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Multi-Site Damage in Aircraft Structural Splice Joint

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Abstract
A significant difficulty of current aircraft industries is to beat the expanding demand of lower weight, higher performance and stability and longer existence of specimen at decreased expense. Much assembly procedure in critical structures like military or commercial aircraft will be based on riveted joints. These joints are inclined to crack initiation which may prompt multi-site damage, therefore lots of exertion will anticipate the life of structure under multi-site damage, in this project a piece of fuselage connection is taken for multi-site damage analysis, this interfacing part is known as splice joint. In this study, a splice joint which contains two splice panels, a doubler and stiffener made up of AL-2024 T3 is taken as the specimen. This assembly comprises of 26 rivets arranged in zigzag pattern. The load on the assembly (Splice joint) is seen to be reason of cabin internal pressurization of fuselage. The entire assembly is practically modelled utilizing MSC PATRAN programming, followed by stress analysis in MSC NASTRAN. The analysis is accomplished for no crack, single crack and multi cracks, for these entire conditions stress intensity factor are built up and crack propagation for each crack is acquired dependent on the stress intensity factor. The change in S.I.F vs. crack length behavior is studied and final failure of splice skin is analyzed.

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I. INTRODUCTION

A significant difficulty of current aircraft industries is to beat the expanding demand of lower weight, higher performance and stability and longer existence of specimen at reasonable cost and at limited time span. The fuselage structure for the most part comprises of skin panels joined directly to the structural members, like frames, stringers for longitudinal splices. In assembling process critical structures like military or commercial aircraft, riveted or bolted joints are essentially utilized as they offer numerous alternatives to the engineer. To satisfy fatigue conditions, the engineer can either keep the resisting force levels below as far as possible or ensure that the moderate fracture development life of the component is larger than the evaluated value in addition to designed factor of safety. The failure reason of aircraft components was mainly started by eternal forces, like fuselage pressurization force, wing lifting force. The force due of the fuselage cabin pressurization can be considered as one of the necessary loads cases for the fuselage structure damage which encounters constant amplitude load cycles. To beat these difficulties, splice joints are used for the fuselage and wing structures.

Splices are generally used when structural members are required in larger lengths than the available member. Multiple Site Damage (MSD) is one of the major threats to airworthiness of any aircraft. MSD is the simultaneous advancement of fatigue cracks at a variety of similar structural details. MSD has been most obvious in fuselage joint structures, and can bring about unexpected

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catastrophic failure of aircraft, as it is hard to identify. Fuselage fatigue strength at joints and splices is likewise interesting. The fatigue crack initiation phenomenon at joint rivet holes - specifically at the upper row in the outer skin of a lap joint splice - should be understood to encourage joint design. Crack growth rates likewise should be resolved, as this decides the crack inspection interval. The link-up of MSD prompting the failure of ligaments is a significant phenomenon to be studied in this context.

Airframe Design: Airframe structures are the members which are subjected to major loads. These airframe members remain at last if all the equipment and systems of the airplane is removed. In modern aircraft, skin plays the main role in carrying loads. Different design approaches used for aerospace structural design are: Safe Life approach, Fail Safe approach and Damage Tolerance approach.

Multisite Damage (MSD): For aerospace community, structural integrity of aging structure like commercial transport aircraft is of great concern. Structural integrity of a structure may reduce or loss because of fatigue cracking, this happens due to long service life of an aircraft. For structural analyst and maintenance operator, particular kind of fatigue cracking was posing special problems and these were known as Multisite damage. Linear Elastic Fracture Mechanics (LEFM) approach is used to predict the fatigue crack growth behaviour.

Importance of this study:This study explores damage tolerance of the splice joint in presence of the crack, crack initiation sequence and propagation life of the part. Finite element analysis approach is completed, in this study for stress analysis model of splice joint is created. Stresses are introduced in the structure because of the applied tensile load.

Mechanism of fatigue:

Total Life = Crack initiation+ Crack growth

Fatigue happen in the members who are exposed to repeated cyclic loading and this phenomenon will prompt development of crack well below its yield strength. Structure under fatigue load is probably going to fail early under fatigue load rather than the constant loading. To handle these issue designers should plan a structure based on certain patterns and these patterns are increased performance requirement, increased operational flexibility and higher operating stress of design.

