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Design and analysis for damage tolerance of the pressurized fuselage cabin of a wide bodied transport aircraft

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Abstract
One of the fail-safe design features is the two-bay crack arrest capability of the airframe. In particular two-bay longitudinal and two-bay circumferential crack arrest feature is the main aspect of design for damage tolerance of the pressurized fuselage cabin. Under fuselage pressurization load cycles fatigue cracks develop at location of maximum tensile stress. There are locations on the airframe which are favorable for the initiation of longitudinal cracks and other locations for circumferential cracks.
This investigation identifies one such location from where a longitudinal crack can initiate and studies the fast fracture and crack arrest features under the action of uni-axial hoop stress. The main crack arresting features are the bulkheads and crack stopper straps.
A finite element modeling and analysis approach will be used for a realistic consideration of bulkheads and crack stopper straps and their role in the two-bay crack arrest capability of the aircraft. In particular through a stress analysis at a hoop stress corresponding to the design limit load, the load carrying ability of the bulkheads and the crack stopper straps will be assessed. For a realistic representation of two-bay cracking scenario it will be examined under what condition a two-bay crack can be arrested.

Keywords: Damage tolerance, circumferential crack, fracture, bulkhead, tear strap, Finite element analysis, fail-safe design.

1. Introduction

Aircraft are vehicles which are able to fly by being supported by the air, or in general, the atmosphere of a planet. An aircraft counters the force of gravity by using either static lift or by using the dynamic lift of an airfoil, or in a few cases the downward thrust from jet engines. An aircraft is a complex structure, but a very efficient man-made flying machine.
Aircrafts are generally built-up from the basic components of wings, fuselage, tail units and control surfaces. Each component has one or more specific functions and must be designed to ensure that it can carry out these functions safely. Any small failure of any of these components may lead to a catastrophic disaster causing huge destruction of lives and property. When designing an aircraft, it's all about ending the optimal proportion of the weight of the vehicle and payload. It needs to be strong and stiff enough to withstand the exceptional circumstances in which it has to operate. Durability is an important factor. Also, if a part fails, it doesn't necessarily result in failure of the whole aircraft. It is still possible for the aircraft to glide over to a safe landing place only if the aerodynamic shape is retained-structural integrity is achieved.
The basic functions of an aircraft's structure are to transmit and resist the applied loads; to provide an aerodynamic shape and to protect passengers, payload systems, etc., from the environmental conditions encountered in flight. These requirements, in most aircraft, result in thin shell structures where the outer surface or skin of the shell is usually supported by longitudinal stiffening members and transverse frames to enable it to resist bending, compressive and torsional loads without buckling. Such structures are known as semi-monocoque, while thin shells which rely entirely on their skins for their capacity to resist loads are referred to as monocoque.
The load-bearing members of these main sections, those subjected to major forces, are called the airframe. The airframe is what remains if all equipment and systems are stripped away. In most modern aircrafts, the skin plays an important role in carrying loads. Sheet metals can usually only support tension. But if the sheet is folded, it suddenly does have the ability to carry compressive loads. Stiffeners are used for that. A section of skin, combined with stiffeners, called stringers, is termed a thin-walled structure.
The airframe of an aircraft is its mechanical structure, which is typically considered to exclude the propulsion system. Airframe design is a field of engineering that combines aerodynamics, materials technology and manufacturing methods to achieve balances of performance, reliability and cost.

1.1 Major aircraft components

Fig 1: Airplane parts and its function

1.2 Fuselage

The main body structure is the fuselage to which all other components are attached. The fuselage contains the cockpit or flight deck, passenger compartment and cargo compartment. While wings produce most of the lift, the fuselage also produces a little lift. A bulky fuselage can also produce a lot of drag. For this reason, a fuselage is streamlined to decrease the drag. We usually think of a streamlined car as being sleek and compact - it does not present a bulky obstacle to the oncoming wind. A streamlined fuselage has the same attributes. It has a sharp or rounded nose with sleek, tapered body so that the air can flow smoothly around it.

