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FATIGUE AND DAMAGE-TOLERANCE SUBSTANTIATION OF THE GALAXY EXECUTIVE JET AIRCRAFT

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Abstract

The paper describes the on-going program for the fatigue and damage-tolerance substantiation of the new Galaxy wide-body executive jet aircraft. An outline is given of the analytical and experimental work which is being conducted in order to meet the fatigue and damage-tolerance substantiation requirements.

Introduction

Development of the Galaxy , wide-body executive jet (Figure 1) is in progress[1-3]. Flight testing of the first prototype aircraft has commenced in December 1997, and has so far progressed according to plan. The Galaxy will have transatlantic range and a maximum cruise speed of Mach 0.85. It will transport up to nine passengers in an executive configuration and up to nineteen passengers in a corporate configuration. The Galaxy is powered by two PW306A turbofan engines. The primary structure is metallic except for ailerons, elevators and rudder which are made from composite materials.

The airframe structure is being substantiated to the FAR-25 and JAR-25 damage-tolerance requirements. In order to ensure a service life goal of 20,000 flight-cycles and 36,000 flight hours, a comprehensive fatigue and damage-tolerance substantiation program is being undertaken as described below.

The analytical part of this program includes, fatigue loads spectrum development, finite-element modeling, detail stress and damage-tolerance analysis of the various critical locations, and the determination of inspection intervals and methods. The experimental part of this program includes, coupon testing for material behavior characterization, component development testing of key structural items, and full-scale fatigue testing of the aircraft structure for a duration of two lifetimes.

Fatigue Substantiation Philosophy

The Galaxy Executive Jet is designed to provide *at least* 20 years of service including a life goal of 20,000 flight-cycles and 36,000 flight-hours. In order to accomplish this goal, adequate fatigue life has been a design requirement of every component of the aircraft.

In order to insure adequate fatigue life, an encompassing fatigue substantiation philosophy has been adopted. This philosophy includes loading spectrum development, substantiation to the damage-tolerance requirements of FAR-25 and JAR-25, coupon tests to establish crack growth parameters under spectrum loading, coupon tests to establish truncation limits, component tests to verify the adequacy of specific structures and a two-lifetime full-scale fatigue test. Clearly, the philosophy is to provide sufficient experimental basis to the analytical damage-tolerance substantiation.

FIGURE 1: Galaxy Wide Body Executive Jet Aircraft

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Multi-site damage (MSD) has, in recent years, been identified as an important mechanism of many fatigue failures that have occurred in several aircraft. Galaxy structures, that may be prone to multi-site damage, have been designed to moderate stress levels in order to minimize the probability of an MSD failure during the design life. In addition, the two-lifetime full-scale fatigue test will be used to verify that MSD is extremely unlikely to occur during the expected service life of the aircraft.

Composite structures are used for several control surfaces of the Galaxy. These have been designed to sufficiently low strain levels in order to preclude fatigue failures in service. In addition, the composite material elevator and rudder will be tested for damage-tolerance under manufacturing and service inflicted damage, as well as under barely-visible impact damage, as part of the empennage component fatigue test. The entire empennage will be tested for two lifetimes, followed by an additional lifetime of damage-tolerance testing, aimed at verifying the adequacy of the composite control surfaces.

The results of the Galaxy damage-tolerance analyses are translated into inspection requirements which will be applied to the principal structural elements (PSE) of the aircraft. At each PSE, a threshold (initial) inspection and an inspection interval will be specified, as well as the NDI method to be used. A goal was established to schedule the threshold inspection at 50% of the design life goal (10,000 flights) and subsequent inspections at 25% intervals (5,000 flights). The NDI method selected for each PSE will be based on the crack growth characteristics of the specific location with an aim of selecting the most cost-effective procedure.

Analytical Substantiation

The process of analytical substantiation consists of the following steps: loading spectrum development, finite-element modeling, damage-tolerance analysis and specifying inspections during service. These steps will be briefly described.

