THE RIB-SKIN INTERFACE FRACTURE TOUGHNESS OF A STIFFENED COMPOSITE SHELL

RIBOTO KOMPOZĪTU ČAULU RIBAS UN APŠUVUMA STARPSLĀŅU PLĪSUMA STIPRĪBA

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INTRODUCTION

With the increased emphasis on reducing the cost of manufacturing composite structures, secondary bonding or co-curing is an attractive option to eliminate the need for mechanically fastened subassemblies. Many composite components in aerospace structures and building constructions consist of flat or curved panels with co-cured frames and stiffeners. Out-of-plane loading such as internal pressure in a composite fuselage or out-of-plane deformations in compression-loaded post-buckled panel may lead to debonding of the frame or stiffener from the panel [1].

In polymer matrix composites, progressive type failures are often observed where catastrophic failure is generally preceded by constituent level damage accumulation. Some investigations efforts propose progressive failure algorithms aimed at capturing the progressive failure process from the initial to final failure [2, 3, 4].

The possibility of accurately predicting the damage and progressive failure in structures produced from composite materials is an important task in the design of lightweight structures as in the aircraft as in the civil engineering industries.

The interlaminar delamination fracture toughness of composite rib-stiffened-shell elements made of carbon/epoxy materials has been investigated. DCB (double cantilever beam) tests and pull-off tests were carried out for this purpose. The knowledge of experimentally observed failure mechanisms is needed to determine the constituent ultimate strength from the micromechanics-based volume averaged stress values.

MATERIALS AND METHODS

Specimens

DCB and pull-off specimens were prepared from the co-cured rib stiffened multilayered shell with radius 938 mm manufactured from CFRP composite material. The symmetric skin lay-up was (0, -45, 45, 90). The stiffener was made of 24 plies of the same material with following sequences (0, 0, 45,-45, 0, 0, 45, -45, 0, 0, 45, -45). The panels were made by one-shot co-curing technology, base panel and stiffener together.

Teflon film insert has been incorporated between rib and shell skin on the one edge of the specimen for DCB test. The specimens have been cut from manufactured stiffened shell. Dimensions of specimens are follows: width b = 100±0.5 mm and length L=250±1.0 mm. DCB tests were performed in order to obtain the mode I delamination fracture toughness at the interface of the present laminated composite material [5, 6].

The pull-off specimens were cut with diamond-tipped tool from manufactured stiffened shell in the form of a co-cured rib-stiffened multilayer shell.

The skin width of the specimens was b = 100±0.5 mm, the width of a glued rib was 60±0.5 mm, and the panel length L = 40±0.5 mm. The thicknesses of the skin and the rib of specimens were respectively 1 and 3 mm. The glued thickness of the rib changed from 1.5 mm near the rib to 0.5 mm at the tip.

Fig.1 Universal testing system Zwick Z100 with temperature chamber

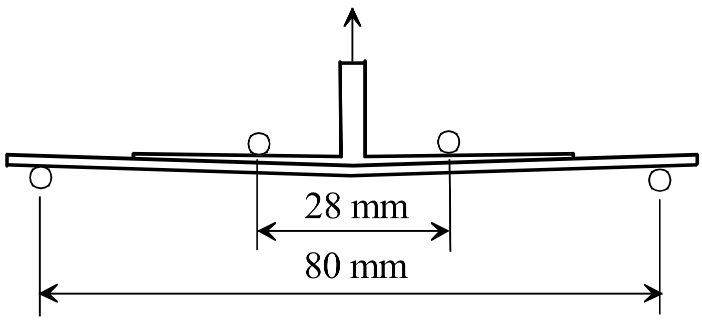


Fig. 2. Boundary conditions of pull-off test

Mechanical experiments

All pull-off tests were performed on a Zwick Z100 universal testing system (Fig.1). The tests were done in a ZWICK test system with temperature controllable chamber. The relative humidity level in the laboratory was between 52-55%.

The constant loading rate was 2.0 mm/min during each test, ether at room temperature, or at high temperatures. The limit load P_c and critical displacement (deflection) δ at midspan of the specimen were measured at the moment of the crack extension. The boundary conditions are shown in Fig. 2.

DCB test for the rib-stiffened panel is performed in the same manner as the standard DCB test [7]. Specially manufactured, to accommodate the panel curvature, aluminium loading blocks are glued both to the panel skin and stiffener flanges, see Fig. 3.

