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Optimization of Composite Fuselage Frame Layup for Energy Absorption under Crash Loading

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ABSTRACT

Considerable research has been carried out on aircraft fuselage design, with a particular focus on frames due to their role in energy absorption during crash impacts. While metallic alloys have traditionally been used for frame construction, the growing popularity of composites has led to a shift towards their utilization. However, the research specifically targeting composites for fuselage frames is limited in scope. This study aims to bridge this gap by conducting a comprehensive analysis using LS-DYNA software. The composites investigated in this analysis include Standard Carbon Fiber, Graphite AS-3501-6 Fiber, E Glass Fiber, and Kevlar Fiber. Parameters such as deformation, energy absorption, maximum normal stresses, and shear stresses are compared against Aluminum Al 7075-T6, a commonly employed metallic alloy. Furthermore, an optimization process is performed, focusing on the laminate orientation of standard carbon fibre, to determine the most favourable orientation for each parameter studied. The results highlight that employing Standard Carbon Fiber, with laminates having 90° ply such as quasi-isotropic, cross-ply and unidirectional 90° laminates lead to superior outcomes in terms of energy absorption and deformation for the fuselage frame under crash loading.

1. Introduction

Aircraft are intricate machines that consist of a wide array of mechanical, electrical, and electronic components. Modern aircraft are equipped with advanced control systems to assist pilots in flying, but this does not guarantee complete safety. The National Transportation Safety Board (NTSB) documented approximately 1499 aviation accidents worldwide between January 2022 and November 2022, resulting in various levels of injuries and fatalities [1]. Mechanical failure is often cited as a leading cause of these incidents. The data from NTSB revealed that the most common phase of flight where accidents or crashes occurred was during the landing process [2].

Ensuring a successful emergency landing requires addressing several critical factors, including the type of landing (landing gears or belly landing) and the landing surface (firm ground or water) [3].

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These considerations are crucial during aircraft design, especially when designing the fuselage and its components. The fuselage, which serves as the main body of the aircraft, is responsible for accommodating the crew, passengers, cargo, and various aircraft systems, and it may also contain fuel. Given its vulnerability during a crash, special attention must be given to its design and strength to withstand impact.

Numerous studies and analyses have been carried out to investigate the ability of aircraft bodies, especially the fuselage, to absorb energy during crash landings [4]. NASA has been actively involved in crash dynamics research since the 1970s, conducting experiments at the Impact Dynamics Research Facility in Virginia [5,6]. One such experiment involved dropping three B707 transport fuselage sections from a height of 70 ft to evaluate data acquisition systems and gather data for model validation as shown in Figure 1.

Fig. 1. Drop Test of Fuselage [5]

The results of the crash analysis on the aircraft fuselage showed that the major deformation occurred at the bottom and sides of the fuselage, particularly in the frame and skin, while the passenger floor had minimal deformation. This suggests that these three components were the primary energy absorbers during impact. However, the experimental method used in the study had some limitations. It did not provide specific values to determine the most efficient energy-absorbing component. Moreover, physical crash dynamic analysis is expensive and requires significant resources and space. As a result, research in this field has shifted towards numerical analysis rather than physical experimentation.

In a recent study on crash analysis of a B737 fuselage during belly landing, computer simulations were conducted [7,8]. Different impact velocities and surfaces were considered, comparing rigid ground and water landings. The analysis revealed that the deformation in the fuselage increased with higher impact velocities, and the frame was the most significant energy absorber during the early stages of impact, followed by the skin, passenger floor, strut, and stringer [9-11]. Therefore, future work in this area are likely to focus on studying the impacts of loads on frames while neglecting other components of the fuselage.

Frames in an aircraft fuselage serve multiple functions, classified into three categories: geometry, loads distributor, and support [12]. Geometrically, frames retain the circular form of the fuselage against compressive stresses and help reduce structural instability. They also provide support functions, such as load distribution, fail safety, and prevent skin crack propagation. Frames also transmit aircraft loads to other components, transferring shear stresses and responding to pressurization loads. Various cross-section shapes are used for aircraft fuselage frames, including I, J, C, and Z shapes [13]. Among these, the Z-frame is commonly used in the fuselage frame. The dimensions of frames vary along the fuselage length, with those closer to the center of the fuselage being larger than those near the tail. Different loads, such as tension, compression, bending, shear, and torsion, act on the fuselage, and frames play a crucial role in supporting the fuselage against these loads [14]. Bending and shear stress are the primary stresses acting directly on the fuselage frames.

The selection of materials for fuselage frame has a significant impact on various aspects, such as structural stiffness, payload capacity, safety, reliability, cost, recyclability, and energy consumption. Several factors must be considered while choosing materials, including high strength-to-weight ratio, stiffness, fatigue endurance, low density, damage tolerance, thermal stability, and corrosion resistance [15]. Aluminum alloys, titanium alloys, steel alloys, and fiber-polymer composites are the four primary types of materials commonly used in manufacturing fuselage frames [16]. Aluminum alloys are the most widely used due to their reasonable cost, light weight, high stiffness, strength, and ease of manufacturing. Titanium alloys offer superior properties compared to aluminum, such as high strength, toughness, and corrosion resistance, but they are less commonly used due to economic factors [17]. Steel alloys were used in the past but are now less favored than aluminum due to their heavy weight. Composites, particularly carbon fiber reinforced polymer (CFRP) and glass fiber reinforced polymer (GFRP), are gaining popularity in aircraft manufacturing due to their high specific strength and stiffness [18].

CFRP and GFRP are the two most commonly used polymer matrix composites in fuselage frames. CFRP offers a high strength-to-weight ratio and stiffness but is more expensive than GFRP. GFRP provides good strength and resistance to corrosion and is relatively less expensive than CFRP [19,20]. The main reason for the limited use of composites in aircraft structures, including fuselage frames, is their higher cost compared to traditional materials like aluminum. However, composites offer significant advantages in terms of weight reduction and performance [21]. Material selection for fuselage frames is critical to ensure that the frames can support the loads applied to them, transfer loads to other aircraft components, and maintain structural integrity and safety.

Kalanchiam and Chinnasamy [22] conducted a comparative study using Finite Element Analysis (FEA) software, analyzing metallic and composite airframe structures. The results indicated that metallic frames, specifically aluminum alloy, had a higher margin of safety in terms of maximum allowable loads compared to CFRP frames. Similarly, in another analysis, Dandekar [12] specifically focused on composite fuselage frames and compared them with metallic frames Al 7075-T6. Four types of composite frames, including carbon unidirectional tape, carbon fabric, E-glass unidirectional tape, and E-glass fabric, were analyzed. The study revealed that carbon unidirectional tape exhibited the highest deceleration and energy absorption capacity. Other composite materials showed similar results to the aluminum alloy. Carbon fabric offered the least stress and provided the highest safety factor. A detailed experimental study [23,24] was performed to evaluate the structural integrity of the airframe structure under impact. An overview of current research and future development of crashworthiness of aircraft fuselage structures has been recently presented by Mou *et al.*, [25,26].

Based on the review, it can be noted that the fuselage frame is considered as the crucial component in absorbing energy during crash-landing impact. Developing effective fuselage frames requires careful consideration of geometry, size, and, most importantly, frame materials. While aluminum alloy has been widely used in frame construction for years, the introduction of advanced composites with superior qualities has brought new perspectives. Research has been conducted to explore the use of composites in aircraft structures, but many studies rely on expensive experimental methods, where actual composite frames are fabricated and tested. However, it should be noted that these works only considered CFRP and GFRP while a broader comparison with other composite materials is necessary for a comprehensive evaluation. To address this, the current study proposes numerical dynamic loading analysis of composite frames and subsequent optimizations of laminate layup to explore their performance of frame on energy absorption and deformation.