Loading Spectrum Development:

Loading spectra development begins with the definition of typical aircraft missions. Based on market surveys and usage-tracking of previous customers, three typical missions have been identified having flight lengths of one, two and three hours. Each mission has then been subdivided into several flight segments such as: taxi, rotation, climb, cruise, descent, flap-extension, approach and landing impact. For each of these segments, the load-factor spectra for gusts, maneuvers, taxi and landing impact are determined. The specific load spectrum for each aircraft component is determined from the load-factor spectrum by computing the aerodynamic and inertial loads corresponding to the specific load condition.

Finite-Element Modeling:

Finite-element modeling is used in the fatigue substantiation procedure in two distinct ways: A finite-element model of the entire aircraft structure is used to determine the stress spectrum corresponding to a specific location on the aircraft. Special purpose finite-element models are used to analyze specific discrete structures in order to determine stress-concentration factors or stress-intensity factors for complicated geometries. NASTRAN is used for the first application while StressCheck, a special purpose p-version finite-element program, is used to generate the discrete models.

Damage-Tolerance Analysis:

The damage-tolerance analysis begins with the selection of specific locations to be analyzed. These locations are selected on the basis of stress level, stress-concentrations, previous test or service experience and criticality of the structural element.

The damage-tolerance analysis is performed, based on the FAR-25 and JAR-25 requirements, using a specific computer program that calculates crack growth and residual strength. Initial cracks are assumed to be of a 0.05 inch size and the number of flights to reach the critical crack size are calculated by the methods of linear elastic fracture mechanics. The crack growth material parameters, including spectrum retardation effects, are based on coupon testing that was performed specifically for the Galaxy loading spectrum. A table-lookup method, based on the Walker model and the Wheeler and Closure retardation models, are used to calculate crack growth. When appropriate, continuing damage calculations are made to calculate the progression of cracking in multi-path designs. Environmental effects such as high humidity or fuel contamination are accounted for empirically.

Determining Inspection Intervals:

The output of the damage-tolerance analysis is used to determine inspection methods and intervals to safeguard the structure throughout its lifetime. For each location that was analyzed, a threshold inspection and inspection interval is specified.

The threshold inspection is taken as 50% of the crack growth life, as determined by the damage-tolerance analysis, but not greater than 50% of the design lifetime (10,000 flights), as is FAA practice in this matter.

The inspection interval is determined using a "cumulative probability of detection" method. This method computes the probability of detection of each scheduled inspection, based on the predicted crack size. The method calculates the cumulative probability, as a function of inspection interval, that the crack will be detected, *at least once*, during *all* the scheduled inspections. The required cumulative probability of detection criterion ranges from 95% to 99.5%, depending on the criticality of the structure. On this basis the inspection interval is determined. Often, several inspection methods (visual, liquid-penetrant, eddy-current, etc.) are investigated and traded-off in order to select the most cost-effective method and interval. In order to minimize aircraft downtime, a goal was established to schedule the *great majority* of inspections not more frequently than 25% of the design lifetime (5,000 flights).

Since the Galaxy aircraft is capable of flight up to a 45,000 foot altitude, FAA regulations require reduced inspection intervals for critical locations on the main cabin, failures of which could result in rapid decompression. These locations are designed so that the cumulative probability of detection is not less than 99.5%.

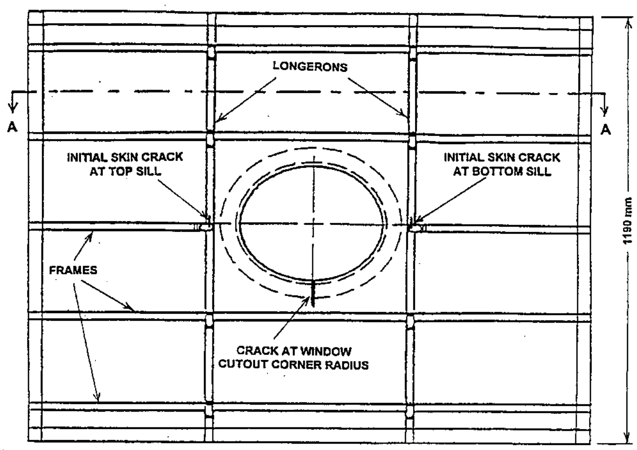
Component Testing

Cabin Window Test:

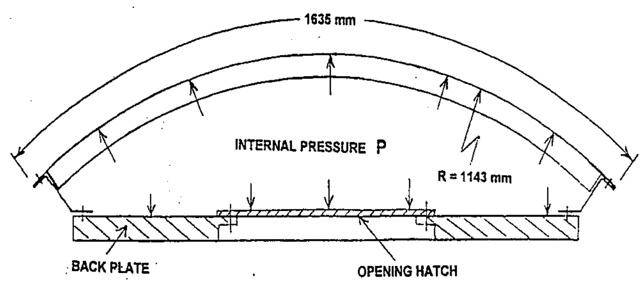
Figure 2 shows the cabin window fatigue and damage-tolerance component test specimen and setup. An improved lower weight one piece aluminum window frame is reinforcing the elliptical window cutout. The objectives of the test are: to show adequate fatigue service life, prove adequate residual strength, obtain data on crack initiation and crack propagation rates for cracks at the most susceptible locations, and to evaluate the appropriate NDI methods for in-service inspections.

A section of the fuselage, containing the structure surrounding the cabin window, was repetitively pressure tested for two lifetimes in the test rig. The loading consists of pressure cycles from zero to 10.35 psi. A 15% increase in the maximum cabin pressure is taken to allow for the exclusion of other types of flight induced loads. At the conclusion of the two lifetimes of fatigue testing, initial flaws, represented by saw cuts, were introduced at three locations having the highest stresses, namely at the window frame (at cutout corner radius), and at two locations of significant skin bending due to bulging near the top and bottom window sills.

As shown in Figure 3, after 12,100 flights the crack at the window frame grew sufficiently to break the window frame. After a total of 14,600 flights, a continuing crack has emerged in the skin and grew to 45 mm. No growth was seen at the other



a) Top View of Test Specimen



b) Section AA Of Test Rig

FIGURE 2: Cabin Window Test Component and Rig

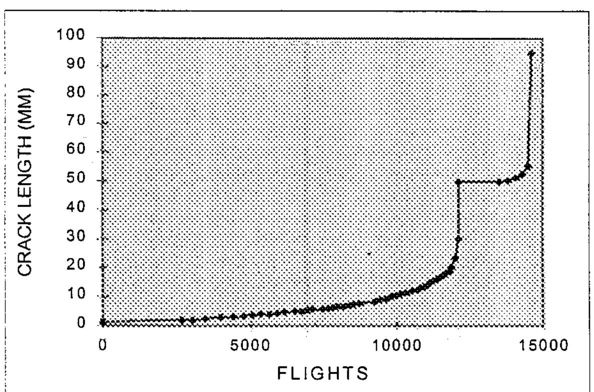


FIGURE 3: Crack Growth at Window Frame and and Continuing Skin Crack

two saw cut locations. A residual strength test was applied to a maximum pressure of 11.9 psi.

Test of Lower Wing Skin Splice at Rib 2 :

The Rib 2 area of the wing lower skin is a complex area that includes the splicing of four skin segments (outboard-to-inboard and center-to-rear skins) and change of contour around the landing gear bay. In addition the rear spar is terminated and its load is transferred to the center spar in this area.

The test specimen included a lower wing skin segment from Rib 0 to Rib 3A and from the rear spar to the longitudinal splice (see Figure 4). The spars and ribs were conservatively represented by their lower flanges, omitting the stabilizing effect of the remaining structure. A typical access panel cutout was included in the specimen.

