JOURNAL OF AIRCRAFT
Vol. 47, No. 2, March–April 2010

# Compression Failure of Carbon Fiber-Epoxy Laminates in Fire

L. A. Burns,* S. Feih,† and A. P. Mouritz‡
*Royal Melbourne Institute of Technology, Melbourne, Victoria 3001, Australia*

DOI: 10.2514/1.45065

**This paper investigates the compression failure of carbon fiber-epoxy laminates when exposed to fire. The investigation gives insights into the softening and failure of carbon-epoxy laminates supporting compression loads in the event of aircraft fire. A thermomechanical model is presented for calculating the compression properties and failure of polymer matrix laminates under combined loading and one-sided heating by fire. The accuracy of the model to predict the failure time of carbon-epoxy laminates at different compression load levels and fire temperatures is determined with structural fire tests performed on woven carbon-epoxy panels. The model predicts a gradual increase in the failure time of the laminate with decreasing compression stress (down to 10% of the room temperature buckling load) and decreasing heat flux (or temperature) of the fire. This was confirmed by the fire tests, which showed good agreement between the calculated and measured failure times. Compression failure of the laminate usually occurred within relatively short times (less than a few minutes) by viscous softening of the polymer matrix. For long failure times, matrix decomposition was shown to influence the failure process. Parametric analysis using the model reveals that raising the glass transition temperature of the polymer matrix increases the compression failure time of laminates in fire. However, only small improvements to the failure time are achieved by raising the glass transition temperature of laminates exposed to high-temperature fires typical of postcrash aircraft accidents.**

## Nomenclature

| | | |
|---|---|---|
| $A$ | = | rate of resin decomposition |
| $C_p$ | = | specific heat of laminate |
| $E_a$ | = | activation energy for resin decomposition |
| $h$ | = | thickness of the laminate |
| $h_c$ | = | enthalpy of laminate |
| $h_G$ | = | enthalpy of volatile gases |
| $k$ | = | thermal conductivity of laminate |
| $M_g$ | = | mass flux of volatiles |
| $m$ | = | number of Simpson integration intervals |
| $n$ | = | reaction order of decomposition process |
| $n_1$ | = | an empirical exponent of the remaining resin content fitted to experimental data to best describe the order of the relationship between polymer mass loss and residual strength |
| $Q_p$ | = | decomposition energy of polymer matrix |
| $R_{rc}(T)$ | = | remaining resin content |
| $T$ | = | temperature |
| $T_g$ | = | chemical glass transition temperature |
| $T_k$ | = | mechanical glass transition temperature |
| $t$ | = | heating time |
| $w_f$ | = | final remaining mass fraction of polymer resin |
| $x$ | = | through-thickness coordinate |
| $\beta$ | = | heating rate of laminate |
| $\rho$ | = | laminate density |
| $\sigma_c(0)$ | = | room temperature compression strength |
| $\sigma_c(R)$ | = | residual compression strength |
| $\sigma_c(T)$ | = | compression strength as a function of temperature |
| $\sigma(x_k)$ | = | strength at position $x_k$ in the through-thickness direction of the laminate |
| $\Theta$ | = | rate of strength decay |

Received 21 April 2009; revision received 3 October 2009; accepted for publication 8 October 2009. 

*School of Aerospace, Mechanical, and Manufacturing Engineering, G. P. O. Box 2476V. Member AIAA.
†School of Aerospace, Mechanical, and Manufacturing Engineering, G. P. O. Box 2476V.
‡School of Aerospace, Mechanical, and Manufacturing Engineering, G. P. O. Box 2476V; adrian.mouritz@rmit.edu.au. (Corresponding Author).

## I. Introduction

MODERN aircraft design incorporates carbon fiber-epoxy composite materials into primary load-bearing structures such as the fuselage, wings, wing-box, and empennage. When exposed to fire, carbon-epoxy composite responds differently to traditional aerospace structural metals such as aluminum and titanium alloys. The structural properties of carbon epoxy decrease more rapidly than those of aerospace metals when exposed to fire. The compression stiffness and strength of carbon epoxy is reduced by 50% when heated to 80–150°C (depending the glass transition temperature of the epoxy matrix). In comparison, the properties of aircraft-grade aluminum alloys (e.g., 2024 Al, 7075 Al) and titanium alloy (Ti-6Al-4V) are reduced by one half when heated to about 200 and 500°C, respectively. In addition to mechanical softening, the epoxy matrix may ignite when exposed to moderate temperature fire. As the epoxy matrix decomposes it releases heat, smoke, and potentially toxic fumes. Unlike metallic structures, composites have the potential to add to the fuel load of the fire due to the combustibility of the epoxy matrix.