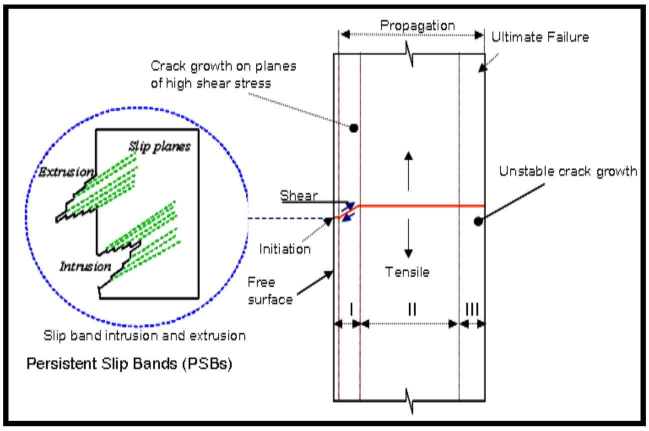

Fig.1 Schematic representation of crack growth

Fracture mechanics principle: In fracture mechanics it is expected that all engineering materials cracks from which cracks initiate and material fails. The presence of crack will prompt high stresses close to crack tip. If this cracked body is loaded, the load is joined by inelastic deformation close to crack tip, with the exception of brittle material. However in some cases the effect of inelastic deformation and nonlinear near crack tip is small compared to crack size. In such cases linear theory is used to address stress distribution in cracked body. There are two methods used in fracture mechanics: Energy criterion approach and Stress intensity approach

Fatigue Crack Growth Rate (FCGR): When a component containing crack will undergo cyclic or repeated loading then the growth of crack is likely to increase with gradual increase in number of cycles. This process is known as fatigue crack growth.

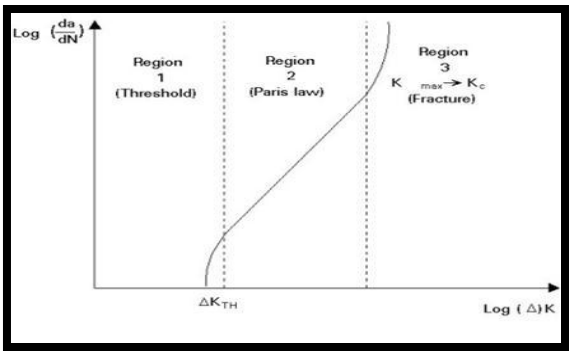

Fig.2 Typical da/dN curve

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Methodology:This project will investigate the fatigue life estimation of a splice joint panel in the presence of the multiple cracks at the riveted locations. In the initial part of the study analysis is carried out to identify the maximum tensile stress location at the rivet locations which are the prone areas for crack initiation. FE tools such as MSC NASTRAN/PATRAN will be employed for carrying out the FE analysis and stress intensity factor estimation for the progressive crack will be evaluated using MVCCI method.

Finite Element Analysis:The fatigue crack will initiate from the locations of the maximum tensile stress. The rivet hole locations are one of the stress concentration regions. Therefore rivet hole locations are the most probable location for the fatigue crack initiation.

Panel Geometry:The splice joint panel consists of two skin splices joined by doubler plate having size of 220 x 56 x 1.5 mm thickness. Skin and doubler plate is joined by rivets having 3mm diameter. The two longitudinal plates are separated by a gap of 2mm. The panel is subjected to tensile loads equivalent for internal pressure of 12 psi.

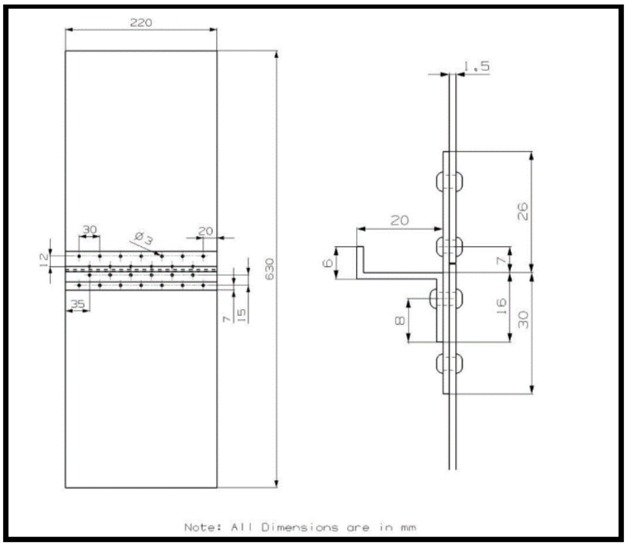

Fig.3 Panel Geometry

II. Finite Element Modelling of Panel

Elements used in model and degree of freedom at each node.