The specimen was clamped at Rib 0 and lower skin loads applied by three servo-hydraulic actuators at Rib 3A as seen in Figure 4. The loading spectrum was the Galaxy gust and maneuver spectrum with a block of 2,000 flights divided into three missions. Nearly 50 strain gages were bonded on the specimen and monitored during the test.

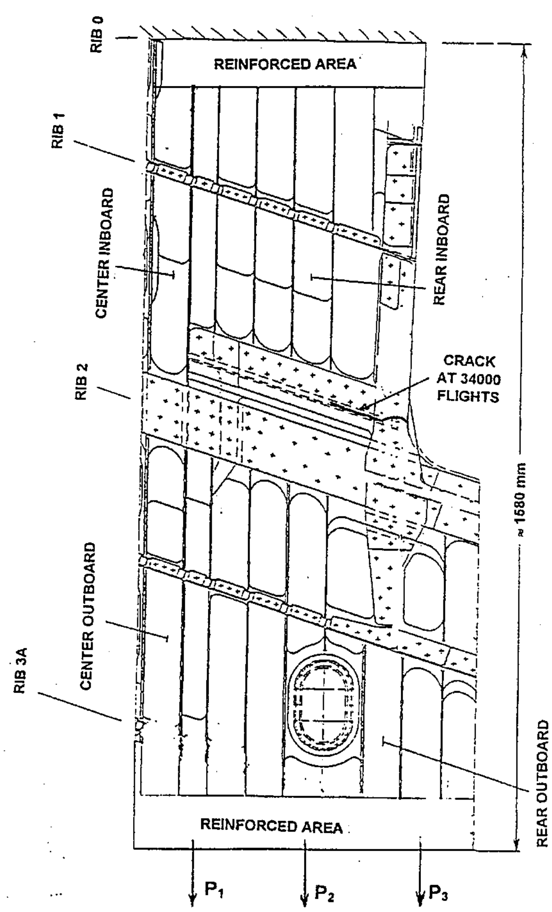


FIGURE 4 : Lower Skin Test Specimen

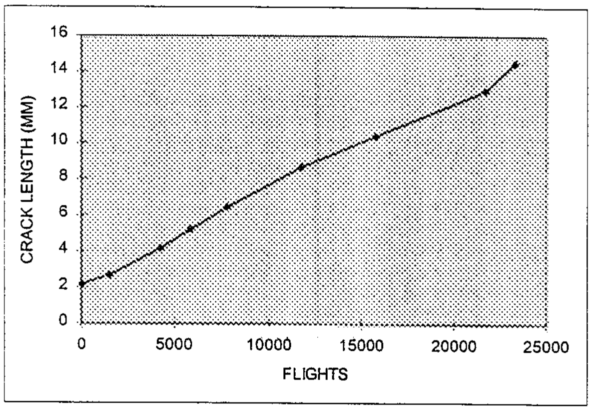


FIGURE 5: Crack Growth at Access Panel

Visual, eddy-current and X-Ray inspections were performed periodically. There was good correlation between the measured and calculated strains.

A number of small cracks (3-10 mm) were detected at the radius of the outboard skin near the center spar after 30,000 flights of fatigue cycling. The cracks joined and grew into a 100 mm crack after 34,000 flights (see Figure 4).

The causes of crack initiation were:

- Sharp edge at skin landing gear bay intersection with the thickness step at the skin splice.
- Small radius at the thickness step at the skin splice.
- Eccentric load transfer at the lap joint forming the skin splice.

The latter effect was more severe in the test article than in the actual wing, due to lack of stabilizing effect of spars and ribs. At 34,000 flights a patch was installed in the cracked area and the testing was resumed. Design changes were introduced in the area as a result of the component test.

At 40,000 flights, eight artificial flaws were introduced at points not influenced by the patch. A crack began to grow from the access panel cutout initial flaw at 50,000 flights and it grew up to 15 mm to the end of the test at 73,000 flights. The crack growth curve is given in Figure 5.

