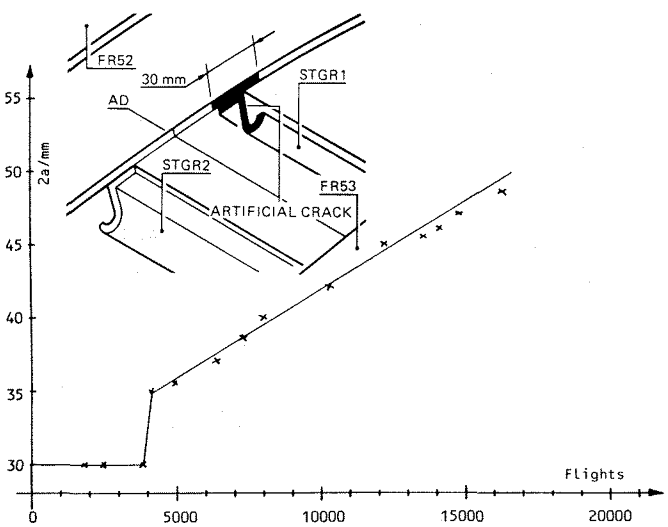

Figure 16 Upper Shell Circumferential Damage

Figure 16 shows the details of a circumferent-ial crack, which was installed in the upper shell at stringer 1 between Frames 52-53 at 56 000 flights. This is an area of maximum longitudinal stress. The crack was a 30 mm skin crack over a completely severed stringer to demonstrate externally inspectable structure according to the damage tolerance recommendations of References (5) and (6). The crack propagation, up to 72 000 flights, also shown in Fig. 16, was slower than predicted by analysis. This was partly due to lower stress levels than used in the analysis.

E. Fuselage Frames -

Two cracks were installed in fuselage frames at 56 000 flights as shown in Figure 17.

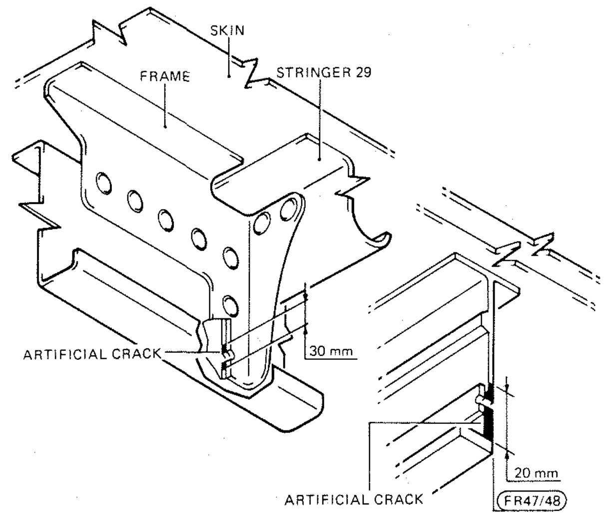

Figure 17 Damages in Fuselage Frames

One was a 20 mm crack in Frame 47 in the coupling area between stringer 4 and the crown center line. This is a major load introduction milled frame at the wing rear spar. Additionally,a 30 mm crack was installed at a typical formed frame at Frame 54.2, stringer 29. The crack propagation tests showed no increase in crack length at Frame 47 up to 72 000 flights and no propagation in Frame 54.2 up to 68 000 flights. The crack in Frame 54.2 was increased artificially up to 44 mm, with no propagation up to 72 000 flights.

F. Aft Pressure Bulkhead -

The artificial cracks were installed in the rear pressure bulkhead at 52 000 flights as shown in Figure 18.

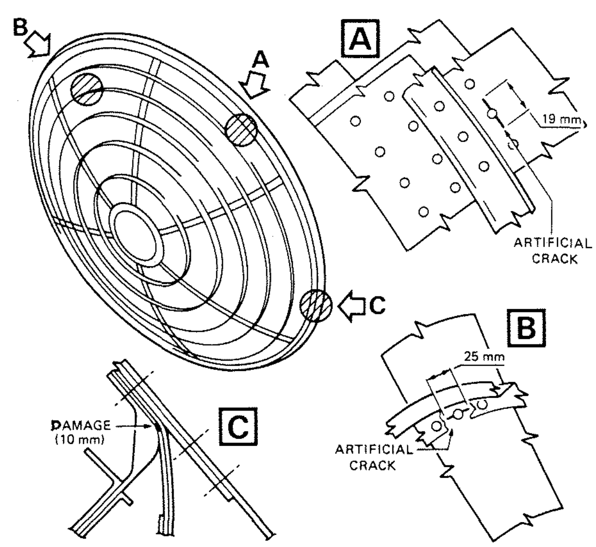

Figure 18 Damages in Rear Pressure Bulkhead

One is a 19 mm crack in the first fastener row of the outer circumferential joint which is a fatigue sensitive area,one 25 mm crack in the membrane at a circumferential stiffener, and one 10 mm crack in the inner attachment angle to the fuselage at Frame 80. No propagation results are available to date.

G. Major Attachment Lugs -

Artificial cracks were installed at 52 000 flights in the major fuselage attachment lugs of the horizontal tailplane and the vertical fin. As shown in Figure 19, one 4 mm corner flaw was installed in the upper corner lugs at frame 91 and a 0.25 mm scratch in the upper lug of the connecting fitting of the horizontal tailplane.