Passengers and crew are exposed to hazards caused by softening and pyrolysis of carbon-epoxy structures in the event of an aircraft fire. The temperature of postcrash aircraft fires depends on several factors, including the fuel load, wind speed, and oxygen availability, although values as high as 1000–1200°C are expected. This may lead to structural weakening, distortion, and eventual failure of the fuselage, which hinders the escape of passengers in a postcrash fire. The fire-reaction properties of carbon-epoxy laminates have been extensively studied, including their ignition times, heat release rates, smoke, gas emissions, and flame spread properties [1–7]. However, the structural response of carbon-epoxy laminates to fire is less well understood. The survival time and failure of composite structures during fire is critical to aircraft safety, but relatively little research into their fire structural properties has been published [8].

This paper investigates the compression properties and failure of carbon-epoxy composite when exposed to fire. Compression loading is important when considering the performance of structural columns such as longerons and fuselage skins. The paper presents a

thermomechanical model developed by Feih et al. [1] to calculate the reduction to the compression strength and predict the onset of failure of fiber-polymer laminates exposed to fire. The accuracy of this model to predict the failure of carbon-epoxy laminate panels is evaluated using experimental data from fire structural tests. A carbon-epoxy laminate was tested under different compression load levels while simultaneously being heated from one-sided radiant heat fluxes of 10 kW/m$^2$ (maximum temperature of 270°C), 25 kW/m$^2$ (480°C), and 50 kW/m$^2$ (650°C). The test data is used to assess the accuracy of the thermomechanical model in predicting compression failure times of carbon-epoxy laminates in fire. The paper also examines the effect of the softening (glass transition) temperature of the polymer matrix on the structural survival time of carbon fiber laminates in fire.

## II. Thermomechanical Model

Feih et al. [1] developed a thermomechanical model to calculate the compression properties and failure of hot decomposing fiber-polymer laminates exposed to one-sided heating, before ignition. A full description of the model is provided by Feih et al. [1], and therefore the model is only briefly described here. The model involves a decoupled, two-step analytical approach. The first step is the thermal model, which calculates the temperature rise and extent of decomposition to the laminate when exposed to fire. Based on the calculated temperature profile (which may also be measured), the second step is the mechanical model, which calculates the reduction in compression strength of the laminate as it softens and decomposes to the point of final failure.

The first step in the analysis involves calculating the temperature profile in the through-thickness direction using the thermal model. Unlike for metals, the temperature distribution in composite laminates exposed to fire is influenced by three energy transfer processes: 1) heat conduction from the fire/laminate interface into the material, 2) absorption of heat by the (endothermic) matrix decomposition reaction, and 3) convective flow of volatiles generated by decomposition of the polymer matrix.

The one-dimensional governing equation to calculate the temperature rise with increasing heating time ($\partial T/\partial t$) is expressed as [9]

$$\rho C_p \frac{\partial T}{\partial t} = \frac{\partial}{\partial x}\left(k\frac{\partial T}{\partial x}\right) - \rho\frac{\partial M}{\partial t}(Q_P + h_C - h_g) - \dot{M}_g \frac{\partial}{\partial x}h_G \quad (1)$$

The three terms on the right relate to heat conduction, resin decomposition, and volatile convection, respectively. The resin decomposition term is negative when the decomposition process is endothermic and positive when exothermic. Most polymers (including epoxies) decompose by endothermic reactions, and so this term is negative in Eq. (1). The last term is also negative because the convection of volatiles has a cooling effect on the laminate. An additional decomposition term for the oxidation of carbon fibers in fire, which is an exothermic process, could be included in Eq. (1). However, the model currently assumes that carbon fibers remain inert during temperature exposure, which led to good predictions for glass fiber composites [1]. Decomposition of the polymer matrix occurs at a faster rate than oxidation of the carbon fibers, and therefore it is assumed that the laminate fails under compression loading before significant fiber oxidation occurs.

