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On the application of metal foils for improving the impact damage tolerance of composite materials

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Abstract: Composite mechanical characteristics can be heavily influenced by impact damages; however, this influence can be reduced by choosing a correct stacking sequence and constituents materials. In this paper, the influence of metal layer placement within the stacking sequence of a carbon/epoxy laminate on impact resistance was studied. Impacts were simulated by means of Quasi Static Indentation tests.

INTRODUCTION

Advanced composite materials have many alluring characteristics: first of all, higher strength-to-weight ratio and better fatigue resistance, compared to metals commonly used in the aerospace field. Another positive characteristic is the tailoring of the mechanical properties of a structure by combining layers of different fibres or with different orientation. Composite manufacturing techniques also permit the fabrication of large integral structures, limiting the need of joints.

Due to these advantages, composite materials have been steadily increasing in their usage within the aerospace industry. The latest wide-body aircraft designs, the Boeing 787 Dreamliner and Airbus A350, have more than 50% in weight of the airframe made of composites. However, there is still a huge lack of knowledge in their mechanical behaviour that leads to more strict regulations to guarantee safety standards.
This has resulted in composites impossibility of reaching their full potential in the aerospace industry and necessity for deeper studies, especially related to damaged structures and their residual mechanical characteristics (impact damaged and aged structures).

Most of FRP drawbacks are related to their brittle behaviour. This led to the idea, developed since '80 at TU Delft, of combining composite materials with metal layers previously treated to obtain the best adhesion possible (Figure 1). Fibre Metal Laminate can combine advantages of both constituents in order to avoid drawbacks of common composites or metals, when used separately. In fact, in '80s, intense use of aircraft, due to the higher and higher number of people travelling, raised problems for aluminium ageing and fatigue properties. With FML materials these issues can be avoided [1].

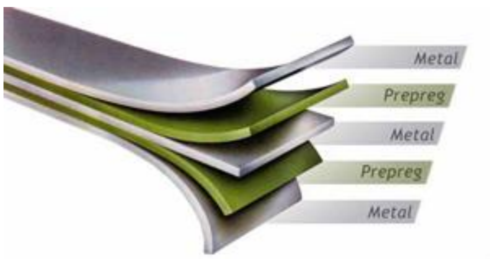


Figure 1: Hybrid material example

Moreover, introduction of metal layers in a composite material could improve impact resistance. This is due to metal plasticity, which can absorb more impact energy, potentially reducing load transfer to other composite plies.
Impact damage is an important issues for composite materials, especially when they are used in the aeronautical field. In fact, airplanes are exposed to a lot of different impact causes: during maintenance or construction, tools could drop and hit structures; during take-off or landing, debris could be thrown against airplanes; during flight,

hail stones and birds could strike on the plane; during boarding, passengers or employees could hit doors cut-outs with luggage, and so on [2] (Figure 2 – Figure 3).
Depending on velocities, energies and structure thickness, impacts could result in visible or barely visible damages (BVID). The last, even if difficult to detect, can result in a huge decrease of mechanical characteristics and in sudden failures [3-4]. This is an issue that needs more studies to achieve a better knowledge of which kind of damage a structure can bear and how damages could evolve in operative conditions.

In order to find out aluminium layers position influence on impact resistance of an Al reinforced carbon/epoxy laminate, four stacking sequences were tested by means of Quasi Static Indentation (QSI) tests.
It was, in fact, demonstrated that QSI tests could be a reliable method to simulate Low Velocity Impacts and, therefore, to study material impact resistance [5-7].

