MAGELLAN
AEROSPACE CORPORATION

ER 1924

Fatigue and Damage Tolerance Certification Report
ROLLS ROYCE – TRENT 500 ENGINE
Nozzle and Plug Exhaust System

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REV	DATE	BY	PAGES	CHANGES	PAGES AFFECTED
A	07-03-02	AW/DS	ALL	Incorporated Updated Acoustic Stresses and Creep Analysis	All
B	10-24-02	DS	All	Weight Reduction Appendix	All
C	11-04-03	DS	All	Add Element Stress Contour Plot and update report to incorporate HH comments	All

To be checked as latest Version to Airbus/Aircelle – email

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1.1 Summary

This report examines the methods and analysis used to justify the fatigue and damage tolerance properties for the A340 Trent 500 exhaust assembly.

A physical description of the exhaust system and the corresponding FEA models used to represent the exhaust system for analysis are in §1.2 and Chapter 3.0 of this report. The analysis carried out using these models was done using loads from reference 5 (ER1921 Loads Report). The analysis results from: reference 6 (ER1923 Static Report), reference 7 (ER1922 Thermal Report), and reference 13 (ER1944 Vibration and Acoustic Report), are used to determine the low cycle fatigue (LCF) and high cycle fatigue (HCF) stresses to be used in the fatigue analysis. Summaries of these stresses are shown in Chapter 6.

The material properties used in the analysis are contained in Chapter 4 of this report. The material fatigue properties are results of material testing carried out in reference 8 (ER1936 Material Test Program). Material properties for crack growth are determined from other source material, refer to §4.2 of this report.

The requirements of Chapter 2 and the fatigue methods of analysis described in Chapter 5 are applied to major components of the exhaust system with a combined LCF and HCF loading. Each section of Chapter 7 examines geometry, stresses, and stress concentration factors for critical locations on each part. A summary table in each section shows the combined damage induced by both LCF and HCF.

Damage tolerances of the highest stressed flange, lap joint, rivet joint, and weld are examined in Chapter 8. These components are specifically chosen in order to give a conservative representation of the exhaust system. Each component is analyzed using AFGROW fracture mechanics software program for single load path – safe life. The component's geometry, stresses, stress spectrum (LCF & HCF), and likely critical crack paths are examined. Damaged areas with an initial crack of 0.005" (0.016" x 0.063" for welds) are demonstrated to withstand ultimate loading after 3 A/C by comparison of ultimate load stress to stress at critical crack growth.

At this time, test data for honeycomb fatigue properties from reference 8 are not available. Until these properties are available, data from the Critical Design Review shall be used, as shown in §4.1.1.

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2 Requirements

1) Fatigue and damage tolerance analysis shall conform to the requirements shown in FAR/JAR 25.571 (a), (b), (c), (d).

2) The fatigue design life goal (DLG) for the Trent 500 exhaust system is 20,000 flight cycles (100,000 cycles with a scatter factor of five applied).

3)..Demonstrate damage tolerance during the DLG of the aircraft with an initial crack sizes of 0.005" in the joints and 0.016" x 0.063" for the welds, with a scatter factor of 3.

Since there is no inspection requirement for the exhaust system, if the crack growth occurs during the 3 A/C it is demonstrated that the damaged allowable stress is greater than the stress in the part under ultimate load conditions.

4)..Additionally we have analyzed the exhaust flange via static analysis to be missing one bolt under the application of ultimate loading (Blade-out Loop2). This analysis has been made in the Static Stress Report ER1923 (ref. 6), §5.2.1.4.1 and §5.2.2.4.1.

Fatigue Mission is ≤ 8 hours

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2.1 Assumptions

1. All dimensions are nominal.
2. Pressure and temperature profiles are axi-symmetric.
3. Fatigue mission is approximately eight hours as determined in Table 2.1-1 (reference 10).

