EVALUATION OF COMPRESSION AFTER IMPACT STRENGTH OF CARBON/EPOXY COMPOSITES USED IN AERONAUTICAL AREA

Jane Maria Faulstich de Paiva
Materials Division (Divisão de Materiais) – AMR - Instituto de Aeronáutica e Espaço - IAE
Centro Técnico Aeroespacial – CTA
Praça Marechal-do-Ar Eduardo Gomes, 50 – Vila das Acácias
CEP: 12228-904 – São José dos Campos / SP – Brazil

UNISAL – Centro Universitário Salesiano de São Paulo
Campus Dom Bosco - Unidade de Americana / SP – Brazil
e-mail: janepaiva@uol.com.br

Sergio Mayer
Departamento de Aeronáutica e Mecânica - Instituto Tecnológico de Aeronáutica - ITA/CTA
Embraer – Empresa Brasileira de Aeronáutica
São José dos Campos / SP – Brazil

Mirabel Cerqueira Rezende
Materials Division (Divisão de Materiais) – AMR - Instituto de Aeronáutica e Espaço - IAE
Centro Técnico Aeroespacial – CTA
Praça Marechal-do-Ar Eduardo Gomes, 50 – Vila das Acácias
CEP: 12228-904 – São José dos Campos / SP – Brazil
e-mail: mirabel@iae.cta.br

Abstract. *Carbon fiber reinforced polymeric composites are used in aeronautical industry for the manufacture of aircraft components attending tight requirements. Moreover, in service, these components can suffer mechanical damages after low energy impact, provoked by hailstorms, stones, bird impacts, etc. Aiming to minimize these damages, studies suggest modifications on the orientation, reinforcement arrangement and/or in the polymeric matrices of the composite. The present work evaluates the effect of the reinforcement arrangement and mainly the polymeric matrix modification on the impact strength of laminates. For this purpose, it was used unmodified and modified epoxy resin matrices correlating their influences, after different impact loads, on the compression after impact strength of the laminates. The laminates were obtained in autoclave by using prepregs with orientation of [(±45°),(0°/90°)]. The prepregs were based on epoxy matrices F155™ (DGEBA type, with modified epoxy), F584™ (modified epoxy with elastomer) and 8552™ (toughened epoxy with thermoplastic) reinforced with carbon fiber fabric "Plain Weave" (PW) and "Eight Harness Satin"(8HS) types. After molding the laminates were inspected by C-Scan technique and cut attending the SACMA and BSS7260 standards. Compression after impact (CAI) and without impact tests were performed. For CAI tests the specimens were previously submitted to the "falling dart" impact test, with energy of 28 J, and inspected by C-Scan to evaluate the damaged areas. The results show that the laminates presented a decrease of nearly 30% in compression strength after impact tests. In spite of this reduction, all strength values (starting from 240 MPa) are higher than the allowable average value (227 MPa) for aeronautical applications. The samples submitted to the CAI test were photographed, revealing typical fracture aspects.*

Keywords: *composites, carbon fiber, compression after impact, CAI*

1. Introduction

Aeronautical structures can suffer damages by impact of low or high energy. Depending on the energy level the performance of the structure can be compromised. Thus, the impact strength might be an important consideration for project and maintenance of such structures, mainly when they are obtained in polymeric composites (Hawyes, Curtis and Soutis, 2001; Moura and Marques, 2002; Schoeppner and Abrate, 2000).

Particularly, low-energy impact may be caused by tool drops, runway debris, hailstorms, strange objects, and other accidental impacts during manufacturing processes, assembly or in service (Gao and Kim, 2001; Schoeppner and Abrate, 2000; Sjogren, Krasnikovs and Varna, 2001). Also, during operational use the aircraft can suffer bird strikes (Wu et al., 1996).

The low-energy or low-velocity impacts are potentially dangerous because they can provoke sub-surface delamination and cracks not visible on the surface (Gao and Kim, 2001). The main problem is that the structure with internal damages can suddenly fail in service, although the external aspect seems to be appropriate.

In general, polymeric laminated composites present high strength-to-weight and stiffness-to-weight ratios when compared with metallic materials. However, laminated composites are susceptible to impact loads because they are constituted by a laminar system with not strong interfaces (Moura and Marques, 2002). The impact damages can appear in the form of matrix cracking, delamination, fiber-matrix debonding, fiber fracture, surface microbuckling and other interfacial phenomena. Such effects can cause considerable decreasing on the mechanical properties of polymeric composite (Ding, Yan and McIlhagger, 1995; Gao and Kim, 2001; Hawyes, Curtis and Soutis, 2001; Moura and Marques, 2002; Roy, Sarkar and Bose, 2001).