2. Methodology

2.1 Modelling the Frame

To validate the analysis employed in this work, a comparative study is conducted with a previous research work by Dandekar [12]. The research methodologies and analysis steps followed in this work closely align with [12], allowing for a meaningful comparison. The design of the 3D model of the fuselage frames mirrors the approach used by Dandekar [12], employing ANSYS DesignModeler software. The dimensions and geometry of the frames are detailed in Figure 2, along with the corresponding values presented in Table 1. These dimensions are vital for constructing the Z cross-section of the frames which is one of the most used shapes for fuselage frame cross section. Later this design is revolved to create a circular shape using the "Revolve" feature where the radius of the circular fuselage frame is L10 = 1879.6 mm. The resulting 3D model is depicted in Figure 3 and Figure 4, with axes orientation specified for subsequent analyses where the x-axis of the system is perpendicular to the front of the frame while, z-axis is upward, and y-axis is the side of the frame. These directions are important for analysis later.

Table 1
Dimension of the Z Cross Section Fuselage Frame [12]

Parameter	Value (mm)
L1	0.9144
L2	0.4572
L3	0.9144
L4	0.9144
L5	7.23
L6	0.9144
L7	90.00
L8	43.40
L9	23.35
L10	1879.6

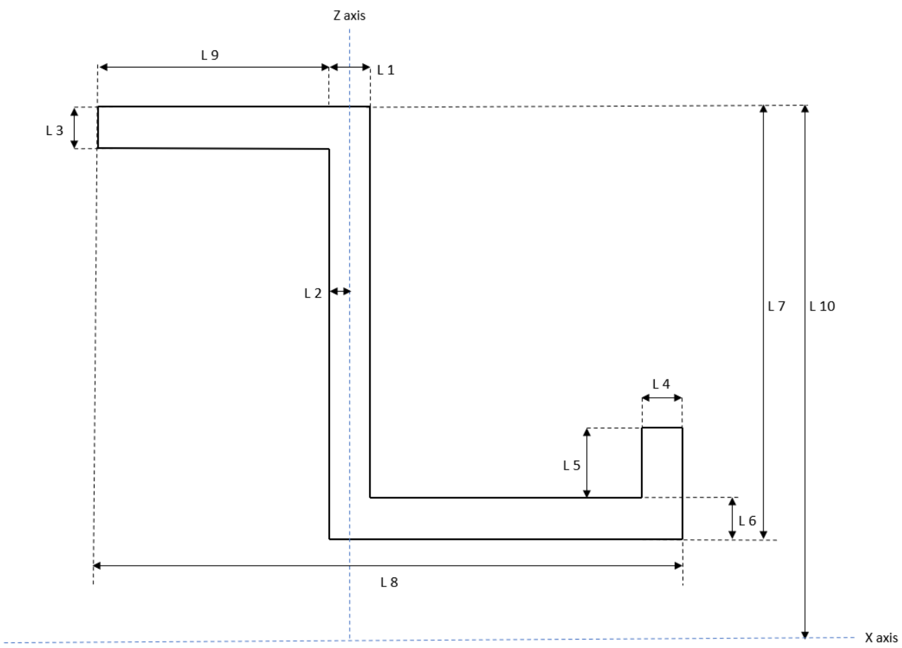

Fig. 2. Dimension of the Fuselage Frame [12]

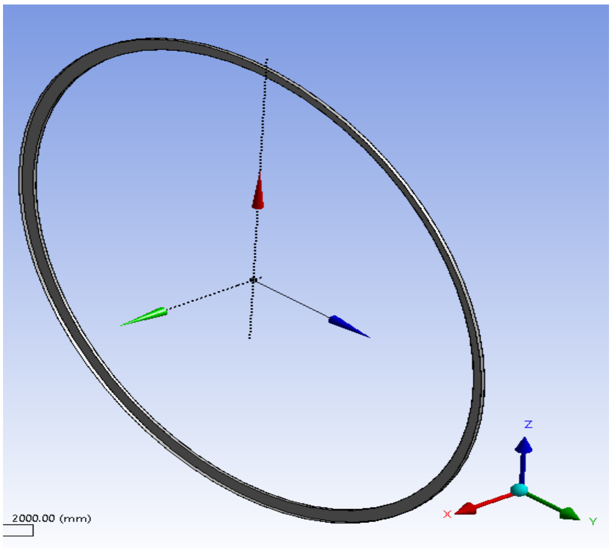

Fig. 3. 3D Model of the Fuselage Frame

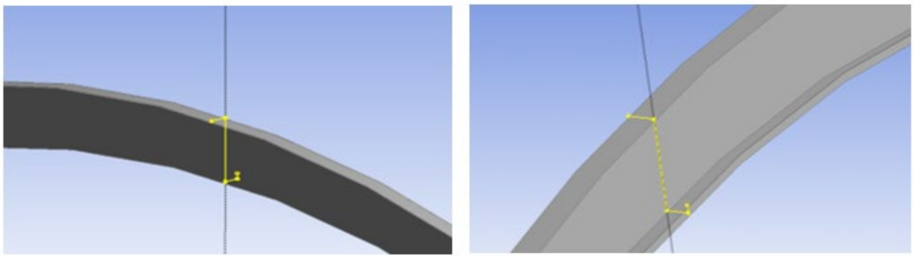
Fig. 4. Z cross section view from 3D Model of Fuselage Frame

2.2 Validation
2.2.1 Analysis of metallic fuselage frame

In this step, the model designed using ANSYS DesignModeler software will be converted into an IGES file for analysis in LS-DYNA software. The analysis will involve dynamic loading, where the model is released from a specific altitude with a given velocity to observe the impact response. The meshing of the shell model is done by using the "Auto Mesher" feature in LS-DYNA. The details of meshing are provided in Table 2.

Table 2
Details of Meshing

Parameter	Value
Element Size (mm)	20
Number of Elements	4037
Number of Nodes	4598

The materials for the model will be set as Aluminum Alloy Al 7075-T6, and the section part will define the shell model with the desired thickness. Boundary conditions, including a rigid wall and initial velocity, will be set up for the analysis. The rigid wall will act as the crushing plate for the frame. The initial velocity will be in the negative z-direction with a magnitude of 9.144 m/s (30 ft/s) as referred to others similar type of study [23,24]. All the details regarding material properties and boundary conditions are presented in Table 3 and Table 4.

Table 3
Properties of Materials

Parameter	Value
Materials	Aluminum Alloy Al 7075-T6
Materials Card	MAT 024 - Piecewise Linear Plasticity
Mass Density, ρ (kg/mm^3)	2.810×10^{-6}
Young Modulus, E (GPa)	71.699997
Poisson Ratio	0.3300000
Yield Stress (GPa)	0.5194800

Table 4
Sections Card and Boundary Conditions

Parameter	Value
Element Formulation Option, ELFORM	16 - Fully integrated shell element
Shear Factor, SHRF	0.833
Thickness (mm)	0.9144
Impact Surfaces	Rigidwall
Termination Time (ms)	100
Time interval (ms)	5
Initial Velocity, (m/s)	-9.144

The analysis will be executed by setting up the "database" and "control" parts in LS-DYNA software. The results obtained from the metallic analysis are shown in Figure 5 until Figure 7 for time analysis of 0.035s, 0.050s, and 0.075s respectively and then compared with existing literature by Dandekar [12] to validate the procedure adopted in this work as shown in Table 5.