Parts of the splice joint	Type of elements	No. of Elements
Skin	Quad4 , Tria3	217229, 1512
Doubler	Quad4	38080
Stiffener	Quad4	27740
Rivet	Beam	86

Table 1

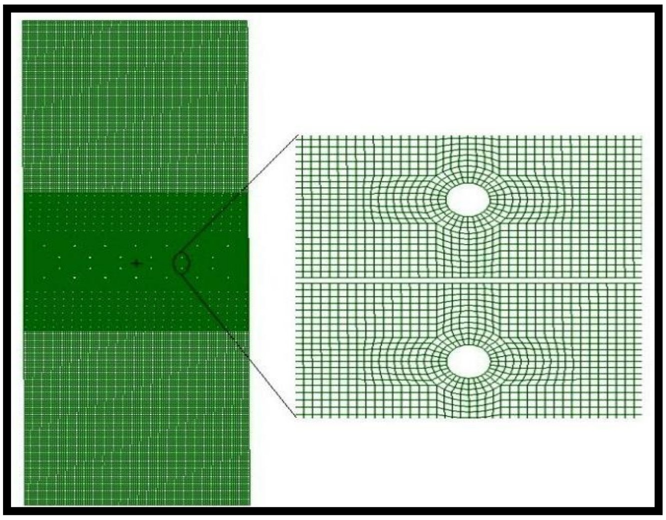

Fig.4 FE model of splice skin

Element size(mm)	**Stress in MPa**
0.3	200.1
0.4	200.3
0.5	200.6
0.6	199.7
0.8	197

Table 2:Stress values for different mesh sizes

Property	Aluminium 2024 T-
Density	2.77 g/cm^3
Young’s modulus	70326 MPa
Tensile yield strengtl	343 MPa
Ultimate tensile streng	475 MPa
Fatigue strength coefficient	1014 MPa
Fatigue strength exponent	-0.114
Fracture toughness	32 MPa√m

Table 3:Material Properties AL 2024 T-3

In the present study the internal cabin pressurization is considered as the load case. It is observation that fuselage will undergoes pressure of 0.0827 N/mm2.

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Load is extracted from this stress and is equivalent to 8855 N, and is applied on the panel. This load will generate 26.83MPa of the stress on plane.

Internal cabin pressurization

Circumferential stress (Hoop stress):-

P= Differential cabin pressure= 12Psi= 0.0827 N/mm

d= Diameter of the fuselage= 973 mm

t= Thickness of the stiffened plate= 1.5mm

The hoop stress= p*d/2t

= (0.0827*973)/(2*1.5)

σ_h = 26.83MPa

Calculation of Damage accumulation and number of cycles:

Here detailed explanation on calculation of crack incremental length, number of cycles for next crack to initiate and damage accumulation for 3rd crack is shown,

$197.297=1014(2N_f)^{-0.114}$

N_f=861168 cycles

$d= (N_1/N_{ab1}) + (N_2/N_{aa1})$

d= (741499/861168) + (40605/772131)

d= 0.9137

Damage for second crack to initiate:-

D= (1-d) = 0.0863

Number of cycles for second crack to initiate = 782104 + 33116 =815220 cycles

Incremental Crack length:

For 1st crack, $da= 3e\text{-}8*(4.581)^{3.2}$

= 0.130mm

For 2nd crack, $da= 3e\text{-}8*(4.557)^{3.2}$

= 0.127mm

III. Results and Discussion

Linear static stress analysis of the splice joint is carried out to determine the stress distribution at the rivet holes and also to determine the maximum tensile stress location. The fatigue crack will initiate from these locations of the maximum tensile stresses.