Table 2.1-1: Hybrid Fatigue Mission Mix

Frequency of Mission	**20%**	**40%**	**40%**	**Hybrid Mission**
Time (min)/ Flight	75	405	705	459
Life time (hrs)	3268	35294	61438	**100000**
Life time (cycles)	2614	5229	5229	13072

Flight Cycle (FC)= 0.20 x (75 min) + 0.40 x (405 min) + 0.40 x (705 min)

4. Two thermal cycles per flight mission.
5. Initial crack size for damage tolerance analysis is 0.005" except at the welds it is 0.016" x 0.063".

SR: 100,000 flying hours

Nacelle Certified for 20,000 flying hours by Aircelle / Airbus through EASA

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4.2 Crack Growth Properties

Although the fracture toughness of the material increases with temperature, which is desirable, the crack growth rate is adversely affected at elevated temperatures. The effect temperature has on the crack growth properties is not significant except at the high stress ratio, see Figures 4.2-4 through 4.2-6 per reference 17.

Based on these Figures, it is shown that the crack growth rate increases with temperature. Table 4.2-1 summarizes this information and calculates ratios for elevated temperature crack rate to room temperature crack rate.

Since the only available data for Beta 21s crack growth rate is at room temperature, the maximum ratio calculated in Table 4.2-1 shall be used to multiply both the static and alternating stresses for Beta 21s. By doing this, it will conservatively account for the effect of temperature upon the room temperature crack growth properties for Beta 21s.

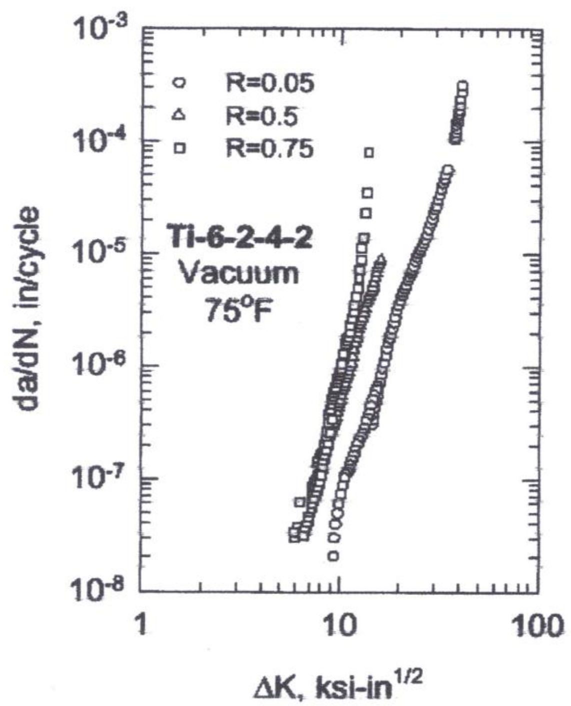

Figure 4.2-4: Vacuum FCG data for Ti-6Al-2Sn-4Zr-2Mo-Si 75°F(24°C)

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5.0 Method of Analysis

The eight hour hybrid mission for fatigue is shown in Table 2.1-1. An example of the flight phases during a typical mission is shown below in Table 5-1. An overall temperature history of the plug is shown in Figure 5-2.

To conservatively treat the LCF stresses contributed from thermal loading, we shall apply the maximum thermal stress twice per one flight cycle. This loading spectrum is shown in Figure 5-3.

Conservatively the highest thermal stresses are used for the plug and nozzle although they do not always occur simultaneously with the highest level of vibration.

Table 5-1: Typical Mission Profile

Condition	Altitude (ft)
Tax Min Idle	0
Tax Min Idle	0
MTO 56k	0
MTO 56k	0
MCL	1500
MCL	10000
MCL	20000
MCL	39000
Cruise 39k ft	39000
Cruise 39k ft	39000
Descent Idle	39000
Descent Idle	31000
Descent Idle	20000
Descent Idle	10000
Descent Idle	1500
Approach Idle	1500
Approach Idle	1500
Max Reverse	0
Max Reverse	0
Taxi Min Idle	0
Taxi Min Idle	0

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