CHARPY IMPACT TOUGHNESS OF CONVENTIONAL AND ADVANCED COMPOSITE LAMINATES FOR AIRCRAFT CONSTRUCTION

Gatti M. C. A., e-mail:mcgatti@sc.usp.br

Tarpani, J. R., e-mail: jrpan@sc.usp.br

Escola de Engenharia de São Carlos (EESC-USP) - Avenida Trabalhador São-Carlense 400, Centro, São Carlos - SP, CEP 13566-590, Brazil

Abstract. *A comparison has been made in a relative (i.e. density or weight) basis in regard to the translaminar Charpy impact toughness of conventional and advanced composite materials devised to aircraft fabrication. These are, respectively, carbon-epoxy (C-Ep) and TiGr (Titanium-Graphite) fiber-metal laminates. 5mm-thick three-point-bend specimens were tested within the temperature range of -70 to 180°C in order to simulate in-service conditions of subsonic and supersonic airliners. Energies required to damage initiation (E_i), propagation (E_p), and the whole fracture process ($E_t=E_i+E_p$), were evaluated at two distinct loading rates, namely, 2.25 and 5.52 m/s in an instrumented Charpy impact testing machine. Unidirectional cross-ply C-Ep laminates competed directly with TiGr hybrid material, whereas bi-directional fabric C-Ep materials exhibited poor performance. The obtained results have been corroborated in terms of fracture topology aspects of tested specimens. Charpy methodology was sensitive enough to clearly discriminate the dynamic fracture behavior of the studied materials.*

Keywords*: Airframe material, Charpy impact testing, fiber-metal laminate, translaminar dynamic fracture.*

1. OBJECTIVE

This prospective work aimed at comparing the translaminar impact toughness of an advanced hybrid fiber metal laminate and concurrent conventional carbon-epoxy laminates, in a weight or density basis, within a wide temperature range and under two loading rates.

2. INTRODUCTION

New technologies require the use of materials with properties not usually satisfied by conventional metallic alloys, especially in the aeronautical industry where high specific property (property/density ratio) is a premium. High structural efficiency leads to pay load maximization, fuel saving, extended fleet autonomy, among others advantages [1].

Composite materials have been claimed to be the best response to this demand, so that nowadays there is an increasing effort in their development.

Traditional carbon-epoxy (C-Ep) laminates, in which carbon fibers strengthen a relatively brittle epoxy matrix, still satisfy the basic requisites of subsonic aircraft designs.

However, in commercial supersonic aircrafts (e.g., High Speed Civil Transport Project [2]), in-flight temperatures as high as 180°C are expected in the fuselage, making the use of laminates C-Ep as well as of aluminum alloys impossible.

Therefore, in order to overcome this limitation, the Boeing Co. and the American Space Agency (NASA), have developed the hybrid fiber-metal laminate (FML) TiGr [3-6]. It is composed of titanium (Ti) foils interspersed with polymeric thermoplastic matrix PEEK (poli-ether-ether-ketone) reinforced with carbon fibers (Graphite). Typically, FML attends to requisites of high stiffness, superior mechanical resistance (especially to fatigue), high static fracture toughness, as well as a good capacity of absorbing transversal impact energies (trans-thickness impact), and high flame and corrosion resistances. These characteristics are highly desirable in modern aeronautical industry [7,8].

On the other hand, translaminar impact toughness of these hybrid materials in the presence of notch-like or crack-like defects has not yet been adequately appreciated.

Over the last decade, some studies [9-11] have been conducted in order to characterize the dynamic properties of structural composite laminates, especially the traditional C-Ep, making use of instrumented Charpy testing.

For instance, Zanetti [9] has recently carried out an experimental program to evaluate the dynamic Charpy toughness under impact rates ranging from 1 to 5.5 m/s in four classes of C-Ep laminates manufactured by Embraer S/A. His results are reproduced in the present study by aiming a direct comparison with TiGr laminate.

Actually, as far as the authors know, this has been a pioneer initiative to characterize the dynamic translaminar fracture in this new class of hybrid composite materials. Interestingly, impact loading rates provided by Charpy testing machines match exactly with vertical speeds of landing aircrafts (from operational to emergency conditions [12]), as well as to the so-called tail-strike event, in which, during landing or take off, the aircraft tail collides with the track, culminating, in the most serious cases, even in the loss of the aircraft [13,14].

According to [10], standard methodologies widely employed to characterize the dynamic fracture of monolithic metals and alloys seem to be perfectly applicable in composite materials, so that the uncertainties generated by the dynamic test on monolithic materials would also be inherent to the composite materials.

In the present work, the impact toughness of structural TiGr and C-Ep laminates of aeronautical use is compared in terms of damage initiation (E_i) and propagation (E_p) energies, where E_i and E_p comprise the whole energy spent in the global dynamic fracture process of the materials.

3. MATERIALS AND TEST SAMPLES

3.1 Hybrid Fiber-Metal Laminate TiGr

Hybrid laminate TiGr was produced in laboratory scale at the University of Liverpool-UK. The laminate is constituted of three sheets of commercial grade titanium, 0.85 mm-thick each, interspersed with two composite PEEK/C laminas, 1.10 mm-thick each. Each PEEK/C lamina is constituted of seven 0.14 mm-thick unidirectional tapes (fibers taking nominally 60% in volume of the composite). TiGr full-thickness was approximately 5 mm.