Particularly, composites containing thermoset matrices, such as epoxy and polyimide ones, are known to be susceptible to internal damages caused by low velocity or low-energy impact (Besant, Davies and Hitchings, 2001; Jang, 1994). Therefore, the capacity to predict the impact damages and the mechanical performance is very important for aeronautical structure area and this evaluation can be carried out by compression-after-impact (CAI) strength test (Larsson, 1997; Schoeppner and Abrate, 2000; Sjogren, Krasnikovs and Varna, 2001).

CAI tests are not very simple to carry out and require more material than the conventional mechanical tests, as tensile, compressive, flexural, etc. (Jang, 1994; Soutis, Smith and Matthews, 2000). This test involves higher costs because the laminated specimens must have large dimensions (SACMA, 1988). During the CAI test the compression load can not cause global or total buckling on the tested specimen. The expected result considers micro-buckling mechanisms followed by fiber breakage, cracks, etc.

Firstly, CAI tests require to perform low-energy impact simulations, where the polymeric composite specimens must be able to resist the impact load which results generally in internal damages (Jang, 1994; Thanomsilp and Hogg, 2003). After the impact tests the same specimens are submitted to the compression tests.

However, before to perform the CAI tests it is important to evaluate the effect produced by the low-energy impact in the polymeric composites considering the damage extension, the position and the geometry of internal and external delaminations. For this, it is commonly used an ultrasonic technique, preferentially C-Scan type (Gao and Kim, 2001; Greenhalgh et al. 1996; Hennecke, 1987; Hennecke, 1990; Schoeppner and Abrate, 2000).

The sub-surface delamination mechanisms can be extensive in polymeric composites and usually the internal delamination grows under the subsequent compressive loading. This extensive damaged area results in a significant strength reduction in post-damage performance. Therefore, it is important to improve the delamination resistance of these composites (Davies et al., 2004; Davies, Hitchings and Zhou, 1996; Larsson, 1997).

Different approaches can be used aiming improvements in the impact damage tolerance of polymeric composites, such as the control of fiber-matrix interfacial adhesion, modification of matrix formulation, modification of lamination design (different orientations and/or sequences of plies), modification of fiber rearrangements, utilization of high-strain fibers, hybridization, interlaminar layers, etc. (Gao and Kim, 2001; Jang, 1994).

This work shows the CAI evaluation of laminated composites manufactured by using three different epoxy matrices a F155™ (DGEBA type), one with epoxy resin combined with elastomeric converters of polymer (code F584™) and other epoxy resin with thermoplastic modifier (code 8552™), reinforced with two different carbon fabrics, i.e., “plain weave” and “eight harness satin” types.

2. Experimental

2.1. Composite Molding:

For composite molding it was used carbon fabrics preimpregnated (prepreg) with epoxy resins. The prepregs were based on epoxy matrices (codes F155, F584, and 8552™) and carbon fiber fabric reinforcements with PW (Plain Weave) and 8HS (Eight Harness Satin) arrangements. The prepregs were arranged attending [(±45°),(0°/90°)] in relation to the fill direction of the fabric. This configuration is required for compression-after-impact tests and it is in accordance with SACMA (1988) and MEP 15-022 (2000) technical specifications.

In agreement with the prepreg supplier (Hexcel Composites, 2003) the F155™ type is an epoxy matrix (DGEBA type, with modified epoxy), the F584™ matrix was modified and combined with novel elastomeric converters of polymer and the 8552 ™ matrix is an epoxy matrix amine cured that was toughened with thermoplastic polymer.

Six laminated families (F155/PW, F155/8HS, F584/PW, F584/8HS, 8552/PW and 8552/8HS) were consolidated in autoclave by using vacuum bags in metal molds (plates) under pressure of 0.70 MPa. For the F155 prepregs, it was used a cure cycle with a heating rate of 2.5 ± 0.2 °C.min^{-1} up to 121°C, holding at this temperature at least 90 minutes. For the F584 and 8552 prepregs it was used a cure cycle with a heating rate of 2.5 ± 0.2 °C.min^{-1} until 177°C, holding at this temperature for a minimum period of 120 minutes. All processed laminates presented 60±1% v/v of carbon fiber reinforcement, determined according to ASTM D3171.