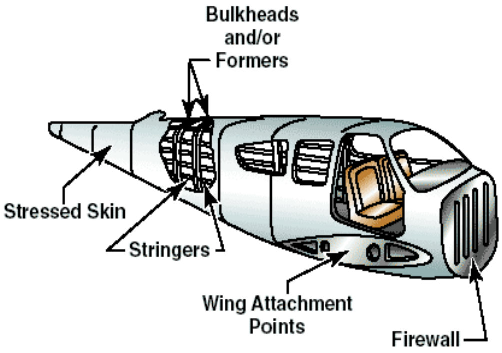

Fig 2: Fuselage

Unlike the wing, which is subjected to large distributed air loads, the fuselage is subjected to relatively small air loads. The primary loads on the fuselage include large concentrated forces from wing reactions, landing gear reactions and pay loads. For airplanes carrying passengers, the fuselage must also withstand internal pressures. Because of internal pressures, the fuselage often has an efficient circular cross-section. The fuselage structure is a semi-monocoque construction consisting of a thin shell stiffened by longitudinal axial elements (stringers and Longerons) supported by many traverse frames are rings (Bulkheads) along the length. The fuselage skin carries the shear stresses produced by torques and transverse forces. It also bears the hoop stresses produced by internal pressures. The stringers carry bending moments and axial forces. They also stabilize the thin fuselage skin.

Fuselage frames often take the form of a ring. They are used to maintain the shape of the fuselage and to shorten the span of the stringers between supports in order to increase the buckling strength of the stinger. The loads on the frames are usually small and self equilibrated. Consequently their constructions are light. To distribute large concentrated forces such as those from the wing structure, heavy bulkheads are needed. A transverse partition or a closed frame in a structure separating one portion from another is called a Bulkhead. Also used to designate solid, webbed or trussed members to dissipate concentrated loads into monocoque or semi-monocoque structure especially a fuselage. Members approximately parallel to the longitudinal axis of a beam or semi-monocoque structure are called longitudinal stiffeners. They are designed to stiffen the skin and assist in resisting shear and bending loads. A stiffener is a member used to reinforce thin sheets. Sometimes they are called stringers. Stringers are longitudinal members in the fuselage to support the skin and to hold the frames in position. It is used to carry direct load in the direction of its length. Longerons are main structural members of the fuselage. It is generally used when there is a big cut-out to be provided. Ex: cockpit.

2. Material Properties

The material considered for the structure is
Aluminum Alloy – 2024-T351, with the following properties.

- Young's Modulus, E = 70,000 N/mm^2
- Poison's Ratio, μ = 0.3
- Ultimate Tensile Strength, σ_u = 420 N/mm^2
- Yield Stress, σ_y = 350 N/mm^2

The following table shows the composition of the material considered.

Table 1: Showing composition of the material [18]

Composition	Wt. %	Composition	Wt. %
Al	90.7-94.7	Mn	0.3-0.9
Cr	max. 0.1	Si	max. 0.5
Cu	3.8-4.9	Ti	max. 0.15
Fe	max. 0.5	Zn	max. 0.25
Mg	5.2-5.8	Others	max. 0.15