Forward Engine Mount Test:

The forward engine mount is the main load carrying element in the engine support structure. It will be tested for two lifetimes of fatigue cycling followed by one lifetime of damage-tolerance loading. The test specimen will include, in addition to the forward engine mount, adjacent structures as pylon skins, part of the crossbeam, bulkhead attachment plate, skin attachment angles, etc. in order to simulate accurately the load transfer from engine to fuselage. The test article in its loading rig is shown in Figure 6. The loading of the specimen includes:

- Vertical inertia and aerodynamic loads on the engine.
- Fore-and-aft engine thrust loads.
- Pitch, roll and yaw moments caused by the engine loads.

The loads will be applied through a loading rig attached to the adapter plate and loaded by three hydraulic jacks (see Figure 6). The loading spectrum will be based on the Galaxy gust and maneuver spectrum and engine thrust loading, with a block of 2,000 flights and nearly 20 events per flight.

At the conclusion of the damage-tolerance cycling a residual strength test, simulating a symmetric up-gust case (n_z = 5.5g), will be performed. Approximately 50 strain gages will be bonded on the specimen.

The test will include the following stages:

- Static calibration consisting of a complete strain survey under a calibration load
- Fatigue test consisting of two lifetimes (40,000 flights) of fatigue cycling, with calibrations and inspections at fixed intervals.
- Introduction of artificial flaws at critical locations, and one lifetime (20,000 flights) of damage-tolerance cycling with crack growth measurement
- Residual load testing
- Tear-down inspection

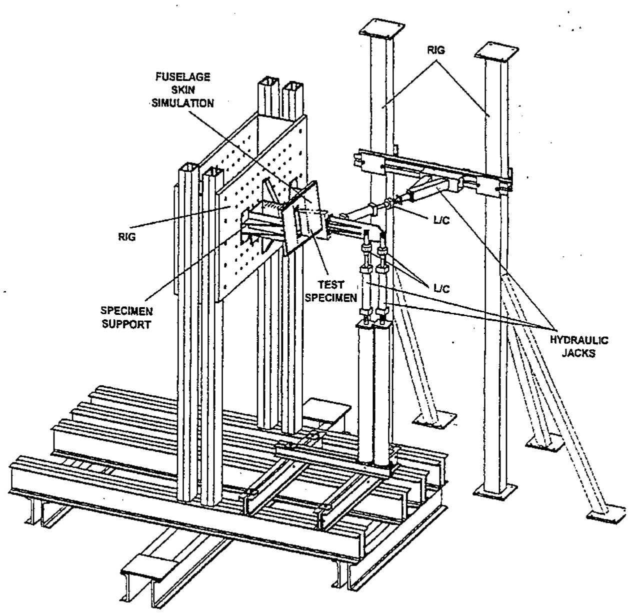


FIGURE 6: Forward Engine Mount Test Set-up

Coupon Testing

Fatigue testing of coupons was used for two purposes: determination of crack growth parameters for specific material and spectra, and to assist in determining spectrum truncation criteria for fatigue testing.

For the principal metallic materials used on the Galaxy structure, center cracked coupons and cracked coupons with a central hole were tested to constant amplitude and spectrum loading. Various Galaxy spectra, which simulate various combinations of gust, maneuver, cabin pressurization, and lateral loading were used in this study. The crack growth behavior for these coupons were determined, for the various spectra and materials, by loading the coupons in a cyclic test fixture and measuring crack growth. By this procedure, the constant amplitude as well as the spectrum retardation parameters were determined for each material and spectrum combination. These parameters were applied to the damage-tolerance analyses that were performed.

During fatigue testing of components or entire airframes, certain cycles of low stress amplitude, but having many cycles of load application, may be eliminated in order to speed-up the test. In order to determine whether these cycles can be eliminated with a minimal loss of test accuracy, a series of coupon tests were performed. Various combinations of spectra were tested and the effect of "load truncation" on crack growth life was determined. These results enabled a rational decision to be taken for load truncation for the component and full-scale fatigue tests.