According to the well-established nomenclature for LMF, TiGr obeys the arrangement $3/2(0°)_7$. The consolidation of the final product was accomplished by hot compression in closed mold, followed by a controlled cooling rate. Carbon fibers and titanium sheets were laid in one direction only (0°), so that the Charpy notch was machined perpendicularly to the maximum mechanical resistance direction of the laminate. Figure 01 shows the final microstructure of TiGr after a thermal stress relief cycle is applied.

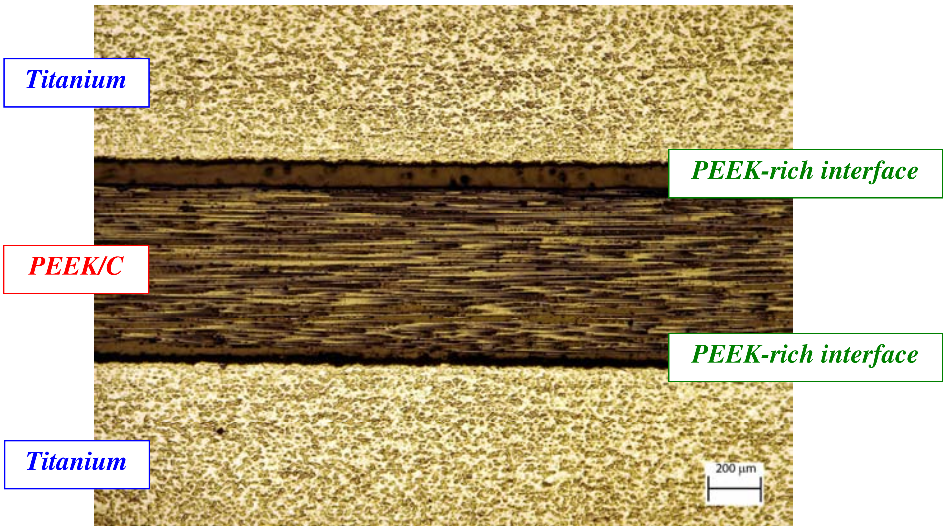


Figure 01 – Longitudinal cross-section of TiGr laminate in which its constituents are identified.

3.2 Carbon-Epoxy Laminates

Four types of carbon-epoxy composite laminates, previously cured in autoclave (vacuum bag) and supplied by Embraer S/A as semi-finished test specimens, were evaluated. In two laminates, twenty-eight unidirectional carbon tapes (TP) were disposed according to the angle-ply $[0/90]_{2S}$ lay-up, whereas fourteen layers of bidirectional eight-harness satin (HS8) weave fabrics (FA) followed the $[0/90]_4$ pattern in the other two laminates. Two different classes of epoxy resin were employed: a standard grade cured at 120°C / 420 kPa, and a rubber-toughened grade cured at 180°C / 700 kPa. The final thickness of the laminates was nearly 5 mm, therefore practically identical to the TiGr. The nomenclature adopted for these materials is as follows:

- TP120 = Tape cured at 120°C
- TP180 = Tape cured at 180°C
- FA120 = Fabrics cured at 120°C
- FA180 = Fabrics cured at 180°C

Typical optical micrographs of cross-sections of the consolidated composite laminates are shown in Figure 02.

