ISSN: 2319-8753

International Journal of Innovative Research in Science, Engineering and Technology
*Vol. 2, Issue 7, July 2013*

# Analysis of the Fuselage Structure for Multi Site Damage

Karthik N[1], Dr. C Anil Kumar[2]

PG student, Department of Mechanical Engineering, KSIT, Bangalore, Karnataka, India[1]

Dean, Department of Mechanical Engineering, KSIT, Bangalore, Karnataka, India[2]

**ABSTRACT:** The current study includes a panel which represents the fuselage splice joint. The fuselage splice joint is a location where it experiences the uniform stress field at many rivet locations in a row. The probability of fatigue cracks initiation at many rivet locations simultaneously is more at splice joint. This paper has relevance in the structural integrity evaluation of aging transport aircrafts structural segment due to multisite damage.Finite element analysis of the Fuselage segment will be carried out to obtain the stress distribution near the joint. Fatigue cracks will emanate from the rivet holes simultaneously as they experience identical stresses due to internal pressure. In service the cracks in the fuselage will grow due to pressurization loading cycle (the difference in internal pressure and the atmospheric pressure at various altitudes). This study reveals the failure mechanics of the net section between the two advancing crack tips.There are two competing mechanism of failure; Failure due to fracture and Failure due to net section yielding (plastic collapse). The mode of failure will depend on which of the above two occurs at a lower load. The stress intensity factor calculations are carried out by using Modified Virtual Crack Closure Integral (MVCCI) method. The stress analysis is done using Nastran & Patran. The results were in compatible.

**Key words:** Fatigue, aircraft, Fuselage, Net section, Stress intensity factor, Finite element analysis

## I. INTRODUCTION

The Boeing 737 of Aloha Airlines has drawn much attention. At an altitude of 7300 meters the aircraft lost a large part of the fuselage skin. It is a wonder that the aircraft could still continue flying to an airport. The failure was caused by a large number of cracks started at many rivet holes in the same lap joint, a phenomenon which is now generally labeled as multiple-site damage (MSD) [1], This MSD phenomenon mode of failure inspired me for this work.

Aircraft is a complex engineering structure. The safety of the aircraft structure is the paramount important issue to be addressed by the designer. Multi site damage (MSD) is one of the important aspects to be studied to ensure the safety of the aircraft structure. Riveted joints are common feature in the built up airframe structure. The fatigue cracks will initiate from the locations of maximum tensile stress. The rivet-hole locations are one of the stress concentration regions. Therefore rivet locations are the most probable locations for fatigue crack initiation.

## II. CONSTRUCTION AND FORMULATION

A fuselage section (Fig. 1-a) shown below, Frame (Fig. 1-b), Stringer (Fig. 1-c) and Doubler or splice Plate (Fig. 1-d) [5], [15]. The three dimensional model is made using CATIA from the two dimensional drawings.

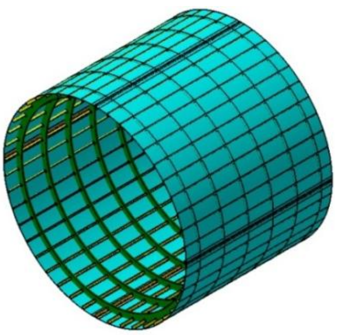

Fig. 1-a Fuselage

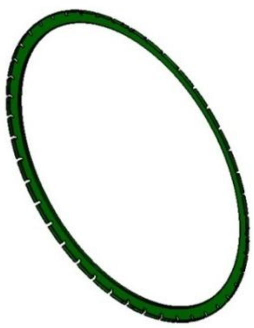

Fig. 1-b Frame

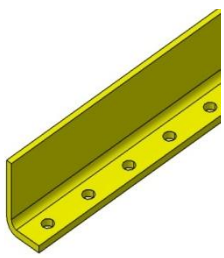

Fig. 1-c Stringer

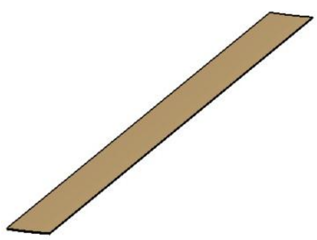

Fig. 1-d Doubler plate

ISSN: 2319-8753

International Journal of Innovative Research in Science, Engineering and Technology
*Vol. 2, Issue 7, July 2013*

