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Case study

Metallurgical failure analysis of a cracked aluminum 7075 wing internal angle

Morteza Soltani Tajabadi *

Left Side, Door No. 6, 74th Alley Mahdieh Avenue, Kerman 7614793499, Iran

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ABSTRACT

Internal angles are used to strengthen Aircrafts center box corners where the wing is attached to the airframe. There are 16 angles in Airbus A300s wing box. On the right side, rear spur, and lower flange area of the center wing box, one of these angles had been cracked with a length of 28 mm. This crack has decreased residual strength of the part under allowed values and resulted to a rupture in the rear spur lower cap. Several reports of the same occurrences in other Airbus A300 air crafts, highlight the importance of finding the causes of this failure. Detailed optical and SEM, plus 4 other metallurgical tests were conducted on the failed angle. Finally, it was concluded that corrosion fatigue was the main reason which itself comes from manufacturing, maintenance, metallurgical, and geometric reasons as were discussed in this study.

1. Introduction

An internal angle fitting related to center box of Airbus A-300 submitted for failure investigation, which is shown in Fig. 1.

This component was a structural component of a 14-year-old commercial jetliner that had carried out approximately 19,000 flight cycles when the crack was detected through NDT implementation. This type of aircraft is mostly utilized by cargo airlines. The aircraft had been grounded for service, and specific NDT operation had been implemented according to manufacturer service bulletin which notifies all operators of A-300 to conduct this test for those aircrafts that their flight cycles exceeds 17,000 cycles [1]. NDT detected a crack between two fastener holes in a way that affect residual strength of the part. Airbus had issued a service bulletin to replace this component with a modified substitute. The location of the component in the airframe is illustrated in Fig. 2.

As an independent failure investigation, metallurgical properties of the angle had been studied with tests, corresponding scientific data had been gathered, and failure causes had been clarified.

2. Chemical quantitative test

With the aim of understanding chemistry of the component, quantitative test was conducted on it. Achieved information about composition of elements that are mentioned in Table 1 had been compared to similar Alloys. It had been concluded that this chemistry is similar to aluminum 7075.

* Tel.: +98 9121354128.
E-mail address: imst3@yahoo.com

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Fig. 1. Image of cracked internal angel after removing from A/C.

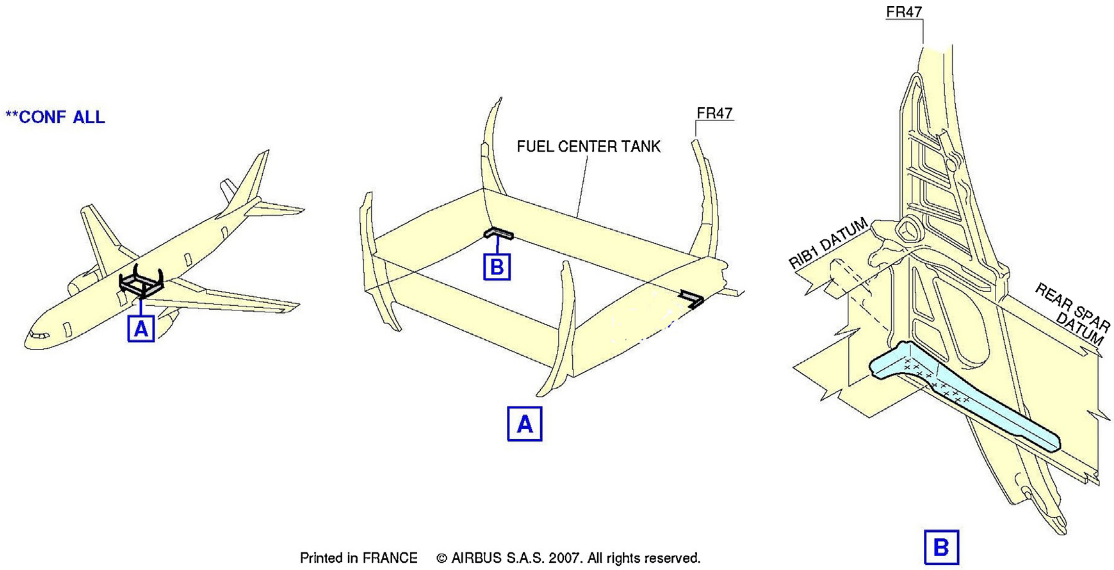

Fig. 2. Illustration of internal angles location [1].