Stiffener flange-skin interface is painted white to improve vision monitoring of the crack growth, metric scale is glued on the both edges of the skin to measure crack increment distance. The distance of the crack increments of 20 mm during round-robin testing is used to measure critical loads and grip displacements. The loading constant rate was 1.0 mm/min during each test at room temperature. The loading scheme of DCB test is shown in Fig. 4.

DSC analysis

The DSC analysis has been made for the investigation of materials of skin and rib. Specimens were cooled till temperature -100°C in order to achieve well exposed glass transition temperatures peak for the composite matrix. These definitions of glass transition temperature allowed making a choice for the testing temperatures of composite specimens. Specimens were heated till temperature +300 °C. Cooling and heating rate was 10°C/min. Possible matrix polymer oxidation peak is observed on the DSC curves (Fig.5-7). According to DSC data five levels of temperatures have been chosen for mechanical investigations: 20º, 50 º, 100 º, 170 º and 250 º C.

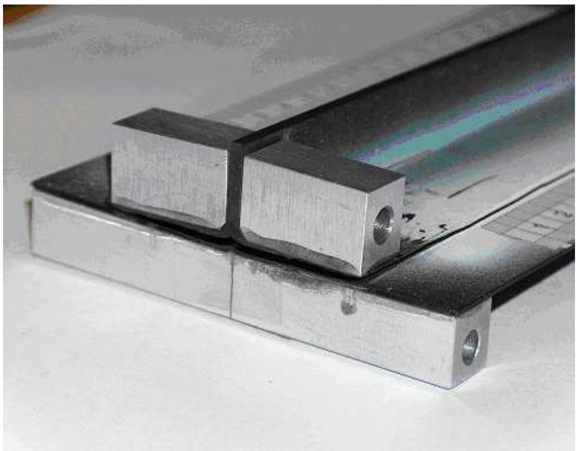


Fig.3. Ribstiffened shell specimen prepared for DCB test

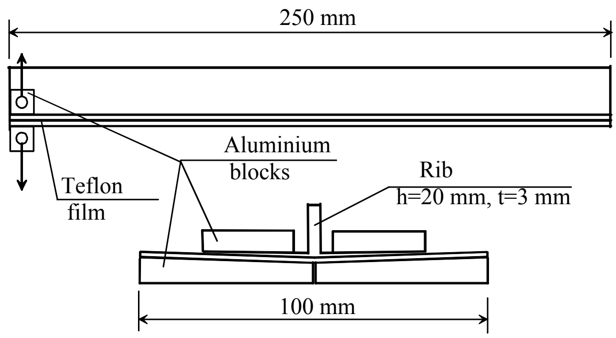
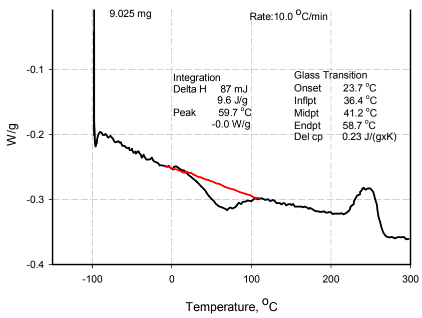


Fig. 4. Loading sheme of DCB test with dimensions of specimen



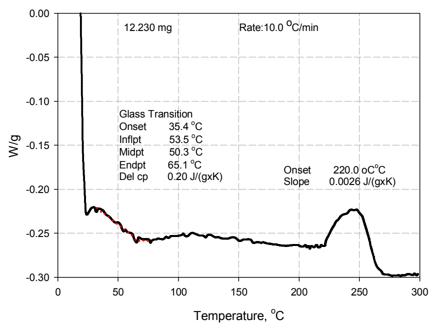


Fig.5. DSC analysis of skin's polymer *Fig.6. DSC analysis of*

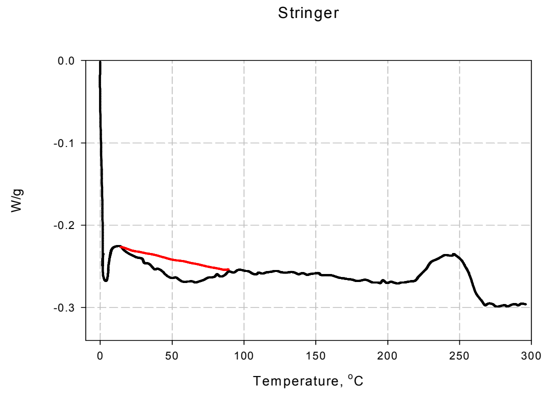


Fig 7. DSC analysis of rib's polymer

RESULTS AND DISCUSSION

Test procedure and failure modes

The pull-off test results of co-cured joints were analyzed in different temperature ranges to understand the failure mode and strength of attaching the frames and bulkheads to the fuselage skin.