After molding, all plates were inspected by an ultrasonic failure detector Reflectoscope S80 with a 0.750’’, 5 MHz transmitter type Automation X19625 and a receiver type Automation X19267. A water squirt was used to transport the ultrasonic beams to reduce surface losses. The laminated plates were mounted in a mid-way between the probes that were carefully aligned to maximize the transmitted signal. The probes were transported by an Automation US640 system that generates a quantized C-scan record of the laminated plate. An actual size map of the plate was generated after scanning associating a color to each attenuation level.

This procedure investigated the quality of the manufactured laminates to assess voids, delaminations, interlaminar cracks, inclusions, foreign object damage, resin rich or resin starved areas, fiber misalignment. After inspection no defects were detected, suggesting that the laminates were processed suitably.

2.2. Low Energy Impact Test

The low impact energy tests were carried out in accordance with SACMA (1988) and MEP (2000) technical specifications. The tested specimens were cut in dimensions of 152.4mm of length, 101.6 mm of width, and approximately 6 mm of thickness. To obtain this thickness it were used plies of PW fabric corresponding to $[(\pm 45°),(0°/90°)]_{8S}$ and plies of 8HS fabric corresponding to $[(\pm 45°),(0°/90°)]_{4S}$ on the layup.

During the low energy impact tests a steel dart (4.5 kgf) with spherical tip impactor (12.7 cm of diameter) was dropped from 60 cm of height until to hit the specimen fixed on the impact equipment, resulting in approximately 28 J of energy (MEP, 2000). The non-instrumented impact equipment was designed and built based on both SACMA (1988) and BSS7260 (1988) technical specifications.

2.3. Ultrasonic inspection of impacted specimens

After the low energy impact tests (28 J) the specimens were evaluated by ultrasonic scanning analyses following similar procedure previously cited, using Reflectoscope S80 equipment with an Automation X19265 receiver. The damaged areas of the impacted composites were inspected and the general configuration of the delamination was recorded with color images.

2.4. Compression Before and After Impact Tests

The compression before and after impact (CAI) tests were carried out in accordance with SACMA (1988) and BSS7260 (1988) technical specifications. The CAI device was adapted to a test machine (Baldwin-Lima-Hamilton Corporation) with BLH Electronics load cell.

Six specimens of each laminate type (F155/PW, F155/8HS, F584/PW, F584/8HS, 8552/PW, 8552/8HS) were tested before and after low energy impact test, as previously described. Therefore, 72 specimens of (152.4 x 101.6 x 6) mm were tested.

The compressive strength values were calculated using Eq. (1) (SACMA, 1988):

$$\sigma_{CAI} = P / b.e \tag{1}$$

Where:

σ_{CAI} = compressive strength, *MPa*
P = ultimate compressive load, *MN*
b = width of specimen, *m*
e = thickness of specimen, *m*

3. Results and Discussion

3.1. Composite Inspection

After molding, the ultrasonic analyses of the inspected laminates presented the attenuation values depicted in Tab. 1 and the images showed in Fig. 1. According to the used color scale, the orange color is related to the non-occurrence of defects, corresponding to the attenuation value of nearly 9 V. The green color indicates the occurrence of defects (voids, delamination, debonding fiber-matrix, etc.) corresponding to attenuation values smaller than 5 V. The blue color reveals more intense defects with attenuation value equal to 1 V.

Table 1 – Attenuation values by ultrasonic inspection of the laminates after molding.

Laminate	*Attenuation (V)*
F155 / PW	8.33 to 9.67
F155 /8HS	8.67 to 9.33
F584 / PW	8.00 to 9.67
F584 / 8HS	8.67 to 9.67
8552 / PW	8.67 to 9.57
8552 / 8HS	6.67 to 9.33

In Fig. 2 are shown the ultrasonic images of the laminate specimens after low energy impact test (approximately 28 J). The impacted specimens were compared to non-impacted specimens used as standard by aeronautical industry. This standard samples present some controlled defects intentionally inserted to facilitate the damage identification in the specimens, called LSP5 and LSP 6 (left and right sides of the image, respectively).

In Fig. 2 it can be observed a small green circle in the center of the specimens, which are related to the internal defects caused by the low energy impact tests. In general, these damages are not detected on the external surfaces of the specimens by a visual inspection, but they provoked sub-superficial delamination in the laminates.

Hawyes, Curtis and Soutis (2001) reported that in general low energy impact tests result in damaged area with circular shape, whereas high-energy impact tests usually result in damaged area with more elliptical shape. C-scan ultrasonic and X-radiograph inspections also showed circular damages after low energy impact tests in some carbon reinforced composites in another work reported in literature (Qi and Herszberg, 1999).