Table 1: Fuselage Structure Dimensions

| Descriptions | Specifications |
|---|---|
| Radius of fuselage | 1250 mm |
| Length of fuselage | 3000 mm |
| Radius of doubler plate | 1246 mm |
| Length of the doubler plate | 3000 mm |
| Pitch of Rivets | 25mm |
| Thickness of fuselage | 1.2mm |
| Thickness of doubler plate | 2mm |
| Rivets diameter | 5 mm |

Table 2: Material Properties [15]

| MATERIAL | Young's Modulus (GPa) | Yield Strength (MPa) | Ultimate Strength (MPa) | SIF ($MPa\text{-}m^{1/2}$) |
|---|---|---|---|---|
| Al 2024 T351 | 73 | 280 | 470 | 26 S-L Direction<br>32 T-L Direction<br>37 L-T Direction |

**Load Case-Aircraft Cabin Pressurization:** Aircraft are flown at high altitudes for two reasons. First, an aircraft flown at high altitude consumes less fuel for a given airspeed than it does for the same speed at a lower altitude because the aircraft is more efficient at a high altitude. Second, bad weather and turbulence may be avoided by flying in relatively smooth air above the storms.

Considering the average flying altitude as 30,000 feet (Max) where the Atmospheric pressure will be around 4.4 psi and atmospheric pressure at sea level will be around 14.7 psi. The pressure variation on the fuselage ranges from 14.7 psi to 4.4 psi, i.e. in an average the differential pressure on the fuselage cabin is taken as 10 psi.

## III. ANALYSIS OF LONGITUDINAL SPLICE JOINT IN THE FUSELAGE SEGMENT

**A. FEM modeling of Fuselage Segment:** After completing the geometric modeling of the fuselage segment the doubler plate is meshed first and these elements are transformed to the fuselage then using tria element connectivity is given for entire model. Finally tria and quad elements are checked and duplicate elements are also checked. Normal's of the elements are made unidirectional. Now it is ready for applying material and elemental properties. The fuselage model is assigned with all degrees of freedom for shell elements. It implies that shell elements are completely fixed. The Beam elements are arrested for translation in z direction (Fig. 2). The fuselage is loaded with internal pressure of 10 psi.

**B. Post Processing:** By using the marker plot randomly 10 elements are selected and hoop stress is observed shown in Fig. 2. When compared with the theoretical values its matching, the theoretical values come around 7.29 $kg/mm^2$.

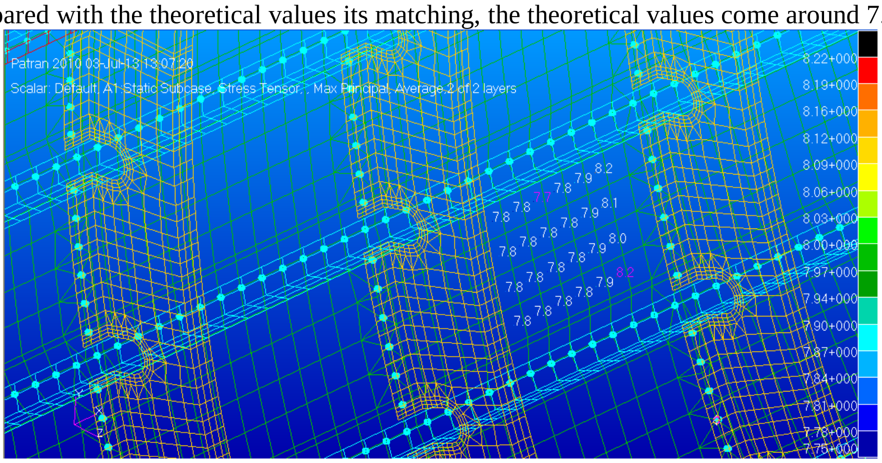


Fig. 2 Hoops Stress Distribution

**C. Local Analysis of the Fuselage Panel:** Here the fuselage skin is treated as a panel and rivets are made using respective rivet diameter instead of 1D beam element. The rivets are assigned multi point constraints and analysis is

ISSN: 2319-8753

International Journal of Innovative Research in Science, Engineering and Technology
*Vol. 2, Issue 7, July 2013*

carried out to find out the maximum elemental stress and exact location of crack initiation. The geometric model is constructed as panel represents fuselage skin with dimensions 800 mm width, 250 mm height and 1.2 mm thickness. The splice joint dimensions as 100 mm width, 250 mm height and 2 mm thickness. The boundary of the fem model is depicted in Fig. 3.