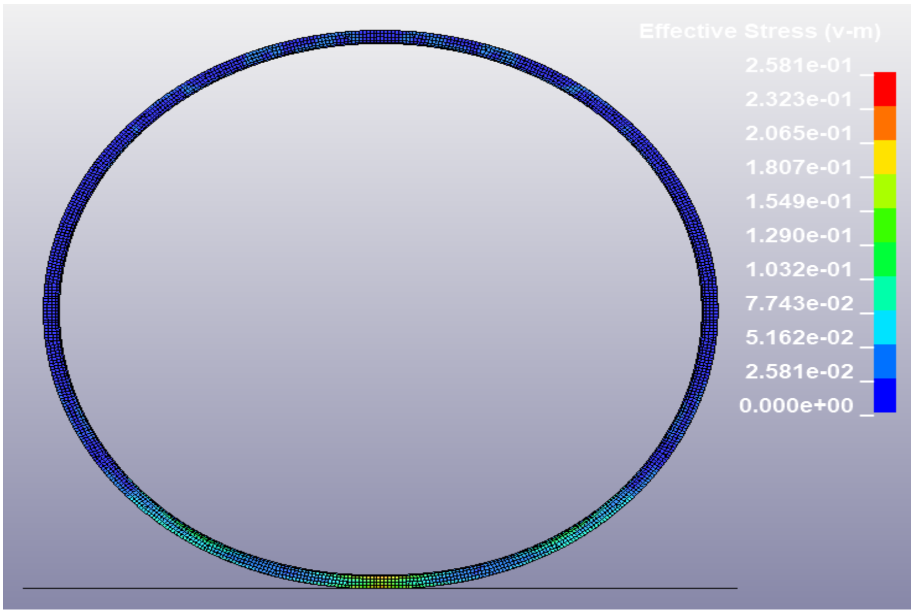

Fig. 5. Von-Mises Stress at time 0.035s

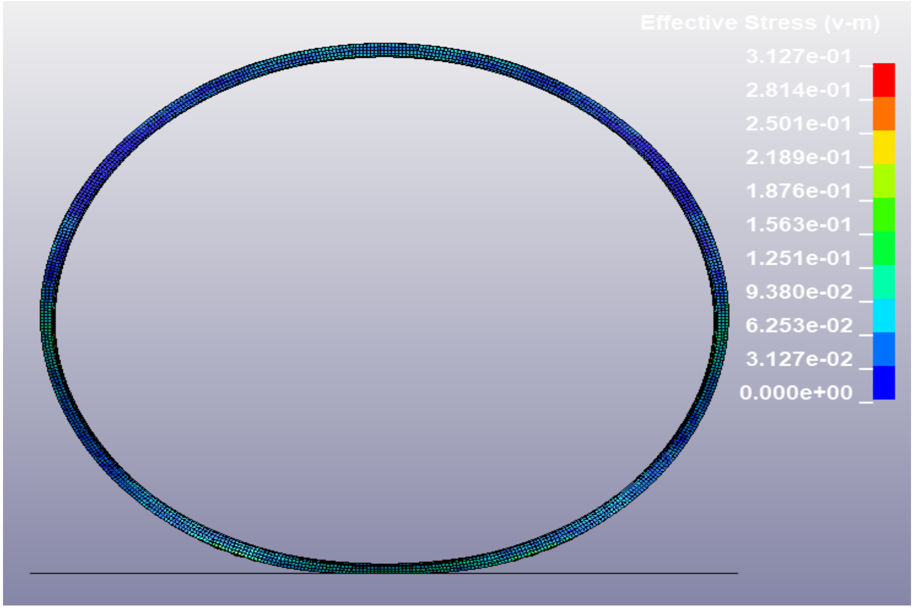

Fig. 6. Von-Mises Stress at time 0.050s

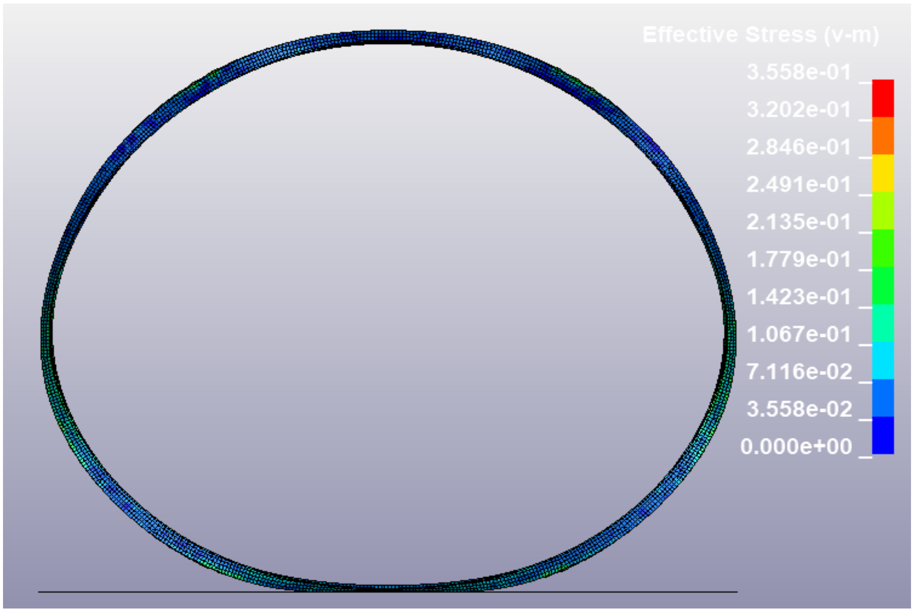

Fig. 7. Von-Mises Stress at time 0.075s

Table 5
Comparison of Metallic Fuselage Frame

Parameter		Simulation	Dandekar	Percentage Error
Maximum von-Mises	t = 0.035s	2.581×10^8	2.863×10^8	9.85%
stress (Pa)	t = 0.050s	3.127×10^8	3.368×10^8	7.15%
	t = 0.075s	3.558×10^8	3.691×10^8	3.61%
Maximum Velocity (m/s)		16.71	15.635	6.88%

The analysis of a metallic fuselage frame yielded promising results. The simulation exhibited a small percentage error of 6.88% compared to references for the model's maximum velocity. Similarly, for the maximum von-Mises stress at various time intervals (t = 0.035s, 0.05s, and 0.075s), the differences were relatively low: 9.85%, 7.15%, and 3.61% respectively. However, given that most results had less than 10% percentage error, it's reasonable to conclude that the analysis was accurately and appropriately conducted. Consequently, this method holds promise for analyzing composite fuselage frames as a next step.

2.2.2 Convergence Study

The meshing element used for the validation part can be considered large. This is because the meshing in the reference's model is large where the number of elements used by Dandekar [12] is only 963 with 1368 number of nodes, and the element size as shown in Table 2 is chosen to reduce the percentage error of the result for validation purpose only. Theoretically, the larger number of elements will give the smaller element size hence providing more accurate results for analysis. However, the smaller element size will result in higher solving time for analysis to be done. Hence, a convergence and mesh independence study are conducted to obtain the most efficient solving time of analysis with the FEA model's result independent of the mesh size. Table 6 shows the data taken from this method meanwhile Figure 8 shows the convergence meshing plot.