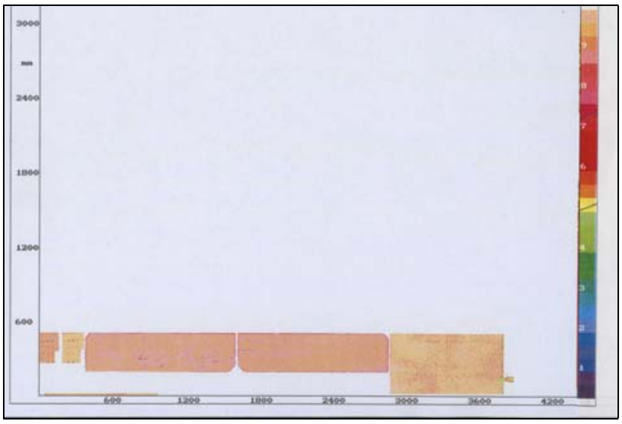


Figure 1. C-Scan ultrasonic images after laminate molding.

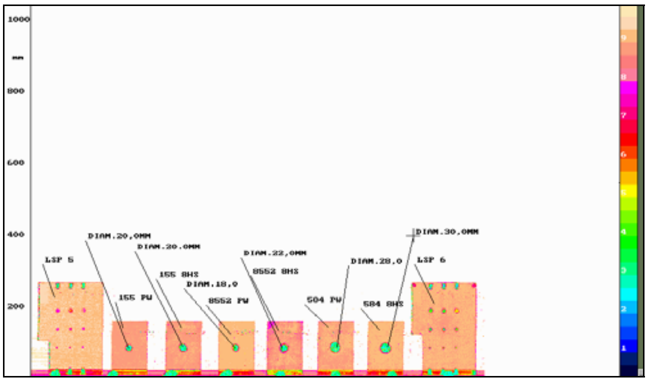


Figure 2. C-Scan ultrasonic images of the laminates after low energy impact (28 J).

The damaged areas of the laminates after low energy impact test presented minimum values of attenuation between 3.0 and 4.3 Volts (Tab. 2). These damaged areas were measured using a software program of the ultrasonic equipment. According to Tab. 2 it can be verified that the impact provoked damage propagation in the central region of all specimens.

Table 2. Extension of damaged areas after low energy impact tests (28 J).

Laminate	*Damaged area (mm)*	*Values of attenuation (minimum)*
F155 / PW	20	3.33
F155 /8HS	20	3.00
F584 / PW	28	4.00
F584 / 8HS	30	3.67
8552 / PW	18	4.33
8552 / 8HS	22	4.33

The F155 laminates presented relative low values of extension of the damaged areas (Tab. 2). However, the visual inspection showed that after impact tests the F155 laminates presented also slight visible defects on the impacted and the opposite surfaces. This fact is attributed to the a little more rigid characteristic of the epoxy matrix F155 comparing with the others, which favors more localized damages in the F155 laminates. For the F584 and 8552 matrices it was observed only sub-superficial delaminations by ultrasonic inspection.

3.2. Compression before and after low energy impact tests

Table 3 shows similar values for the compressive strength tests of the non-impacted specimens. The values were calculated by using six specimens of each laminate type.

The 8552/PW that contains an epoxy matrix toughened with thermoplastic polymer, presented the highest average value, but in the same way, the highest standard deviation. Considering the carbon fiber arrangements it is observed an increasing tendency of the compressive strength values for the PW laminates.

Table 4 shows the compressive strength values after the low energy impact tests. Table 5 shows the decreasing behavior of the compressive strength values before and after low energy impact tests.

Table 3. Compressive strength values of specimens non-impacted. Average results of 6 specimens.

Laminate	*Compressive Strength Before Impact (MPa)*
F155 / PW	381.2 ± 26.6
F155 /8HS	345.6 ± 22.6
F584 / PW	391.3 ± 46.3
F584 / 8HS	380.7 ± 33.8
8552 / PW	424.9 ± 65.8
8552 / 8HS	399.1 ± 35.4

The residual compressive strength values after impact (CAI) of the laminates with orientation [(±45°),(0°/90°)] must attend the minimum value of 207 MPa and the average compressive strength must be equal to 227 MPa, at room temperature, in agreement with technical specification of aeronautical industry (MEP 15-022, 2000). So, the values of all tested laminates are in accordance with the requirements demanded for the aeronautical specification.

