AN ANALYSIS OF THE DAMAGE TOLERANCE OF LIGHT AIRCRAFT LANDING GEAR

Pu-Woei Chen[1], Shu-Han Chang[2] and Jyun-Yuan Siao[1]
[1]*Department of Aerospace Engineering, Tamkang University, Taiwan*
[2]*General Education Center, Hsiuping Institute of Technology, Taiwan*
E-mail: pchen@mail.tku.edu.tw

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ABSTRACT
This study uses the finite element software, FRANC2D, and the life-time analysis software, AFGROW, to perform simulation analyses of the damage tolerance of the landing gear of light aviation vehicles. This study explores the effect that the initial crack positions and different materials have on the life cycle of landing gear under long-term loads. This study also compares the relationship between stress intensity factors and crack growth for four types of aluminum alloys, titanium alloy and alloy steel. The relationship between residual strength and life cycle, in the presence of existing cracks, is also investigated.

Keywords: damage tolerance; crack propagation; finite element analysis; light aircraft.

ANALYSE DE LA TOLÉRANCE AUX AVARIES DU TRAIN D'ATTERRISSAGE D'UN AÉRONEF DE FAIBLE TONNAGE

RÉSUMÉ
Pour cette étude, nous utilisons le programme d'élément fini FRANC2D, et le programme d'analyse AFGROW, pour exécuter des analyses de simulation de la tolérance aux avaries du train d'atterrissage d'un aéronef de faible tonnage. Cette recherche explore les effets que les positions de la première fissure, et les différents matériaux, ont sur le cycle de vie d'un train d'atterrissage sous des charges permanentes. Cette étude compare la relation entre l'intensité des facteurs de stress, et la propagation de fissures sur quatre types d'alliage : les alliages d'aluminium, de titane, et l'acier d'alliage. La relation entre la force résiduelle et le cycle de vie, en la présence de fissures existantes, est aussi investiguée.

Mots-clés : tolérance aux avaries ; propagation de fissure ; analyse des éléments finis ; aéronef à faible tonnage.

1. INTRODUCTION

Small cracks or stress concentration points may be present in aircraft structural elements or parts after manufacture. During operation and use, aircraft are subject to long-term reciprocating forces, vibrations and environmental changes, any of which can lead to crack propagation and fatigue damage. In 1954, two BOAC Comet aircraft developed severe stress concentrations at the corners of the square passenger windows and the resulting cracks expanded or propagated abruptly, under long-term load, causing fatigue damage. In 1977, as a Boeing 707-321C airplane bound for Nairobi (Kenya) approached its destination airport, the starboard horizontal stabilizer became detached. This caused the pilot to lose control of the aircraft, which then crashed. The results of the accident investigation [1] showed that the accident was caused by a fatigue crack that developed on the rear spar of the starboard horizontal stabilizer. The remainder of the front spar could not bear the entire load, causing the accident. In 1988, Aloha Airlines Flight 243, a Boeing 737 airplane, was cruising at a height of 24,000 ft when an 18-ft section of the aircraft's external skin over the front passenger cabin suddenly peeled away. In the investigation report following the accident [2], the National Transportation Safety Board noted that because of a high frequency of take-offs and landings, small cracks had initialed at the junction between the external skin and the main structure of the fuselage. These cracks reduced the structural strength and, when the main cracks developed, they combined with the smaller cracks, causing the cracks to grow and propagate rapidly, which ultimately led to the accident. The recurrence of similar accidents has focused the attention of the aviation industry and civil aviation authority on aircraft design and the inspection of fatigue cracks, during maintenance.