3. Geometrical and Finite Element Modeling Of Stiffened Panel

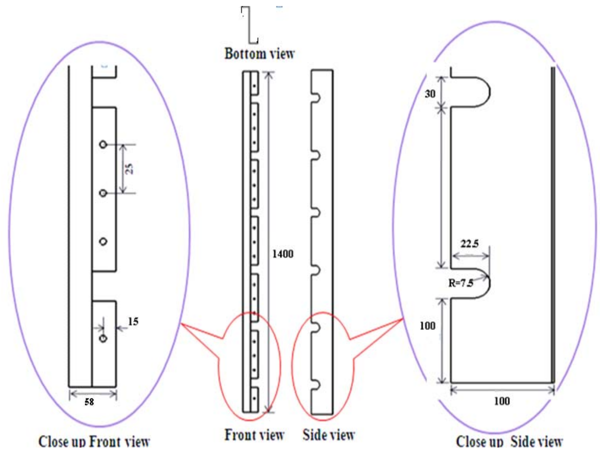

Fig 3: Front, side and bottom view of the Bulkhead with dimensions

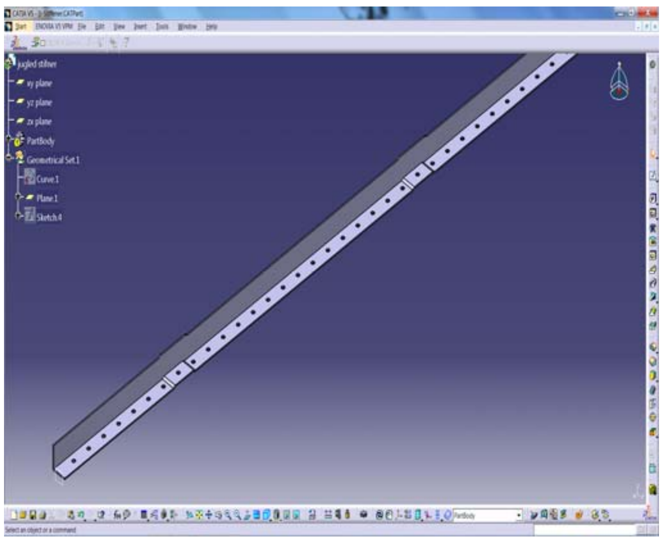

Fig 4: CAD Model of the Longeron

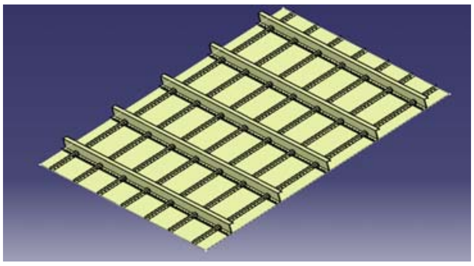

Fig 5: CAD Model of the stiffened panel

4. Results Obtained From the Finite Element Analysis of the Stiffened Panel

Pre-processing and post-processing is carried out by using MSC Patran software and Solved by using MSC Nastran (solver) software. The response of the stiffened panel in terms of displacements and stresses due to loads and boundary conditions described in the previous sections are explained in the following sections.

4.1 Displacement contour of the stiffened panel

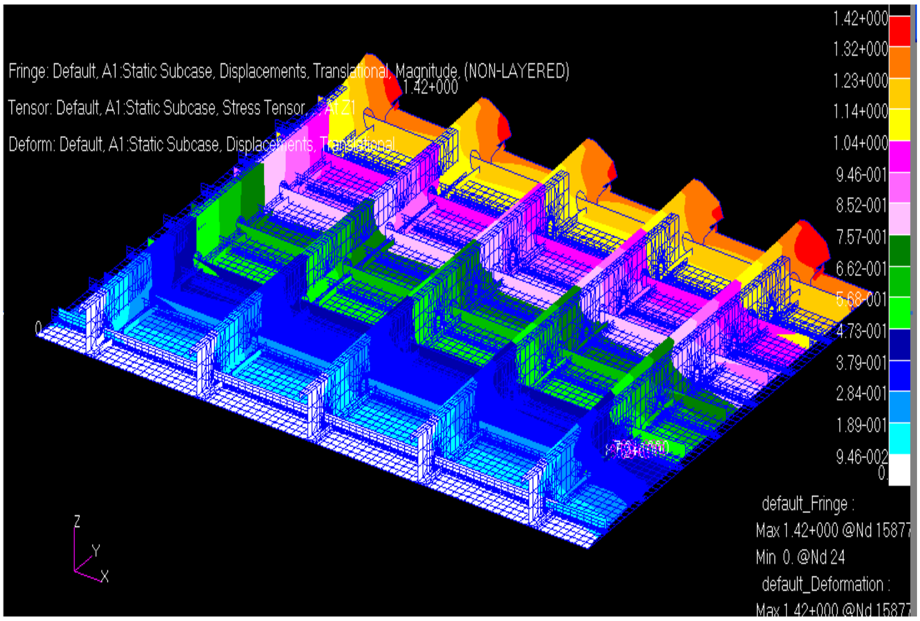

Fig 6: Displacement contour of the stiffened panel

The Fig. 6 shows the displacement contour of stiffened panel. Displacement contour increases from fixed end to loading end and it is shown by different colors fringes where white color showing minimum magnitude of displacement while red color showing maximum magnitude of displacement. The panel is constrained with all degrees of freedom at the bottom edge.