Table 1
Chemical composition of the angle gained by quantitative test.

Si%	Fe%	Cu%	Mn%	Mg%	Cr%	Ni%	Zn%	Ti%	Pb%	V%	B%	Al%
0.06	0.18	1.45	0.01	2.57	0.18	0.011	5.70	0.007	0.004	0.008	0.003	Base

3. Tensile test

Table 2 shows tensile test results for all three specimens. According to very results, material of the component is isotopic. There was not found any kind of coating on the surface of the part.

4. Hardness test

Surface hardness of the component was taken and the results are illustrated in Table 3. As it is shown, there is not any remarkable change in hardness all over the part.

5. Qualitative assessment of loading and stress distribution

Precise analysis of loads was requiring a complete CAD model of the structure. Providing this model was not possible at this study. Therefore, it is attempted to provide a similar configuration of corresponding structure in order to simulate effect of Wing loading on stress distribution of the angle. Although the realistic magnitude of stresses over the angle would not

Table 2
Tensile test results.

	Diameter	Section area (mm^2)	Yield strength 0.2% (MPa)	Ultimate strength 0.2% (MPa)	Relative elongation %A20
1	3.96	12.32	480	547	22.04
2	3.98	12.44	487	551	22.24
3	3.91	12.01	439	511	22.49
AL7475-T761 nominal limits			462	524	12

Table 3
Hardness test results.

Testing point	Affected load (kgrf)	Hardness (HB)			
		1st specimen	2nd specimen	3rd specimen	Average
Center	62.5	150	146	150	150

calculated but such simulation could reveal Potential locations that have higher possibility of cracking due to similar geometry of loadings. According to [10] the loading under Airbus A-300 was calculated as below:

- The pressure load is determined by the calculating the load factor from the arccosine of 17 degrees.
- The maximum climb angle for A-300 from any airport is 17 degrees.
- $N = 1/\text{arccosine } 17 = 1.04569$
- The maximum take-off weight of A300–600 is around 170,000 kg.

From the basics of aerodynamics;

- Lift force = load factor * weight of aircraft
- As we are interested to calculate the structural parameters during take-off and climbing phase, lift must be greater that weight.

Thus;

- The total lift force required to climb through 17 degrees, the aircraft should be able to generate the lift force 1751.531 kN (with its couple of wings). Therefore, the force developed by each wing is 875.765 kN.
- This force is converted into the pressure load, which is in the form of uniformly distributed load by dividing this force by semi wing area of 130 m^2. Therefore, the total pressure load applied from the bottom of the surface is 6736.65 Pa.

This load is affected under lower wing area of the wing as a triangular load distribution.

Contour plot of stress analysis over the angle is indicated in Fig. 3. Apparently, most severe stresses took place around a filet. It seems geometrical shape of angle's installing place had made designers to include this filet in the geometry of the angle. Changing in section area of the part resulted in some kind of stress concentration around the filet zone [3,11]. Meanwhile, crack path is along with filets axis. Therefore, stress concentration along the filet is able to be a cracking reason.

The place in which the angle is located has a step shaped geometry at its lower surface. Hence, geometry of the angle is shaped in consistent with that. Including this step results in a change in the parts section and creates stress concentration at this place. Fig. 4 shows the place in absence of angle to describe the condition effectively.