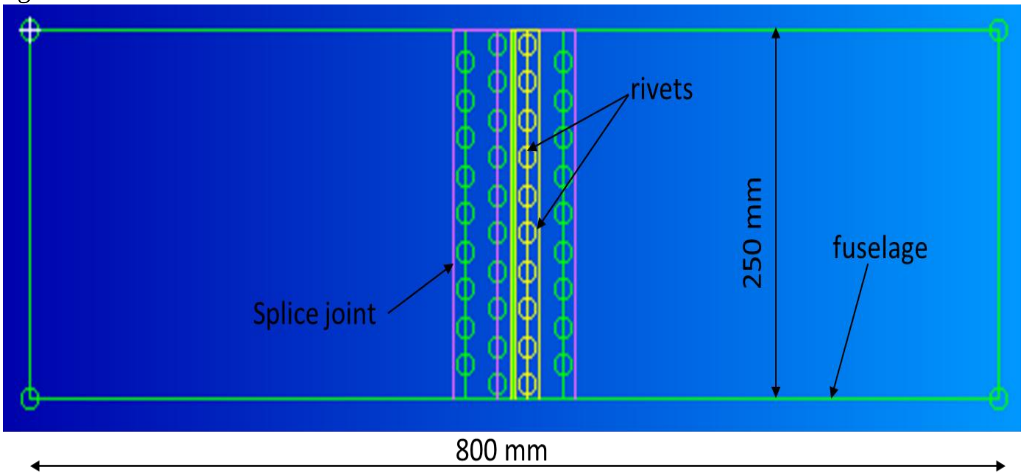


Fig. 3 Specification of Fuselage Panel for local Analysis

**D. Finite Element Modeling of Fuselage Panel:** The finite element model consists of 2D shell elements and Multi Point Constraints (MPC). The splice joint is meshed first then transform the corresponding elements to the fuselage skin, afterwards using tria elements meshing is completed and connectivity is maintained shown in Fig. 4.

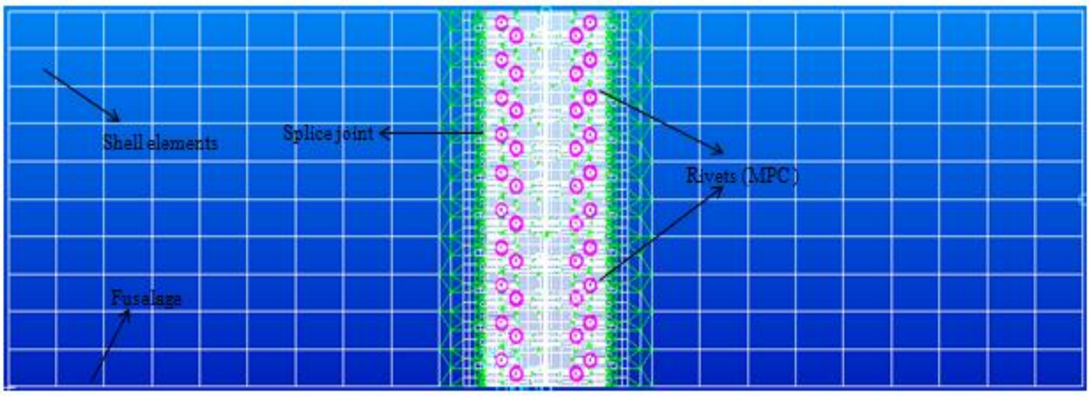


Fig. 4 Meshing of Fuselage Panel Along With MPC

**E. Load and Boundary conditions for Fuselage Panel:** The material and element properties are same for the FEM model. For local analysis the load acting on the fuselage skin is calculated using hoop stress. The hoop stress is equated with tensile stress.

$\sigma_{hoop} = \sigma_{tensile} = \frac{P}{A}$ P = Load on the fuselage (kg) and A = Area of the Plate ($mm^2$).

$P = \sigma_{hoop} \text{ x } A$

P = 7.29 x 250 x 1.2

P = 2187 kg

The single load can be converted to distributed load by equation

$UDL = \frac{P}{L}$

UDL = 8.75 kg/mm

ISSN: 2319-8753

International Journal of Innovative Research in Science, Engineering and Technology
*Vol. 2, Issue 7, July 2013*

The 8.75 kg/mm uniformly distributed load is applied .To avoid the bending of plate we are going to arrest translational motion in z direction. Now the FEM model is ready for analysis.