In order to increase flight safety, following various failures and the flight accidents that have resulted, aircraft design and maintenance has been continually developed. However, initial safe-life design and subsequent fail-safe design cannot effectively guarantee in-flight structural safety. After the 1977 accident involving a Boeing 707-321C, which incorporated the fail-safe concept in its design, the concept of damage tolerance was incorporated into aircraft structural design specification FAR 25.571 in the following year by the Federal Aviation Administration (FAA). Damage tolerance analyzes the effect of crack propagation by assessing the structure's current condition and repairing structural defects promptly, as part of an appropriate maintenance plan, in order to ensure that the structure strong enough to allow safe flight. Since the Aloha Airlines accident, the FAA has revised clauses in the FAR 25 regulations related to fatigue and damage tolerance. For example, FAR 25.571 amdt 25-96 [3], established in 1998, stipulates the implementation of a special safety procedure and the verification of components that are prone to multi-site fatigue damage or multi-element damage (MSD/MED). Subsequently, in 2010, in order to address the operational safety issues of aging aircraft, widespread fatigue damage (WFD) regulations [4] were promulgated. These stipulate that metal structures are extremely likely to develop cracks, because of the reciprocating stresses generated by pressurization and depressurization during flights. Thus, aircraft manufacturers and airlines are required to establish a limit for the number of take-offs and landings, or flight hours, for transport category aircraft currently in service that have a take-off weight of more than 75,000 lbs. The effects of these regulations are extremely far-reaching; 4,198 current aircraft types in the U.S. are expected to be affected. The enactment of these regulations reflects the FAA's focus on aircraft structural fatigue issues.

According to current FAA regulations, any structural parts whose failure would cause a major accident must meet fatigue and damage tolerance design specifications. Because landing gear is an essential structural element of an aircraft, the analysis of its fatigue and damage tolerance is extremely important. FAR 25.571 and FAA AC 23-13A [5] contain the main specifications for aircraft structural damage tolerance and fatigue assessment. However, these provisions mainly apply to large aircraft and do not detail appropriate constraints for light aircrafts. A survey conducted by the General Aviation Manufacturers Association in 2010 [6] showed that the average age of light aircraft in the U.S. was over 30 years, so most of these light aircraft are prone to possible structural fatigue damage. However, in the ASTM 2245-11 [7] specifications for light aircraft design no fatigue standards are specified. The CS-VLA's [8] AMC Subpart B-Standard Specification for the Design and Performance of a Light Sport Aeroplane was issued in June 2011 by the European Aviation Safety Agency (EASA), emphasizing the need for material fatigue tests or calculations in light aircraft design. Consequently, the civil aviation authority has recognized the importance of light aircraft structural design, in terms of fatigue, and a number of requirements have since been established in the form of rules and regulations.

2. REVIEW OF THE LITERATURE

Because of the significant effect of crack growth on an aircraft's structural safety, the aviation industry has developed numerous requirements for damage tolerance design, such as FAR 23, FAR 25 and the damage tolerance design manual [9] issued by the U.S. military. Currently, the main analysis tools for the assessment of aircraft damage tolerance are the finite element analysis program, FRANC2D/L, and AFGROW. FRANC2D/L was developed by the Cornell Fracture Group [10], after a commission from the National Aeronautics and Space Administration (NASA), to analyze the aging and fatigue failure characteristics of airframes and airfoils. This software has since been adopted by Boeing and the U.S. Air Force for the assessment of structural damage tolerance. In addition to these analysis tools, other software, such as ANSYS, also perform crack-related computation commands. However, in order to calculate crack growth and propagation, more complex operational procedures are required. Additionally, many restrictions exist regarding the use of such tools [11].

In 2002, Trego and Cope [12] used FRANC2D/L to conduct damage tolerance analyses of the riveted lap joints between the external aircraft skin (aluminum alloy 2024-T3) and the airframe or fuselage structural rivets (aluminum alloy 2117-T4). Their results indicated that FRANC2D/L can be used to accurately calculate the stress parameters at the crack tips of joints, with values conforming to the experimental results. In order to determine the safety life cycle, they also used AFGROW to calculate the life cycle of fatigue crack propagation. For small aircraft, Vega and Quiroz [13] used FRANC2D/L and AFGROW to assess the tolerance to wing structure damage of an ENAER T-35 PILLAN stunt airplane (which conformed to FAR 23). In their study, finite element analysis was used to simulate the relationship between the stress intensity factor (SIF) under the flight envelope curve load and the speed of the aircraft. The results are of use in the assessment of aircraft maintenance plans.