Stress analysis of splice joint

Linear static stress analysis was carried out for the splice joint panel for an applied load of 8.8KN (26.8N/mm2) maximum tensile stress and deflection was determined. The fuselage splice joint with Z-Stiffener is the location where it experiences the uniform stress field at many rivet locations in a row. It is observed that maximum stress location occurs near rivet on skin. For the far field applied stress of 26.8N/mm2 on the panel, maximum stress of 200N/mm2 is observed at one of the edge rivet locations of the extreme row of the rivets

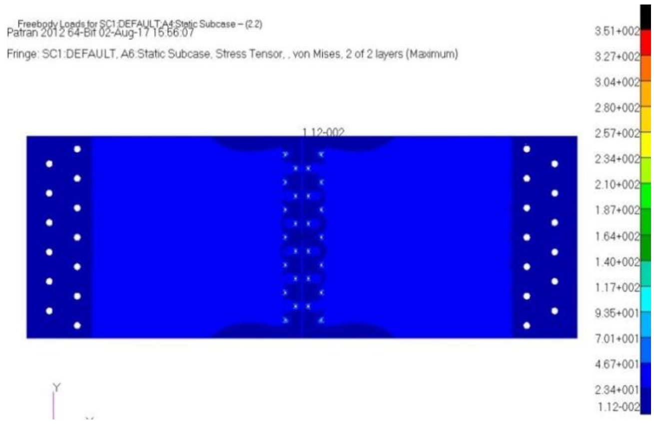

Fig.5 Stress distributions on splice joint

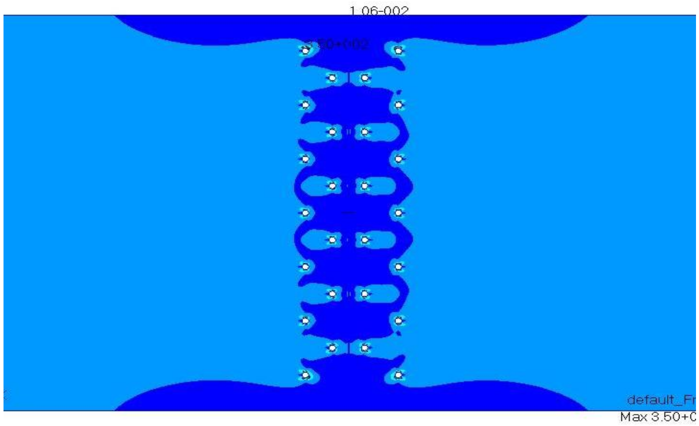

Fig.6 Maximum stress locations on skin rivet holes

A. Estimation of the location and initiation life of first crack

In the first step, stress analysis of model is carried out under no crack condition; stress values for this run are noted.

$\sigma_1=200.691 N/mm^2$ N_1=741499 cycles.

Because of this number of cycles, damage is accumulated at each and every rivet location. Initially 0.5mm of crack is assumed and is simulated by releasing the nodes of model.

Second crack

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After first crack initiation created in model, again analysis is carried out. Stress in the panel will distribute and next maximum stress location is identified. Number of cycles for this maximum stress is calculated.

Damage for second crack to initiate: D=0.0533

Number of cycles for 2nd crack =D*N_{2a1}=40605

Number of cycles for 2nd crack to initiate =741499+40605=782104 cycles

Notations	Stress (N/mm²)	Notations	Number of cycles(N)	Damage accumulation
σ_{2b1}	199.44	N_{2b1}	783307	0.9467
σ_{2a1}	200.105	N_{2a1}	760765	

Table 4 -Values of stress, no. of cycles and damage accumulation

For 2nd crack

	SIF (N/mm²√m)	Da (mm)
Crack 1	4.568	0.157

Table 5 -Values of SIF and crack increment for 2nd crack

In presence of first and second crack of 0.5mm is initiated, analysis is carried out and damage accumulation is calculated.

Damage for third crack to initiate: D=0.0863

Number of cycles for 3rd crack =D*N_{3a1}=33116

Number of cycles for 3rd crack to initiate=782104+33116=815220 cycles.

Notations	Stress (N/mm²)	Notations	Number of cycles(N)	Damage accumulation
σ_{3b1}	197.297	N_{3b1}	861168	0.9137
σ_{3b2}	199.767	N_{3b2}	772131	
σ_{3a1}	216.636	N_{3a1}	383409	

Table 6 Values of stress, no. of cycles and damage accumulation for 3rd crack

	SIF (N/mm²√m)	dv (mm)	Crack length after increment (mm)
Crack 1	4.581	0.13	0.784
Crack 2	4.557	0.127	0.627

Table 7 Values of SIF and crack increment for 3nd crack

Fourth crack

First and second crack are incremented correspondingto the crack length. Third crack is initiated to 0.5mm and analysis is carried out.