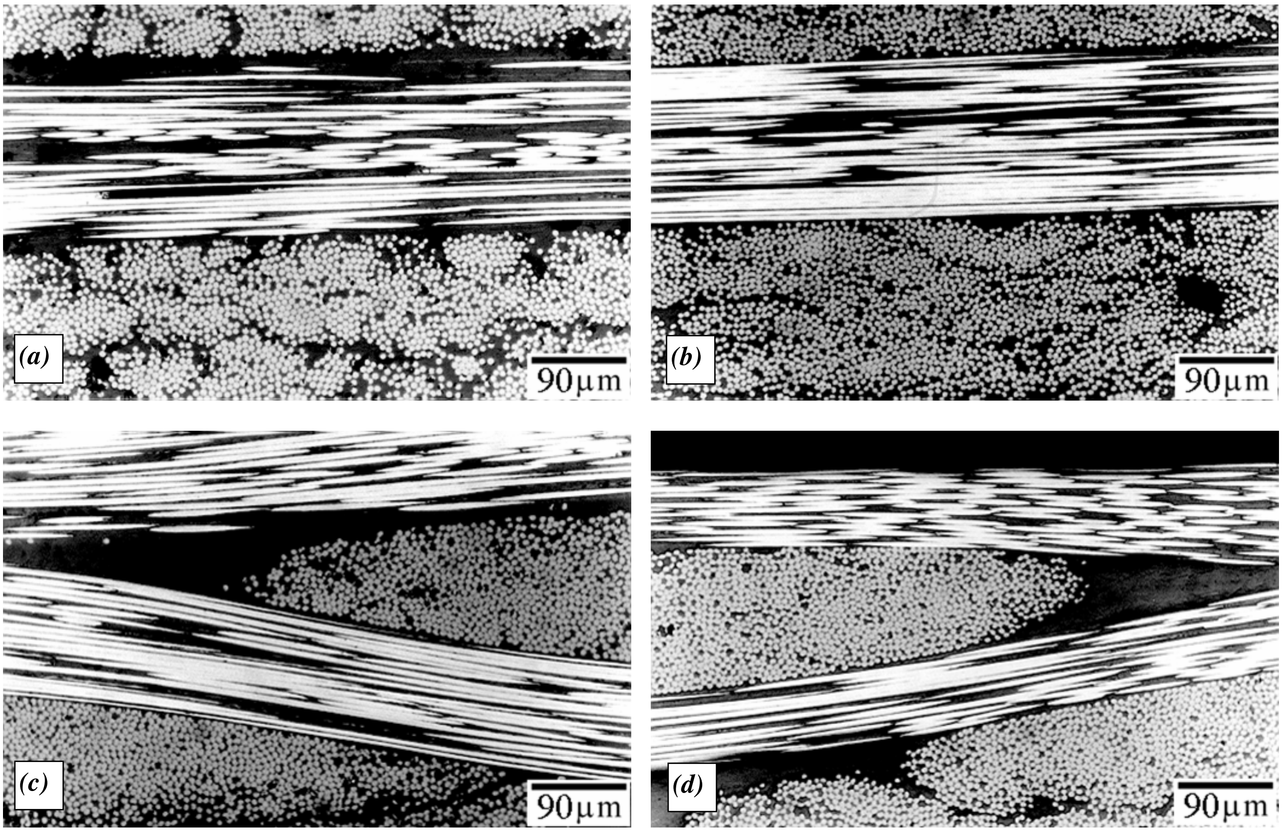


Figure 02 – Microstructure of the C-Ep laminates: (a) TP120, (b) TP180, (c) FA120, and (d) FA180.

3.3 Test Specimens

TiGr testpieces were extracted from the original plates by means of water jet cut, while, in the case of the C-Ep laminates, the specimens were provided by Embraer S/A already in their ready-to-test size.

Nominal dimensions of the test specimens (Figure 03) were of $5x10x55mm^3$. Notch machining was accomplished by a means of a 0.5mm-thick diamond disk, under low speed rotation and water refrigeration. Notch positioning was selected in order to enforce the composite laminates to fracture in a translaminar manner, i.e., full-thickness specimens.

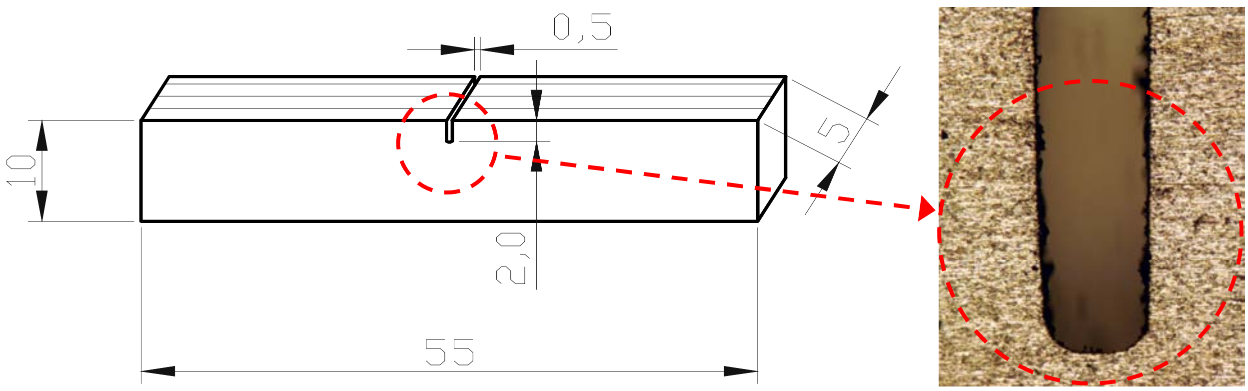


Figure 03 – Charpy impact specimen geometry according to the ASTM D5045 standard [15] (sizes in mm). The approximately 0.25 mm root radius of a notch in a TiGr specimen is shown in detail.

4. EXPERIMENTAL

Instrumented Charpy impact tests were conducted in an Instron-Wolpert® PW30 system with a maximum full-scale of 300 Joules (5.5 m/s hammer speed), integrated to a microcomputer operating with Instron® and National Instruments® software to read, treat and interpret finals results.

The impact system hammer is instrumented with extensometers that register the load signal (P) in an oscilloscope, enabling the visualization of load (P) variation with time (t). The displacement or specimen load-line deflection (s) is monitored by an optical transducer.

P-t and P-s graphs supply precise information on yield and ultimate loads, onset and arrest of unstable cracks, among other parameters.

The partial energies absorbed in the initiation (E_i) and in the propagation (E_p) of the crack (generally speaking: damage) during the impact, were estimated by integrating the P-s curve up to maximum load and after maximum load, respectively, as illustrated in Figure 04.

