much distance transform of body from one position to another position it is different value along the length of structure. In this structure maximum deformation is at free end of structure and the magnitude of maximum deformation is mm.

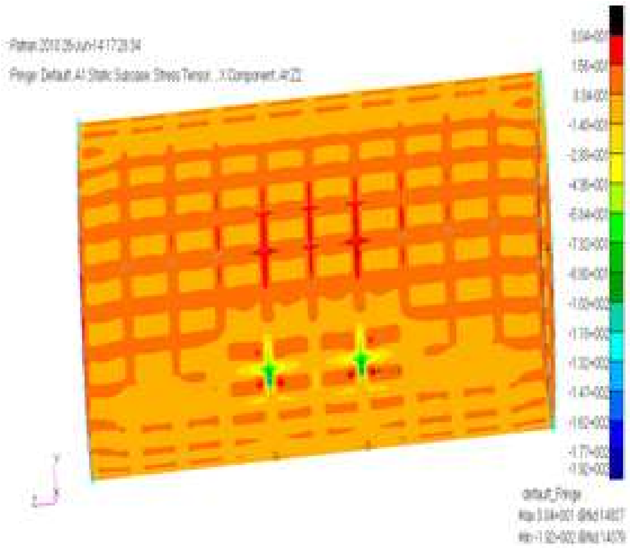

Figure 4.3: Stress Counter

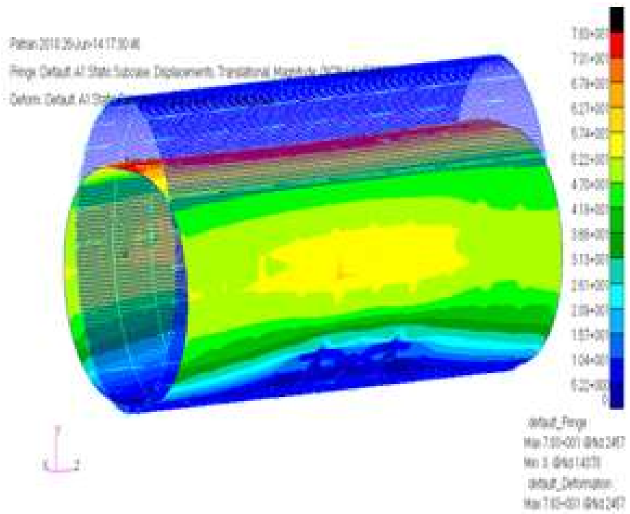

Figure 4.4: Deformation

5. Local Analysis of Stiffened Panel

5.1 Finite Element Model

The stiffened panel has taken from the fuselage structure where the maximum stress in the fuselage structure occurred from the global analysis. The stiffened panel taken from between two bulkheads and centre of two longenors. The longenors are connected to the skin of the fuselage with rivets. One end of stiffened panel structure nodes are fixed and another end apply the axial load for both skin and longenors. The structure is fixed at nodes of the skin and longenors are constrained in all six degree of freedom (three translations and three rotations), and also the stiffened panel constrained in z-direction.