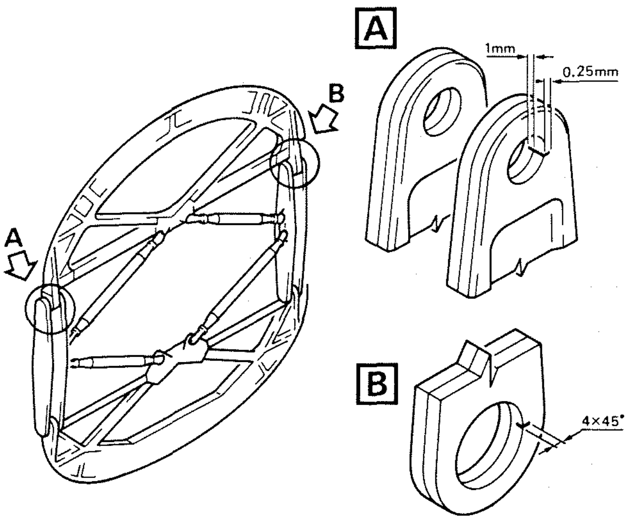

Figure 19 Damages in Fuselage/Horizontal Tailplane Attachment Lugs

This latter damage simulates possible scratches resulting from improper installation of the interference bushings used in all major attachment lugs to increase their fatigue lives.

In the case of the horizontal tailplane attachment lugs, which are bonded double lugs, the crack propagation in the first lug half will be monitored and the time to crack initiation and subsequent crack growth in the second lug half including load redistribution will be recorded for comparison with analysis assumptions.

Fig.20 shows the 4 mm corner flaw installed in the aft fuselage attachment lug for the vertical fin. No crack popagation results are available to date.

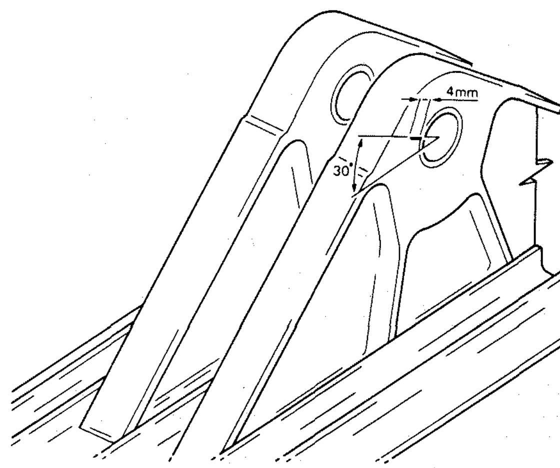

Figure 20 Damage in Fuselage/Fin Attachment Lug

For all of the above damages, periodic inspections will be made every 1000 - 3000 flights using visual inspection methods and microfractographic analysis of selected crack surfaces will also be performed.

H. Residual Strength Tests

Most of the residual strength tests were performed at the end of the full scale static test by CEAT (France).Due to space limitations these results are not given in detail in this paper. The results confirmed previous analyses performed for certification. For those areas subjected to internal pressure stresses only, residual strength tests will be performed on the center and aft fuselage fatigue specimen at the conclusion of the crack growth test.

VI. Conclusions and Recommendations

The results of the major A310 fuselage full scale fatigue tests to date show that the full durability design goals are met with only a few redesigns/modifications or alternative inspection programs. The damage tolerance test results to date show actual crack growth rates to be equal to, or less than, those resulting from the analysis, which was used for structural certification.
If this tendency continues until the end of the test programs (90 000 flights),it may be possible to increase the current planned inspection intervals or the required inspectability (minimum detectable crack lengths) locally and to maintain the current intervals and requirements for developed versions of the A310 having higher required performance. These results verify the original basic design concepts and analysis methods used for the A310 primary fuselage structure.

The results of these tests will be used for future design and analysis, and to establish more efficient fatigue and damage tolerance test programs for such aircraft as the A320. Specific recommendations include installation of initial quality flaws at selected locations early in the test program (a = 0.1 - 1.0 mm) to establish inspection thresholds,which are currently based only on fatigue life considerations and do not include flaws or damages due to material processing, manufacturing procedures, accidents, etc. It is also recommended to begin the crack propagation tests earlier in order to obtain more information and to develop test load spectra,that eases the task of future microfractographic investigations by establishing the sequence of peak loads in the spectra in order to form easily recognizable "marker striations" on the crack surfaces while not effecting the durability or damage tolerance behaviour of the structure. Additional tests are planned for the A320 to establish such test spectra and also to study the limits of load truncation of high loads and omision of high cycles without effecting the crack initiation or crack growth. Finally, it is recommended to measure local stresses at selected locations for all flight types of the test program to be used directly to increase the accuracy of certification analyses without significant increases in cost.

References

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

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

(3) FAA Advisory Circular AC 25.571-1, "Damage Tolerance and Fatigue Evaluation of Structure"; September 28, 1978.

(4) JAR Part 25, Section 2, Para. ACJ 25.571, "Damage Tolerance and Fatigue Evaluation of Structure (Acceptable Means of Compliance)," Amendment 7, November 1980.

(5) T.Swift,'Verification of Methods for Damage Tolerance Evaluation of Aircraft Structures to FAA Requirements', Presented to 12th ICAF in Toulouse, France; 25th May 83.

(6) Joint Airworthiness Requirements,Study Group Structures,Document No.125 'Interpretative Manual on the Substantitation of Fatigue Life and Fail-Safe Behaviour of Aircraft Structures', Issue 2, November 1976.