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Static and Fatigue Analysis on Repaired Fuselage Skin

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ABSTRACT OF PROJECT WORK
This project focuses on Static and Fatigue Analysis of repaired fuselage skin at the pitot tube location. The fuselage is an aircraft's main body section. Skin is damaged around the pitot tube on fuselage. It is removed by trimming out cracked area and installing the repair doubler to the skin.
As part of the certification process, an aircraft manufacturer performs tests or analysis to demonstrate compliance with Federal Aviation Regulations 25.571. This analysis is generally based upon an implicit assumption of isolated cracking, i.e., the effect of a single crack is considered with respect to the issues of detectable or initial size, fracture-critical size, and rate of growth. The subject repairs need to be evaluated for Static strength and fatigue evaluation. Fatigue analysis is performed for the subject repair and fatigue life is evaluated based on High Cycle Fatigue (HCF), through S-N curves.

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I. INTRODUCTION

This report evaluates the static and fatigue strengths for repair /rework performed on airplanes. The analysis carried out for the repair uses generic fatigue and fracture methods obtained from standard books as well as public domain data.

The satisfactory performance of an aircraft requires continuous maintenance of aircraft structural integrity. It is important that metal structural repairs be made according to the best available techniques because improper repair techniques can pose an immediate or potential danger.

Damage to metal aircraft structures is often caused by corrosion, erosion, normal stress, and accidents and mishaps. Sometimes aircraft structure modifications require extensive structural rework. The problem of repairing a damaged section is usually solved by duplicating the original part in strength, kind of material, and dimensions.

The purpose of this report is to provide repair evaluation for fleet safety which comprises Static and Fatigue analysis. In general, when an airplane damage tolerant fail safe structure is subjected to repair /modification, the airplane needs to be evaluated for the two main stages static strength and fatigue as a part of airworthiness compliance, structural fleet safety and passenger safety which are the important criteria for any aerospace industry standards.

1.1 Damage Description

Operator found damage to the skin around the pitot tube on aircraft as shown in the

Figure 1and Figure 2. The distortion starts about 75 mm behind the radome and extends back 110 mm. Two cracks were identified at the aft end of the skin at the pitot mount cut out. No distortions were found on the internal stringers below or above the pitot tube attached. Skin thickness is 0.8 mm made of Al 2024-T3 Clad Sheet and Fasteners are 100^0 flush head MS20426D aluminum rivets

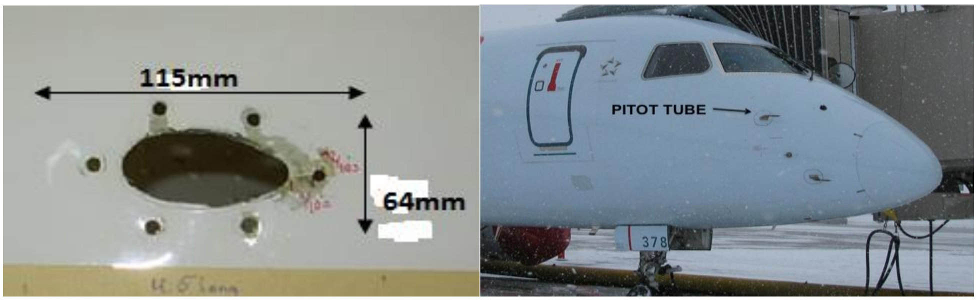


Figure 1: Damage Description and Repair Location on Fuselage Skin

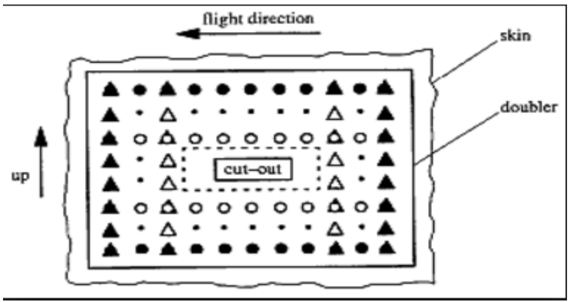


Figure 2: Repair Sketch for the Crack Damage

II. OBJECTIVES

This repair is mainly divided into two stages in terms of Static strength & Fatigue evaluation calculations mentioned below.