Because of concern for this issue, the FAA also provides relevant information to the transport and general aircraft manufacturers [14] to facilitate the assessment of the damage tolerance characteristics of aircraft structural parts and materials. Whitney and White [15] published a report, compliant with FAR 23, regarding the fatigue and damage tolerance analysis of the structural parts of six-seater Cessna 210 airplanes that have been in service for an average of approximately 32 years. In their study, finite element analysis was used to calculate and analyze crack propagation in aerofoil jointing parts and lugs. The results showed that variances exist when large aircraft structural damage tolerance procedures and requirements are used to analyze the structures of small aircraft. For example, small aircraft typically experience lower loads and, unlike large passenger airplanes, small aircraft do not have multiple heavy load path designs.

Despite its relatively simple structure, the landing gear of light aircraft is a critical structural element (CSE). Possible causes and types of landing gear damage include foreign object damage (FOD), fretting, environmental corrosion and crack growth caused by material defects and stress concentrations, which eventually lead to fatigue failure or damage under long-term loads. Therefore, in addition to regular inspections, appropriate materials, for example those with high yield strength and damage tolerance must be used to provide sufficient residual strength to satisfy the design limit load (DLL), after cracks begin to expand. Currently, most structural damage tolerance analyses still focus on the landing gear of large aircraft. However, studies have been conducted by the U.S. military on the requirements for helicopter landing gear, which

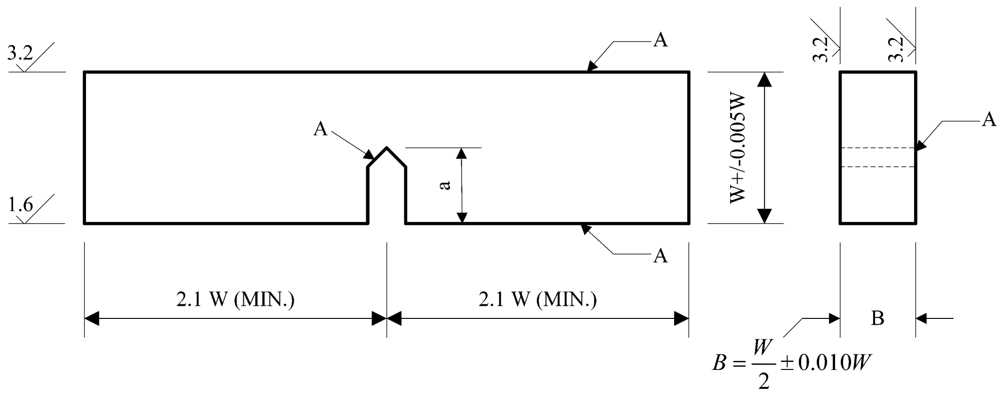

Fig. 1. SENB based on ASTM E399-09 [17].

are similar to those for light aircraft landing gear [16]. Considering the extensive personal use of light aircraft and light sport aircraft (LSA), the structural damage tolerance of the landing gear of aircraft in these categories is worthy of investigation.

3. NUMERICAL AND ANALYTICAL MODELLING

This study performs simulation analyses of the damage tolerance of light aircraft landing gear using finite element analysis software and life cycle simulation software. The experimental process for the study was as follows: (1) 2D models of the landing gear of STOL CH 701 light aircraft were created using the CASCA program; (2) the constructed model was imported into FRANC2D and the parameters for static analysis and SIF (K) calculation were established, in order to determine the relationship between the crack length, a, and SIF; (3) using FRANC2D, the relationship between the beta modification factor, β, which is required for the subsequent execution of AFGROW, and SIF, was determined; (4) the SIF and corresponding β value were imported into AFGROW to calculate the landing gear life cycle, in order to determine the relationship between crack length and the number of load cycles (N), and (5) using the relationships between the crack length, critical stress intensity factor (K_{IC}), and β, the effect that crack propagation in various landing gear materials and load cycles has on the residual strength, S_r, was determined.