Once the temperature profile has been obtained, the next step in the analysis is to apply this information to the mechanical model to calculate the reduction to the compression strength in the through-thickness direction. Compression strength degradation is a two-step process that is dominated by viscous softening and decomposition of the epoxy matrix. The carbon fibers also lose strength at high temperatures, but the fiber softening kinetics are ignored in the mechanical analysis because compression strength is a matrix-dominated property. However, fiber softening is an important consideration for the degradation of tension strength and other fiber-dominated properties in fire [10]. The reduction in compression strength with increasing temperature of the carbon-epoxy laminate used to validate the model is shown in Fig. 1. This figure shows the loss in strength under isothermal conditions, and not transient thermal conditions experienced in fire. The laminate begins to lose strength at a critical softening temperature due to viscous softening of the epoxy matrix $T_{so}$. The rate of decline is greatest around the mechanical glass transition temperature $T_k$ and the compression strength then stabilizes at a low residual value at temperature $T_{sf}$. At higher temperatures (typically 350–500°C) the residual compression strength falls to zero due to decomposition of the epoxy matrix. Gibson et al. [9] expressed the relationship between compression strength and temperature of fiber-polymer laminates using the semiempirical expression

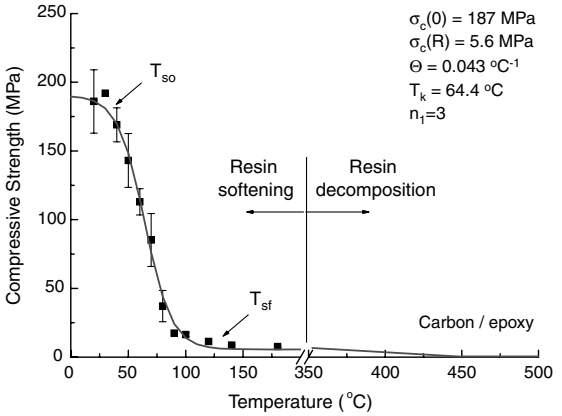



**Fig. 1 Softening of the carbon-epoxy laminate with increasing temperature. The data points are the compression strength measured under isothermal conditions. The curve is fitted using Eq. (2).**

$$\sigma_c(T) = \left[\frac{\sigma_c(0)+\sigma_c(R)}{2} - \frac{\sigma_c(0)-\sigma_c(R)}{2}\tanh(\Theta(T-T_k))\right] \cdot R_{\mathrm{rc}}(T)^{n_1} \quad (2)$$

Room temperature strength $\sigma_c(0)$ can be calculated using laminate theory or measured. Residual strength $\sigma_c(R)$, mechanical glass transition temperature $T_k$, and the rate of decay $\Theta$ must be fitted to the elevated temperature compression test data for the material. The remaining resin content $R_{\mathrm{rc}}$ is used as a scaling function to account for strength loss due to decomposition of the polymer matrix. The exponent $n_1$ is an empirical value fitted to experimental data to best describe the order of the relationship between polymer mass loss and residual strength.

The compression strength is calculated using Eq. (2) as a function of temperature at a number of locations in the through-thickness direction of the laminate. The average bulk compression strength is then determined by integrating these values over the thickness of the laminate using Simpson numerical integration according to

$$\int_{-h/2}^{+h/2}\sigma(x) = \frac{h}{3m}[\sigma(x_0) + 4\sigma(x_1) + 2\sigma(x_2) + \ldots 2\sigma(x_{k-2}) + 4\sigma(x_{k-1}) + \sigma(x_k)] \quad (3)$$

$$\sigma_{\mathrm{av}} = \frac{1}{h}\int_{-h/2}^{+h/2}\sigma(x)\,\mathrm{d}x \quad (4)$$

where $m$ is the number of Simpson integration intervals and $h$ is the laminate thickness.

The mechanical model assumes that all plies fail simultaneously once the residual compression strength of the hot laminate falls below the value of the applied compression stress. Because of the relatively short failure times of carbon-epoxy laminate in hot fires, the mechanical model assumes that time-dependent viscoelastic softening effects (such as creep) do not influence the failure. As mentioned, it is also assumed that softening and oxidation of the

carbon fibers does not affect the failure process, which is controlled by the matrix properties of the laminate.