Table 6
Convergence and Mesh Independence Study

Parameter	Case 1	Case 2	Case 3	Case 4	Case 5
Element Size	25	20	15	10	5
Number of Elements	2985	4037	7697	16162	66849
Number of Nodes	3446	4598	8446	17285	69099
Solve Time (hr)	0.02	0.03	0.06	0.15	1.36
Maximum Deflection (mm)	-600	-603	-608	-611	-615
Maximum von-Mises stresses (Pa)	3.534×10^8	3.558×10^8	3.568×10^8	3.542×10^8	4.101×10^8

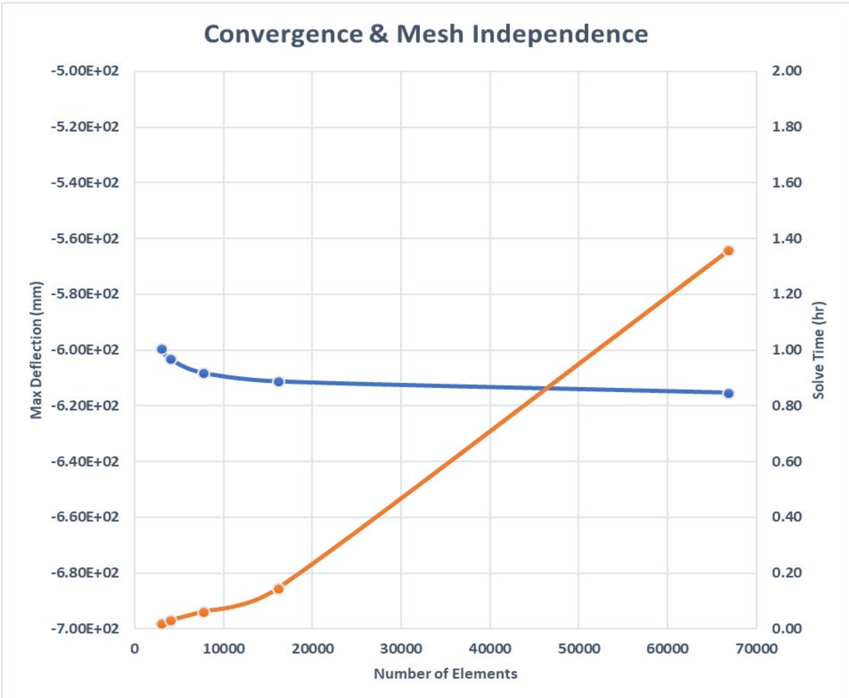

Fig. 8. Convergence and Mesh Independence

In this study, the maximum deflection of the metallic fuselage frame and the number of elements in the model were analyzed and plotted. Figure 8 shows that as the number of elements increases, the value of maximum deflection changes continuously. However, when the element size decreases from 10 to 5, the change in maximum deflection becomes small. It is important to note that the time for analysis of models with element sizes of 5 and 10 differs significantly, with an element size of 5 requiring 1.36 hours and an element size of 10 taking around 0.15 hours to solve.

Based on this analysis, it is concluded that an element size of 10 is sufficient for the analysis. The difference in the desired parameter (maximum deflection) was not significantly affected when the number of elements increased, and the solving time with an element size of 10 is efficient. Therefore, in the subsequent analysis, all the meshing properties will be changed to use an element size of 10 as shown in Table 7, instead of the element size mentioned in the previous part.

Table 7
Details of New Meshing

Parameter	Value
Element Size	10
Number of Elements	16162
Number of Nodes	17285

2.3 Analysis and Optimization of Composite Fuselage Frame

The optimization process focuses on enhancing the energy absorption capabilities of composite fuselage frames compared to their metallic counterparts. The analysis of composite frames follows a similar methodology as the metallic frames, with the only difference being the materials used. The composite materials consist of fixed epoxy matrix and varying carbon, graphite, Kevlar, and E-glass fibers. The material and shell card details are presented in Table 8. The properties of each composite obtained from Ansari *et al.,* [27] and Performance Composites Ltd. [28] are presented in Table 9.

Table 8
Sections Card and Boundary Conditions

Parameter	VALUE
Materials Card	MAT 054 - Enhanced Composite Damage
Number of Layers, NIP	8
Thickness (mm)	0.9144
Orientation of the Ply	$[0/90/45/-45]_s$
Material axes option parameter, AOPT	3

Table 9
Properties of Composite Materials

Materials		Standard Carbon Fibre (Unidirectional)	Graphite AS-3501-6 (Unidirectional)	E Glass Fibre (Unidirectional)	Kevlar Fiber (Unidirectional)
Mass Density (kg/mm^3)		$1.60x10^{-6}$	$1.61x10^{-6}$	$1.90x10^{-6}$	$1.40x10^{-6}$
Young's Modulus	Longitudinal E_1	135.00	142.73	40.00	75.00
	Transverse E_2	10.00	13.79	8.00	6.00
Poisson Ratio	ν_{12}	0.30	0.30	0.25	0.34
	ν_{21}	0.02	0.03	0.05	0.03
Shear Modulus (GPa)	G_{12}	5.00	4.64	4.00	2.00
	G_{23}	3.70	3.03	2.80	1.80
Compressive Strength (GPa)	Longitudinal X_C	1.20	1.45	0.60	0.28
	Transverse Y_C	0.25	0.20	0.11	0.14
Tensile Strength (GPa)	Longitudinal X_T	1.50	1.45	1.00	1.30
	Transverse Y_T	0.05	0.05	0.03	0.03
Shear Strength (GPa)	Inplane	0.07	0.09	0.04	0.06

All the properties of the composites mentioned used for analysis are obtained directly from the references except for Poisson ratio, ν_{21}. However, the Poisson ratio, ν_{21} can be obtained directly by using Eq. (1) since Poisson ratio ν_{12}, longitudinal Young's modulus and transverse Young's modulus are known [29].

$$\nu_{21} = \nu_{12}\frac{E_2}{E_1} \tag{1}$$

3. Results

3.1 Effect of Different Materials of Composite Fuselage Frame

The study focuses on the impact resistance of various composite materials, including Carbon/ Epoxy, Graphite/Epoxy, Glass/Epoxy, and Kevlar/Epoxy. The frames are designed with a symmetrical laminated structure consisting of eight plies, following a stacking sequence of quasi-isotropic laminates $[0/90/45/\text{-}45]_s$. The key parameters analyzed include maximum global displacement, specific internal energy, and normal and shear stresses.

The composite fuselage frames with the $[0/90/45/\text{-}45]_s$ stacking sequence are subjected to an impact test, similar to the metallic fuselage frame in validation part. The results are shown in Figure 9 until Figure 13 comparing the Aluminum Alloy (Al 7075-T6) with the composite materials for all parameters mentioned.