I. Static strength Evaluation **-** (Stage I) :

- ✓ It evaluates the Static strength at the critical detail.

II. Fatigue Evaluation **-** (Stage II) :

- ✓ It evaluates the repair life at the critical detail where it is subjected to cyclic loading.

III. DAMAGE EVALUATION AND SELECTION OF REPAIR METHODS.

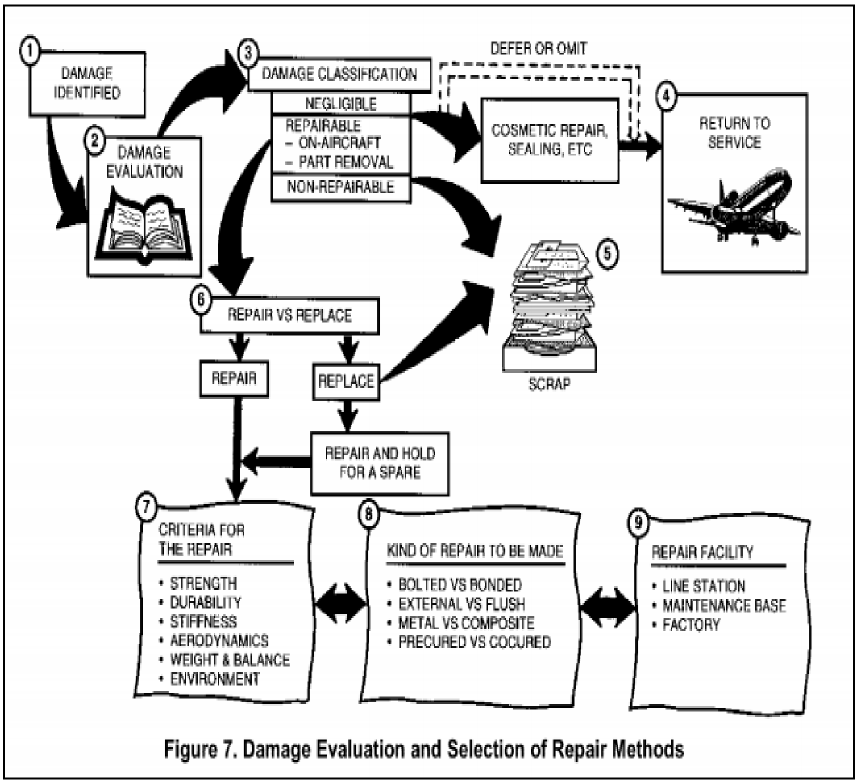


Figure 3: Figure showing the Damage Evaluation and selection of Repair Methods.

3.1 REPAIR INSTRUCTIONS

3.1.1 General Repair Instructions to Pressurized Area:

The skin of aircraft that are pressurized during flight is highly stressed. The pressurization cycles apply loads to the skin, and the repair to this type of structure requires more rivets than a repair to a non-pressurized skin (See

Figure 1) and [3, 6]

1. Remove the damaged skin section.
2. Radius all corners to 12.5mm.
3. Fabricate a doubler of the same type of material as, but of one guage greater thickness than, the skin. The size of the doubler depends on the number of rows, edge distance, and rivets spacing.
4. Fabricate an insert of the same material and same thickness as the damaged skin. The skin to insert clearance is typically 0.4mm to 1mm.
5. Drill the holes through the doubler, insertion, and original skin.
6. Spread a thin layer of sealant on the doubler and secure the doubler to the skin with Clecos.
7. Use the same type of fastener as in the surrounding area, and install the doubler to the skin and the insertion to the doubler. Dip all fasteners in the sealant before installation