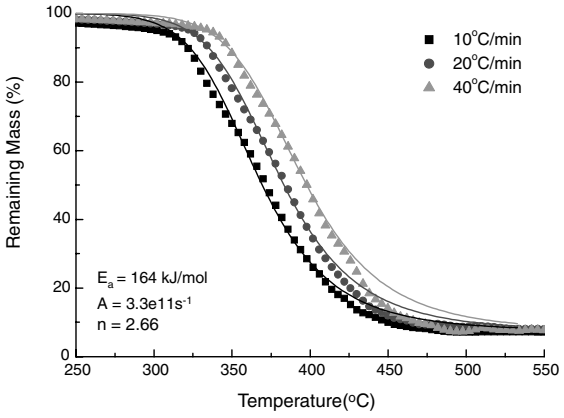
## III. Experimental Procedure

The composite used to assess the accuracy of the thermomechanical model was a [0/90] woven carbon-epoxy laminate. The carbon fabric used in the laminates contained 24 K polyacrylonitrile (PAN)-based carbon fibers woven in a [0/90] pattern with an areal density of 600 g/m$^2$. The epoxy was a low-temperature cure resin (Kinetix® R118). The epoxy was infused into the carbon fabric using the vacuum-assisted resin transfer molding process. After infusion, the laminate was cured at room temperature and then postcured at 120°C for two hours. The fiber volume content of the cured laminate was 55%. The carbon epoxy is not an aerospace-grade material due to the type of epoxy used as the matrix phase. The glass transition temperature $T_g$ of the cured laminate was only 65°C, which is much lower than the $T_g$ values of 120–180°C for aerospace-grade carbon-epoxy materials. While the test laminate is not an aircraft-grade material, it is still relevant because the fire structural model is applicable to both aerospace and nonaerospace laminates.

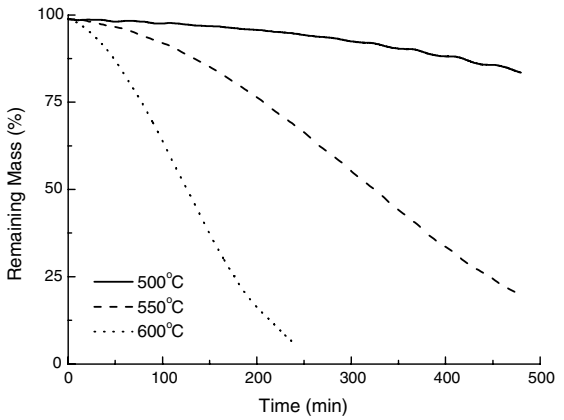
Thermogravimetric analysis (TGA) was undertaken to study the decomposition behavior of the epoxy resin and PAN-based carbon fibers (Fig. 2). TGA revealed that the decomposition temperature range of the laminate increased with the heating rate, and most of the mass loss occurred between 320 and 450°C. TGA also revealed that carbon fibers oxidize in the presence of oxygen and decompose within a temperature range of 500–600°C. These decomposition temperature ranges are typical for the epoxies and carbon fibers used in aircraft structural composites.

Fire structural tests were conducted to validate the thermomechanical model for carbon-epoxy laminate using a novel fire-under-compression test, which is shown schematically in Fig. 3. The laminate specimens were axially loaded in compression and simultaneously heated on one side using a radiant heater. Tests were performed at incident heat fluxes of 10, 25, and 50 kW/m$^2$, which heated the laminate surface to maximum temperatures of about 270, 480, and 650°C, respectively. The temperatures were measured using thermocouples located at the heated and unheated sides of the laminate specimen together with a thermocouple located in the center of the laminate. The specimens were 9.5 mm thick, 50 mm wide, and 560 mm long, although only a 100 mm long section at the middle was heated during testing. Tests were conducted at compression stress levels between 5 and 60% of room temperature Euler buckling load. The specimen was deemed to have failed after it could no longer carry the applied compression stress, and this coincided with unstable microbuckling and collapse.

The fire structural tests were performed using heat flux levels ($\leq$ 50 kW/m$^2$) typical of an in-flight fire, but below that experienced in postcrash fires involving burning jet fuel when the heat flux can exceed 125 kW/m$^2$. Performing the fire tests at high flux values (>125 kW/m$^2$) would have resulted in very short failure times for the test laminate (less than a few seconds). Such short times have limited value in assessing the accuracy of the model for longer failure times. Similarly, the stress levels applied to the laminate during testing were lower than structural stresses on aircraft. The application of high stresses on the laminate would also have resulted in very short failure times, thereby limiting the validation of the model.




a)




b)

**Fig. 2 Thermogravimetric analysis of the a) epoxy resin in nitrogen atmosphere at three different heating rates of 10, 20, and 40°C/min (solid lines are fitted curves), and b) carbon fibers measured under isothermal conditions in air.**