Notations	Stress (N/mm²)	Notations	Number of cycles(N)	Damage accumulation
σ_{4b1}	191.828	N_{3b1}	1102298	0.742
σ_{4b2}	192.061	N_{4b2}	1090324	
σ_{4b3}	192.602	N_{4b3}	1063750	
σ_{4a1}	193.088	N_{4a1}	1040492	

Table 8: Values of stress, no. of cycles and damage accumulation for 4th crack

	SIF (N/mm²√m)	dv (mm)	Crack length after increment(mm)
Crack 1	4.365	1.094	1.881
Crack 2	5.044	1.433	2.06

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Crack 3	5.046	1.436	1.936

Table 9: Values of SIF, crack increment and crack length for 4th crack.

Fifth crack
Damage for 5th crack to initiate: D=0.1680
Number of cycle for 5th crack=D*N_{5a1}=83010
Number of cycles for 5th crack to initiate =1084647+83010=1167657 cycles.

Notations	Stress (N/mm^2)	Notations	Number of cycles(N)	Damage accumulation
σ_{5b1}	187.831	N_{5b1}	1325552	0.832
σ_{5b2}	188.004	N_{5b2}	1314891	
σ_{5b3}	188.504	N_{5b3}	1284611	
σ_{5b4}	189.13	N_{5b4}	1247789	
σ_{5a1}	210.2	N_{5a1}	494029	

Table 10: Values of stress, no. of cycles and damage accumulation for 5th crack.

	SIF ($N/mm^2\sqrt{m}$)	dv (mm)	Crack length after increment(mm)
Crack 1	3.128	0.096	1.977
Crack 2	4.129	0.25	2.31
Crack 3	4.287	0.263	2.199
Crack 4	4.477	0.302	0.802

Table 11: Values of SIF and crack increment for 5nd crack

Sixth crack
After incrementing all the four cracks corresponding to the crack length, fifth crack is initiated to 0.5mm and analysis is carried out.
Damage for 6th crack to initiate: D=0.1151
Number of cycle for 6th crack =D*N_{6a1}=127925
Number of cycles for 6th crack to initiate =1167657+127925=1295582 cycles.

	SIF ($N/mm^2\sqrt{m}$)	dv (mm)	Crack length after increment(mm)
Crack 1	3.13	0.148	2.125
Crack 2	4.26	0.396	2.706
Crack 3	4.3	0.408	2.607
Crack 4	4.398	0.439	1.241
Crack 5	3.764	0.267	0.767

Table 12: Values of SIF, crack increment and crack length of 6th crack.

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<table>
<tr><th>Notations</th><th>Stress (N/mm^2)</th><th>Notations</th><th>Number of cycles(N)</th><th>Damage accumulation</th></tr>
<tr><td>σ_{6b1}</td><td>187.468</td><td>N_{6b1}</td><td>1348238</td><td rowspan="6">0.8849</td></tr>
<tr><td>σ_{6b2}</td><td>187.613</td><td>N_{6b2}</td><td>1339125</td></tr>
<tr><td>σ_{6b3}</td><td>187.929</td><td>N_{6b3}</td><td>1319501</td></tr>
<tr><td>σ_{6b4}</td><td>188.289</td><td>N_{6b4}</td><td>1297535</td></tr>
<tr><td>σ_{6b5}</td><td>190.75</td><td>N_{6b5}</td><td>1155713</td></tr>
<tr><td>σ_{6a1}</td><td>191.648</td><td>N_{6a1}</td><td>1111108</td></tr>
</table>

Table 13: Values of stress, no. of cycles and damage accumulation for 6th crack.

<table>
<tr><th rowspan="3">Condition no.</th><th colspan="6">Location of maximum stress (N/mm^2)</th></tr>
<tr><th colspan="2">Rivet no.1</th><th colspan="2">Rivet no.2</th><th colspan="2">Rivet no.3</th></tr>
<tr><th>L</th><th>R</th><th>L</th><th>R</th><th>L</th><th>R</th></tr>
<tr><td>No crack</td><td></td><td>200.7</td><td>197.3</td><td>199.4</td><td>187.8</td><td>191.8</td></tr>
<tr><td>1st crack</td><td></td><td rowspan="6">1</td><td>199.8</td><td>200.1</td><td>188.0</td><td>192.1</td></tr>
<tr><td>2nd crack</td><td></td><td>216.4</td><td rowspan="5">2</td><td>188.5</td><td>192.6</td></tr>
<tr><td>3rd crack</td><td></td><td rowspan="4">3</td><td>189.1</td><td>193.1</td></tr>
<tr><td>4th crack</td><td></td><td>210.2</td><td rowspan="3">4</td></tr>
<tr><td>5th crack</td><td></td><td rowspan="2">5</td></tr>
<tr><td>6th crack</td><td></td></tr>
</table>