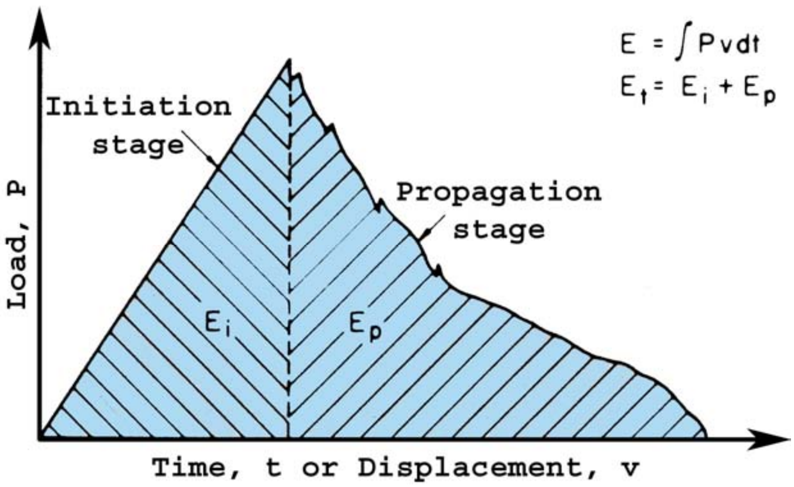


Figure 04 – The two stages of the instrumented Charpy impact test curve [9].

Thus, the total absorbed energy in the complete fracture process (E_t) is comprised of two parts (E_t=E_i+E_p). E_t values were automatically corrected for losses due to friction between specimen and the anvil, air resistance, as well as fractured specimen tossing. The obtained values of net energy were then divided (i.e., normalized) by the respective densities (specific weight) of the five evaluated materials, which are listed in the Table 01, so that specific impact toughnesses in a weight basis were derived, as it is preferred in aeronautical applications.

Original oscillations of the load signals were filtrated for their elimination, or minimization, by means of a Matlab routine, via Mobile Average method.

Charpy impact tests were carried out for TiGr-FML at -196°C, -70°C, 23°C and 180°C, respectively, at two different impact speeds, respectively of 2.25 m/s and 5.52 m/s. The dynamic tests of the C-Ep laminates were performed at -70°C, 23°C and 100°C, only, under the same strain rates as for TiGr.

Table 01 – Relative density of the several tested laminates.

Laminate	Specific weight (in respect to water)
TiGr	3.095
TP120	1.492
TP180	1.460
FA120	1.472
FA180	1.442

5. RESULTS AND DISCUSSION

Figure 05 plots total absorbed energy (E_t) and its two parcels, namely, initiation (E_i) and propagation (E_p) energies, considering the various test temperatures and applied loading rates. From these graphs the following conclusions can be drawn, which have been separated in topics in order to simplify the reader's task:

5.1 General Behavior

5.1.1 Global toughness (E_t: Figs.05a e 05b)

- TiGr and C-Ep TP laminates are tougher than C-Ep FA ones. It can be argued that TiGr's good performance is related to the presence of both metallic and thermoplastic phases, which are intrinsically ductile and tough, not to mention the complete alignment of the strengthening carbon fibers in the direction of the applied loads (0°). In fact, C-Ep TP laminates compete directly with TiGr at -70 and +100°C, under a loading rate of 2.25 m/s, while they are beat by the latter at the intermediate temperature of +25°C. Previous work [16] showed that at -70°C residual thermal stresses arising from differential contraction between two consecutive plies (0° and 90°, respectively) in C-Ep TP architectures facilitates delamination, which constitute a powerful mechanism of energy consumption; on the other hand, at +100°C epoxy resin softening plays a fundamental role in donating plasticity to the laminate, originating, therefore, an extra toughness. Under a loading rate of 5.52 m/s, both TiGr and C-Ep TP laminates show practically the same level of mechanical performance at room temperature;