## IV. Results and Discussion

Figure 4 shows the temperature profiles measured at the heated surface, center, and unheated surface of the carbon-epoxy laminate when exposed to radiant heat fluxes of 10, 25, and 50 kW/m$^2$. These heat flux values were selected to induce different types and amounts of softening to the laminate specimens. The laminate reached a maximum temperature of ~270°C when tested at 50 kW/m$^2$, which is above the temperature range for viscous softening of the polymer matrix (50–100°C) but below the decomposition temperature of the matrix (300–450°C) and fibers (>500°C). Therefore, the specimens tested at 10 kW/m$^2$ experienced viscous softening of the epoxy matrix, but decomposition of the matrix or fibers did not occur. The

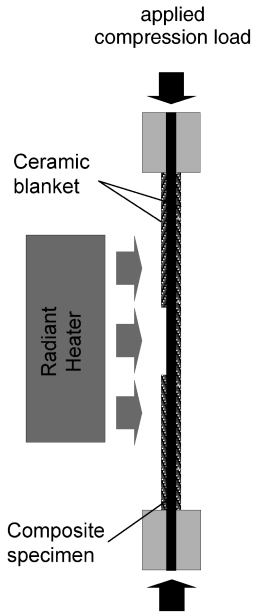
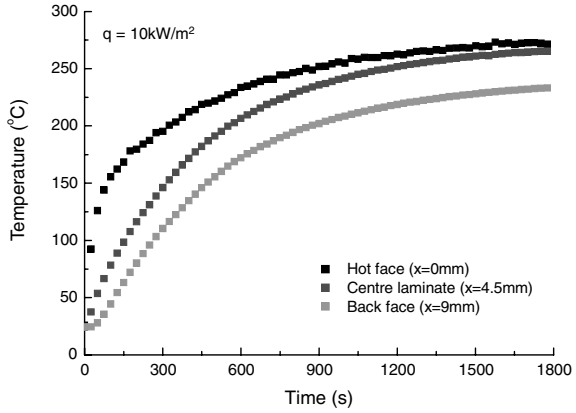
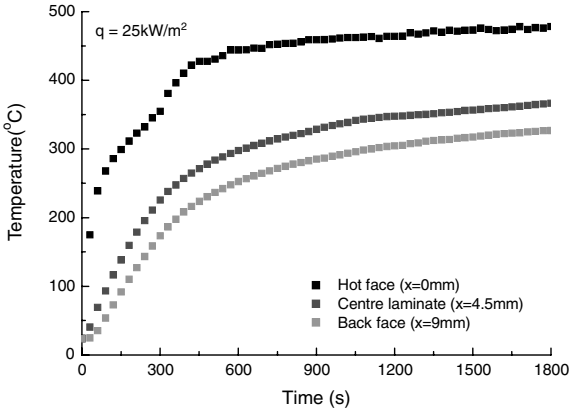


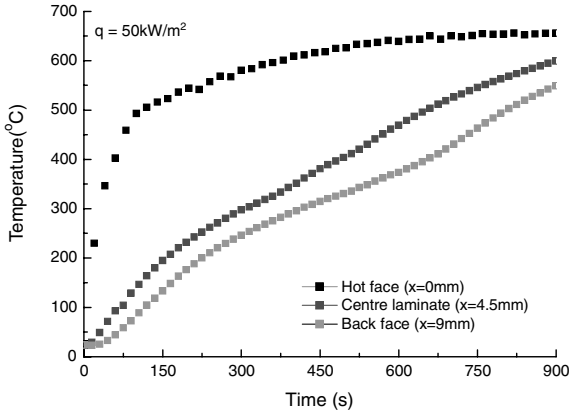

**Fig. 3 Schematic of the fire structural testing of the laminate specimens under combined compression loading and one-sided heating.**




a)




b)




c)

**Fig. 4 Experimental temperature profiles at the heat flux $q$ values of a) 10, b) 25, and c) 50 kW/m$^2$.**

heat flux of 25 kW/m$^2$ heated the laminate surface to a maximum temperature of 480°C after several minutes, which is above the matrix decomposition temperature. However, the unheated surface remained below the decomposition temperature. Therefore, the specimens tested at 25 kW/m$^2$ experienced viscous softening and decomposition of the epoxy matrix close to the heated surface and viscous softening only (without decomposition) toward the unheated surface. The temperatures at 25 kW/m$^2$ were too low to cause fiber oxidation. The highest heat flux of 50 kW/m$^2$ heated the laminate surface to ~650°C, which would cause softening and decomposition of the polymer matrix and oxidation of the carbon fibers. The