6. Visual and light microscope examination

Fig. 5 illustrates condition of the part and the crack prior to performing tests. As is shown, crack had been grown between two holes. These holes are related to fasteners. Apparently, applying load under the wing triggers further stress concentration around the holes [2,3]. If we suppose the principal stress, which came from loads under wing surface it leads to a tensile stress would be happened under angles lower surface which comprises the crack. Obviously, crack path is normal to direction of tensile stress from wing lower surface. This cracking direction is to be expected because according to fracture mechanics principles crack path is normal to principle stress direction [3].

7. Scanning electron microscopic test

After performing SEM test, two important evidences were revealed. Firstly, as it is also discussed in [4], general appearance of fracture surface is consistent with fatigue as there are numerous beachlines and striations on surface that are illustrated in Fig. 6.

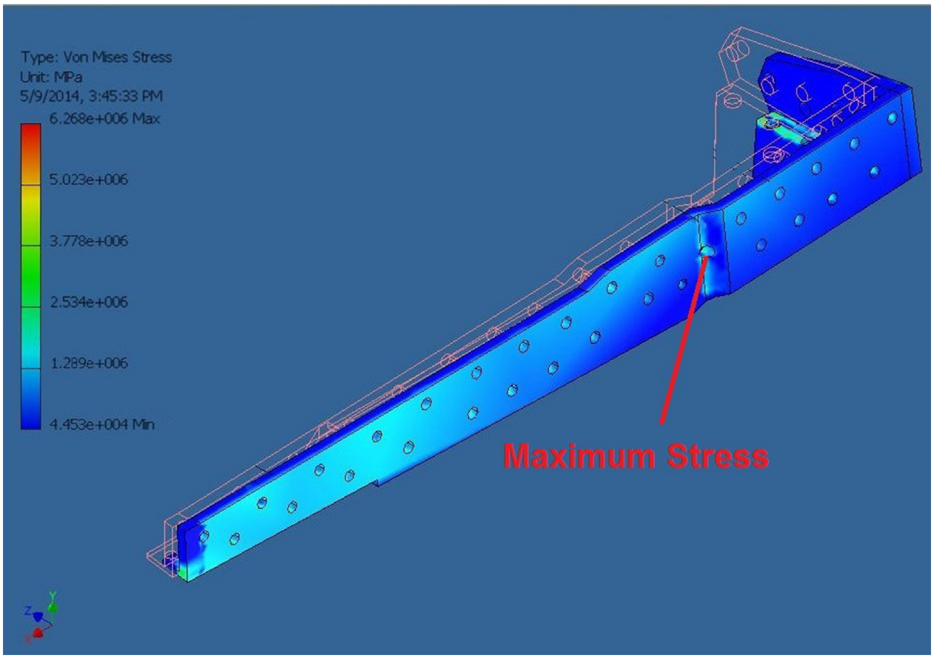

Fig. 3. von misses stress distribution analysis.

Fig. 4. Installation place of the angle at the corner of center box (step shaped geometry is highlighted in a red rectangular). (For interpretation of the references to color in this figure legend, the reader is referred to the web version of this article.)

Greater magnitude of striations pictures is also shown by Fig. 7. This picture confirms that the crack is propagated through fatigue that comes from cyclic stresses of wing aerodynamic loads [4].

Secondly, a shiny crack origin was obvious. The reason of this initiation was not clear at this stage but this crack origin is located at the limb of a fastener hole. According to SEM test results, it is proved that some kind of damage or material defect resulted in creation of crack origin (which will be investigated at Section 7 of this paper). Meanwhile, a cyclic loading propagates the crack between two holes.