The form of fracture in the zone of joining between the rib and skin was obtained by experimental investigation of specimens and are typical for all range of temperatures (Fig.8-11).

The DCB tests were performed in order to obtain the mode I delamination fracture toughness at the interface of the laminated composite material. The experiments have been performed using standard method elaborated for unidirectional laminated composite materials.

Pull-off tests

Results of pull-off tests at the different temperatures 20^{o}C, 50^{o}C, 100^{o}C, 170^{o}C, and 250^{o}C are presented in the Table 1. The mean value of ultimate load is decreasing with increasing of temperature, excluding 170^{o}C, when first ultimate load isreached for the composite.

Typical load/displacement curves are presented in Fig.12-16 for all range of temperatures. The initial audible and visual crackings are in according to ultimate load and can be seen in Fig.12-16. Displacement-temperature curve, obtained from the tests results is presented in Fig.17 and tab.1.

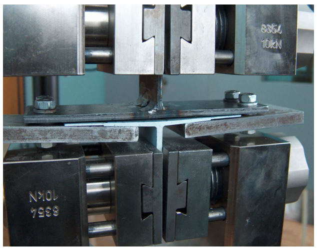


Fig.8. Beginning of the test

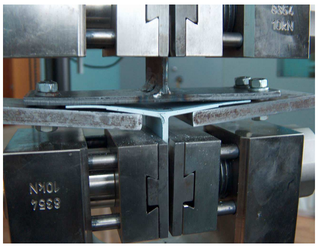


Fig.9. 2nd step of the test

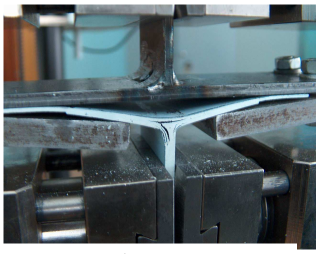


Fig.10. 3rd step of the test

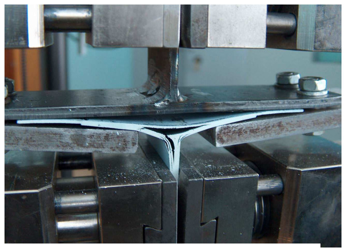


Fig.11. 4th step of the test

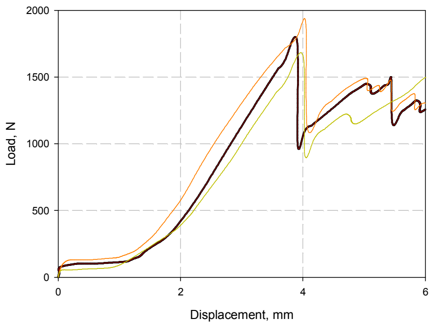


Fig.12. Load/displacement curve, 20°C

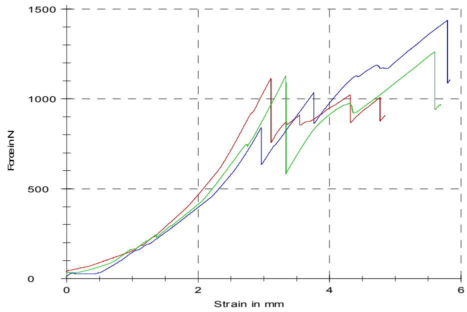


Fig.13. Load/displacement curve, 50°C

Table 1. Load mean values dependence on temperature.

Temperature	20°C	50°C	100°C	170°C	250°C
Load (mean value), N	1569	1232	793	1134	354