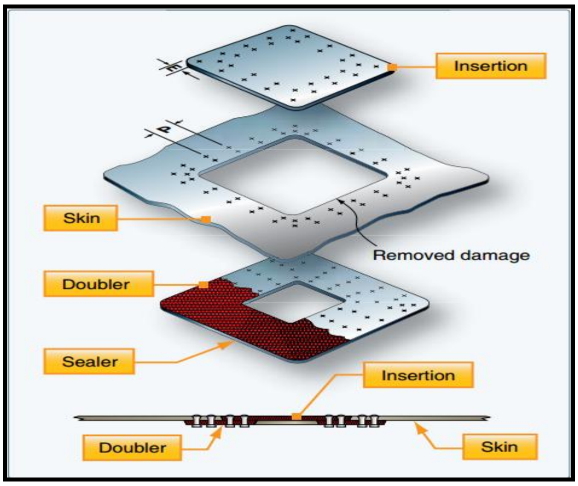


Figure 4: Pressurized Skin Repair

Assumptions

- The damage is in the skin, there is no damage to adjacent structure.
- Internal pressurization effects are the dominant cause of stresses in the pitot tube region.
- Airplane is subjected to constant amplitude loading.

3.1.2 Repair Instructions for Subject Crack Damage:

- Trim out the damage in the skin as shown in the Figure 2. HFEC inspect the edge of each trim and enlarge and maintain a minimum radius to avoid any sharp edges. Maintain a minimum of 2D minimum edge margins to the trim edges.
- HFEC inspect all the existing fastener holes common to the skin repair to ensure no crack exist.
- Fabricate a repair doubler made of 1.143mm thick and 2024-T3 clad aluminum sheet of 215mm X 165mm as shown in Figure 2
- For pressurized fuselage cabins and lower wing skins - two areas particularly prone to fatigue through the long-continued application and relaxation of tension stresses - the standard material is an aluminum alloy designated 2024-T3.
- Do not use the repair doubler on a stringer or other structure. If the last row of a fastener comes over a stringer then add an additional fastener row to have a better internal inspectability and repair durability.
- Install all repair fasteners at all repair doubler corners.
- Pre-form the repair doubler to the fuselage contour. Drill the appropriate fastener holes. Use a drill stop to restrict penetration through skin.

IV. GEOMETRY AND MATERIAL DETAILS

Geometry and material properties are shown in Table 1 and **[**6, 8**]**

Table 1: Geometric and Material Details

S. No.	Part	Geometry Thickness (mm)	Material
1	Skin	0.80	Al 2024-T3 Clad Sheet
2	Repair Doubler	1.143	Al 2024-T3 Clad Sheet

4.1 Material Properties

Material Allowable is listed in below Table 2

Table 2: Material Allowable[1]

Material Al 2024-T3 CLAD SHEET	
F_{ty} (Tensile Yield Strength)	324MPa
F_{tu} (Ultimate Strength)	469MPa
K_A (Fracture Toughness)	37 MPa-m$^{1/2}$
E (Young's Modules)	73.1 GPa

4.3 Fastener Details

Fasteners: 100^0 flush head aluminum rivets are used [6, 8]

Table 3: Fastener Details

Fastener	Diameter	Material	Young's Modulus
MS20426D	4mm	Al 2017-T3	72 GPa

V. LOAD DATA EVALUATION:

In order to predict the crack-growth behavior of an aircraft structure, the designer needs to know the sequence of stress cycles applied during the life of the structure. This stress history for a new design is developed from the service life requirements and the mission profile information specified by the procurement activity. Based on this information a repeated load history due to ground handling, flight maneuvers, gusts, pressurization, landing, store ejection, and any other load source is developed.

5.1 Types of Load Cases

Airplanes are being subjected to a variety of types of loading, such as pull-up manoeuvres, rolling manoeuvres, gusts, taxi loads etc. Each of these types of loading can be associated with a variety of airplane and flight conditions with regard to airplane configuration, weight, speed, altitude etc. The occurrence of a specific type of loading with specified magnitude in a fully defined flight condition may be indicated as a loading case [9].