8. Corrosion on the surface of the angle

Over lower surface of the angle, a large number of micro cracks were detected through optic microscope inspection (Fig. 8) which were results of corrosion. These places are not accessible for cleaning at the time of maintenance because this

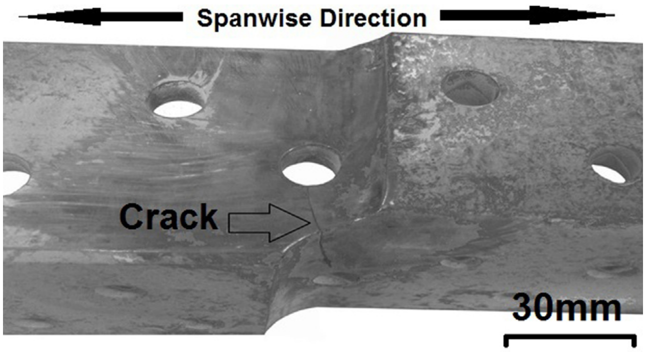

Fig. 5. Closer image of the crack and illustration of A/C Span wise direction (this direction is normal to cracking path as it is shown in the picture).

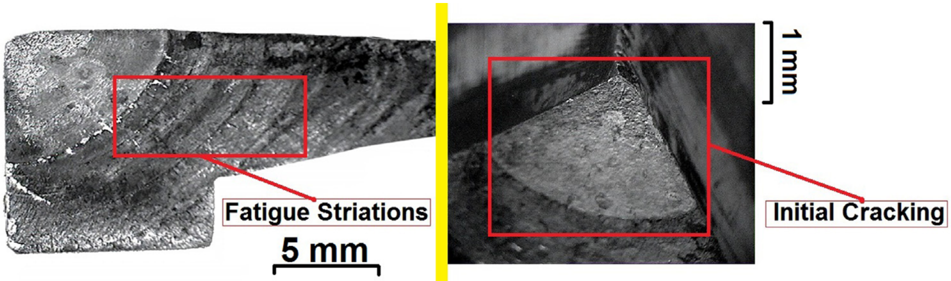

Fig. 6. SEM low magnitude images of fracture surface (at the left image, shiny initial crack is obvious and at the right side, apparent fatigue beach lines are visible).

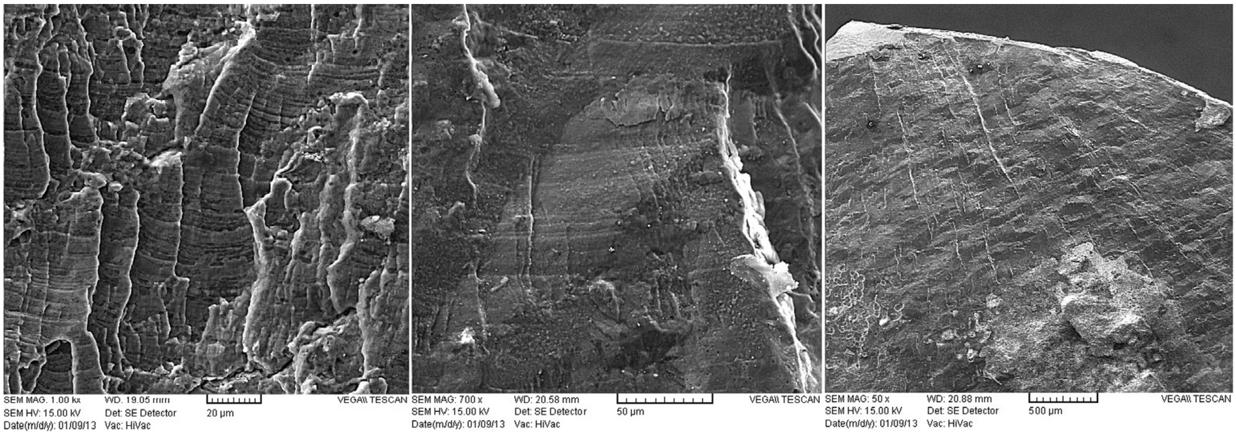

Fig. 7. SEM images in three different magnitudes from fracture surface.