<table>
<tr><th colspan="8">Location of maximum stress (N/mm^2)</th></tr>
<tr><th colspan="2">Rivet no.4</th><th colspan="2">Rivet no.5</th><th colspan="2">Rivet no.6</th><th colspan="2">Rivet no.7</th></tr>
<tr><th>L</th><th>R</th><th>L</th><th>R</th><th>L</th><th>R</th><th>L</th><th>R</th></tr>
<tr><td>187.5</td><td>187.5</td><td>191.8</td><td>187.8</td><td>199.4</td><td>197.3</td><td>200.7</td><td></td></tr>
<tr><td>187.6</td><td>187.6</td><td>192.1</td><td>188.0</td><td>200.1</td><td>199.8</td><td rowspan="6">1</td><td></td></tr>
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<tr><td>190.8</td><td>190.8</td><td rowspan="3">4</td><td>210.2</td><td></td></tr>
<tr><td>191.6</td><td>191.6</td><td rowspan="2">5</td><td></td></tr>
<tr><td>6</td><td>6</td><td></td></tr>
</table>

Table 14 Stress values at rivets holes for each run

<table>
<tr><th rowspan="3">Condition no.</th><th colspan="6">Damage Estimation</th></tr>
<tr><th colspan="2">Rivet no.1</th><th colspan="2">Rivet no.2</th><th colspan="2">Rivet no.3</th></tr>
<tr><th>L</th><th>R</th><th>L</th><th>R</th><th>L</th><th>R</th></tr>
<tr><td>1st crack</td><td rowspan="6"></td><td rowspan="6">1</td><td>0.861</td><td>0.947</td><td>0.559</td><td>0.673</td></tr>
<tr><td>2nd crack</td><td>0.985</td><td rowspan="5">1</td><td>0.579</td><td>0.698</td></tr>
<tr><td>3rd crack</td><td rowspan="4">1</td><td>0.298</td><td>0.360</td></tr>
<tr><td>4th crack</td><td>0.834</td><td>1</td></tr>
<tr><td>5th crack</td><td rowspan="2"></td><td rowspan="2"></td></tr>
<tr><td>6th crack</td></tr>
</table>

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Damage Estimation							
Rivet no.4		Rivet no.5		Rivet no.6		Rivet no.7	
L	R	L	R	L	R	L	R
0.550	0.550	0.673	0.559	0.947	0.861	1	
0.568	0.568	0.698	0.579	1	0.985		
0.291	0.291	0.360	0.298		1		
0.802	0.802	1	0.834				
0.407	0.407		1				
1	1						

Table 15 Damage estimation for each step at all critical locations of rivets.

	1^{st} crack	
	Crack length (mm)	SIF ($N/mm^2\sqrt{m}$)
1^{st} run	0.5	4.568
2^{nd} run	0.657	4.581
3^{rd} run	0.784	4.635
4^{th} run	1.881	3.128
5^{th} run	1.997	3.13
6^{th} run	2.125	3.16
7^{th} run	2.721	3.102
8^{th} run	3.161	3.143
9^{th} run	3.747	3.396
10^{th} run	4.242	3.546
11^{th} run	4.304	4.485

	3^{rd} crack	
	Crack length (mm)	SIF ($N/mm^2\sqrt{m}$)
1^{st} run		
2^{nd} run		
3^{rd} run	0.5	5.406
4^{th} run	1.936	4.287
5^{th} run	2.199	4.3
6^{th} run	2.607	4.299
7^{th} run	4.203	4.523
8^{th} run	6.056	5.125
9^{th} run	9.105	6.959
10^{th} run	10.595	7.513
11^{th} run	14.404	9.255

	5^{th} crack

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	Crack length (mm)	SIF (N/mm$^2\sqrt{m}$)
1st run		
2nd run		
3rd run		
4th run		
5th run	0.5	3.764
6th run	0.767	4.705
7th run	2.897	4.176
8th run	4.35	4.354
9th run	6.012	5.646
10th run	6.775	6.738
11th run	9.489	11.63

Table 16 Cumulative crack lengths and stress intensity factor for each run.