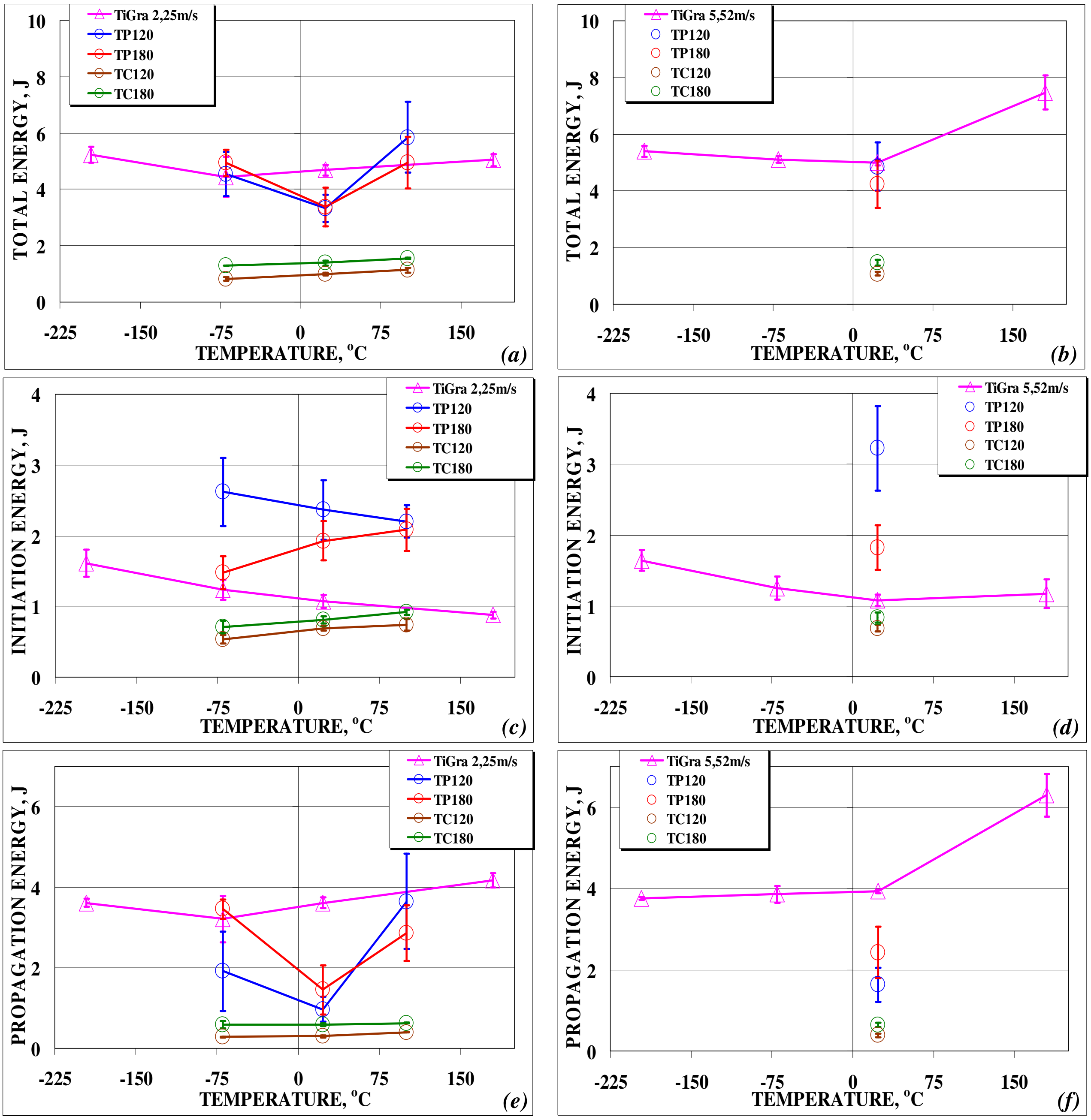


Figure 05 – Test results at different temperatures and applied loading rates for the studied laminates: (a) and (b) Total absorbed energy - E_t; (c) and (d) Initiation energy - E_i; (e) and (f) Propagation energy - E_p.

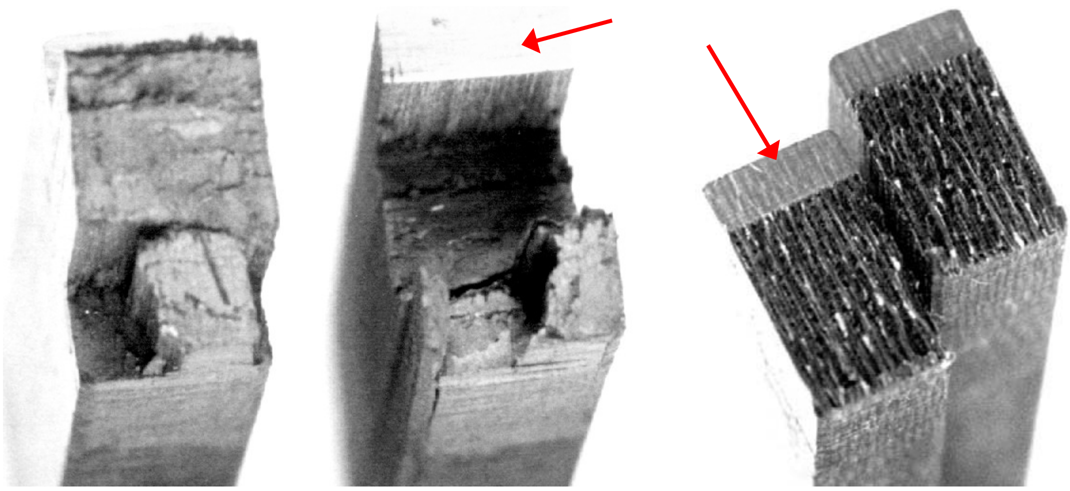
- C-Ep TP laminates are much tougher than FA ones. This is probably due to the existence of many more interfaces in the former materials (27 against 13 of FA arrays), which are therefore much more prone to delamination, which constitutes a powerful toughening mechanism. E.g., Figure 06 exhibits the noteworthy differences between topological aspects of fracture in TP and FA laminates;
- The nature of the employed epoxy resin affects only slightly the mechanical behavior of the C-Ep laminate, and its influence seems to depend on the arrangement of the reinforcing fibers.

5.1.2 Initiation toughness (E_i: Figs.05c and 05d)

- C-Ep TP laminates invariably present higher values than fabric architectures and hybrid fiber-metal TiGr;
- As for global tenacity, E_t, C-Ep FA laminates TC are the least tough (or the most brittle) in terms of crack/damage initiation;
- The epoxy resin affects much more significantly the TP array than the FA architecture. Likewise observed for E_t, the non-toughened resin, cured at 120°C, seems to affect positively the TP performance, whereas the rubber toughened resin, cured at 180°C, is more beneficial to the FA laminate. To some extent, this may be closely connected to number of interfaces present in one and another type of C-Ep laminate.