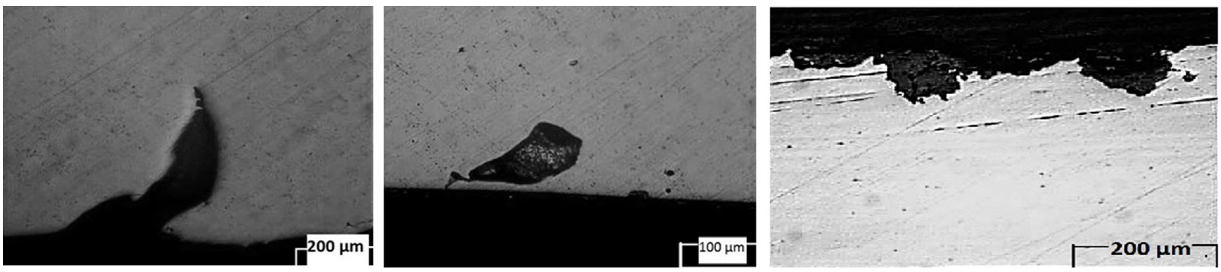

Fig. 8. Microscopic crack detected on the surface of the angle.

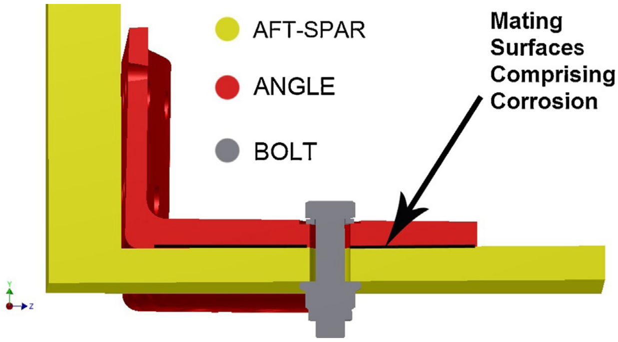

Fig. 9. Illustration of corrosion area between angle and spur.

surface is tangent to the lower cap of aft-spar. Thus, penetration of fuel, water, and other corrosives might result in corrosion over corresponding surface much more than the others. Fig. 9 illustrates the area of major corrosion. Based on fracture principles, such flaws bring about severe stress concentration if they become at exposure of tensile stress [7,8] (lift force of the wing generates tensile stress along lower surface of the wing and compressive stress along upper surface). Therefore, existence of these flaws should take into account as another reason of corrosion fatigue cracks in the component.

9. Hole surface inspection

Evaluation of this region indicated several scratches on it. It had found that a special maintenance operation named cold working was the reason of these scratches. In this job by applying special tools an axial force was affected into the hole walls in order to convert potential cracks to notches and prevent crack progression. Therefore, ultimate life of the component would be extended. However, wrong implementation of this operation can damage the hole surface as it shown in Fig. 10 and creates scratches which can result in crack initiation as found in [5]. According to maintenance records, this component had detached 2 times during aircrafts service life and fasteners removal- installation could not make such severe and serial scratches. On the other hand, numerous holes on the component could make the technicians fatigue and lessen maintenance performance in a way that some holes became damaged. Therefore, another cause of producing crack origin could be this human error to cold work the hole of crack initiation. The fasteners should locate at the holes with push fit condition during assembling process and it can increase potentiality of making initial cracks [6].

10. The reason of initial flaw

As the flaw was located at the limb of the hole, two surfaces should be inspected at the vicinity of the flaw. One was the internal surface of the hole; the other is external surface of angles main body. Important information had been collected from these surfaces. Fig. 11 shows both surfaces.