Following a component or full-scale fatigue test, fractography is used to track crack growth. In order to correlate crack growth with the number of flights, certain "marker loads" are introduced into the loading spectrum. The coupon tests were used to check the *visibility* of these marker loads as well as to enable an evaluation of the effect of these marker loads on the fatigue life.

Full-Scale Fatigue Test

The aircraft structure will be full-scale fatigue tested for a duration of two lifetimes. The Full-Scale Fatigue Test (FSFT) is being performed in order to fulfill the following objectives:

- Substantiate the airframe structure for two lifetimes of fatigue loading representative of anticipated service.

- Detect critical locations where cracks could initiate.
- Establish more accurate crack growth rates at critical locations which account for redistribution of stresses and interaction between structural components. This data will be used to update the damage-tolerance analysis.
- Define effectiveness of Non-Destructive Inspection (NDI) methods and set criteria for detectable cracks in the analysis.
- Obtain data on extent of secondary cracking for multiple site cracking analysis.
- Demonstrate the effectiveness of any repairs or redesigns introduced to rectify fatigue problems.

The test-article for the full-scale fatigue test consists of a standard production aircraft having all the structural members of the fuselage and both wings, leaving out all non-essential components and systems such as, mechanisms, cabin interior trim and furnishings, radome assembly, aft cone, and landing gear doors. The empennage is being fatigue tested separately, as described below.

Figure 7 shows the Galaxy aircraft mounted in its loading fixture. In this test the aft empennage from the static test article is employed. Dummy engines, landing gears, actuators, and left hand Kruger flap are used. The system and equipment supports and brackets, that are structurally effective are included in the test article. The remaining smaller brackets are omitted, but the fastener holes used for their attachment are drilled on the airframe structure.

The test aircraft is mounted to the test fixture on rigid constraint supports, as follows:

- *Nose landing gear*: constraint in the vertical direction.
- *Engine mount fittings*: constraint in the vertical and lateral directions.
- *Fuselage floor beams*: constraint in the fore and aft direction

The reactions at these support points will be monitored by means of load-cells throughout the tests. The fatigue test spectrum loading consists of randomly selected flight-by-flight sequences, reflecting the anticipated usage of the aircraft. The various flight and aircraft configurations are represented, where each flight is arranged to consist of all the various airborne and ground events that the aircraft experiences in service.