To define a loading case, for a certain aircraft, the following data has to be set:

- Payload and its distribution
- Fuel load and its distribution
- Speed
- Altitude
- Acceleration (linear and rotational)
- High lift devices setting
- Primary and secondary flight control angles

Cabin Pressure Case:

Cabin Pressure Case [9, 10]: The cabin pressure is not an external load for the airplane taken as a body. However for the structure, the cabin pressure is a significant external load source creating internal hoop load and longitudinal load in its members. The hoop load is always more important than the longitudinal load. For a perfect cylindrical fuselage, the running loads are: Hoop load and Longitudinal load

Where p is the cabin differential pressure and R is the fuselage radius. The regulations states that a maximum cabin altitude of 1520m must be maintained at the maximum operating altitude of the aircraft. What it means is that at the maximum operating altitude of the aircraft, the cabin pressure must be equivalent to the ambient air at 1520m. Most of the pressurized aircraft have their cabin pressurization system designed exactly for that requirement. Very few aircraft have a more stringent design standard providing a more comfortable environment for the passengers. Therefore the relief pressure valve of the pressurization system is set to the design settings, based on the maximum cabin altitude selected for passenger comfort and the maximum operating altitude selected for aircraft performance, ensuring that the design cabin differential pressure will not be exceeded in service.

The stress spectrum is considered to have a remote stress due to cabin pressurization. Cabin pressurization primarily causes hoop tension in the fuselage shown in Figure 9. The GAG pressurization load is based on Federal Aviation Regulations 25.571. Hence for the subject deviation location, generation of load spectrum is avoided and assumed only to loaded by hoop stress.

The circumferential stress acting in the Fuselage is calculated as follows:
The atmospheric pressure at sea level =101 kPa
Air pressure at the altitude of 16000m =10.1kPa (10% of Atmospheric pressure) [11]
It is assumed that the cabin pressure of airplane is maintained at 1520m when cruising at 16000m. The pressure inside the cabin is calculated is as follows:
Air pressure at the altitude of 1520m = 85 kPa [11]
Cabin pressure = Air pressure at the altitude of 1520m - Air pressure at the altitude of 16000m
Cabin pressure = (85-10.1) kPa
Cabin pressure = 75 kPa

VI. STATIC STRENGTH EVALUATION - (Stage I):

This analysis evaluates whether proposed repair configuration restores the production strength in line with the standard repair instructions.
All standard strength checks will be performed to insure positive (+ve) Margin of Safety (MoS)

Static Analysis

Subject repair is installed as per the instructions provided in text book [3], the repair doubler is considered 1.143 mm thick.
Material replacement:
The doubler is thicker and larger than the cutout. The materials are same. The minimum margin safety factor is:
MS = 1.143/0.080 – 1 =0.4285 (+)
The repair area and load capability are restored.

Fastener and joint capability

Hoop Direction:
F hoop = PR/t = $(75\times0.864)/0.82\times10^{-3} = 80\times10^{3}$ kN/m^{2} =80 MPa
Where
R = Fuselage radius = .864 m [14]
P = Cabin Pressure =75 MPa (Refer section 7; for load calculation)
t = Thickness of skin

A loss = 15.875*0.812=12.9 mm^2
P req = f_{hoop}*A loss
P req = 80*12.9 =1031 N
Assumed there are 3 MS20426D fasteners,
P_{allow} = 2668*3 =8000 N
M.S. = (P_{allow}/P_{req}-1)
M.S. = (8000/1031-1)
=7.8 (+)

VII. FATIGUE ANALYSIS

Durability is the ability of structure to resist fatigue and environmental damage during the airplane's expected operational service use.

Fatigue calculations are performed by evaluating based on reference "Airframe Stress Analysis and sizing [12]. Following are the steps for calculating fatigue life.

1. Severity Factor (SF), which accounts for
- Fastener type, method of installation, interference, hole preparation, etc
- Detail Design
- Fastener load distribution to avoid 'peaking effect'
- Minimization of the stress concentration caused by both local load transfer at a fastener and bypass load

2. Discrepancy Factor
3. Fatigue quality index
4. Finally fatigue life from S-N Curve.

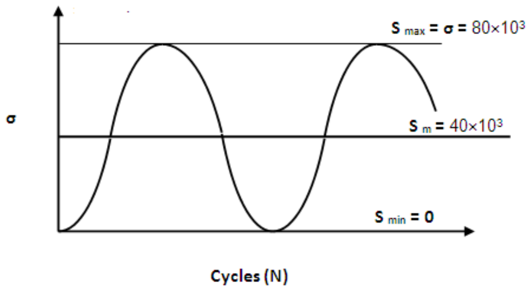
Fatigue loading considering as 'Constant Amplitude loading':
Subject repair is installed as per the instruction provided in text book [3], the repair doubler is considered 1.143mm thick.
Hoop Direction:
F_{hoop} = P R/t = (75×0.864)/0.82×10^{-3} = 80×10^3 kN/m^2 =80 MPa
Where
R = Fuselage radius = 864mm [14]
P = Cabin Pressure =75kPa (Refer section 6 for load calculation)
t = Thickness of skin