**F. Review of results for Fuselage Segment:** The results are requested for grid point force balance while solving the model. After solving maximum elemental stress is noted at the rivet location and reaction forces are observed at the rivet locations in loading direction. The reaction forces are shown in Fig. 5. It is clear that the reaction forces are equal and opposite in direction and the sum of all the reaction forces will give the value of applied load.

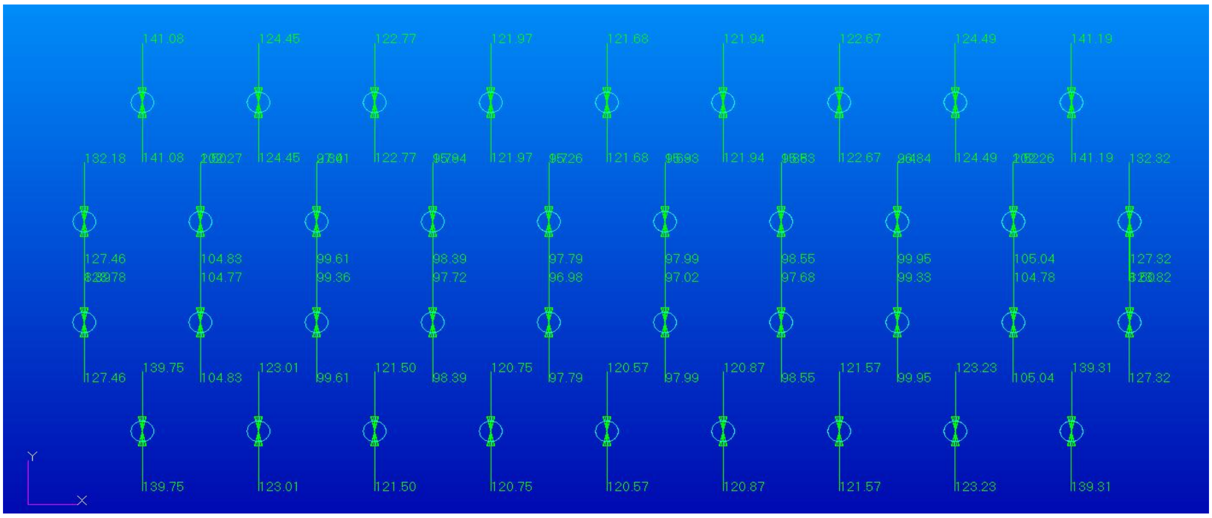


Fig. 5 Reaction Forces at Rivet Location in Loading Direction

One can clearly notice that there is a chance for emerging of multiple site cracks in the fuselage structure because everywhere in the rivet location identical stresses are observed. This indicates that when the structure is experiencing the fluctuating loads because of the pressurization cycles fatigue cracks will get initiated from multiple locations in a row of rivets. The crack must initiate from the outer row of rivets because stresses are higher in the outer row of rivets than inner row of rivets. Now the value of maximum elemental stress is used for shoot up the load in Local Analysis. The maximum elemental stress is found to be 30 kg/mm$^2$ and this result is reviewed in marker plot shown in Fig. 6.