4.2Stress contour of the stiffened panel

4.2.1 Skin

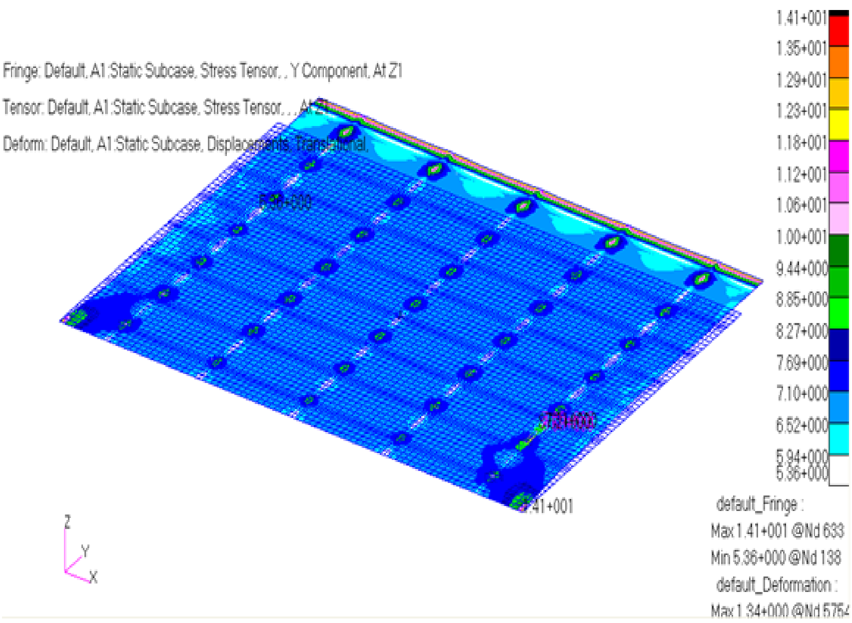

Fig 7: Stress contour for skin

Fig.7 shows the stress contour on the skin from global analysis results. It is clear that the maximum stress on skin is at the rivet location where the rivets are used to fasten the tear strap, bulkheads, longerons and skin. The magnitude of maximum tensile stress is 14.1kg/mm^2 in the loading direction can be observed from the fig.7. The maximum stress locations are the probable locations for crack initiation. Invariably these locations will be at rivet locations in the skin. Representation of layered structure is important in identifying critical stress locations, integral representation will miss lead as for as critical locations are concerned.

4.2.2 Tear strap

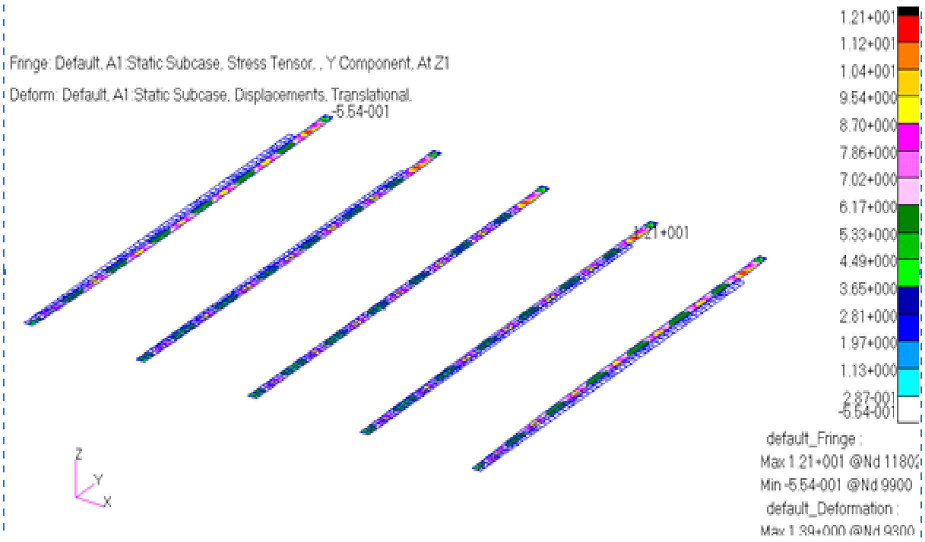

Fig 8: Stress counter for tear strap

Fig.8 shows the stress contour on the tear strap from global analysis results. It is clear that the maximum stress on tear strap is at the rivet location where the rivets are used to fasten tear strap, bulkhead and longeron on skin. The magnitude of maximum tensile stress is 12.1kg/mm^2 in the loading direction can be observed from the fig.8. The maximum stress will be at rivet locations in the tear strap. Invariably these maximum stress locations are the probable locations for crack initiation.