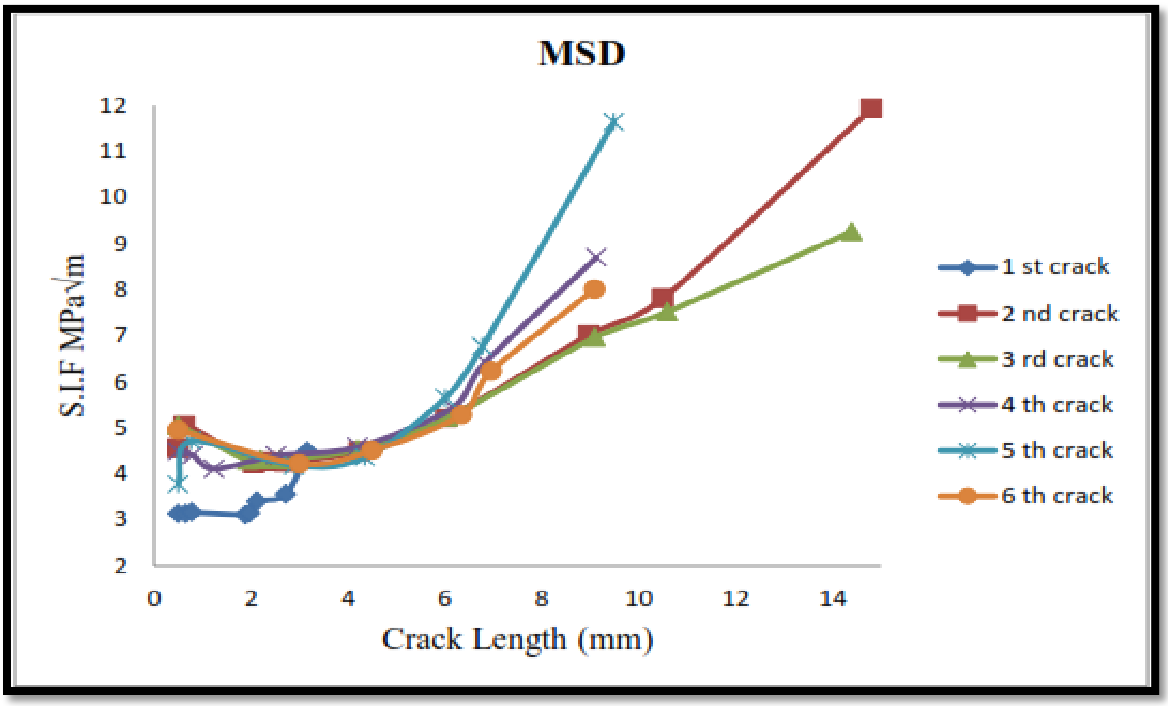

Fig. 7 All crack lengths vs. SIF

CONCLUSION

1. *In the present study, a longitudinal splice joint panel of the fuselage is analysed under tensile loads obtained from hoop stresses developed in the fuselage for internal pressurization loads. The stress analysis of the riveted section of the fuselage splice joint is carried out and the uniform stress distribution was observed at all the rivet holes indicating the possibilities of Multisite Damage with initiation of crack. Crack is introduced in splice joint panel proceeded by stress analysis. Change in stress distribution can be observed and next maximum stress is found out, this procedure is repeated until crack is initiated at all rivet locations. For the different half crack lengths at the rivet holes, stress intensity factor was estimated using MVCCI method. After all the cracks initiated, the splice*

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panel is further checked for final failure, for this net section failure $K_I > K_{IC}$ criteria is used.

2. *It is observed that SIF does not exceeded the fracture toughness of the material but plastic collapse takes place at the crack length of 16mm that is the net section average yield stress between the cracks was 347.91N/mm^2, whereas yield strength of the component was 345N/mm^{2}and SIF at the tip was 11.922N/mm$^2\sqrt{m}$, whereas the fracture toughness for 2mm material is 80N/mm$^2\sqrt{m}$ the yield strength of the material. The section between the two advancing crack tip failure was due to the net section yielding (plastic collapse) of the material. Fatigue life at the progressive crack lengths at the rivet holes is estimated and the final failure was estimated 3095582 to be cycles.*

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