Soutis, Smith and Matthews (2000) reported average residual compressive strength values of 102.4 and 168.2MPa for laminated composites, previously impacted with 28 J of energy, after CAI tests. These laminates were molded with epoxy resins (codes 922 and 924 ™, from Hexcel Composites) and carbon fiber (code T800, Toray) with orientation of $[(45^o/0^o/-45^o/90^o)]_{4S}$. Davies et al. (2004) presented that residual compressive strength after CAI tests can be less than 40% of the undamaged structure.

Table 4. Compressive strength values of specimens impacted (28 J). Average results of 6 specimens.

Laminate	*Compressive Strength After Impact (MPa)*
F155 / PW	253.7 ± 8.8
F155 /8HS	245.5 ± 5.4
F584 / PW	258.7 ± 2.4
F584 / 8HS	255.5 ± 12.0
8552 / PW	297.8 ± 8.3
8552 / 8HS	280.7 ± 18.9

Table 5. Decreasing behavior of the compressive strength values of the laminates before and after low energy impact tests (CAI).

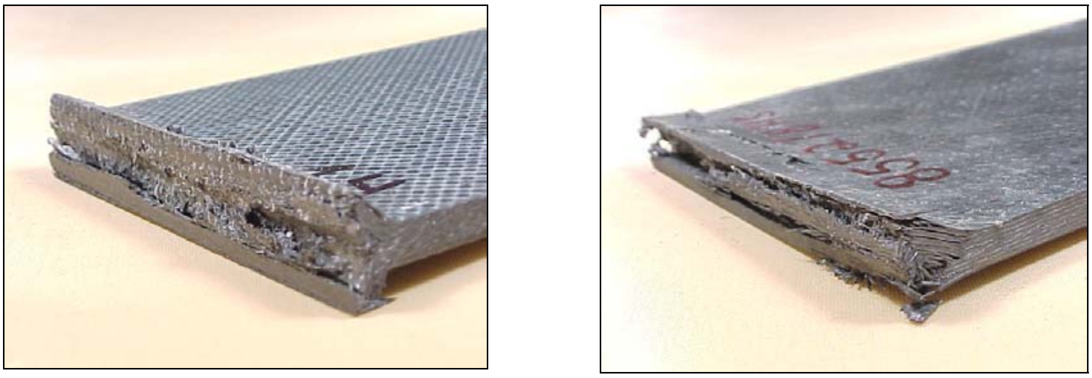
Laminate	*Decreasing (%)*
F155 / PW	33.5
F155 /8HS	29.0
F584 / PW	33.9
F584 / 8HS	32.9
8552 / PW	29.9
8552 / 8HS	29.7

Figures 3-6 show some photographs of the studied laminates after CAI tests. All failure modes are valid and required for the evaluated test.

Usually during CAI tests occur, in a first step, the fiber microbuckling followed by the kink band formation. In sequence, it occurs the growth of the microbuckling region and the kink bands provoking instability of the specimen and finally the material failure. Typical inclination values for fiber microbuckling are between 2 and 3°, and for kink band inclinations are between 5 e 25°. In general, the fiber instability failure mode occurs in all types of unidirectional and multidirectional laminates with or without impact damages (Hawyes, Curtis and Soutis, 2001). The main difference when the laminate is impacted is that the failure mode is located preferentially in the damaged region.

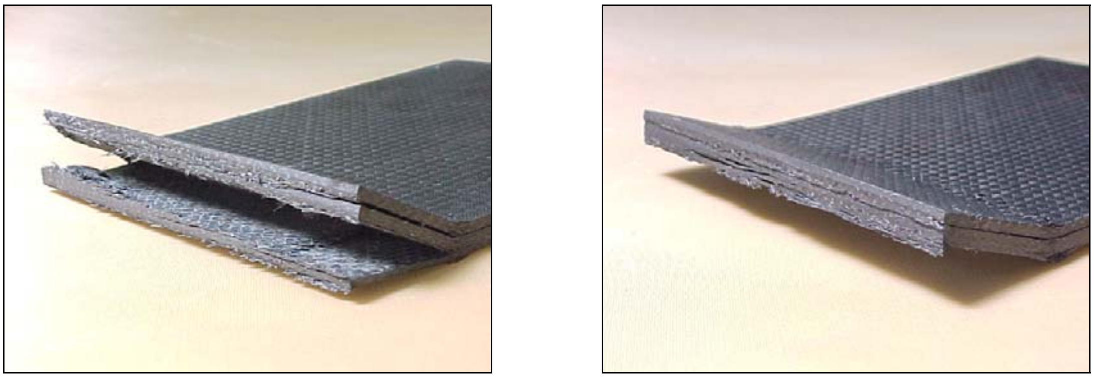
In this work it is observed that for the majority of non-impacted laminates the failure occurred on the inferior or on the superior side of the specimens (Fig. 3). This behavior is expected because the non-impacted laminates not presented defects, as voids, delamination, fiber-matrix debonding, etc.