5.1.3 Propagation toughness (E_p: Figs.05e and 05f)

- The crack propagation energy of TiGr hybrid laminate is always superior to its damage initiation energy, while for the C-Ep laminates no rule has been found regarding the proportion between E_i and E_p;
- The propagation energy of TiGr is generally superior to the C-Ep laminates. In the case of TiGr, by virtue of the similarity of the magnitude order of E_p (with $E_p >> E_i$) and E_t (see Fig.05a) parameters, as well as the corresponding graphic curve shapes, it may be argued that the propagation stage controls the global tenacity of the hybrid laminate. This is usually observed in metals and ductile alloys, indicating that the presence of titanium and of thermoplastic phases are the main responsible for this behavior in TiGr;
- In absolute terms, the effect of the epoxy resin in the C-Ep laminates is almost insignificant for the FA array, while it is much more evident in the TP architecture;
- C-Ep TP laminates perform better than FA ones; again, and very likely, it is related in some extent to the higher potential for delamination exhibited for the former materials. In fact, as shown in Figure 07, the noticeable ability of TP laminates for fracturing along translaminar planes oriented perpendicularly machined notch must also be considered, as long as it demands large energy apportion as well.



(a) (b)

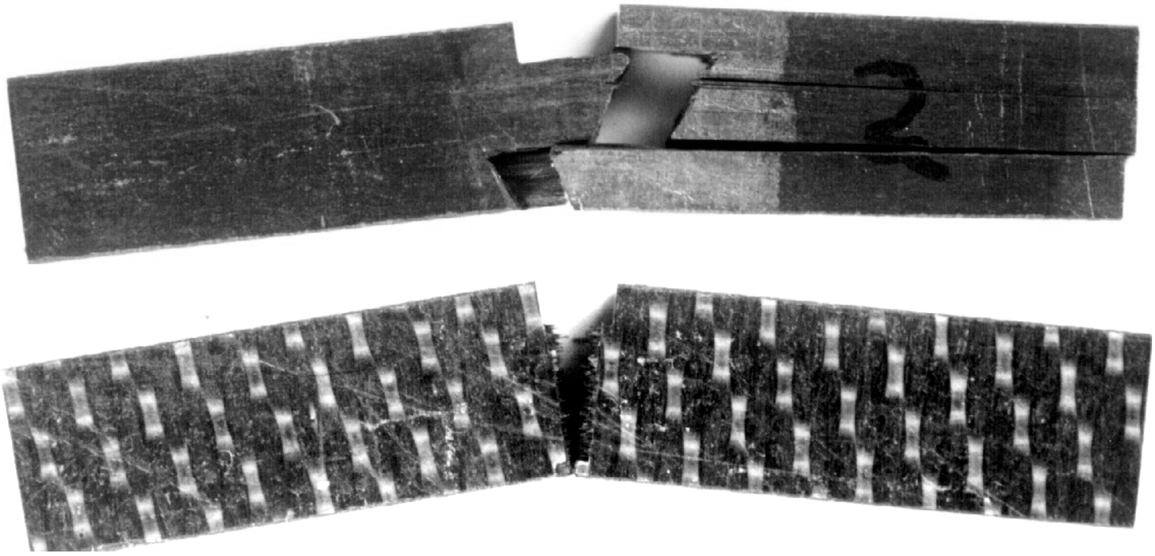
Figure 06 – Fracture surfaces of C-Ep laminates tested under identical conditions: (a) Tape, and (b) Fabric array of reinforcing carbon fibers. Note the flat topology in (b), signalizing a very low capacity of energy absorption during the specimen rupture.

5.2 Test Temperature Effect

5.2.1 Loading rate of 2.25 m/s

5.2.1.1 Global toughness (E_t: Fig.05a)

- TiGr laminate can be considered practically insensitive to thermal changes within the expected temperature envelop of operational service conditions;
- C-Ep FA laminates seem to be very little affected by this variable as well;
- C-EP TP laminates, however, display an atypical behavior, in which at -70°C and 100°C its global toughness is higher than at the intermediate temperature of 25°C. As it has been previously referred to, at -70°C the differential thermal contraction between two adjacent plies orthogonally oriented to each other gives rise to residual stresses that facilitate delamination, therefore increasing the capacity of the material to absorbed energy. Figure 08 corroborates this assumption. On the other hand, at 100°C, one may argue that a toughening mechanism relying on epoxy matrix softening become active, especially in that one cured at 120°C, promoting a certain degree of ductility in the composite laminate as a whole.



(a)

(b)

Figure 07 – Laminates TP120 (a) and TC120 (b) tested under identical conditions and exhibiting quite different cracking patterns.

Figure 08 – Tespiece halves of TP180 laminate tested at -70°C, where numerous delamination planes were induced by differential thermal contraction in-between simultaneous composite plies.

5.2.1.2 Initiation toughness (E_i: Fig.05c)

- As temperature is risen, the absorbed energy decreases in identical proportion for TiGr and C-Ep TP120 laminates, while for others C-Ep materials the opposite trend is clearly noticed. It is difficult, at this stage of the work, to infer the reason why materials behave this way. Efforts must, therefore, be undertaken in this regard.

5.2.1.3 Propagation toughness (E_p: Fig.05e)

- There is a remarkable tendency of consumed energy to increase as the temperature is increased, contrary to what is observed for the initiation energy. This is an indicative that, at least for FML, distinct mechanisms govern one and another of these two complementary fracture processes;
- For C-Ep laminates, as a result of the great similarity between the curve shapes and magnitude order of parameters E_p (Fig.05e) and E_t (Fig.05a), it is possible to conclude that, as already done for TiGr laminate, the propagation stage largely controls the global toughness, at expenses of the initiation stage. In this sense, the reasoning previously developed to explain the minimum peak attained by the global tenacity (E_t) at ambient temperature, comparatively to the obtained values at -70°C and 100°C, respectively, also seems be fully valid for E_p.