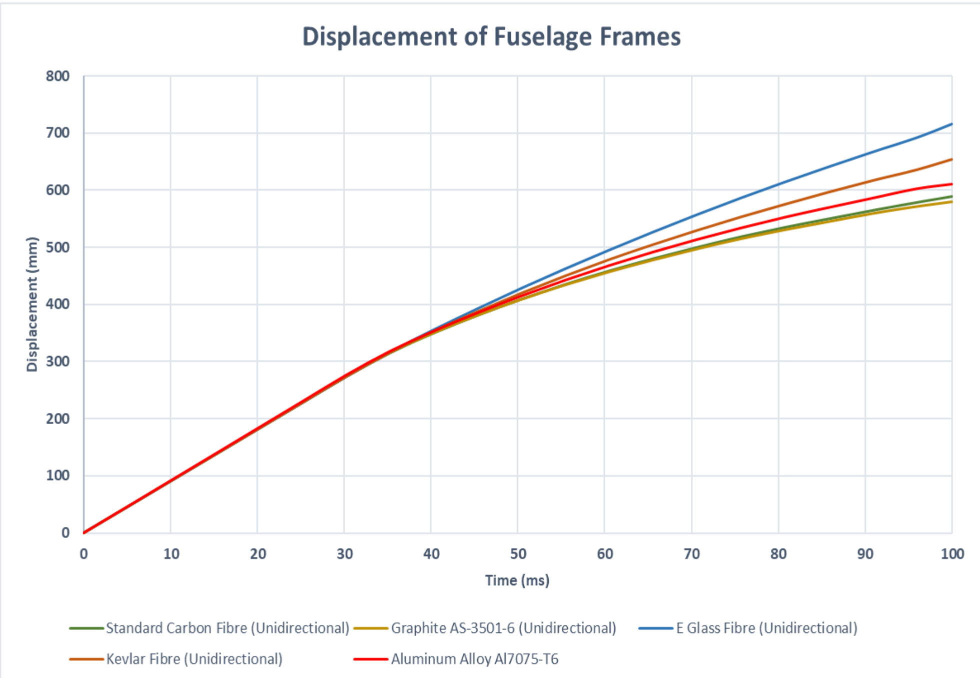

Fig. 9. Comparison of Displacement for Different Materials of Fuselage Frames

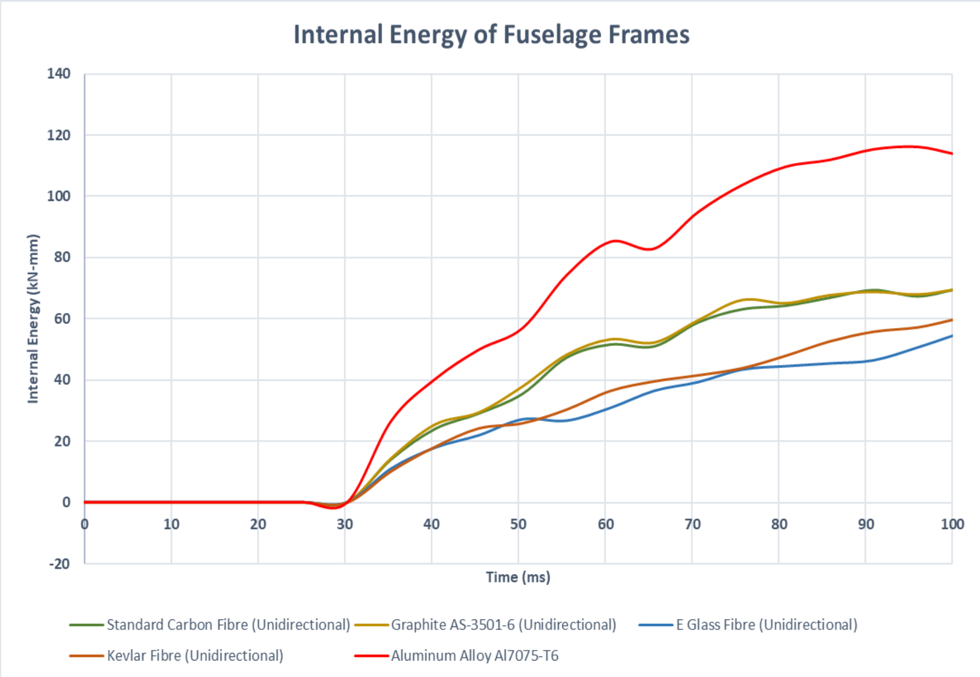

Fig. 10. Comparison of Internal Energy for Different Materials of Fuselage Frames

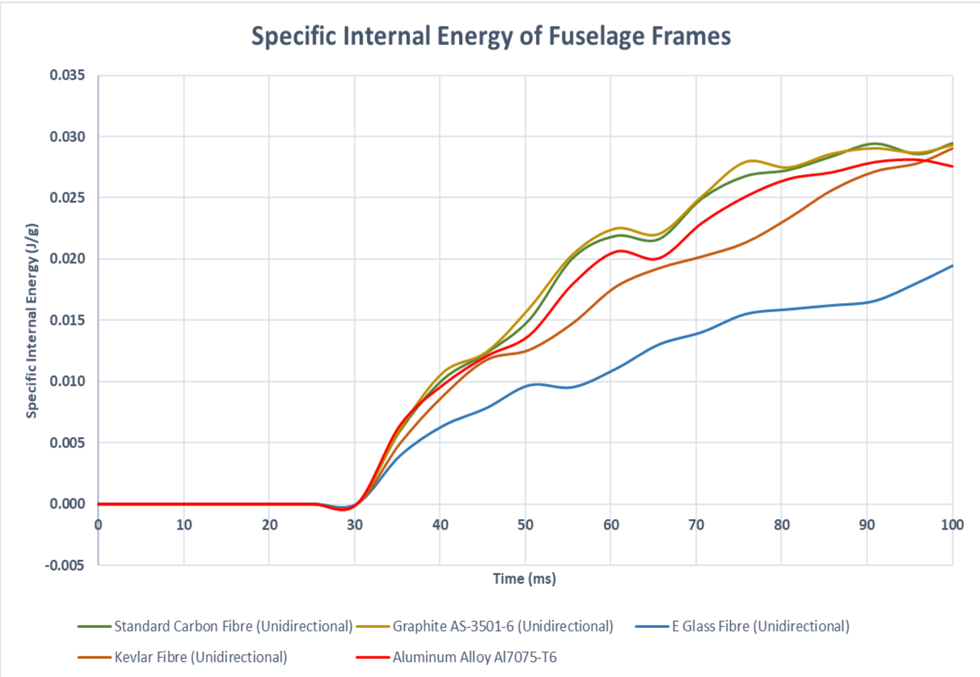

Fig. 11. Comparison of Specific Internal Energy for Different Materials of Fuselage Frames

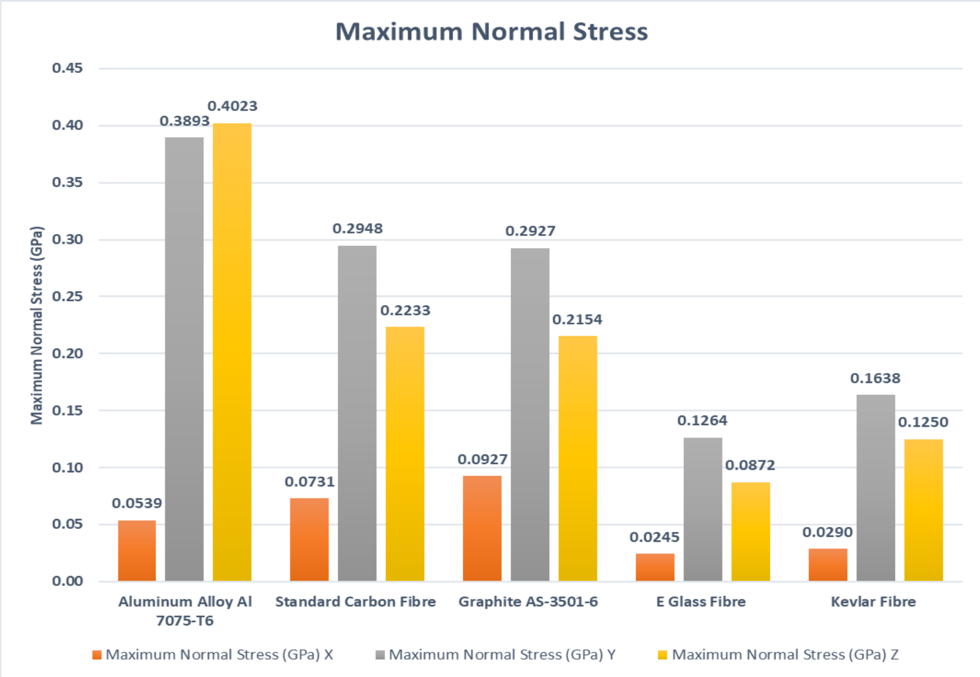

Fig. 12. Comparison of Maximum Normal Stress for Different Materials of Fuselage Frames