11. Dimensional analysis on fastener holes

Similar to a work done through reference [12], fastener holes were inspected to investigate consequences of cold working. According to concerned service bulletin [1], it is necessary to carry out this job on the fastener holes. Through this work, potential micro cracks at the holes proximity turn into notches because of radial force affected by cold expanding tools. Thus, probability of crack initiation will be decreased. However, the manufacturer has clarified precise specific limits for hole expansions. If structure technicians do not observe the limits, it would lead to damage such as micro cracks. Nine sample

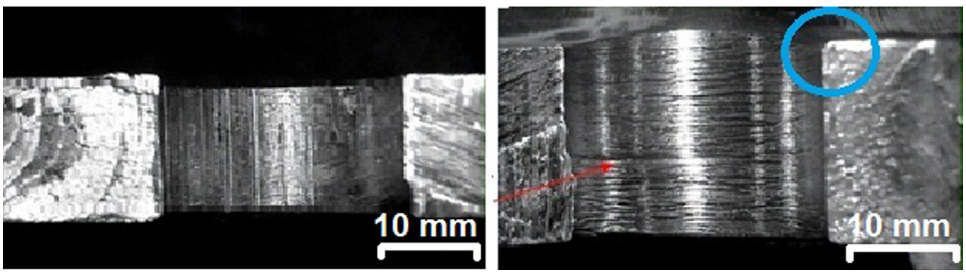

Fig. 10. Illustration of hole internal surface (crack initiation is shown by blue ring and scratches are pointed by red arrow). (For interpretation of the references to color in this figure legend, the reader is referred to the web version of this article.)

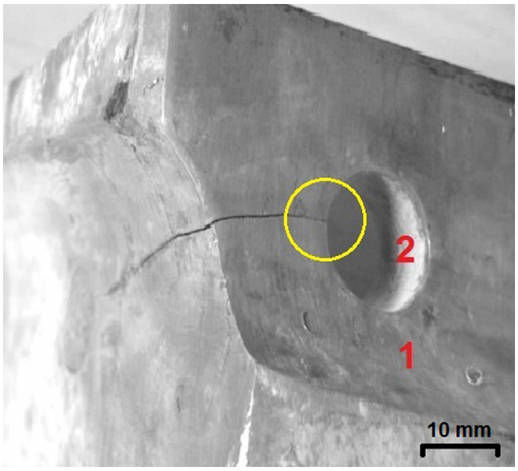

Fig. 11. Hole surface and main body surface (crack has initiated at the limb of this hole which is indicated by yellow ring). (For interpretation of the references to color in this figure legend, the reader is referred to the web version of this article.)

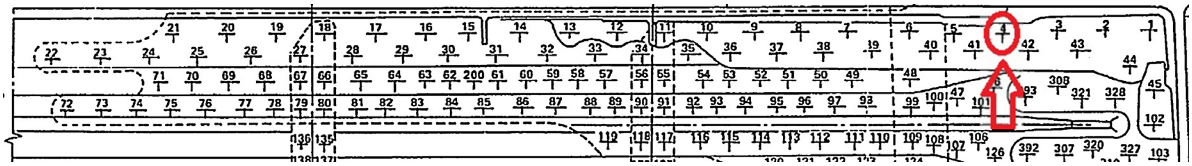

Fig. 12. Hole numbering of the angle in S/B 53-0282 [1].

HOLE	ZONE	DIAMETER BEFORE COLD EXPANSION		
		STARTING REAMER	MIN	MAX
1	B	CBSR-20-2-N-1	16.03 mm (0.631 in.)	16.10 mm (0.633 in.)
2	B	CBSR-20-2-N-1	16.03 mm (0.631 in.)	16.10 mm (0.633 in.)
3	B	CBSR-18-2-N-1	14.43 mm (0.568 in.)	14.50 mm (0.570 in.)
4	B	CBSR-16-2-N-1	12.83 mm (0.505 in.)	12.90 mm (0.507 in.)
5	B	CBSR-16-2-N-1	12.83 mm (0.505 in.)	12.90 mm (0.507 in.)
6 KITS A03, A04	B	CBSR-16-2-N-1	12.83 mm (0.505 in.)	12.90 mm (0.507 in.)
6 KIT A05	B	CBSR-18-0-N-1	13.64 mm (0.537 in.)	13.72 mm (0.540 in.)
7	A	CBSR-16-2-N-1	12.83 mm (0.505 in.)	12.90 mm (0.507 in.)
8	A	CBSR-14-1-N-1	11.10 mm (0.437 in.)	11.18 mm (0.440 in.)
9	A	CBSR-14-1-N-1	11.10 mm (0.437 in.)	11.18 mm (0.440 in.)