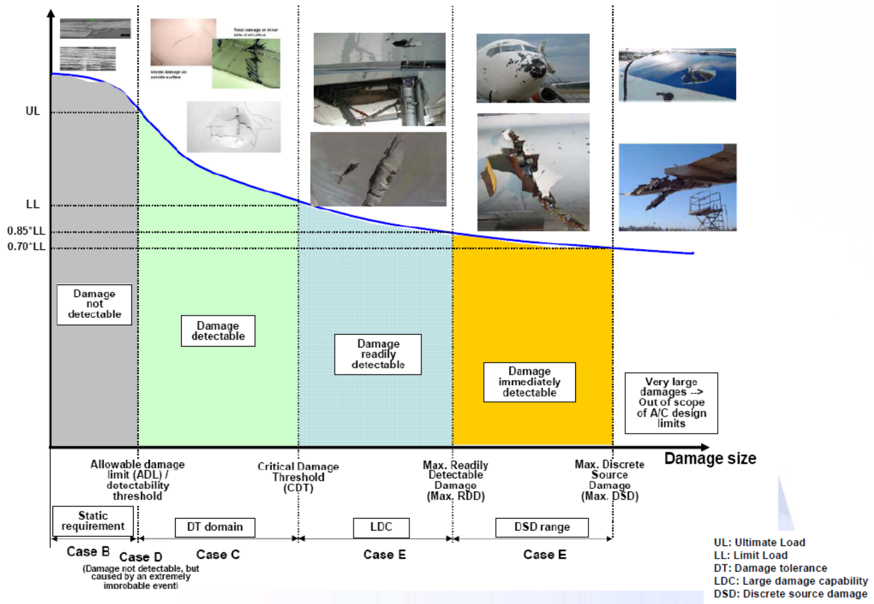


Figure 2: Load bearing capability based on impact damage size [3-4]

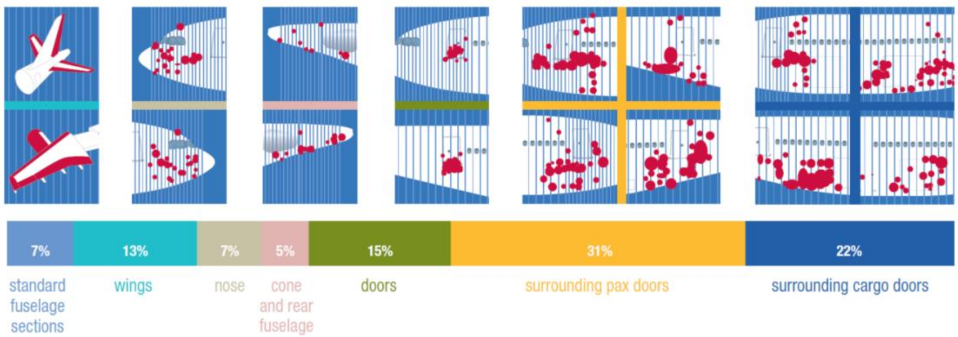


Figure 3: Global percentage of impacts by zones on the aircraft (example on A320) [2]

Specimens and Material
Four laminates were manufactured: in each of these, Al foils were in different but symmetrical position with respect to laminate middle plane (Table 1). Laminates have the same bending stiffness, calculated by means of Classical Laminate Theory for carbon/epoxy sections and Elastic theory for Aluminium sections.

The chosen materials were carbon/epoxy pre-preg Hexcel M18/1 43% G939 Fabric (Table 2), Al 2024 T3 (Table 3) and 3M Scotch Weld resin (Table 4). The latter was used for INT stacking sequence in order to achieve adhesion between two Al layers.

Table 1: Specimens staking sequences and bending stiffness

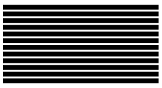
Metal Location	**Stacking sequence**		**Bending Stiffness [Pa*m^3]**
EXT		Al/(0/90)$_9$/Al	132
MID		(0/90)$_2$/Al/(0/90)$_4$/Al/(0/90)$_2$	98
INT		(0/90)$_4$/Al/resin/Al/(0/90)$_4$	107
No-Metal		(0/90)$_{12}$	108

Table 2: Carbon/epoxy pre-preg characteristics

Property	**43% G939 Fabric**
Fibre density	1.78 g/cm^3
Resin density	1.22 g/cm^3
Fibre areal weight	220 g/m^2
Nominal ply thickness	0.227 mm
Nominal fibre volume	55%
Tensile strength	800 MPa
Compressive strength	800 MPa
Tensile modulus	65 GPa
Compressive modulus	64 GPa
In-Plane shear strength	100 MPa

Table 3: Al 2024 T3 Properties

Property	**Al 2024 T3**
Ultimate Tensile Strength	483 MPa
Tensile Yield Strength	345 MPa
Elongation at Break	18 %
Modulus of Elasticity	73.1 GPa
Poisson's Ratio	0.33
Fatigue Strength	138 MPa
Shear Modulus	28 GPa
Shear Strength	283 MPa

Table 4: AF 191U resin

Property	**AF 191U**
Thickness	0.0625 mm
Foil Weight	73±24.4 g/m^2
Cure temperature	177°C
Cure time	60’
Stress (at 23°C)	13 MPa
Strain (at 23°C)	2.11 %
Young Module (at 23°C)	0.71 GPa

Specimens were manufactured at Delft Aerospace Structures and Materials Laboratory at TU Delft by hand layup and autoclave curing. They were cut manually, with a diamond saw, with dimensions 150x100mm based on ASTM D7136M/D7136M-15 [8] for future comparison with other studies involving impacts (Figure 4).