5.2.2 Loading rate of 5.52 m/s

5.2.2.1 Global toughness (E_t: Fig.05b)

- For TiGr laminate, this parameter is practically independent of temperature until about 25°C, from which there is an abrupt increment of this variable up to 180°C, which corresponds to the maximum temperature expected for this material application in real service conditions.

5.2.2.2 Initiation toughness (E_i: Fig.05d)

- For TiGr-FML, there is a decrease in the initiation energy up to room temperature; from this point on, E_i trends to stabilization.

5.2.2.3 Propagation toughness (E_p: Fig.05f)

- For TiGr hybrid laminate, E_p displays the same tendency observed for the total energy (E_t), shown in Fig.05b. The similarity between E_p and E_t was already related to the predominance of the crack propagation stage in the global process of dynamic fracture of this FML material;
- For both evaluated loading rates, TiGr's E_p/E_i ratio varies from about 2.5 to 5 as the temperature increases from -70°C to +180°C. For C-Ep laminates, this relationship is most frequently lower than the unit, with the initiation stage controlling the dynamic fracture process. However, as the temperature approaches 100°C, the propagation stage begins to dominate it, so that the E_p/E_i ratio assumes higher values than, but still next to 1.

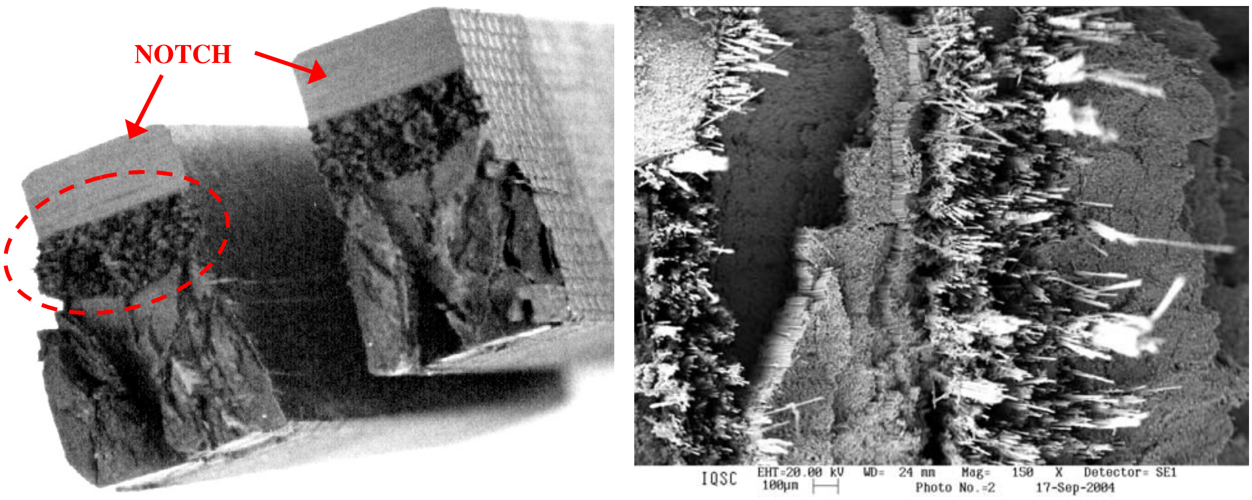
5.3 Loading Rate Effect

5.3.1 Global toughness (E_t: Figs.05a and 05b)

- TiGr is clearly sensitive to the applied loading rate at temperatures above 25°C, when a significant increase in E_t is observed. This is a typical behavior of ductile materials, whose fracture process is strain-controlled, and is in full agreement with the presence of titanium and thermoplastic PEEK in the FML microstructure;
- Regarding most of the C-Ep laminates studied, nothing can be stated in respect to the loading rate effect, since at room temperature the standard-deviation ranges at the two impact velocities are overlapped. Exception is the TP120 laminate, which seems to be favorably affected by higher applied strain rates.

5.3.2 Initiation toughness (E_i: Figs.05c and 05d)

- For TiGr laminate, a beneficial effect of increasing loading rate in E_i can be noted at temperatures above 25°C, resembling previous analysis of E_t (Figs.05a and 05b);
- C-Ep TP laminates are better than TiGr-FML regarding the criterion E_i. As earlier noticed for TiGr at temperatures above 25°C, TP120 laminate is favorably affected by a temperature increase, whereas the temperature influence on TP180 laminate is insignificant, if any. Figure 09 shows typical fracture features consisting of fiber debonding and fiber pullout close to the notch root in a C-Ep TP specimen, which strongly corroborate high crack initiation toughness values experimentally determined for this reinforcing fiber arrangement.



(a) (b)

Figure 09 – Fracture surfaces of TP180 laminate tested at 100°C: (a) General view; (b) Localized view by using scanning electronic microscopy techniques.