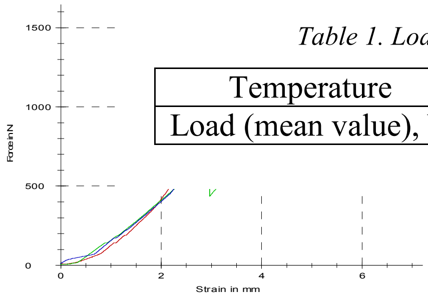


Fig.14. Load/displacement curve, 100°C

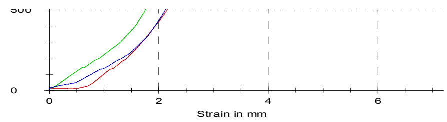


Fig.15. Load/displacement curve, 170°C

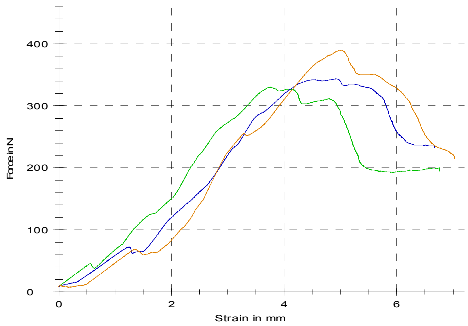


Fig.16. Load/displacement curve, 250°C

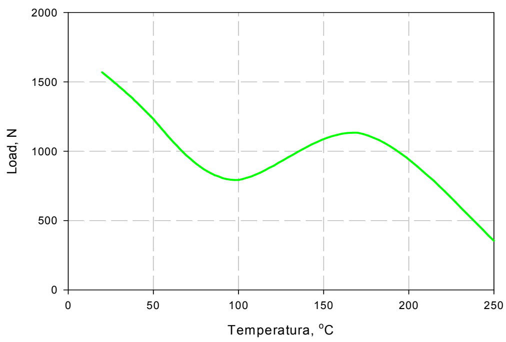


Fig.17. Ultimate load/temperature curve

DCB test

Typical load-displacement curve obtained during round-robin test of the rib-stiffened panels is shown on Fig. 18. It is similar to the standard DCB test by nature for the unidirectional specimens.

Critical energy release rates for the rib-stiffened panels are calculated as for standard DCB test employing Berry's method (1).

$$G_I = \frac{nP\delta}{2ba} \qquad (1)$$

Where P is load in N, δ is displacement in mm, a is length of crack in mm and b – width of specimens in mm, that have been taken equal to front of crack 60 mm.

Values of the calculated energy release rates of the crack initiation and propagation calculated by with n=2.67 (1) are presented in Table 2. Critical energy release rate versus crack growth for the average values is shown on Fig. 19.

The curve has been shown for the range from 0.92 till 1.12 kJ/m^2 of critical energy release rate for better visualisation of differences between initial critical energy release rate and propagation critical energy release rate.

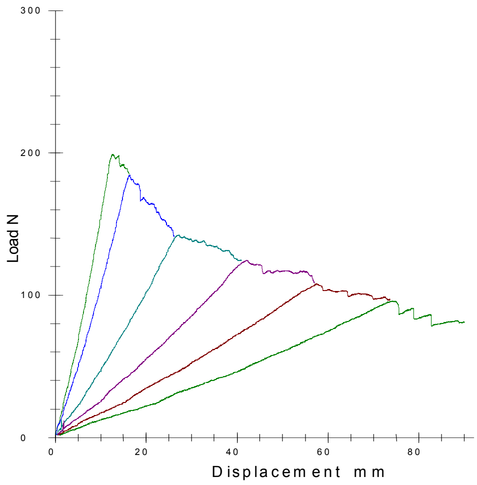


Fig. 18. Load-displacement curve of the DCB test

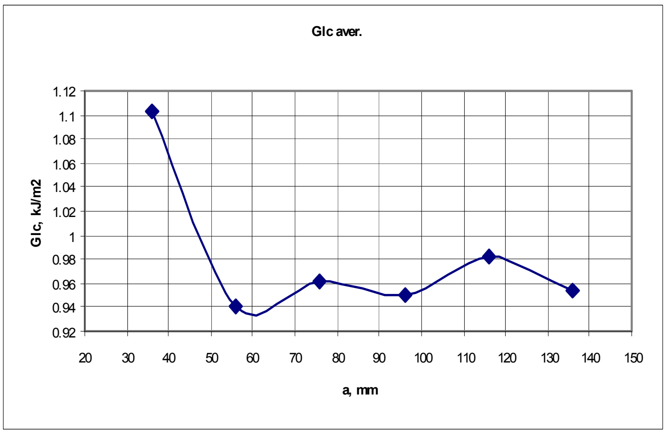


Fig. 19. Critical energy release rate versus crack length

Table 2. Values of the critical energy release rates for the rib-stiffened panels

a	G_I, kJ/m2								
mm	1	2	3	4	5	6	7	8	Average
36	1.054	0.704	0.990	1.285	1.141	0.947	1.019	1.152	1.103
56	0.736	0.459	0.796	0.946	0.931	1.117	0.860	1.144	0.940
76	0.784	0.429	0.802	0.888	0.947	0.931	0.875	1.138	0.961
96	0.790	0.408	0.777	0.951	0.878	0.903	0.852	1.111	0.950
116	0.877	0.406	0.741	0.935	0.823	0.949	0.789	1.087	0.982
136	0.790	0.362	0.674	0.911	0.935	0.881	0.977	1.118	0.954

CONCLUSIONS

Mode I interlaminar fracture toughness properties for interface between rib flanges and shell skin of carbon/epoxy material has been obtained. Initial critical release rate value is a little bit higher than other values and this can be explained by the absence of pre-crack into the specimens. Other factors can be incorrect using of the critical energy release rate extraction elaborated for the standard unidirectional specimens in the case of stiffened panels, differences in manufacturing conditions of the unidirectional laminate and rib-stiffened panels, specimen symmetry (for the rib-stiffened panels there are observed crack front rotation from the perpendicular of the stiffener, explained by the presence of the nearest to the crack interface plane 45^0 layer).