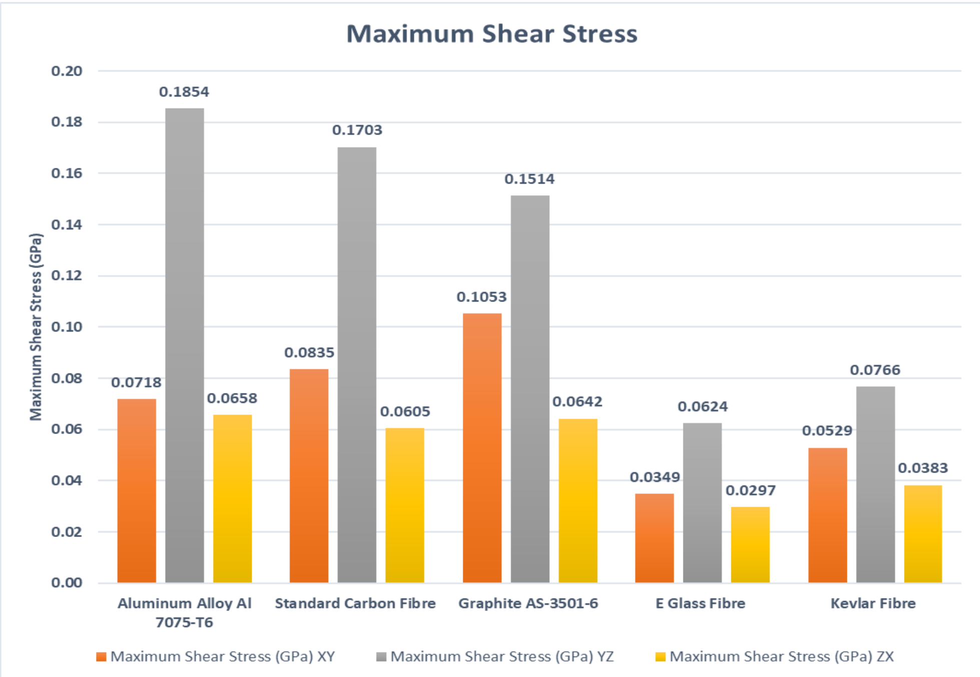

Fig. 13. Comparison of Maximum Shear Stress for Different Materials of Fuselage Frames

The analysis reveals that the E Glass Fiber composite exhibits the highest displacement, followed by Kevlar Fiber and Aluminum Alloy. The Standard Carbon Fiber and Graphite Fiber composites display similar displacement values which is better than others. Higher displacement represents higher deformation of the fuselage frame structure due to the impact.

Figures 10 and 11 depict the internal energy and specific internal energy of each frame. Aluminum Alloy shows higher energy absorption capability due to its higher mass. Standard Carbon Fiber and Graphite Fiber exhibit similar energy absorption characteristics, while Kevlar Fiber and E Glass Fiber have lower energy absorption capacities. The specific internal energy analysis reveals that Standard Carbon Fiber and Graphite Fiber outperform Aluminum Alloy in energy absorption. Kevlar Fiber surpasses Aluminum Alloy in energy absorption after a certain time, while E Glass Fiber absorbs the least energy among all materials due to its low material properties.

The maximum normal and shear stresses are studied and presented in Figure 12 and Figure 13 for each fuselage frame. The composite frames generate lower stresses compared to Aluminum Alloy under the same load conditions. E Glass Fiber and Kevlar Fiber exhibit the lowest stresses in all directions. Standard Carbon Fiber and Graphite Fiber produce stresses better than Aluminum Alloy but still considerably high. Lower stress levels are desirable for ensuring structural safety.

Based on the analysis, Standard Carbon Fiber and Graphite Fiber fuselage frames demonstrate better energy absorption and deformation characteristics compared to Aluminum Alloy and other composites. In terms of stress generation under impact loading, E Glass Fiber, and Kevlar Fiber show promising results. However, it is important to note that the specific orientation of composite plies can significantly affect the results due to the anisotropic nature of composites. The next part of the analysis will explore the effects of varying ply orientations on the performance of the composites.

3.2 Optimization of Composite Fuselage Frame with Different Laminates Orientations

Since in previous analysis, standard Carbon Fiber exhibit the best results for energy absorption and deformation compared to other materials hence, in this section the optimization in terms of laminates orientation is conducted specifically on Carbon Fiber frame. The different stacking sequences for each laminate orientation used in this analysis are presented in Table 10. There are namely 12 orientations studied in this analysis to observe the relation between composite fuselage frame behaviour in crash loading with its ply orientations.

The previous quasi-isotropic laminate is assumed to be the first orientation for this analysis, followed by unidirectional laminates with 0°, 15°, 30°, 45°, 60°, 75°, and 90° angles. Additionally, cross-ply laminate $[0°/90°]_s$, as well as symmetrical-balanced laminates consists of $[15°/75°]_s$ and $[30°/60°]_s$ and angle-ply laminate $[45°/-45°]_s$ are analyzed separately. The number of plies and their thickness remain constant, as in the previous analysis, with 8 plies and a thickness of 0.1143 mm for each ply. The objective of this part is to examine the effects of orientation on the behavior of the composite structure.

Table 10
Stacking Sequences used for Different Laminates

Parameter	Stacking Sequences	Parameter	Stacking Sequences
Orientation 1	$[0/90/45/-45]_s$	Orientation 7	$[75/75/75/75]_s$
Orientation 2	$[0/0/0/0]_s$	Orientation 8	$[90/90/90/90]_s$
Orientation 3	$[15/15/15/15]_s$	Orientation 9	$[0/90/0/90]_s$
Orientation 4	$[30/30/30/30]_s$	Orientation 10	$[15/75/15/75]_s$
Orientation 5	$[45/45/45/45]_s$	Orientation 11	$[30/60/30/60]_s$
Orientation 6	$[60/60/60/60]_s$	Orientation 12	$[45/-45/45/-45]_s$

The results of this analysis are presented in Figure 14 until Figure 19.

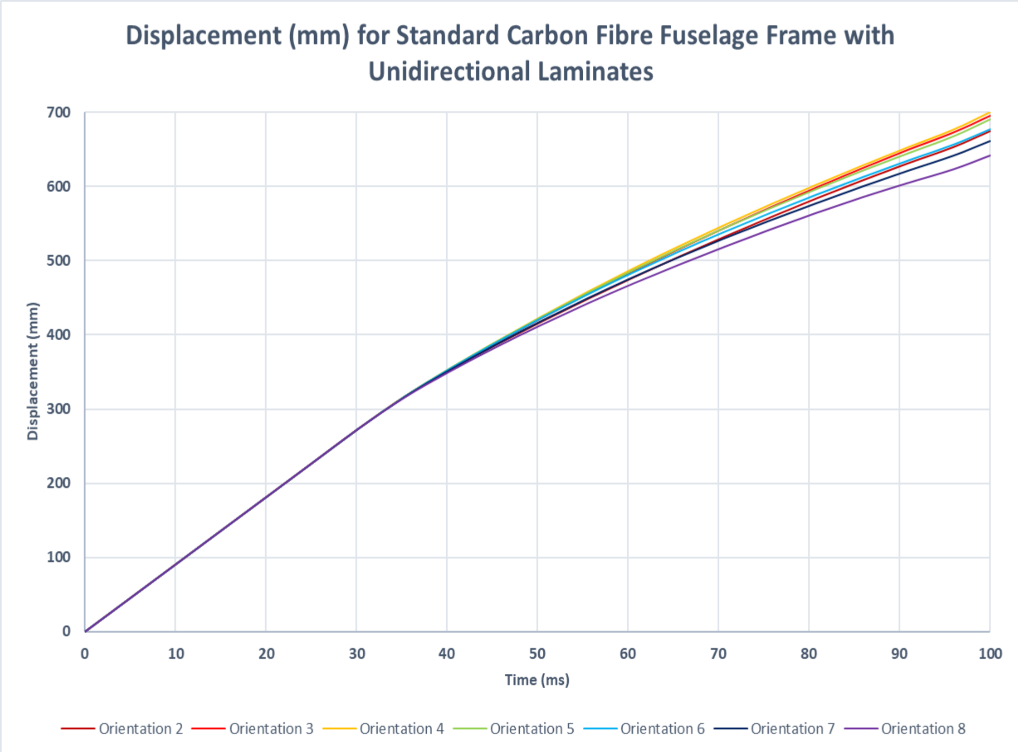

Fig. 14. Comparison of Displacement for Standard Carbon Fiber Fuselage Frames with Unidirectional Laminates