4.3.3 Bulkhead

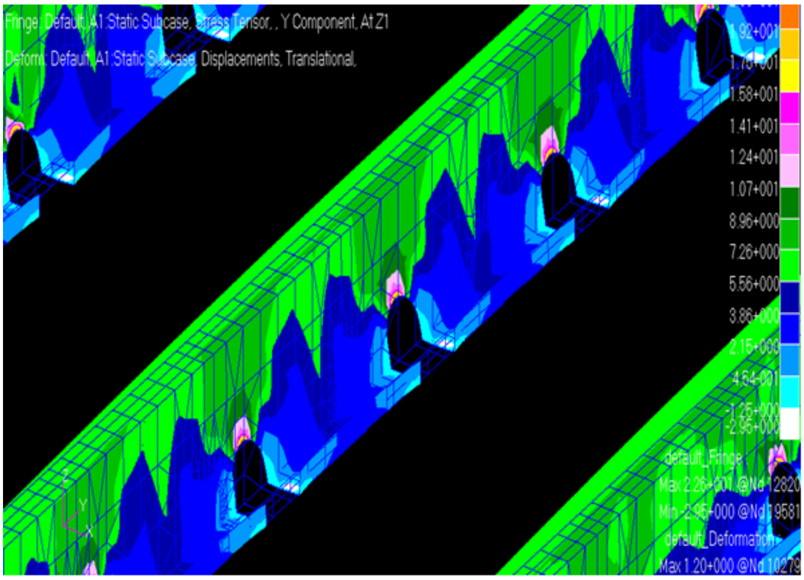

Fig 9: Stress counter for Bulkhead

Fig.9 shows the stress contour on the bulkhead from global analysis results. It is clear that the maximum stress on bulkhead is at stringer cut-out (mouse cut-out) and this maximum stress is uniform in all the stringer cut-outs. The magnitude of maximum tensile stress is 22.6 kg/mm^2 in the loading direction can be observed from the fig.9, which is more than the stresses in all other components of the stiffened panel. In the bulkhead the maximum stress will be at the stringer cut-out (mouse hole) which is shown in fig.9 and the maximum stress locations are the probable locations for crack initiation. Invariably these locations will be at stringer cut-out locations in the bulkhead.

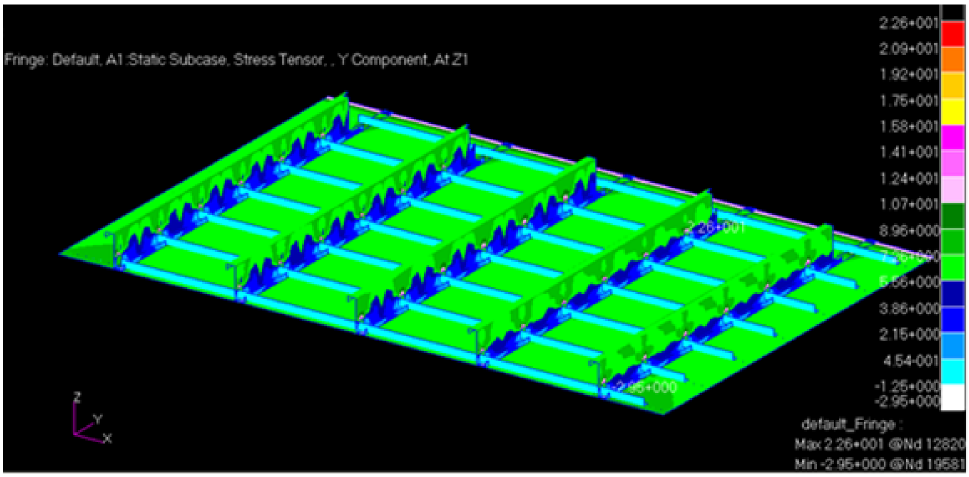

Fig 10: Stress counter stiffened panel without centre crack

Fig.10shows the stress contour on the stiffened panel from global analysis results. It is clear that the maximum tensile stress on stiffened panel is at stringer cut-out (mouse cut-out) and this maximum tensile stress is uniform in all the stringer cut-outs.