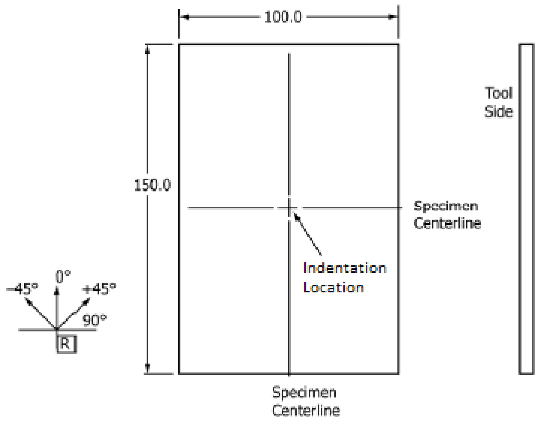


Figure 4: QSI specimens dimensions

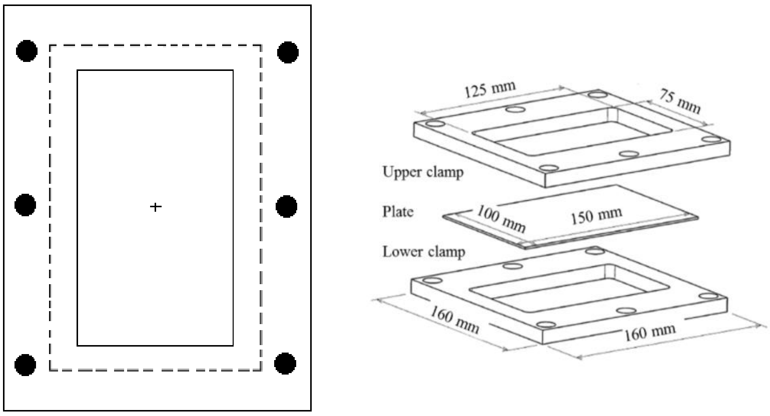


Figure 5: Fixture dimensions and scheme

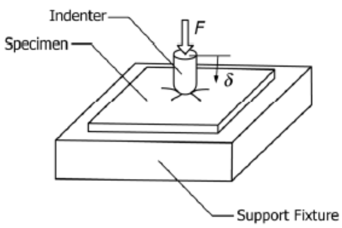


Figure 6: QSI test scheme

QSI tests [8] were performed by means of a 20 kN Zwick electro-hydraulic testing machine. Specimens were fixed in a proper fixture (Figure 5). Indenter was a metallic cylinder with a 12.5 mm diameter, a 25 mm length and a hemispherical head. During tests, the indenter is pushed orthogonally against specimen, with a velocity of 2mm/min. Tests end was set at 80% load drop, which resulted also in a complete specimen perforation. Tests data were recorded by Zwick software and then processed by means of Office Excel.

Results

Three specimens per each stacking sequence were tested. In Table 5, average indentation results are reported; in particular, maximum indentation strength has been normalised on the basis of laminate bending stiffness in order to obtain more comparable values.

Looking at normalised indentation resistance results, the INT coupons show higher values due to presence of high toughness carbon/epoxy layers in the outer part. Conversely, the composite fragile behaviour brings lower resistance values for No-Metal coupons without Al layers. Indeed, combining the CFRP with Al results in a more plastically deformable material (Al is in this case carrying shear loads even when carbon fibres are already failed).

Table 5: Laminates ultimate load results

Stacking sequence	F_{MAX} [N]	$t_{average}$ [mm]	Bending Stiffness [Pa*m^3]	F/EI [1/m]
EXT	6194.47	3.12	132	46.93
MID	5139.67	2.95	98	52.45
INT	5879.53	2.94	107	54.95
No-Metal	4560.77	3.09	108	42.23

In the following figures, Force-Displacement graphs are shown. Each graph is analysed in details to understand failure modes [10-15].

Looking at No-Metal laminate behaviour in Figure 7, there is a little load drop at instant named point A: given that 'fixture dimensions/specimen thickness' rate is quite high, at that point matrix failure starts and evolves until point B, where the fibre damage is involved; damage progression is then pretty stable until point C where the maximum failure strain is reached.