5.3.3 Propagation toughness (E_p: Figs.05e and 05f)

- For TiGr laminate, a positive effect of increasing loading rate at temperatures above the ambient can be noticed. This finding is analogous to previous results on E_t (Figs.05a and 05b) and E_i (Figs.05c and 05d) for the same material;
- C-Ep TP laminates appear to be positively affected by higher loading rates, whereas for FA composites no statement is possible in this regard since low absolute results and just small relative shifts in E_p values were observed by varying the impact velocity.

6. CLOSING REMARKS

In this prospective work, the translaminar Charpy impact toughness of advanced metal-fiber hybrid laminate (TiGr) and conventional carbon-epoxy laminates (C-Ep) has been compared in a specific (weight) basis for two distinct applied loading rates within a wide range of temperatures. The following main conclusions have been drawn:

- For temperatures ranging from -70°C to 100°C, C-Ep TP architecture competes directly with TiGr-FML when the global tenacity behavior (initiation + propagation stages) is considered;
- If damage initiation and growth stages are separately considered, C-Ep TP laminates surpass TiGr performance in terms of initiation criterion (damage resistance approach), while the former are correspondently surpassed by the latter, and in identical proportion, in terms of propagation criterion (damage tolerance approach);
- Among the four tested C-Ep laminates widely employed in primary and secondary subsonic commercial aircraft structures (except wings and fuselages), unidirectional cross-ply tape (TP) array cured at 120°C generally exhibited the best results. Interestingly, this is the cheapest laminate presently evaluated;
- In general, a tendency has been noted on increasing impact toughness by raising both temperature and loading rate, a behavior which is compatible with very most structural engineering materials;
- Instrumented Charpy impact methodology has successfully differentiated the translaminar fracture behavior of several studied laminates, so that it can be very useful to select structural composite materials for which resistance and tolerance to dynamic loads may be determinant to aircraft design;
- C-Ep FA laminates displayed a very disappointing translaminar Charpy impact performance.

Acknowledgements: To Capes for the Doctorate scholarship of M.C.A.G., Embraer S/A for supplying the carbon-epoxy laminates, Prof. James Wesley Cantwell for the facilities made available for the preparation of metal-fiber TiGr-FML in the Laboratory of Composite Materials at the University of Liverpool (UK), the post-graduated student Douglas Serpa Zanetti from Materials Science and Engineering Program at Interunidades-USP, and the undergraduate student Guilherme A. Teti from Aeronautical Engineering Course at SMM-EESC-USP.

7. REFERENCES

[1] NIU M.C.Y. (1992) Composite Airframe Structures, Hong Kong Conmilit Press Ltd.
[2] HIGH SPEED CIVIL TRANSPORT (HSCT) Project (2004) NASA Report.
[3] BURIANEK D.A., GIANNAKOPOULOS A.E., SPEARING S.M. (2003) Modeling of facesheet crack growth in titanium-graphite hybrid laminates - part II: experimental results, Engng Fract. Mech., v.70, p.799-812.
[4] RHYMER D.W., JOHNSON W.S. (2002) Fatigue damage mechanisms in advanced hybrid titanium composite laminates, Int. J. Fat., v.24, p.995-1001.
[5] CORTÉS P., CANTWELL W.J. (2002) Interfacial fracture properties of carbon fiber reinforced PEEK/titanium fiber-metal laminates, J. Mater. Sci. Lett., v.21, p.1819-1823.
[6] LI E., JOHNSON W.S. (1998) An investigation Into the fatigue of a hybrid titanium composite laminate, J. Comp. Tech. Res., v.20, p.3-12.
[7] VLOT A., VOGELESANG L.B., de VRIES T.J. (2001) Towards application of fiber-metal laminates in large aircraft, <www.glareconference.com>.
[8] VLOT A., KROON E., LA ROCCA G. (1998) Impact response of fiber metal laminates, Key Engng Mater., v.141-143, p.235-276.
[9] ZANETTI D. (2005) Tenacidade à fratura translaminar dinâmica de laminados compostos carbono-epóxi de grau aeronáutico. Dissertação de Mestrado. Escola de Engenharia de São Carlos, Universidade de São Paulo.
[10] FERNANDEZ-CANTELLI A., ARGÜELES A., VIÑA J., RAMULU M., KOBAYASHI A.S. (2002) Dynamic fracture toughness measurements in composites by instrumented Charpy testing: influence of aging, Comp. Sci. Tech., v.62, p.1315-1325.
[11] BURZIC Z. (2001) The effect of fiber orientation on impact toughness and fracture properties of carbon fiber-epoxy composite, In: Charpy Centenary Conference, Poitiers, France.
[12] ALVES M., BIRCH R. (2003) Impact on aircraft, Relatório interno Embraer S/A.
[13] Http://aviation-safety.net/database/record.php?id=19850812-1.
[14] Http://aviation-safety.net/database/record.php?id=20020525-0.

[15] ASTM D5045-96 (2001) Standard Test Methods for Plane-Strain Fracture Toughness and Strain Energy Release Rate of Plastic Materials, American Society for Testing and Materials Handbook.
[16] GATTI M.C.A., TARPANI J.R (2007) Desempenho sob impacto Charpy de laminados compostos convencionais e avançados da indústria aeronáutica, In: 62° Congresso Internacional da ABM, Julho 2007, Vitória-ES, Brasil, Anais, 2007, 1CD.

8. RESPONSIBILITY NOTICE

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