Some non-impacted laminates, as 8552/PW, presented “more open” failures (Fig. 4). This characteristic can be related to the percentage of toughness agent in the 8552 epoxy formulation and/or some interfacial adhesion problems, that facilitated the crack propagation that evolved to the layer delamination (Fig. 4(a)). Therefore, some 8552/PW specimens did not presented extensive layer delaminations (Fig. 4(b)). Probably, this difference observed for the failure modes caused the increase of the standard deviation for the 8552/PW laminates during the compressive tests before impact (Tab. 3).



(a) (b)

Figure 3. Failure region of laminates non-impacted: (a) F155/PW; (b) 8552/8HS specimens.



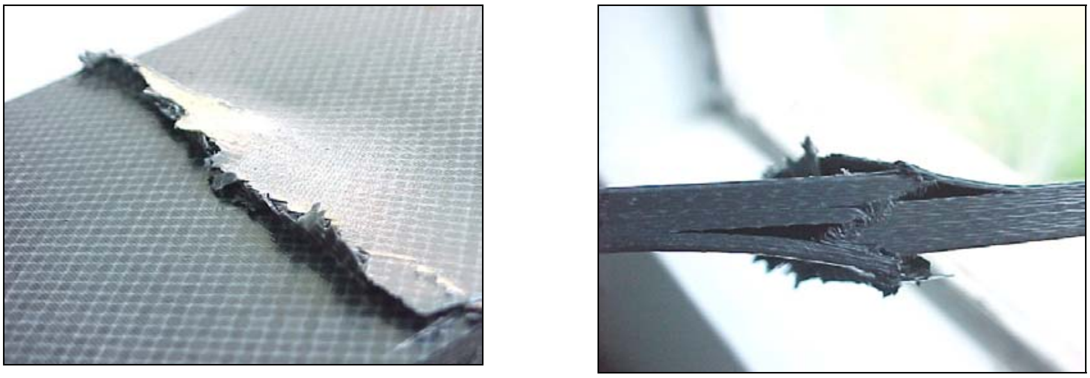
(a) (b)

Figure 4. Failure region of the 8552/PW laminates non-impacted: (a) open failure; (b) normal failure.

Figure 5 shows typical failures of the F155/PW laminates after CAI tests, previously impacted with 28 J. The laminates presents a small region showing kink band and several regions with carbon fiber breakage, mainly, in the center of the impacted specimens. The final failure mode was shear (Fig. 5(b)), resulted by the propagation of the delamination.

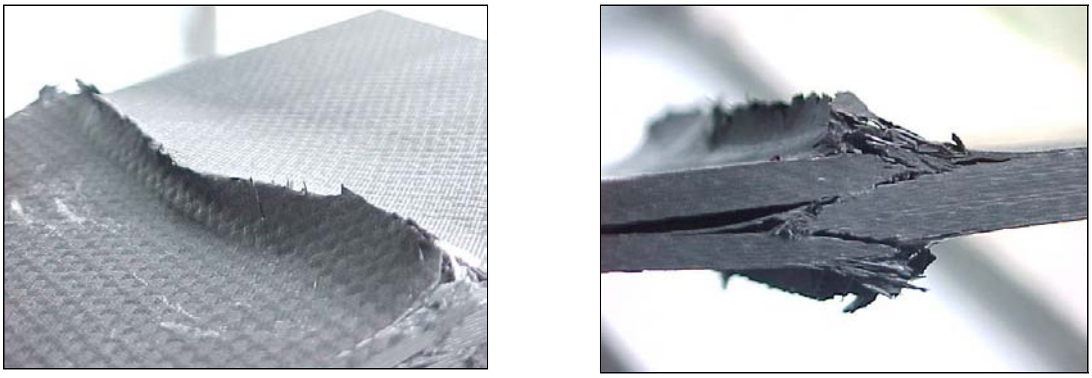
For the impacted F584/PW laminates (Fig. 6(a)) it can be observed that the failure mode shows less fragile aspect and accentuated than that one observed for the F155 laminates (Fig. 5(a)). These F584/PW specimens also present kink band and regions with carbon fiber breakage, mainly, in the center of the impacted specimens. The final failure mode also occurred by shear (Fig. 6(b)).