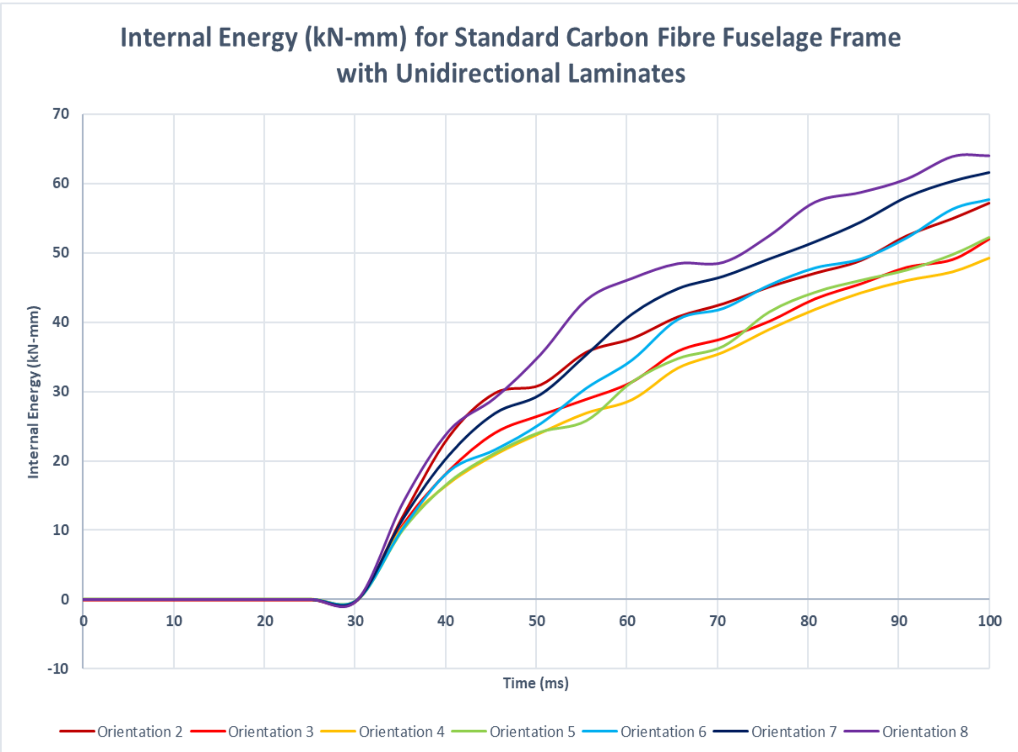

Fig. 15. Comparison of Internal Energy for Standard Carbon Fiber Fuselage Frames with Unidirectional Laminates

Figure 14 and Figure 15 depict the comparison of unidirectional Carbon Fiber laminates for fuselage frames under crash loading in terms of displacement and energy absorption respectively. Based on the graph, it can be observed that the difference in deflection of the frames for each unidirectional laminates are insignificant despite laminate with orientation 8 shows better results compared to others. However, when it comes to energy absorption, the difference of each laminate is clearly visible where orientation 8 still dominates the chart followed by orientation 7, 6 and 2. It can be observed also that the higher the angle of ply in each laminate results in better deflection and energy absorption except for orientation 2 (0°) case. Next, the comparison of composite laminates for orientation 9 until orientation 12 are studied and observed.

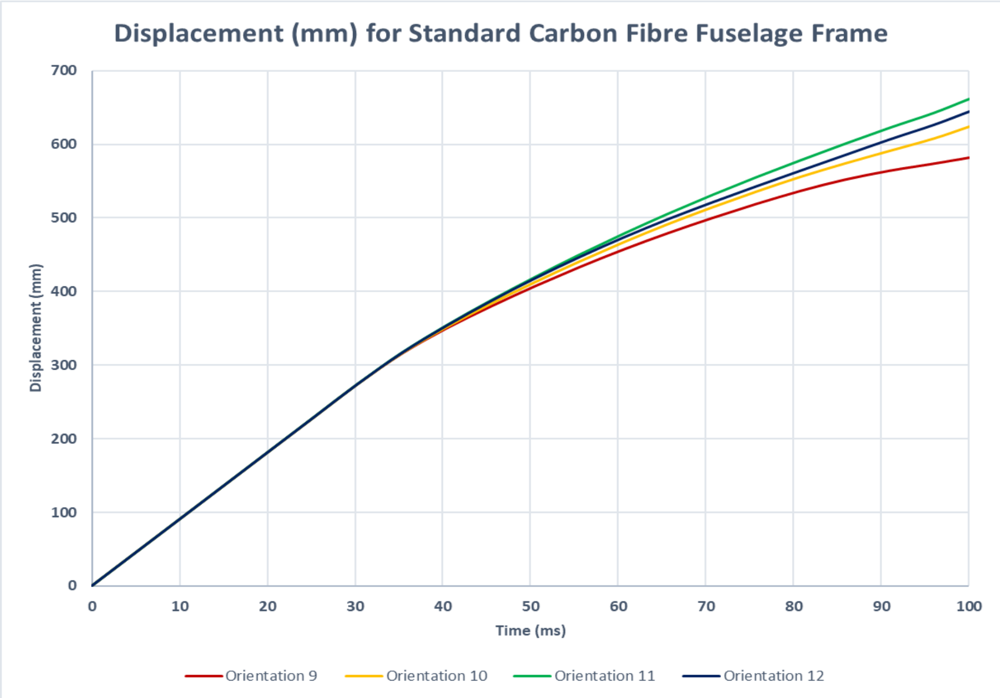

Fig. 16. Comparison of Displacement for Standard Carbon Fiber Laminated Fuselage Frames

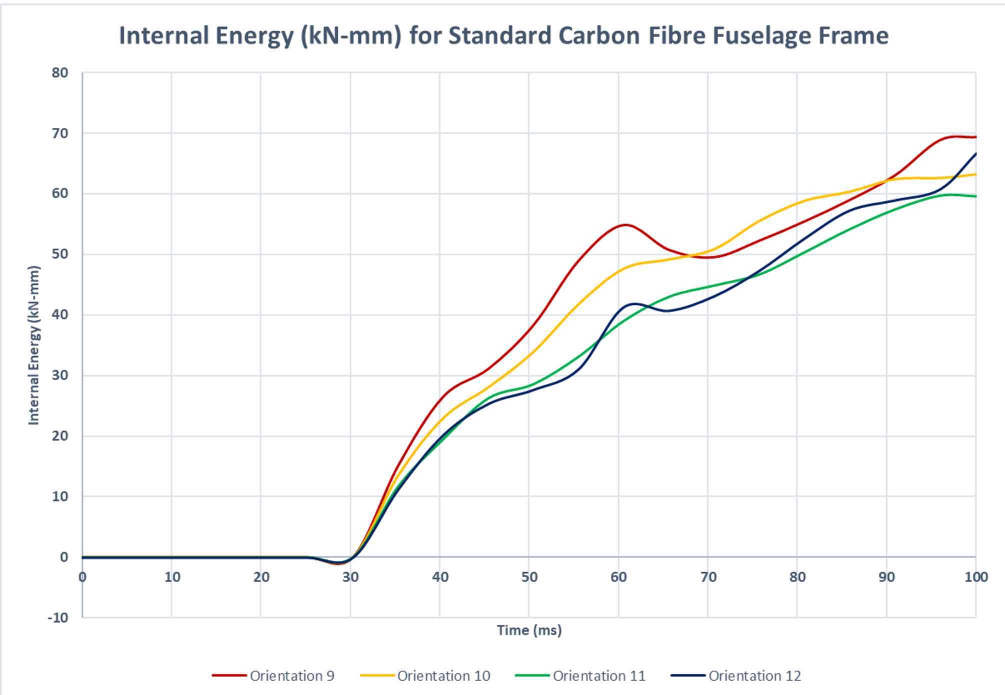

Fig. 17. Comparison of Internal Energy for Standard Carbon Fiber Laminated Fuselage Frames

In this part, four different composite laminates are studied and analyzed which is cross-ply, angle-ply, and two symmetrical-balanced laminates. Figure 16 shows that cross-ply laminate is better in deflection compared to other laminates followed by symmetrical-balanced laminates of $[15°/75°]_s$ and angle-ply laminate. Laminate with orientation 11 exhibit the highest deformation among all the other laminate.