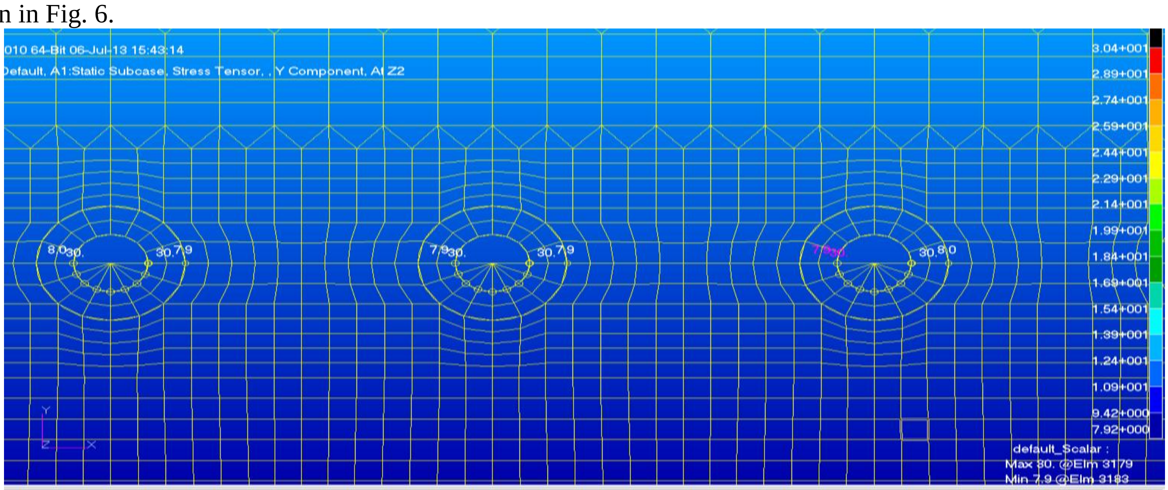


Fig. 6 Location of Maximum Elemental Stress

**G. Analysis of Fuselage Panel**: From the above analysis we are getting maximum elemental stress at rivet locations. The middle row rivets shows minimum forces, hence adjacent to the middle row rivets are selected for Local analysis. To get exact results only one side of the fuselage plates are taken for Local analysis and loading is been extracted from the global analysis and been applied. The same mesh will be used for further MSD analysis also. Now the modified finite element model for local analysis is shown in Fig. 7.

ISSN: 2319-8753

International Journal of Innovative Research in Science, Engineering and Technology
*Vol. 2, Issue 7, July 2013*

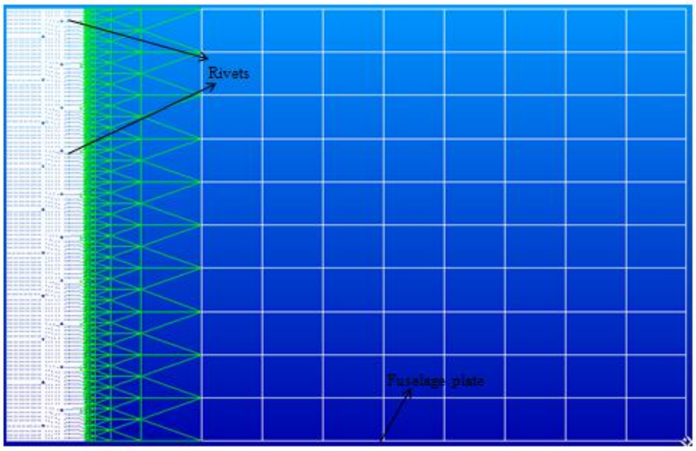


Fig. 7 Finite Element Model for Maximum Elemental Stress Analysis.

**H. Results and Discussions:** The Local analysis of fuselage segment discussed in previous section is analyzed with multiple cracks emanating at the rivet hole. In this section we are going to initiate multiple cracks at the rivet hole and evaluate the net section stress values between the crack tips keeping the maximum elemental stress constant at the rivet hole. The Fig. 8 shows that multiple cracks emanating from the rivet hole in a fuselage segment.

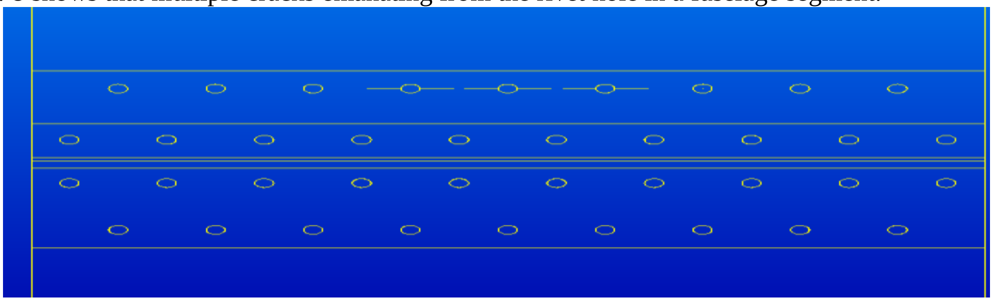

Fig. 8 Initiation of Multiple Site Cracks in Fuselage Segment.

**I. Tabulation of Results for Fuselage segment:** In this section we are tabulating the average elemental stress values between the crack tips and calculating the fracture toughness $K_I$ by MVCCI method against the crack length. Initially we start from multiple site crack length of 0.776 mm on both side of the rivet hole and continue the iterations with increase of .776 mm of crack length. At each iteration we are evaluating fracture toughness $K_I$ and average elemental stress values.