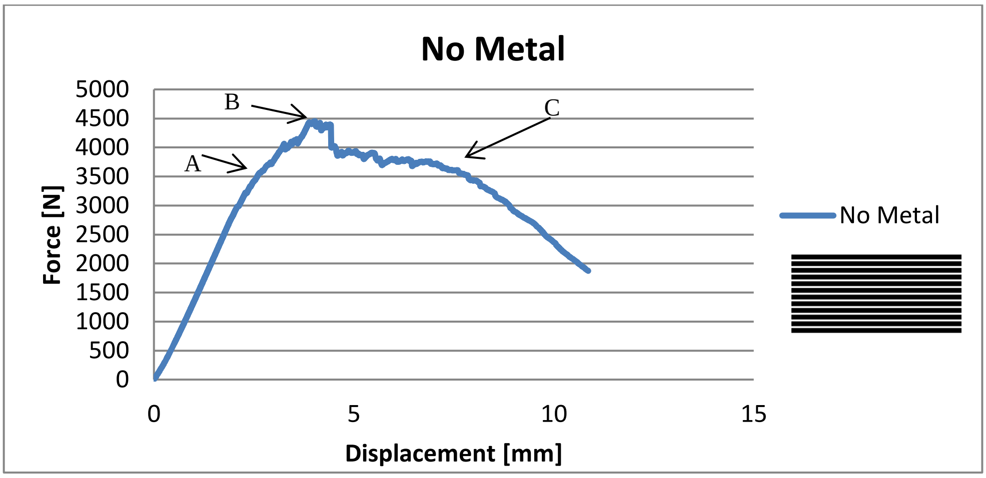


Figure 7: No metal QSI test behaviour

The behaviour of EXT coupons (Figure 8) shows a change of slope (point A) attributable to yield of the Al layer at the specimen back-face, opposite to the indenter. This is due to the small thickness of specimen and relative small bending stiffness of the laminate. Hence, yield point is firstly reached on the specimen back face.

Point B shows another change of curve slope related to matrix cracks onset. Matrix crack growth ends at point C where fibres failure occurs. From this point until maximum failure force (point D), failure is stable.

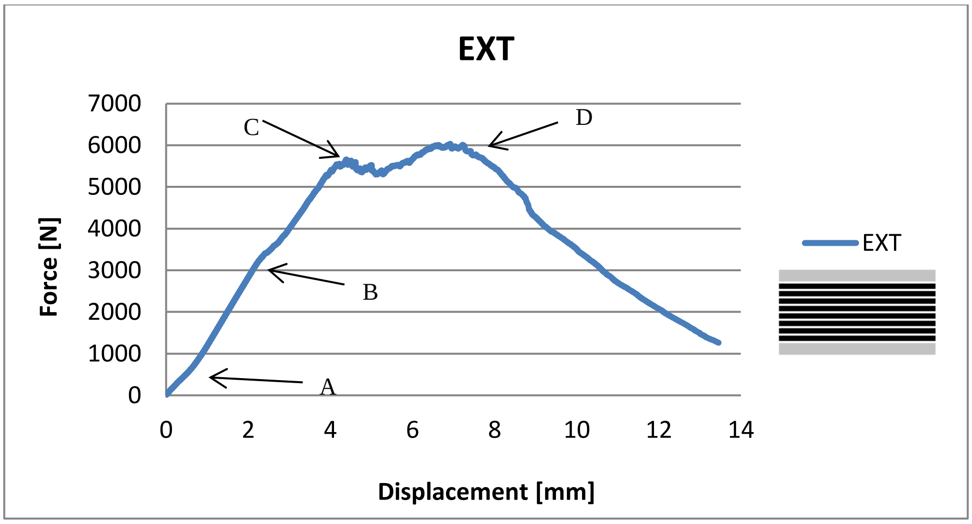


Figure 8: EXT specimen QSI behaviour

MID specimens (Figure 9) show a slightly different behaviour: there is a first yield point (A) where first Al layer yields while matrix crack onsets are at points B and C. This is due to the peculiar stacking sequence that results in more groups of carbon/epoxy plies: point B refers to external back-face carbon/epoxy layers cracking, while point C is linked to internal carbon/epoxy layers cracking. Point D describes max load and max displacement that the specimen can carry and the point where fibre failure takes place.