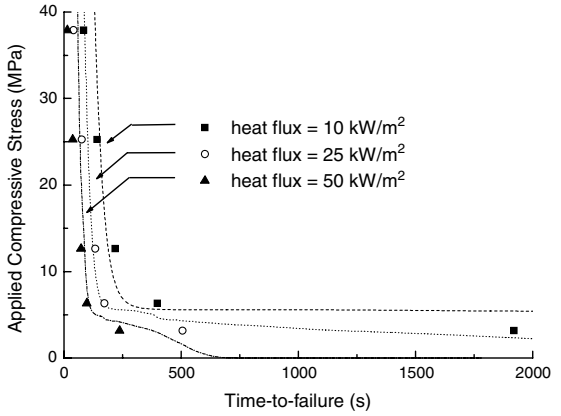



**Fig. 5 Predicted (curves) and measured (data points) failure times of the carbon-epoxy laminate for different heat flux levels. The buckling stress at room temperature is 64 MPa.**

temperatures at the center and unheated surface were also high enough to cause matrix decomposition but too low for fiber oxidation.

Figure 5 shows the effect of compression load on the failure time of the carbon-epoxy laminate when exposed to the heat fluxes of 10, 25, and 50 kW/m$^2$. As expected, the failure time decreases with increasing compression load or heat flux. This behavior has been found previously for nonaerospace types of polymer laminates, such as glass-vinyl ester and glass-phenolic materials (e.g., [1,2,9,11,12]). The failure times for the carbon epoxy are very short (under several hundred seconds) when subjected to compression loads more than about 10% of the buckling stress. The short failure times are observed for the three heat flux levels, even with the lowest value of 10 kW/m$^2$. The heat flux of hydrocarbon fuel fires in postcrash aircraft accident can exceed 50 kW/m$^2$ (>700°C) due to the high fuel load, and the results presented in Fig. 5 indicate that this carbon-epoxy material will have short survival times before failing under compression loading.

The curves in Fig. 5 show the failure times calculated using the thermomechanical model, and there is reasonable agreement with the measured times. The failure times were calculated based on the temperature rise from the measured thermal profiles (Fig. 4). A gradual increase in the failure times when the compression stress is reduced down to 10% is predicted by the model. The model also predicts that the laminate will have relatively long failure times when the load is under 10%, and this is in agreement with the measured times. It appears, therefore, that the model can estimate the survival time of compression-loaded carbon-epoxy laminates when exposed to fire.

Figure 6 shows the typical appearance of a failed specimen following fire structural testing above 10% load. The material is extensively delaminated and the heated surface plies have failed by microbuckling. There is no evidence of widespread decomposition of the epoxy matrix to carbonaceous char, mainly because the failure times were too short to allow significant heating and pyrolysis. Failure occurred by viscous softening of the epoxy matrix close to the

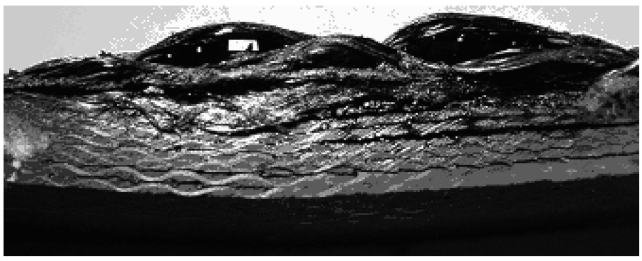
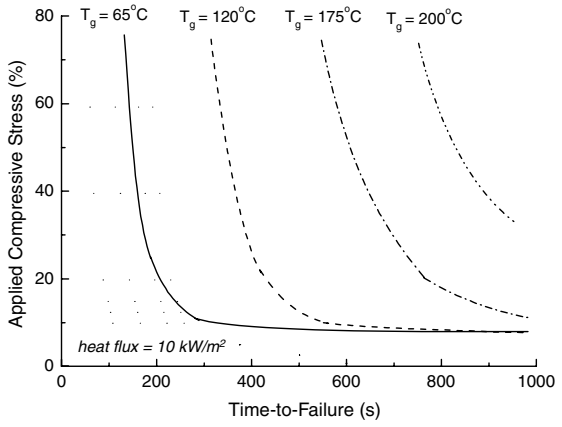
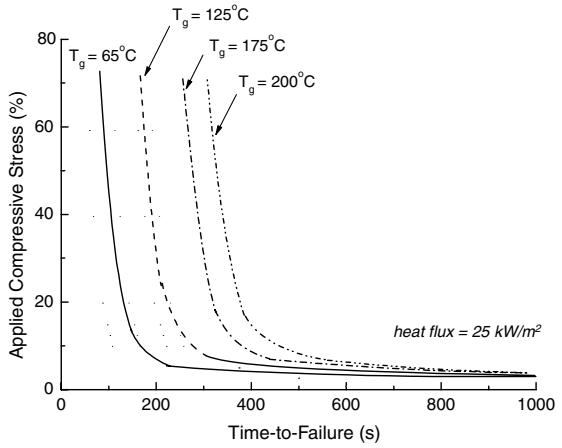