S_{max} = 80×10^3 kN/m^2
S_{min} = 0 kN/m^2 [Assuming]

$$S_m = \frac{S_{max} + S_{min}}{2}$$

$$S_m = \frac{80 \times 10^{\wedge}3 + 0}{2}$$

= 40×10^3 kN/m^2 or 40 MPa



Severity Factor (SF):
The severity factor (SF) is a fatigue factor that accounts for local peak stress caused by load transfer through fastener and by pass load transfer through plate.

$$SF = \frac{(\alpha \times \beta)}{\sigma_{Ref}} \times (\sigma_1 + \sigma_2)$$

$$\sigma_1 + \sigma_2 = \left(\frac{K_{tb} \times \Delta P}{Dt} \theta + \frac{K_{tg} \times P}{Wt} \right)$$

Where:

α		- Surface Condition Factor
β		- Hole Filling factor
K_{tb}		- Bearing Stress Concentration Factor
K_{tg}		- Stress Concentration Factor
θ		- Bearing distribution factor
D		- Diameter of the fastener
t		- Thickness of the skin
W		- Width of the plate

Bearing distribution factor ϴ = 1.25

$$SF = \frac{(\alpha \times \beta)}{\sigma_{Ref}} \times \left[\left(\frac{K_{tb} \times \Delta P}{Dt} \right) \theta + \left(\frac{K_{tg} \times P}{Wt} \right) \right]$$

$$SF = \frac{(1 \times 0.75)}{\left(\frac{P}{W \times t} \right)} \times \left[\left(\frac{1.2 \times 0.35 P}{0.00396 \times 0.000812} \right) \times 1.25 + \left(\frac{3.1 \times 0.65 P}{0.0240 \times 0.000812} \right) \right]$$

$$SF = (1 \times 0.75 \times 0.0240 \times 0.000812) \times \left[\left(\frac{1.2 \times 0.35}{0.00396 \times 0.000812} \right) \times 1.25 + \left(\frac{3.1 \times 0.65}{0.0240 \times 0.000812} \right) \right]$$

SF = 3.89
Discrepancy factor is assumed to be 1.2.
K = 1.2×3.89 = 4.662

S-N Curve based on mean stress for K_t = 5

The classical approach to fatigue also referred to as Stress Controlled Fatigue or High Cycle Fatigue (HCF), through S-N curves. In order to determine the strength of materials under the action of fatigue loads, specimens with polished surfaces are subjected to repeated or varying loads of specified magnitude while the stress reversals are counted up to the destruction point. The number of the stress cycles to failure can be approximated by the WOHLER or S-N DIAGRAM.