From the stress analysis of the stiffened panel it can be observed that a crack will get initiated from the maximum stress location. There are three structural elements at the rivet location near the high stress location. Crack will either get initiated from the bulkhead at stringer cut out or from the nearby rivet location from the rivet hole. This eventually will lead to the failure of the bulkhead in the perpendicular direction to loading. Once the bulkhead is broken, simultaneously cracks will appear on the tear strap and the skin.

5. Results and Discussion

For the evaluation of crack arrest capability of bulkheads in the stiffened panel with and without the presence of tear strap is studied. This crack present in the stiffened panel is assumed to be caused due to the discrete sources of damage. It may be due to bird hit, foreign object hit etc. The internal pressurization of the cabin is considered to be 9 psi.

5.1 Stress intensity factor plots

Table 2: Comparison of K_I FEA values of stiffened panel for given loads and boundary conditions

Crack length (a) in mm	K_I FEA without Tear strap (MPa√m)	K_I FEA with Tear strap (MPa√m)
100	**41.53**	**61.00**
200	50.80	51.54
300	58.46	58.68
400	65.74	65.28
500	71.43	70.67
600	76.77	75.54
700	80.28	77.57
800	76.15	69.96

Table.2 shows the SIF value for different crack length initially only for stiffened panel without tear strap and then considering stiffened panel with tear strap of 1.5mm thickness. Fracture toughness for aluminum alloy 2024 T3 is K_{Ic}=98.9MPa√m (Ref.18)

In the structure the maximum stress value obtained at the two bay bulkhead rivits region, which is 8.58kg/mm^2.This is the obtained value only for pressure load condition. but in the actual condition of an aircraft while it is operating at higher altitudes it will experience both pressure load and aerodynamic loads.

The maximum obtained stress=3*8.58=25.74 kg/mm^2

As the problem is defined for the pressure load the obtained value is not the design limit load. By using strength of material approach we can obtain the design limit load. The stress concentration considering the nominal stress value of 8.58 and SIF of three of one can get the maximum stress at rivets location. i.e.= 1.35

We know that aluminum maximum yield strength is 35 kg/mm^2. The structure is normally designed in such a way that the maximum stress developed at design limit load will be equal to the yield strength of the material. By using incremental ratio, which is ratio of aluminum yield strength to the obtained maximum stress.

i.e. So we can increase the stress to 1.35 times the originally obtained values. Those values are calculated and tabulated below.

Table 3: Comparison of K_I FEA values with tear strap and without tear strap of stiffened panel for actual loads and boundary conditions

Crack length (a) in mm	K_I FEA without Tear strap(MPa√m)	K_I FEA with Tear strap (MPa√m)
100	56.06	82.35
200	68.58	69.57
300	78.92	79.21
400	88.74	88.12
500	96.43	95.40
600	103.63	101.97
700	108.37	104.71
800	102.80	94.44

6. Conclusion

- Damage tolerance philosophy is widely used in the aircraft structural design to reduce the weight of the structure.
- A stiffened panel which is generic structural element of the fuselage structure is evaluated analytically for its crack arrest capability.

- The internal pressure is one of the main loads that the fuselage needs to hold. In the current project also pressurization load case is considered for the analysis.
- Finite element analysis (FEA) approach is used for structural analysis of the stiffened panel.
- Stress analysis is carried out to identify the maximum tensile stress location in the stiffened panel. The magnitude of maximum tensile stress in loading direction is 23.3 Kg/mm^2 (228.59 MPa) which is in the bulkhead at the stringer cut-out. The maximum stress locations are the probable locations for crack initiation. Invariably these locations will be at stringer cut-out locations in the bulkhead.
- There are other possibilities of crack initiation at different locations in the stiffened panel due to discrete source of damage. It may be due to bird hit, foreign object hit. For the analysis centre cracked stiffened panel with central broken bulkhead and tear strap is considered which is due to discrete source of damage.
- Modified virtual crack closure technique (VCCT) along with FEA analysis results are used for calculation of stress intensity factor (SIF).
- The effect of tear strap in arresting two-bay crack is studied.
- From the residual strength calculation it indicates that without Tear strap (crack stopper straps) a two bay crack will not get arrested in the stiffened panel.
- Tear straps (crack stopper straps) with thickness 1.5 mm shows that a two bay crack is arrested it the stiffened panel.
- These results were obtained for the rivet pitch of 25mm in the bulkheads by varying the pitch of the rivet may alter the crack arrest capability of the stiffened panel.

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