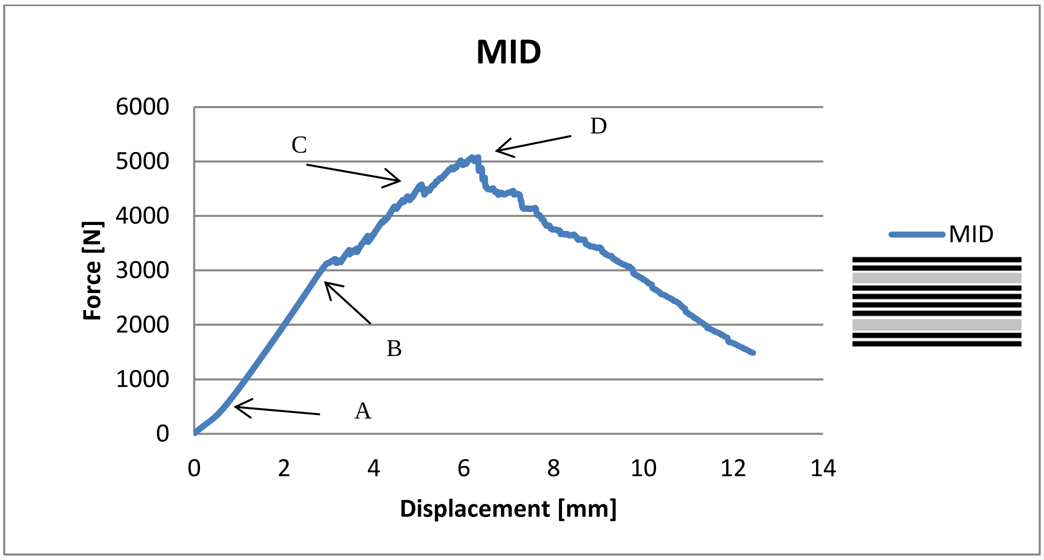


Figure 9: MID specimens QSI behaviour

For INT specimens (Figure 10) only a matrix cracks onset, at point A, is noticeable. Cracks propagate until point B, where fibre failure occurs. This is also the max displacement: after this point, all back-face carbon/epoxy layers are heavily damaged and hence load bearing capacity is reduced.

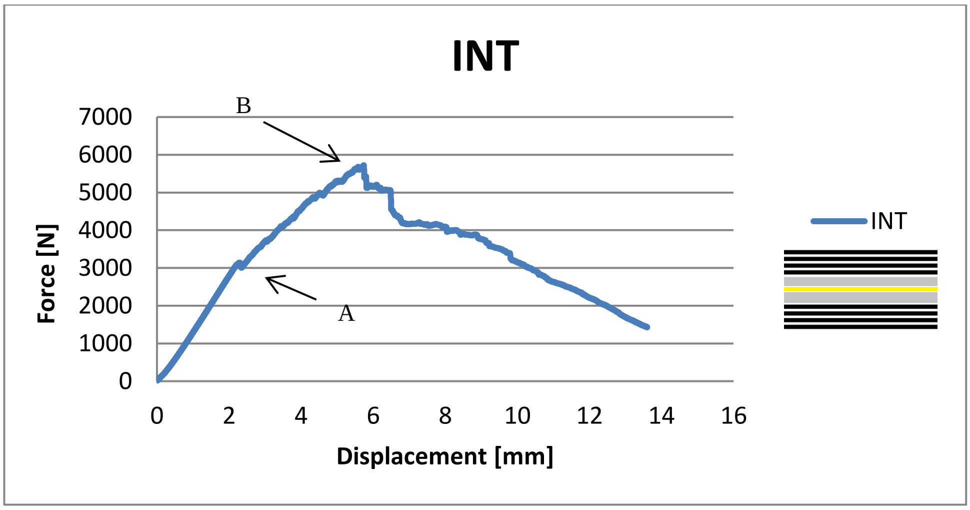


Figure 10: INT Specimens QSI behaviour

Average values per each stacking sequence were calculated and a plot of averaged QSI behaviour is showed in Figure 11. In Figure 12 the same graph is proposed with normalised values, for a better comparison of different stacking sequences behaviour. Normalization was obtained dividing loading force by laminates bending stiffness.

The most important feature raising from these plots, is failure progression. No-Metal stacking sequence has a more stable failure growth. Also EXT has a quite stable damage propagation (better load bearing also after first fibre damage).
Moreover, yield loads are almost the same, while failure load is higher in INT and MID laminates (looking at normalised graph, where results are independent from slightly different bending stiffness).