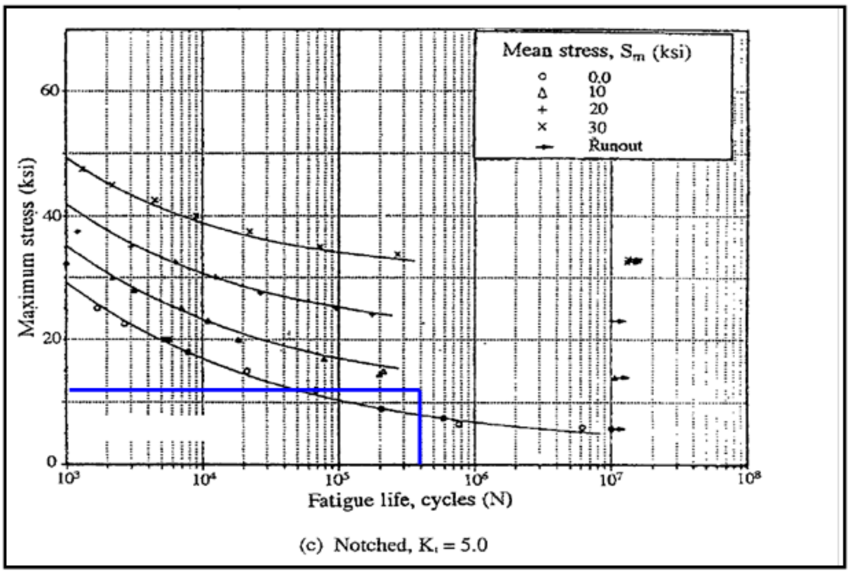


Figure 2: Life Cycles for K_t=5

80MPa=11.6 ksi
Calculated Life Cycles = 5, 50,000 Flight Cycles.
Scatter factor - A factor used to reduce the calculated fatigue life, time interval of crack growth, or verification testing of a safe-life structure (it is generally 3.0-5.0). The scatter factor is affected by the following factors:

- ✓ A confidence level factor due to the size of the test sample establishing the fatigue performance
- ✓ Number of test samples
- ✓ An environmental factor that gives some allowable for environmental load history
- ✓ A risk factor that depends on whether the structure is for safe-life or fail-safe capability

Scatter Factor = 4.0
Estimated life cycles = $N_{4.66}$/4= (5, 00,000)/4
= 1, 25,000 Flight Cycles.

VIII. CONCLUSIONS

- Two cracks were identified at the aft end of the skin at the pitot mount cut out. No distortions were found on the internal stringers below or above the pitot tube attached.
- The repair is checked for static strength to insure production strength is restored.
- Fatigue life is determined at the critical detail by using standard procedures and methodologies and found to be 1, 25,000 flight cycles.
- As per the results shown, installed repair is good at two aspects i.e. static strength & Fatigue.

Further Work

Good details design is the most important mean to decrease the stress concentration factor which will significantly increase the fatigue life of the joint. Below are the parameters that can be looked in to for better fatigue strength.

- Reduce Stress Concentration

- Interference-fit fastener hole condition
- Reduce end fastener load
- Cold work fastener hole

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REFERENCES

[1]. Damage Summary Information.
[2]. Repair Options for Airframes by Mohan M. Ratwani, Ph D. https://www.faa.gov/regulations_policies/handbooks_manuals/aircraft/amt_airframe_
[3]. handbook/media/ama_Ch04.pdf-Aircraft Metal Structural Repair, Chapter 4.
[4]. Airframe Structural Design by "Michael Chun –Yung Niu" Second Edition, Jan 1989(Chapter 15)
[5]. Damage Tolerance Analysis of Aero Structural Components by tcs.aerospace
[6]. Airframe Stress Analysis and sizing by "Michael Chun –Yung Niu", Second Edition, Jan 1999 (pg. no.696)
[7]. Material Property; www.matweb.com http://www.matweb.com/search/DataSheet.aspx?MatGUID=781ce4adb30c4d548320b0ab262a5d28
[8]. Fracture Toughness; https://www.makeitfrom.com/material-properties/2024-T3-Aluminum http://asm.matweb.com/search/SpecificMaterial.asp?bassnum=MA2024T3
[9]. MECH 6471 Aircraft Structures External Loads By Dr. Mohammed Abdo https://www.faa.gov/regulations_policies/handbooks_manuals/aircraft/amt_airframe_
[10]. handbook/media/ama_Ch04.pdf-Aircraft Metal Structural Repair, Chapter 4.
[11]. Pressure Altitude: http://wiki.answers.com/Q/Air_pressure http://www.ext.colostate.edu/pubs/foodnut/p41.html
[12]. Airframe Stress Analysis and sizing by "Michael Chun –Yung Niu", Second Edition, Jan 1999 (Chapter 9.12, page 382)
[13]. Elementary Engineering Fracture mechanics by "David Broke" 3rd printing, 1984.
[14]. Airplane Information: http://www.aviamarket.org/reviews/civil-aircraft/515-cessna-560xl-citation-excel.html
[15]. Elements of Fracture Mechanics by Prasant Kumar (pg.no. 65,192).
[16]. https://www.faa.gov/documentLibrary/media/Advisory_Circular/AC_25_571-1D_.pdf