Table 3: Average Stress and Fracture Toughness against different Crack Length

| Iterations | Half of Crack length | SIF | Average stress |
|---|---|---|---|
| | mm | Mpa √m | Mpa |
| 1 | 0.776 | 10.56347 | 84.7 |
| 2 | 1.552 | 10.76634 | 95.5 |
| 3 | 2.328 | 11.02827 | 99.5 |
| 4 | 3.104 | 11.15577 | 102.0 |
| 5 | 3.880 | 12.1836 | 110.0 |
| 6 | 4.656 | 12.61341 | 121.0 |
| 7 | 5.432 | 13.19244 | 134.0 |
| 8 | 6.208 | 13.86495 | 152.0 |
| 9 | 6.984 | 14.7705 | 174.0 |
| 10 | 7.760 | 15.88561 | 211.0 |
| 11 | 8.536 | 17.50616 | 258.0 |
| 12 | 9.312 | 19.94303 | 351.0 |

ISSN: 2319-8753

International Journal of Innovative Research in Science, Engineering and Technology
*Vol. 2, Issue 7, July 2013*

## IV. CONCLUSION

Comparing the values from Table 2 and The Table 3, We can conclude the average stresses at the crack reaches the yield value therefore the mode of failure of fuselage panel is by Net section yielding criteria. The average elemental stress values exceed the yield strength of the material at the multiple site crack length iteration. Hence mode of fuselage failure is found is Net section yielding. The elemental stress at the crack tip will exceeds the yield strength of the material between 8.536 mm and 9.312 mm of crack length. Now comparing the values of Fracture toughness from Table 2 and from Table 3, the results reveals that the fuselage panel is not failed by Fracture criteria. The two validations of results depicts that the mode of failure of fuselage panel is by Net section yielding criteria.

## REFERENCES

[1] Jaap Schijve, Fatigue Damage in Aircraft Structures, not wanted, but Tolerated? Delft University of technology, faculty of aerospace engineering kluyverweg 1, 2629 hs, the Netherlands.
[2] J.Schijve, Multiple-Site Damage in Aircraft Fuselage Structures. Report LR-729 July 1993.
[3] Vitaly Pavelko, Julija Timoshtchenko, Model of the Multi-Site Fatigue Damage in the Thin- Walled Structure. Proceedings of the 5th International Conference RelStat'05.
[4] T.S.Ramamurthy, B.Dattaguru, Elangovan.R and V. Selladurai, Analytical determination of Residual Strength and linkup Strength for Curved Panels, with Multiple Site Damage. International Journal of Engineering Science and Technology (IJEST).
[5] Michael Chun-Yung Niu, Airframe Structural Design. California U.S.A march 1988.
[6] Stevan Maksimović, PhD, Fatigue Life Analysis of Aircraft Structural Components. Scientific-Technical Review, Vol.LV, No.1, 2005.
[7] R. Jones, L. Molent and S.Pitt, Study of Multi-Site Damage of Fuselage Lap Joints. Theoretical and applied Fracture Mechanics, Volume 32, Issue2, September-October 1999, Pages 81-100.
[8] L. Wang, W.T. Chow, H. Kawai and S.N. Atluri, Residual Strength of Aging Aircraft with Multiple Site Damage/Multiple Element Damage.AIAA Journal Vol. 36, No.5, may 1998.
[9] P. M. S. T. de Castro, S. M. O. Tavares, V. Richter-Trummer, P. F. P. de Matos, P. M. G. P. Moreira, L. F. M. da Silva, Damage Tolerance of Aircraft Panels. Vol 18, Pg 35-46, 2010.
[10] Andrzej LESKI, Numerical Calculation of the Aircraft Skin with Multi-Site Damage. Issue 25, pp. 37÷47, 2009 r.
[11] David Broek, The Practical Use of Fracture mechanics, Paperback Edition 90-247-0223-0.
[12] J. Schijve, Some elementary calculations on secondary bending in simple lap Joints.NLR TR 72036, Amsterdam, 1972.
[13] J. Schijve, Riveted lap joints with a staggered thickness in the overlap of the joint. Calculations of secondary bending. Faculty of Aerospace Engineering, Delft, Document B2 - 06 - 02, 2006.Vlot and J. Gunnink (Eds), Fibre Metal Laminates. An Introduction. Kluwer Academic Publishers.
[14] A. Vlot and J. Gunnink (Eds), Fibre Metal Laminates. An Introduction. Kluwer Academic Publishers
[15] E.F Bruhn, B.S, M.S.C.E, Dr. Engg, Analysis and design of flight Vehicle Structures. 3rd Edition, June 1973.