After Cold Expansion

HOLE	FINAL REAMING		
	MIN	MAX	
1	16.629 mm (0.654 in.)	16.668 mm (0.656 in.)	16.64
2	15.836 mm (0.623 in.)	15.876 mm (0.625 in.)	
2	16.629 mm (0.654 in.)	16.668 mm (0.656 in.)	15.85
3	or)39 mm (0.592 in.)	15.078 mm (0.593 in.)	14.45
4	13.462 mm (0.530 in.)	13.501 mm (0.531 in.)	12.85
5	13.462 mm (0.530 in.)	13.501 mm (0.531 in.)	12.90
6 KITS A03, A04	13.462 mm (0.530 in.)	13.501 mm (0.531 in.)	13.48
6 KIT A05	14.247 mm (0.561 in.)	14.287 mm (0.562 in.)	14.26
7	13.462 mm (0.530 in.)	13.501 mm (0.531 in.)	13.50
8	11.872 mm (0.467 in.)	11.912 mm (0.468 in.)	11.17
9	11.872 mm (0.467 in.)	11.912 mm (0.468 in.)	11.90

Dimensions are extracted from S/B 53-0282 [1].

Fig. 13. Hole expansion limits to utilize in cold working [1].

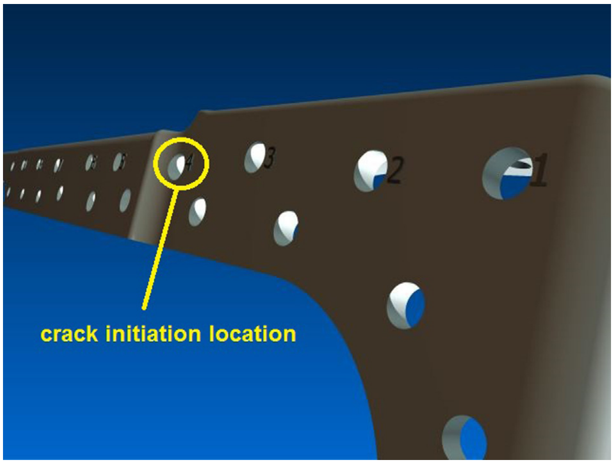

Fig. 14. Illustration of hole location on the angle.

holes were elected for inspection including hole no. 4 which was location for crack initiation. In Fig. 12, hole no. 4 is detectable with a red circle around it.

Factual diameters of the holes are measured through caliper with precision of 0.01 mm. At the left side, a table shows the dimensions of the holes before cold working. The right table illustrates the dimension after final cold working. The results of caliper measurement are indicated at the right side of Fig. 13. Reddened values are not at the limit when greened values are consistent with defined limits. An apparent discrepancy has occurred at some holes including no. 4. It seems cold work expansion was not performed properly. It was not possible to obtain information about cold work from previous owner of the aircraft. However, according to professional structural repair engineers of temporary owner, the filet at the location of hole no. 4 would be a contributing factor to disturb technician to mount cold working equipment on the hole and do the job adequately. A detailed explanation of cold working is available at reference [13]. Nevertheless, this error is seen at other holes that are not closed to a filet. Fig. 14 illustrates hole numbers.

12. Conclusion

According to failure investigation of the angle, corrosion-fatigue is apparently the main reason of the failure but there are some other obvious side reasons for this failure.

- Surface corrosion is obviously cause of microscopic flaws in which very high values of stress intensities were created. So that, these flaws behave like initial cracks and become able to grow.
- Incorrect or careless cold work maintenance job brought about scratches in the cracking region and is the second potential cause of cracking.
- Existence of a filet at the vicinity of crack location created high values of stress intensity and stress concentration that can help crack propagation. Thus, part geometry is the third contributing factor of this failure

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