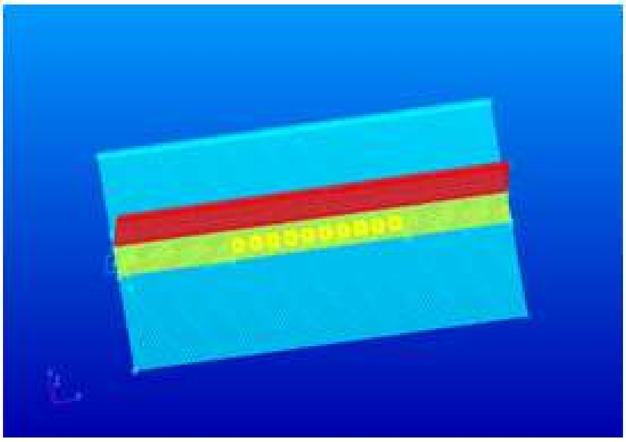

Figure 5.1: Finite Element Modelling of stiffened panel

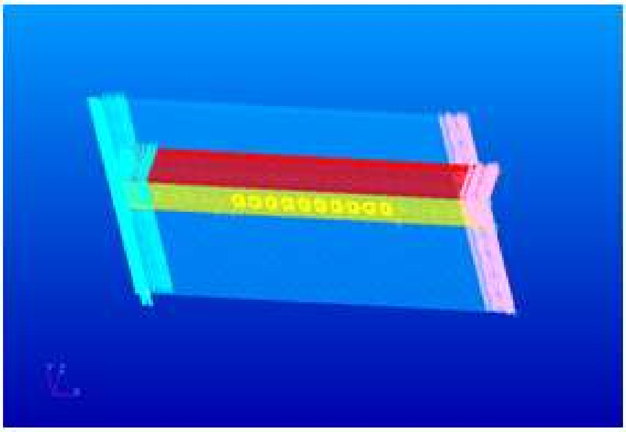

Figure 5.2: Load and boundary condition of stiffened panel

5.2 Stresses and Deformation

The maximum stress developed near the rivet holes of both skin and longenores and nominal stress all over the stiffened panel. The stress near the hole is three times of nominal stress. In figure the red colour shows the maximum stress. At the rivet hole the localization stresses because the area reduces and also stress concentration

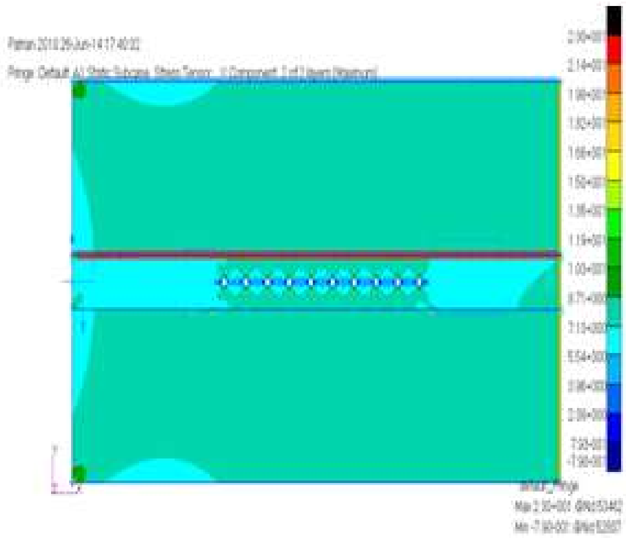

Figure 5.3: Stress distribution in stiffened panel

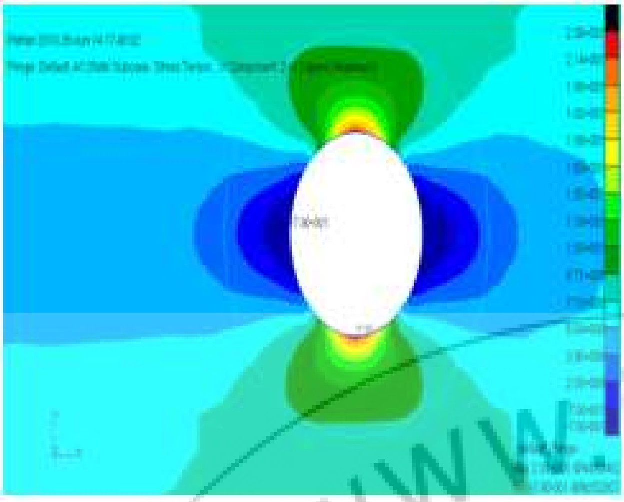

Figure 5.4: Close-up view Stress distribution in stiffened Panel

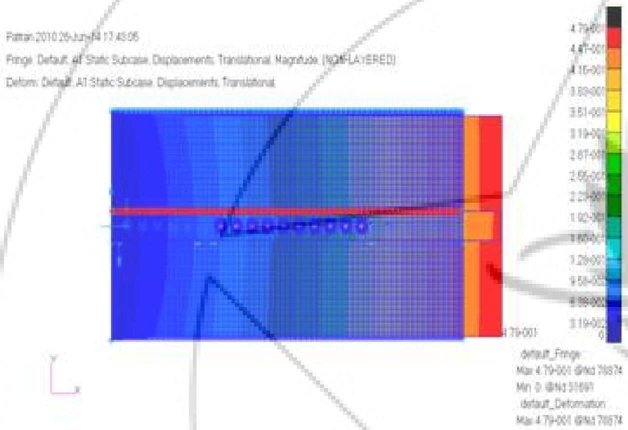

Figure 5.5: Deformation of stiffened panel

6. Results and Discussion

6.1 Calculation of Stress and deformation

1. Stress calculation:

Load on the skin = 3315.2 kg
Load on the longenores = 1036 kg
Cross section are = w × t mm2
Cross section area of skin = 224 × 2 = 448 mm2
Cross section are of longenores = (40×2) + (30×2) = 140 mm2
Total load on stiffened panel = 3315.2 + 1036 = 4351.2 kg
Total area of the stiffened panel = 448 + 140 = 588 mm2
Stress on the stiffened panel = $\frac{\text{Load}}{\text{Area}}$

$$\sigma = \frac{L}{A}$$

$$\sigma = \frac{4351.2}{588}$$

$$\sigma = 7.4 \text{ kg/mm2}$$

The nominal distributed over the stiffened panel is 7.4 kg/mm2 except near the rivet hole. At the rivet hole the stress is maximum and three times of the nominal stress is 20 kg/mm2.