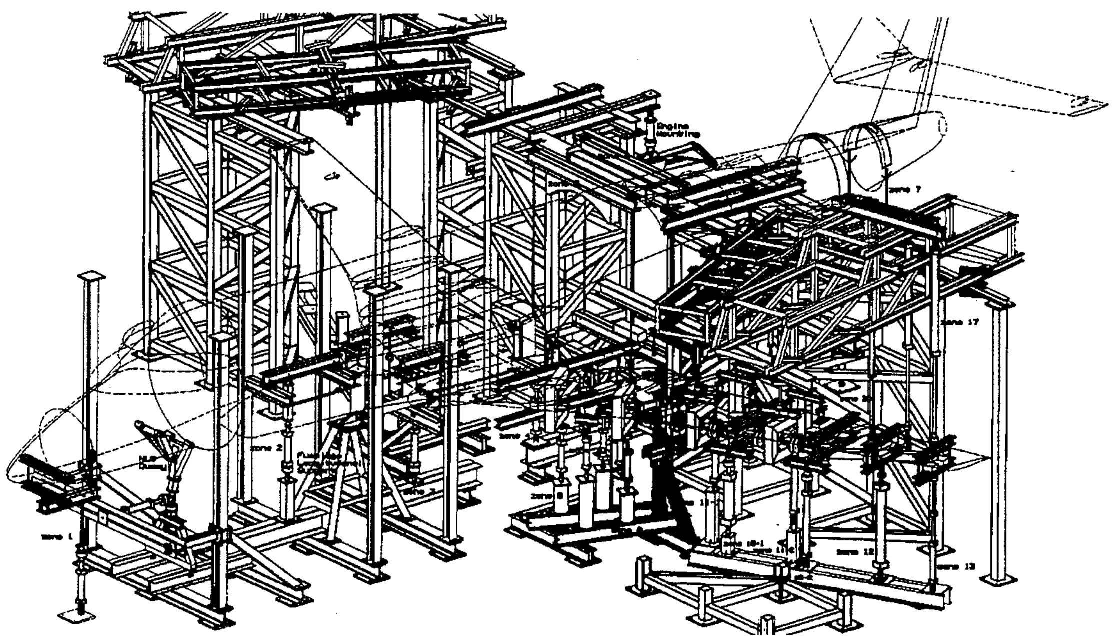


FIGURE 7 : Galaxy Full-Scale Fatigue Test Aircraft Mounted in its Loading Fixture

The spectrum is arranged in blocks of 2,000 flights having three types of missions: 900 short flights, 600 medium flights and 500 long range flights. The 21 events included in each flight (11 air events, 3 transition events, and 7 ground events) are performed in the following sequence:

		No. of Events / flight
1.	Taxi before take-off	2
2.	Braking before take-off	1
3.	Take-off run / Rotation	1
4.	Climb gust	2
5.	Climb maneuver	1
6.	Cruise gust	3
7.	Cruise maneuver	1
8.	Descent	1
9.	Flap extension	1
10.	Approach gust	1
11.	Approach maneuver	1
12.	Landing impact (4 steps)	1
13.	Reverse thrust / 1g after landing	1
14.	Taxi after landing	2
15.	Braking after landing	1
16.	Transition 1g landing / 1g BTO	1
	Total:	21

Randomization of events is performed in each stage, and events are picked randomly for each stage to form one flight. At the end of each block, marker loads are introduced consisting of the 21 highest events in the block.

Testing consists of: a strain survey followed by two lifetimes of spectrum loading. Approximately 500 strain-gages will be mounted to the test-article. At least 200 strain gages shall be monitored every block, and 60 strain gages are to be monitored continuously. Wherever cracks may appear, the crack growth will be monitored using crack gages.

The test starts with the test specimen under zero external load, with the dead weight of test article and test rigs extracted to give zero reactions at the engine and nose landing gear supports.

The test-article is divided into 26 loading zones and four reaction constraints as follows:

No. of Channels	*Zone No.*	*Number of jacks*	*Description of Areas*
LOADS			
4	1,2,3,4	4	Forward fuselage
3	5,6,7	3	Aft Fuselage (F_z)
3	8,9,10	6	Fixed wing, inboard (F_z)
4	11,12,13,14	8	Fixed wing, outboard (F_z)
1	15	2	Inboard flap (resultant)
1	16	2	Outboard flap (resultant)
1	17	2	Kruger flap
1	18	2	Aileron attachment (F_z)
3	19,20,21	6	Slat (resultant)
1	22	1	Horizontal tail (F_z)
1	23	2	Engine thrust
1	24	2	Main landing gear (F_z)
1	25	2	Main landing gear (F_x)
1	26	-	Fuselage pressure
	Total jacks:	42	
REACTIONS			
1	-	-	Nose landing gear reaction (F_z).
2	-	-	Engine mount support structure (F_x) & (F_z).
1	-	-	Fuselage longitudinal support (F_y).

Each loading zone is independently loaded during each event of the spectrum, using servo-hydraulic actuators. The zone loading for each event was determined using a "constrained least-square error method" which minimizes deviations of the important structural parameters. In addition, the passenger cabin and baggage compartment is pressurized once per flight, using compressed-air during the airborne events of the spectrum (pressure varies from zero to 8.7 psi for mission 1, and 9.0 psi for missions 2 and 3).