**Fig. 6 Cross-sectional view of a failed carbon-epoxy laminate specimen showing microbuckling of the near-surface plies and delamination cracking between the plies. The upper surface was exposed to the radiant heat flux while the specimen was loaded in compression.**

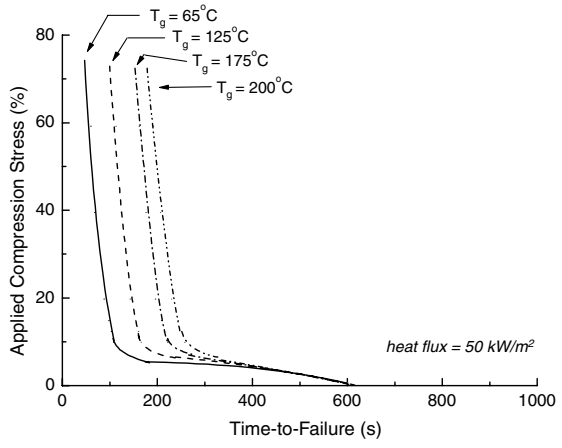
heated surface, which reduced the compression stress needed to induce microbuckling and collapse of the load-bearing plies. Thermal softening of the matrix also reduced the interlaminar fracture toughness that resulted in extensive delamination cracking. The thermomechanical model does not consider delamination cracking and microbuckling in the calculation of the failure time; the analysis is based on the reduction in strength due to resin softening and decomposition. Nevertheless, the model can estimate the failure times with reasonable accuracy. This is the case as delaminated plies have negligible strength at high temperature and therefore do not contribute significantly to the load-bearing cross section.




a)




b)




c)

**Fig. 7 Effect of the glass transition temperature on the compression failure time of carbon-fiber laminates exposed at heat fluxes of a) 10 kW/m², b) 25 kW/m², and c) 50 kW/m².**

The observation that matrix softening controls compression failure suggests that the failure time for laminates should increase with the glass transition temperature of the polymer matrix. Parametric analysis using the thermomechanical model was performed to theoretically assess the influence of glass transition temperature on the failure times for carbon fiber laminates subjected to different compression stress levels and heat fluxes. The glass transition temperatures selected are representative of different polymers used in aerospace structural composites: bismaleimide ($T_g = 200$°C), high-temperature cure epoxy ($T_g = 175$°C) and low-temperature cure epoxy ($T_g = 125$°C), as well as the epoxy used in the laminate specimens ($T_g = 65$°C). In the analysis, it is assumed that the room temperature compression strength and thickness of the different composite panels are the same values. The analysis is presented in Fig. 7 for heat fluxes of 10, 25, and 50 kW/m$^2$. The failure times increase progressively with the glass transition temperature for all compression load levels. This proves that increasing the glass transition temperature will extend the structural survival time of laminates carrying compression loads when exposed to fire. The analysis also reveals that increasing the glass transition temperatures results in a greater improvement to the failure times for lower temperature (heat flux) fires. The failure times increase more rapidly with the glass transition temperature when the heat flux of the fire is reduced. This implies that large improvements to the survival time of structural composites with high $T_g$ resins will occur in relatively low-temperature fires because of the slow heating rate, but only small improvements will occur in very high-temperature fires because the heat-up rate is much faster. Therefore, the use of high-temperature polymers in structural laminates in aircraft is not expected to significantly improve the failure time in the event of postcrash fires involving high temperatures.

## V. Conclusions

The compression properties and failure time of carbon-epoxy laminates when exposed to fire can be estimated using the thermomechanical model developed by Feih et al. [1]. The model predicted an improvement to the failure time of the laminate when the applied compression stress or heat flux was reduced, and this was confirmed using experimental data from fire structural tests. The model predicted a slow increase in the failure times when the applied compression stress was reduced from high values down to 10% of the Euler buckling stress of the laminate specimen. The model also predicted much longer failure times when the compression stress was under 10% and the heat flux was relatively low. These trends were validated with the experimental data. Compression failure of the composite occurred by microbuckling of the load-bearing plies and delamination cracking between the plies, which was caused by viscous softening of the epoxy matrix. Failure can occur before any widespread decomposition of the matrix phase or oxidation of the carbon fibers occurs in the laminate.