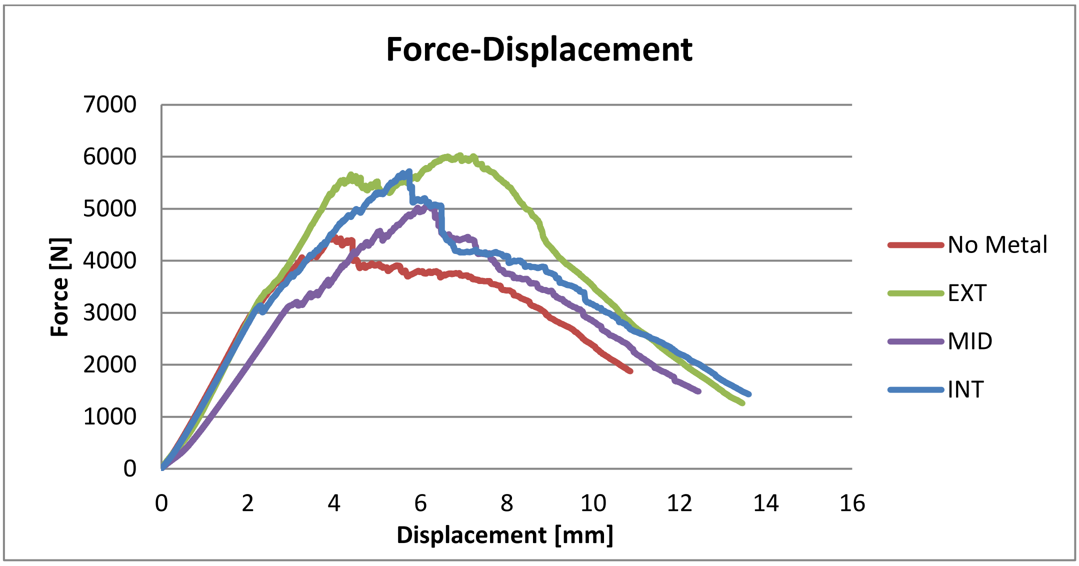


Figure 11: QSI tests average results

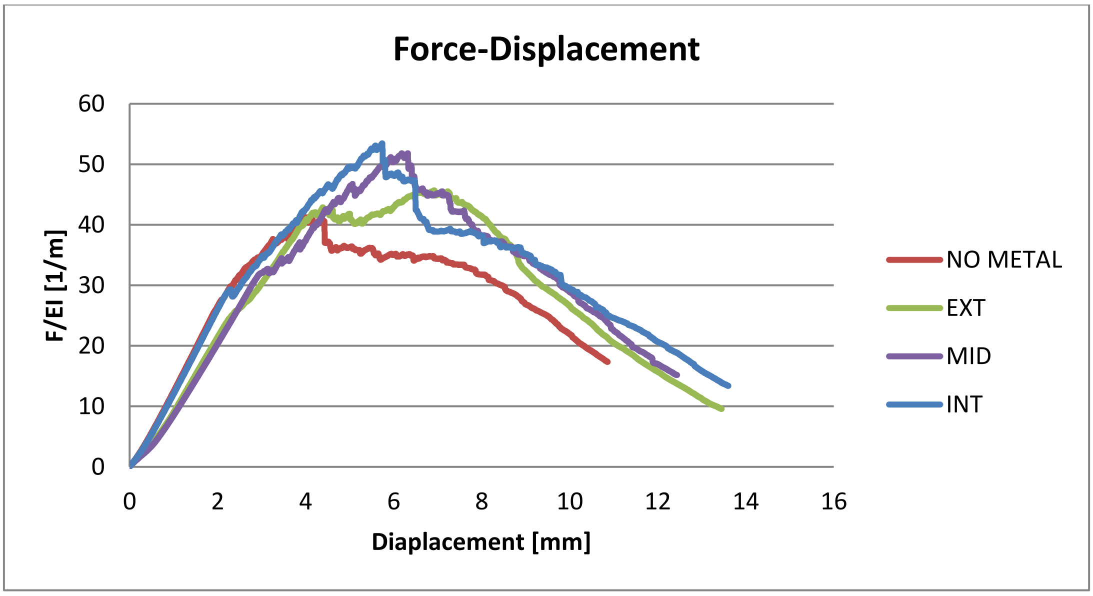


Figure 12: QSI tests normalized average results

It is also evident the elastic modulus insensitivity to bending stiffness: in raw data graph, MID laminate shows a less rigid behaviour compared to the others. Taking into account the MID bending stiffness in the normalised graphs, this difference is not present anymore.
The analysis of the coupons plots after maximum force show that No-Metal and INT specimens have the same behaviour, i.e. a reduced stiffness compared with MID and EXT specimens.