In order to verify the correctness of the parameter settings for damage tolerance simulation, a sample model was first established in accordance with the SIF specified in the ASTM E399 [17]. With reference to the ASTM E399 specifications, the dimensions of the single-edge notched beam (SENB) specimen are as shown in Fig. 1, and the thickness of the SENB specimen is defined using Eq. (1). In the experiments, aluminum alloy 6061-T6, which is used to produce light aircraft landing gear, was used as the test material for the simulation. The critical SIF of the material is 28.57 MPa$\sqrt{\mathrm{m}}$, and the yield stress (σ_{ys}) is 282 MPa. Therefore, the width of the specimen must exceed 0.0257 m. Thus, the thickness (B) specified in the simulation parameter setting is 0.03 m, the width (W) is 0.06 m and the length (S) is 0.252 m. The load applied was 10,000 N. A comparison of the simulated results obtained against the SIF Eq. (2) provided in ASTM E399 was then performed.

$$B \leq 2.5 \left(\frac{K_{\mathrm{IC}}}{\sigma_{\mathrm{ys}}} \right)^2 \tag{1}$$

$$K_Q = \frac{PS}{BW^{3/2}} \cdot f\left(\frac{a}{W}\right) \tag{2}$$

Table 1. Material characteristics used in this work.

	6061-T6	6061-T651	075-T6	2024-T3	4340	Ti-6Al-4V
Density (g/cm^3)	2.7	2.7	2.81	2.78	7.85	4.43
Yield strength (MPa)	282	248	517	365	1655	793
Young's modulus (MPa)	6.89×10^4	6.89×10^4	7.17×10^4	7.31×10^4	20.68×10^5	11.03×10^5
Poisson ratio	0.33	0.33	0.33	0.33	0.33	0.31
K_{IC} (MPa$\sqrt{\mathrm{m}}$)	28.57	29.67	29.67	36.26	60.44	87.91
ΔK_{th}	3.846	4.395	3.297	3.187	3.846	3.846

where

$$f\left(\frac{a}{W}\right) = 3\sqrt{\frac{a}{W}} \cdot \frac{1.99 - \left(\frac{a}{W}\right)\left(1 - \frac{a}{W}\right)\left[2.15 - 3.93\frac{a}{W} + 2.7\left(\frac{a}{W}\right)^2\right]}{2\left(1 + 2\frac{a}{W}\right)\left(1 - \frac{a}{W}\right)^{3/2}} \tag{3}$$

Upon validating the correctness of the ASTM E399 standard sample and the established FRANC2D simulation parameters, the relationship between crack propagations and SIF was explored for light aircraft landing gear under various conditions. The parameters required for the FRANC2D landing gear analysis experiment include the following:

1. Material parameters: besides the aluminum alloy 6061-T6 material originally used for CH 701 light aircraft landing gear, this study also examined aluminum alloy 6061-T651, aluminum alloy 7075-T6, aluminum alloy 2024-T3, alloy steel 4340 and titanium alloy Ti-6Al-4V for comparison purposes. The material characteristics are listed in Table 1, wherein KIC represents the critical SIF and ΔK_{th} is the threshold SIF.

2. The load parameters for this study are as follows: the maximum takeoff weight for CH 701, that is 450 kg, is used as the load condition and the load was applied to the junction between the landing gear and the tire (as shown in Fig. 2).

3. The crack parameters are as follows: in order to investigate the growth of cracks in various positions on the landing gear, three initial crack positions (Fig. 2) were specified in the experiment, including (i) the crook of the landing gear (point a), (ii) the junction between the landing gear and fuselage of the light aircraft [18, 19] (point b), where the stress is maximal, and (iii) the center of the landing gear (point c). These parameters were put into FRANC2D and the initial crack length was established as 0.001 m, with an incremental growth of 0.001 m each time. Using the FRANC2D calculation, the relationship between the crack propagation and the SIF of various materials under a load of 450 kg was obtained. The results were converted into beta modification factor, β, which are required for AFGROW simulation analyses.