6.2 Fatigue Life Estimation

6.2.1 S –N Curve

From typical constant life diagram for un-notched fatigue behaviour of 2024- T3 Aluminium alloy High-Master diagram is shown in below figure. The reference test condition R=0 used for obtain fatigue properties. For this condition smin=0 is called 'pulsating tension' under constant amplitude loading or Zero to tension loading. The numbers of cycles to failure from graph.

Table shows the alternating stress level below which the material has an infinite life. For most engineering purposes, infinite is taken to be 1 million cycles. According to Palmgren-miner's rule the stress amplitude is linearly proportional to the ratio of number of operation cycles to the number of cycles to failure from the graph gives the damage accumulated.

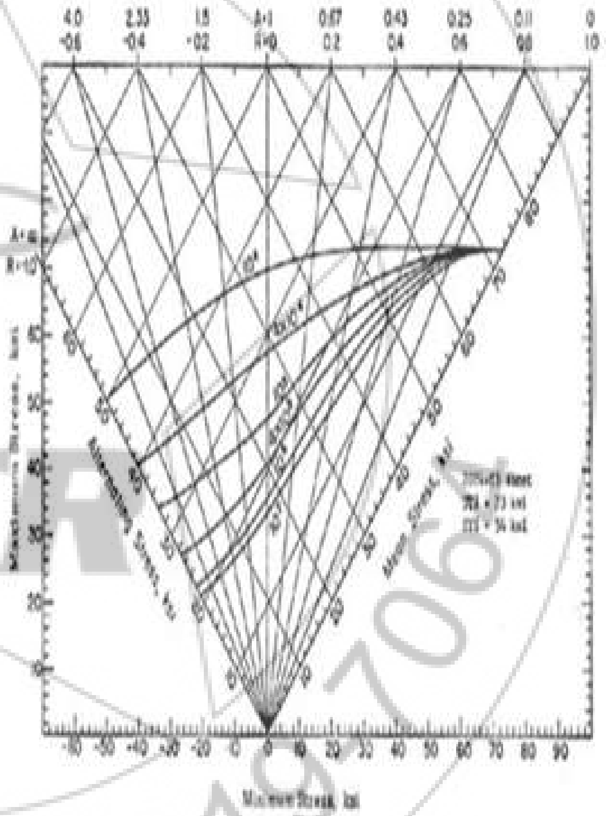

Figure 6.1: S-N curve

Table 6.1: Values of Damage Accumulated

Serial No.	Different G Condition	Alternative Stress (ksi)	Ratio R	No. cycle Induced Ni	No. cycles to failure Nf	Damage accumulated Di
1	0.5G to 0.75G	0.85	0.666	15000	10^7	0.0015
2	0.75G To 1G	0.8335	0.75	11000	10^7	0.0011
3	1G To 1.25G	0.8333	0.8	10000	10^7	0.001
4	1.25G To 1.5G	0.8335	0.833	8000	10^7	0.0008
5	0 To 1.75G	5.8335	0	20	10^7	0.000006
6	0 To 2G	6.667	0	1	10^7	0.0000001
7	-0.5G To 1.5G	6.665	-0.33	100	10^7	0.00001

The total damage accumulated is Di= 0.0044< 1 which is less than unity, therefore the crack not initiate

7. Conclusion

Linear static Stress analysis of the Fuselage structure was carried out and maximum stress was identified on the skin. Around the maximum stress location, we have taken one cut-out of the fuselage called stiffened panel. Local analysis of the stiffened panel was carried out by applying average tensile load on skin and maximum tensile load on longenore. The maximum stress found around rivet holes of both skin and longenore is 20kg/mm2.

Fatigue life estimated of the fuselage structure considering the maximum stress of the stiffened panel with the help of S-N curve and Miner's rule. The damage accumulated of the Fuselage structure is 0.0044 from this it is observed that, the remaining life of the structure is 0.9966. The fuselage structure lost 44121 fatigue cycle and remaining life is 9993406cycles.

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