NDI will be performed at specific intervals at all major structural items. An evaluation of the inspection methods and results in the NDI program will assist in the preparation of Instructions for Continued Airworthiness per FAR 25.1529.

If cracks are detected, crack-gages will be bonded to the crack-tip in order to monitor crack growth rates.

Upon completion of two lifetimes, additional testing may be carried out as required. At the completion of cycling, a teardown inspection of selected areas will be carried out, and fractographic analysis of cracked areas will be performed.

Empennage Test

An empennage fatigue and damage-tolerance test including the horizontal stabilizer, vertical tail, elevators, rudder and a section of the aft fuselage will be performed. The aim of this test is to substantiate the empennage structures for fatigue and damage-tolerance under a combination of symmetrical, asymmetrical and engine thrust reverser buffeting loads.

The test article and its installation to a rigid rig is shown in Figure 8. The aft fuselage section extends up to the aft engine support frame at fuselage Sta. 15891. The elevator will be installed at an angle of 27° upward relative to the horizontal stabilizer reference plane. This angle corresponds to the most severe fatigue event for the elevator.

Three types of loads will be applied on the test article:

- Vertical loads on the horizontal stabilizer and elevator. They include the lift-off, gust, maneuver and ground loads.
- Side loads on the vertical tail and rudder. They include lateral gust and yaw maneuver loads.
- Buffeting loads caused by engine reverse thrust during landing. They will be applied as roll and yaw moments on the horizontal stabilizer.

The loading spectrum is based on a block of 2,000 flights with three missions; it includes nearly 20 gust and maneuver events and approximately 40 buffeting load cycles per flight.

The loading set-up for the empennage gust and maneuver loads will include 5 hydraulic jacks on the vertical tail and 6 jacks on the horizontal stabilizer. The buffeting loads will be applied by a separate set of 4 hydraulic jacks; 2 for the roll moments and 2 for the yaw moments.

The residual strength test to be performed at the end of the cyclic loading will include two limit load cases:

- A combined horizontal stabilizer/ vertical tail limit load case
- A limit load case for the horizontal stabilizer and the elevators.

Approximately 150 strain gages will be bonded on the test article. NDI inspections will be performed periodically during the test, at fixed intervals. A zero inspection will be performed after the assembly of the test article. A teardown inspection will be performed after the test is completed. Embedded defects will be incorporated in the composite material elevators and rudder during their production. Artificial flaws will be introduced on the metallic structures after two lifetimes of fatigue cycling. The test will include the following stages:

- Static calibration consisting of a complete strain survey under a calibration load
- Fatigue test consisting of 2 lifetimes (40,000 flights) of fatigue cycling, with calibrations and inspections at fixed intervals.
- Introduction of artificial flaws at critical locations, and one lifetime (20,000 flights) of damage-tolerance cycling with crack growth measurement
- Two cases of residual load testing
- Tear down inspection

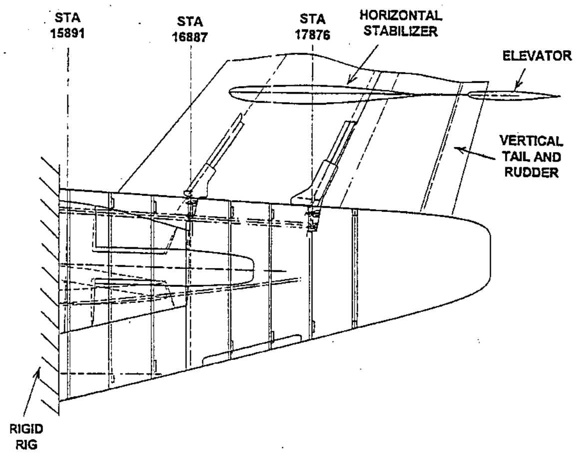


FIGURE 8 : Empennage Test Rig

Summary

This paper has described briefly the experimental and analytical elements that are part of the substantiation program for the structural fatigue durability and damage-tolerance of the Galaxy aircraft.

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