Table 6: Energy Absorption results

		Absorbed Energy		Specific Absorbed Energy		
Coupon		**Max**	**at Max Force**	**Areal Density**	**Max Specific Energy**	**Specific Energy at Max Force**
		Nmm	**Nmm**	**g/mm^2**	**Nmm^3/g**	**Nmm^3/g**
INT	**1**	47056.3	22424.1	0.005205	9040590.9	4308184
	2	43250.9	19426.3	0.005283	8186800.6	3677137
	3	46688.2	19036.0	0.005198	8981953.5	3662179
MID	**1**	43480.6	18327.3	0.005244	8291494.9	3494907
	2	36588.4	17711.3	0.005216	7014638.4	3395570
	3	36428.6	16453.0	0.005110	7128884.6	3219771
EXT	**1**	52263.1	27486.2	0.005583	9361106.7	4923191
	2	48882.8	28811.9	0.005589	8746248.4	5155116
	3	48989.5	25243.4	0.005493	8918526.5	4595562
No Metal	**1**	39603.9	11601.7	0.004549	8706075.2	2550395
	2	35024.5	10018.0	0.004532	7728270.4	2210513
	3	32623.3	14077.3	0.004347	7504788.9	3238390

Each specimen was visually inspected to analyse failure modes (Figure 15). All coupons with carbon/epoxy on the outer side have a cross shape crack that follows fibres directions, while Al foils show common metal failure shape (following rolling direction, as shown in Figure 13 d. for EXT coupon).

Failure happens on the surface opposite to the indenter due to the laminate low thickness and related membrane behaviour.

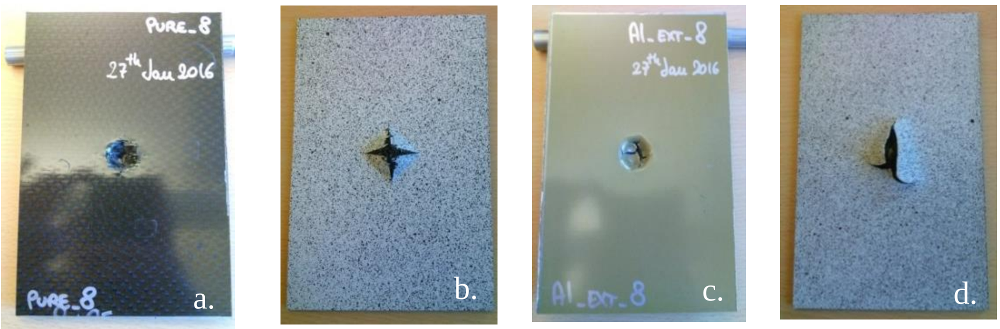


Figure 13: failure shapes examples (specimen with composite in the outside: front and back; specimen with Al on the outside: front and back face)

Conclusions

In conclusion, in the research of a solution for carbon/epoxy composites sensitivity to Low Velocity impacts, addition of metal foils has been investigated. In particular, research focused on influence of metal layers position inside the stacking sequence. Therefore, a Quasi Static Indentation tests campaign has been performed on four different FML laminates.

It was find out that EXT laminate and, therefore, Al layers on the outer part of the specimen, led to a stable failure mode, with a quite high resistance value. Together with an easier detectability of the impact location, due to Al plasticity, these are the reasons why it could be said that having metal layers in the outer part of a composite stacking sequence can be the best configuration.

This stacking sequence has other advantages that do not really come from this experimental campaign but they represent important matters that cannot be ignored during an aircraft design project: lightning strike and surface painting.

A composite structure is usually not conductive and, therefore, in order to avoid issues related to lightning strikes, metallic nets are embedded into the composite material. This approach could be avoided using a laminate with metallic foil on the outside; hence, EXT is once again pointed out as the best configuration.

Another convenience in this laminate application could come from the painting requirements: composite structures have to be treated to have a good adhesion between its surface and varnish, while this is easier when it comes to metallic structures.

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