Meanwhile, in terms of energy absorption, cross-ply laminate shows significant difference compared to other laminates only in the early phase of crash. The angle-ply laminate and symmetrical-balanced laminate of $[15°/75°]_s$ almost exceed the cross-ply laminates around after 70 ms of the impact. Similarly, laminate with orientation 11 exhibit the lowest result among all the other laminate. Hence, it can be stated that cross-ply laminate provides better deformation and energy absorption compared to other laminates.

In addition, by comparing the results obtained for quasi-isotropic, unidirectional, and cross-ply laminates, the results obtained between quasi-isotropic laminates and cross-ply laminates is almost the same. Figure 18 and Figure 19 show the difference between quasi-isotropic laminate and cross-ply laminate.

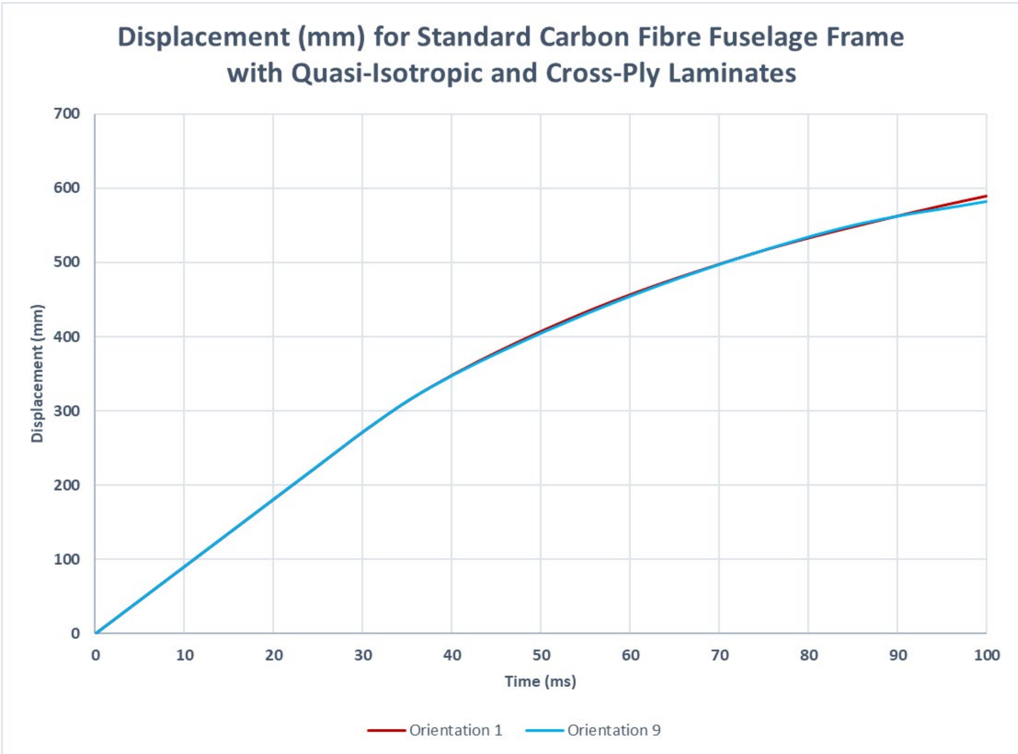

Fig. 18. Comparison of Displacement for Standard Carbon Fiber Fuselage Frames with Quasi-Isotropic and Cross-Ply Laminates

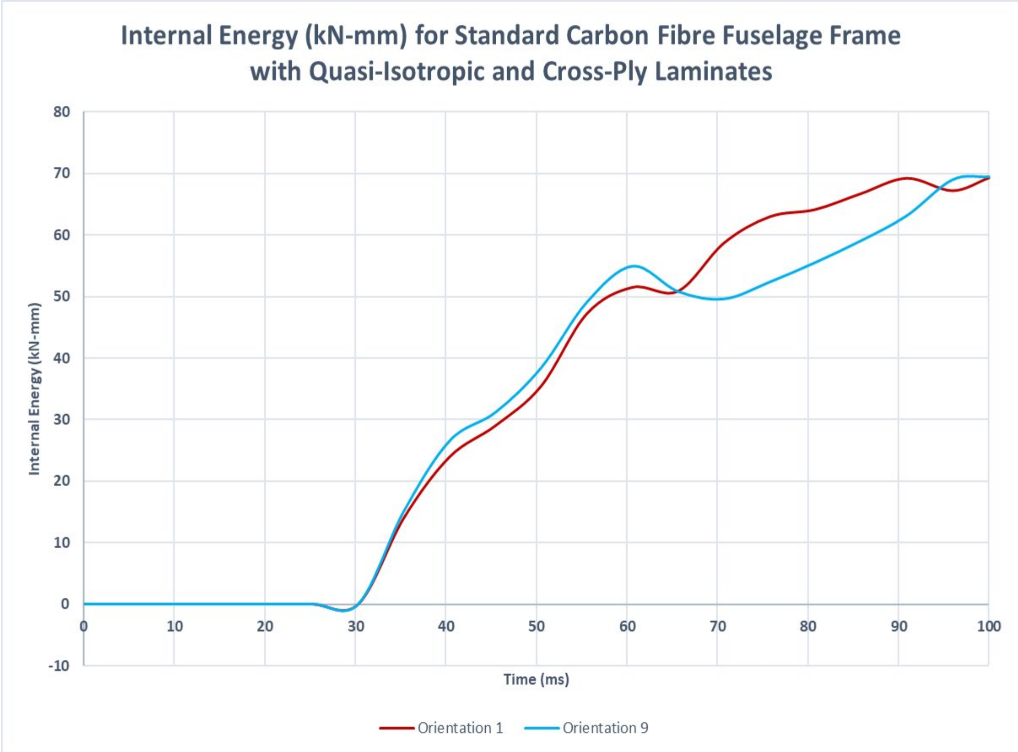

Fig. 19. Comparison of Internal Energy for Standard Carbon Fiber Fuselage Frames with Quasi-Isotropic and Cross-Ply Laminates

Based on these figures, it can be stated that the effects on composite fuselage frame behaviour for quasi-isotropic and cross ply laminates are almost similar to each other. However, as discussed previously, composites that have more directions covered by its ply tend to be better. In this case, since the load applied is only in one direction hence limiting the actual effect of the laminate's orientation on composites behaviour.

4. Conclusions

The study focused on analyzing the effects of different materials and laminate orientations on composite fuselage frames under impact loads. The materials studied included Aluminum Alloy, Standard Carbon Fiber, Graphite Fiber, E Glass Fiber, and Kevlar Fiber. The results showed that each material had its own advantages. Standard Carbon Fiber and Graphite Fiber performed well in terms of energy absorption and deformation, while E Glass Fiber and Kevlar Fiber excelled in minimizing stress generation. The study also investigated the impact of varying laminate orientations on the energy absorption and deformation, and it is found that composite with laminates having 90° ply such as quasi-isotropic, cross-ply and unidirectional 90° laminates provide better results for both parameters. Future research directions could include optimizing cross-section shape and thickness, exploring different manufacturing processes, varying fibre volume percentage, and incorporating hybrid composites or fibre metal laminates to further enhance the strength-to-weight ratio of composite frames.

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