For the impacted 8552 laminates it was observed similar failure modes, with some kink bands and some regions with carbon fiber breakages. Therefore, it was observed microbuckling of carbon fibers (Fig. 7(b)) and that the delamination was not extensive also presenting less fragile aspect (Fig. 7(a)) than the F155 laminates.



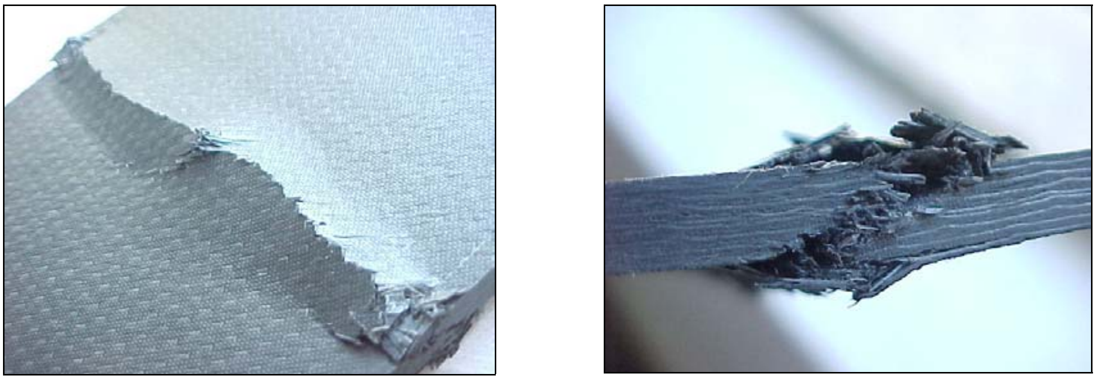
(a) (b)

Figure 5. Failure region of F155/PW laminates, impacted with 28 J: (a) failure in center of the specimen; (b) lateral side of the failure region.



(a) (b)

Figure 6. Failure region of the F584/PW laminates impacted with 28 J: (a) failure in center of the specimen; (b) lateral side of the failure region.



(a) (b)

Figure 7. Failure region of the 8552/8HS laminates impacted with 28J: (a) failure in center of the specimen; (b) lateral side of the failure region.

4. Conclusions

The C-Scan inspections showed to be an important tool, mainly after the impact tests (28 J of energy) because it allowed detecting sub-superficial damage propagation from the central region of the impacted laminates.

The residual compressive strength results of all carbon fabric reinforced epoxy matrix laminates show that they are adequate materials (high performance) for application in aeronautical industry, because they attend the aeronautical requirements even after being impacted (28 J).

The failure mode analyses showed that some 8552/PW non-impacted specimens presented more open failure regions. This aspect can be related to the percentage variation or dispersion of the toughness agent in the 8552-epoxy formulation and/or interfacial adhesion problems. But, after CAI tests, the impacted laminates with 8552 epoxy matrix presented higher average values of the residual compressive strength than the other laminates (F155 and F584) and a failure mode with less fragile aspect than the F155 laminates. This behavior is attributed to the higher toughness of the

8552 matrix, which was modified with an thermoplastic polymer, suggesting that the toughening of the epoxy matrix improve both impact strength and residual compressive strength properties.

In relation to the carbon fiber arrangements it was observed an increasing tendency of the compressive strength values for the PW laminates. All tested laminates showed the occurrence of valid failure modes according to expected for the carbon fiber reinforced epoxy composites for aeronautical use. For all impacted laminates the failure mode occurred in the central region (damaged) of the specimens.

The present work is in progress looking for study the residual compressive strength after impact with other energy values, for example 40 J, and after hygrothermal conditioning (80^{o}C of temperature and 90% of relative humidity), aiming to evaluate the decreasing of compressive strength of the six laminate families.

5. Acknowledgments

FAPESP (Proc: 02/02057-5); *CNPq* (Proc: 303528/2003-6) and *Embraer*.

6. References

ASTM D3171-76. Standard test method for fiber content of resin-matrix composites by matrix digestion. American Society for Testing Materials, CD-ROM. 2004.