The fiber bridging was not observed in the specimens. This can be explained by the use of the prepreg manufacturing technology. The fracture of specimens was caused by the complex stress conditions between the layers at the place of rib curvature.

The analysis of T pull-off test of stiffened shell had shown the step by step failure of detail after ultimate load. The failure is caused by tension stress between unidirectional layers.

The weak point of the present structure can be insufficient interlaminar toughness of the stiffener web.Skin-stiffener separation test can be modeled by FEM analysis, but still hardly dependent of the real structure material properties, which are the kind of unknown variables in current configuration. The complicated laminate structures can be modeled and analyzed by computer code for prediction of ultimate load and fracture toughness.

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Korjakins A., Ozoliņš O., Pizele D. Riboto kompozītu čaulu ribas un apšuvuma starpslāņu plīsuma stiprība.
Oglekļa šķiedras un epoksīdsveķu kompozītos bieži tiek novēroti progresējošie bojājumi, kuri pirms katastrofālās sabrukšanas pakāpeniski tiek akumulēti defekti. Normāla slodze pret fizelāžas virsmu, kas ir veidota no kompozīta materiāla, ir tāda pati kā iekšējais spiediens vai deformācijas panelī pie spiedes var novest pie atslāņošanas starp paneli un ribām. Iespēja precīzi prognozēt defektus un sabrukuma attīstīšanos kompozītu materiālu struktūrā ir svarīgs uzdevums vieglo konstrukciju projektēšanā. Šīm mērķim tika veikti konsoles sijas ribas atslāņošanas pārbaude un T-veida tests uz atslāņošanos stiepē ar T-veida stiprināšanu. Izskatīta temperatūras izmaiņas ietekme uz atslāņošanos. Iegūta starpslāņu sabrukšanas stigrība starp apšuvuma paneli un stinguma ribām, veidotiem no kompozītmateriāliem. Detalizēti, soli pa solim parādīta paraugu sabrukšana ar T-veida stiprināšanu stiepē līdz kritiskās slodzes sasniegšanai.

Korjakins A., Ozolins O., Pizele D. The rib-skin interface fracture toughness of a stiffened composite shell.
Progressive type failures are often observed in carbon-epoxy composites, where catastrophic failure is generally preceded by constituent level damage accumulation. Out-of-plane loading such as internal pressure in a composite fuselage or out-of-plane deformations in compression-loaded post-buckled panel may lead to debonding of the frame or stiffener from the panel. The possibility of accurately predicting the damage and progressive failure in structures produced from composite materials is an important task in the design of lightweight structures. DCB tests and pull-off tests were carried out for this purpose. Temperature dependence measurements have been considered. Mode I interlaminar fracture toughness properties for interface between rib flanges and shell skin of carbon/epoxy material has been obtained. The analysis of T pull-off test of stiffened shell had shown the step by step failure of detail after ultimate load.

Корякин А, Озолиньш О, Пизеле Д. Межслойная вязкость разрушения ребристой композитной оболочки между ребром и обшивкой.
Прогрессирующие повреждения часто наблюдаются в угле-эпоксидных композитах, где катастрофическому разрушению часто предшествует постепенное накапливание дефектов. Нагрузка, нормальная к поверхности фюзеляжа, выполненного из композитного материала, такая, как внутреннее давление или деформации в панели при нагрузке на сжатие могут привести к расслоению между панелью и стрингерами. Возможность точного прогнозирования дефектов и развития разрушения в структуре композитных материалов является важной задачей при проектировании легких конструкций. Для этой цели были проведены тест открытой трещина в консольной балке(мода 1) и тест на расслаивание при растяжении с Т-закреплением. Рассмотрено влияние измерения температуры на расслоение. Получена межслойная вязкость разрушения между панелью обшивки и ребрами жесткости, выполненными из композитных материалов. Детально показано пошаговое разрушение образцов с Т-закреплением при растяжении до достижения критической нагрузки.