Parametric analysis using the model revealed that improvements to the compression failure time of carbon fiber laminates can be achieved by increasing the glass transition temperature of the polymer matrix. The failure times increase with glass transition temperature at a faster rate when the heat flux of the fire is reduced, due to the slower heating rate and softening rate of the matrix phase. The modeling reveals, however, that for postcrash aircraft accidents involving hydrocarbon fuel fires where the flame temperatures can exceed 800°C, the improvement in the failure time from composites with a high $T_g$ resin will be only slightly better than a low $T_g$ matrix.

## Acknowledgments

The authors thank T. Rosewarne (Royal Melbourne Institute of Technology) for technical assistance in manufacturing the composites specimens, P. Tkatchyk and T. Browne (Royal Melbourne Institute of Technology) for assistance with the fire structural testing, and G. Mathys (Defence Science and Technology Organisation) for

the thermogravimetric analysis. The research was performed as part of Project P2.1.2 of the Cooperative Research Centre for Advanced Composite Structures Ltd.

## References

[1] Feih, S., Mathys, Z., Gibson, A. G., and Mouritz, A. P., "Modelling the Compression Strength of Polymer Laminates in Fire," *Composites*, Vol. 38A, No. 11, 2007, pp. 2354–2365.

[2] Mouritz, A. P., and Gibson, A. G., *Fire Properties of Polymer Composite Materials*, Springer–Verlag, Dordrecht, 2006.

[3] McLean, B., Glicksberg, S., and Coulliard, K., "Qualification and Certification of a New Aerospace Material with FAA Fire Property Requirements," *SAMPE Journal*, Vol. 40, No. 5, Sept.–Oct. 2004, pp. 6–12.

[4] Hshieh, F.-Y., and Beeson, H. D., "Flammability Testing of Flame-Retarded Epoxy Composites and Phenolic Composites," *Fire and Materials*, Vol. 21, No. 1, 1997, pp. 41–49. doi:10.1002/(SICI)1099-1018(199701)21:1<41::AID-FAM595>3.0.CO;2-G

[5] Koutides, D. A., "Processing and Flammability Parameters of Bismaleimide and Some Other Thermally Stable Resin Matrices for Composites," *Polymer Composites*, Vol. 5, No. 2, 1984, pp. 143–150. doi:10.1002/pc.750050205

[6] Sastri, S. B., Armistead, J. P., Keller, T. M., and Sorathia, U., "Flammability Characteristics of Phthalonitrile Composites," *Proceedings of the 42th International SAMPE Symposium*, Anaheim, CA, May 1997, pp. 1032–1038.

[7] Mouritz, A. P., *Fire Safety of Advanced Composites for Aircraft*, Australian Transport Safety Bureau, Rept No. B2004/0046, 2004.

[8] Mouritz, A. P., "Fire Resistance of Aircraft Composite Laminates," *Journal of Materials Science*, Vol. 22, No. 21, 2003, pp. 1507–1509. doi:10.1023/A:1026103231041

[9] Gibson, A. G., Wu, Y., Evans, J., and Mouritz, A. P., "Laminate Theory Analysis of Composites Under Load in Fire," *Journal of Composite Materials*, Vol. 40, No. 7, 2005, pp. 639–658. doi:10.1177/0021998305055543

[10] Feih, S., Mathys, Z., Gibson, A. G., and Mouritz, A. P., "Tensile Strength Modelling of Glass Fibre-Polymer Composites in Fire," *Journal of Composite Materials*, Vol. 41, No. 19, 2007, pp. 2387–2410. doi:10.1177/0021998307075461

[11] Case, S., Boyd, S., and Lesko, J. J., "Compression Creep Rupture Behavior of Glass/Vinyl Ester Composite Laminate Subject to Fire Loading Conditions," *Composites Science and Technology*, Vol. 67, Nos. 15–16, 2007, pp. 3187–3195. doi:10.1016/j.compscitech.2007.04.009

[12] Lattimer, B. Y., Ouellette, J., and Trelles, J., "Thermal Response of Composite Materials to Elevated Temperatures," *Modeling of Naval Composite Structures in Fire*, edited by L. Couchman, and A. P. Mouritz, Acclaim Printing Services, Australia, 2006, ISBN:0-646-46193-1, Chap. 1