Besant, T.; Davies, G. A. O.; Hitchings, D., 2001, "Finite element modelling of low velocity impact of composite sandwich panels", Composites: Part A, Vol.32, pp. 1189-1196.

BSS7260, 1988, "Boeing Specification Support Standard", pp. 1-26.

Davies, G.A.O.; Hitchings, D.; Besant, T.; Clarke, A.; Morgan, C., 2004, "Compression After Impact Strength of Composite Sandwich Panels", Composite Structures, Vol.63, pp.1-9.

Davies, G.A.O.; Hitchings, D.; Zhou, G., 1996, "Impact Damage and Residual Strengths of Woven Fabric Glass/Polyester Laminates", Composites: Part A, Vol.27A, pp. 1147-1156.

Ding, Y. Q.; Yan, Y.; McIlhagger, R., 1995, "Effect of Impact and Fatique Loads on the Strength of Plain Weave Carbon-Epoxy Composites", Journal of Materials Processing Technology, Vol.55, pp.58-62.

Gao, S.-L.; Kim, J.-K., 2001, "Cooling Rate Influences in Carbon Fibre/PEEK Composites. Part III: impact damage performance", Composites: Part A, Vol.32, p. 775-785.

Greenhalgh, E.; Bishop, S. M.; Bray, D.; Hughes, D.; Lahiff, S.; Millson, B., 1996, "Characterisation of impact damage in skin-stringer composites structures", Composites Structures, Vol.36, pp. 187-207.

Hawyes, V.J.; Curtis, P.T.; Soutis, C., 2001, "Effect of Impact Damage on the Compressive Response of Composite Laminates", Composites: Part A, Vol.32, pp.1263-1270.

Hennecke, E. G., 1987, "Destructive and Nondestructive Tests", in: Composites. Engineered Materials Handbook**.** Metals Park, OH, USA, pp.774-778.

Hennecke, E. G., 1990, "Ultrasonics", in: Nondestructive Testing of Fibre-Reinforced Plastics Composites, Vol.2, ed. J. Summerscales, Elsevier, Barking, U. K., pp.55-159.

Hexcel Composites, 2003. www.hexcel.com.

Jang, B. Z., 1994, "Advanced Polymer Composites: Principles and Applications". ASM International, Materials Park, OH, USA, pp.167-198.

Larsson, F., 1997, "Damage Tolerance of a Stitiched Carbon/Epoxy Laminate", Composites: Part A, Vol. 28A, p. 923-934.

MEP 15-022, 2000, "Technical Specification of Embraer (Brazilian aeronautical industry)", pp.1-38.

Moura, M.F.S.F.; Marques, A.T., 2002, "Prediction of Low Velocity Impact Damage in Carbon-epoxy Laminates", Composites: Part A, Vol.33, pp.361-368.

Qi, B.; Herszberg, I., 1999, "An Engineering Approach for Predicting Residual Strength of Carbon/Epoxy Laminates after Impact and Hygrothermal Cycling", Composite Structures, Vol.47, pp.483-490.

Roy, R.; Sarkar, B.K.; Bose, N.R., 2001, "Impact Fatigue of Glass Fiber-Vinylester Resin Composites", Composites: Part A**,** Vol.32, pp. 871-876.

SACMA SRM2-88, 1988, "Recommended Test Method for Compression After Impact Properties of Oriented Fiber-Resin Composites", Suppliers of Advanced Composite Materials Association, pp. 1-10.

Schoeppner, G. A.; Abrate, S., 2000, "Delamination Threshold Loads for Low Velocity Impact on Composites Laminates", Composites: Part A., Vol.31, pp. 903-915.

Sjogren, A.; Krasnikovs, A.; Varna, J., 2001. "Experimental Determination of Elastic Properties of Impact Damage in Carbon Fibre/Epoxy Laminates". Composites: Part A, Vol.32, p.1237-1242.

Soutis, C.; Smith, F.C.; Matthews, F.L., 2000, "Predicting the Compressive Engineering Performance of Carbon Fibre-Reinforced Plastics", Composites: Part A, Vol.31, pp.531-536.

Thanomsilp, C., Hogg, P. J., 2003, Composites Science and Technology, Vol. 63, pp. 467-482.

Wu, K-W.; Lee, C-L.; Chang, Y-C.; Ong, C-L., 1996, "Compressive Strength of Delaminated and Repaired Composite Plates. Materials Science Comunication", Materials Chemistry and Physics, Vol.43, pp.173-177.

7. Responsibility notice

The authors are the only responsible for the printed material included in this paper.