This study used AFGROW to analyze the life cycle of landing gear with crack growth. The procedure for AFGROW analysis was as follows: (1) Select materials and crack propagation equations: the crack propagation equation used for this experiment is the NASGRO equation, which defines the relationship between crack growth and life cycle as

$$\frac{da}{dN} = C\left[\frac{1-f}{1-R}\Delta K\right]^n \frac{\left(\frac{1-\Delta K_{\mathrm{th}}}{\Delta K}\right)^p}{\left(1 - \frac{K_{\max}}{K_c}\right)^q} \tag{4}$$

(2) The boundary conditions are as follows: (i) the thickness of the landing gear (B) is 0.082 m, the width (W) is 0.02 m and (ii) the initial crack length is 0.001 m. (3) The load conditions are as follows: the stress

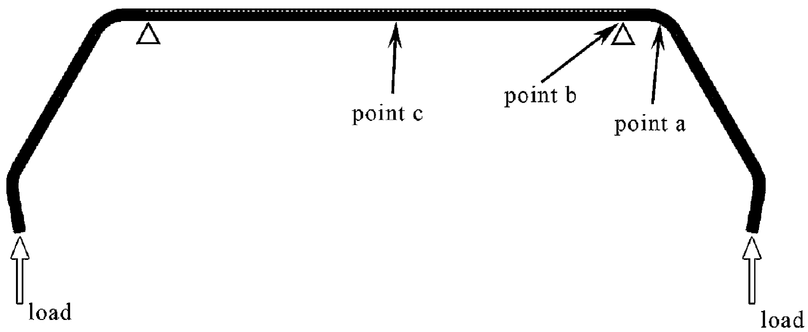

Fig. 2. The loads and specified positions of the initial cracks (points a, b and c).

multiplication factor (SMF) is established as 4,410 N and the stress ratio $R = 0$. (4) Inputting the β in the experiment, it is specified that when the SIF exceeds the critical value (K_{IC}), the load cycle frequency (N) is the life cycle of the landing gear. In order to determine the maximum load sustainable by the various materials under different crack conditions, the residual strengths of the landing gear under different conditions was also investigated. The residual strength of the structure is expressed as

$$S_r = \frac{K_{\mathrm{IC}}}{\beta\sqrt{\pi a}} \tag{5}$$

The residual strength experiment comprised the following steps: (1) the maximum stress sustainable for landing gear without cracks was determined through static load analysis; (2) changes in the residual strength of the structures containing cracks in response to propagating crack lengths was calculated using the residual strength equation; and (3) combined with damage tolerance life cycle analysis, changes in the residual strength of the structure in relation to the load numbers or frequencies were determined. The results of this experiment are expressed as the relationship between the residual strength and cracks (S_r versus a), and between the residual strength and number or frequency of load cycles (S_r versus N).

4. RESULTS AND DISCUSSION

Figure 3 shows a comparison between the theoretical formula of the SENB specimen of ASTM E399 and the SIF determined in this study, using FRANC2D. The results indicate that the deviation between the two is less than $\pm 1\%$, which confirms the correctness of the parameter settings in FRANC2D. Figure 4 shows SIFs that correspond to crack growth under load at various crack positions. Figure 4 shows that the growth of cracks at various positions, under load, follows the same trend as the SIF. Specifically, the SIF at the crook of the landing gear changes according to the crack growth at a rate less than either that at the junction between the landing gear and the fuselage or that at the center of the landing gear.

Figure 5 shows the sustainable life cycles (N) when the critical crack length is reached at various positions. Because the initial crack at the crook (point a) of the landing gear was 1 mm in length, the SIF at this position does not reach the crack propagation threshold (ΔK_{th}) of aluminum alloy 6061-T6, so this is not shown in Fig. 5. The critical length of the joint between the landing gear and the fuselage and that of the landing gear center, both approximately 10.94 mm, with life cycles of 4.8×10^4 and 6.9×10^4, respectively. From the figure, it is also seen that when the initial length of the crack increases from 1 to 2 mm, the joint between the landing gear and the fuselage has undergone approximately 2×10^4 cycles. However, the landing gear center can only withstand 4×10^4 cycles of load when the same crack length is propagated. Figure 6 also shows that the crack at the junction of the landing gear and the fuselage grows rapidly after 4×10^4 cycles. Because the location of this crack is the crucial point at which the landing gear bears fatigue loading, the

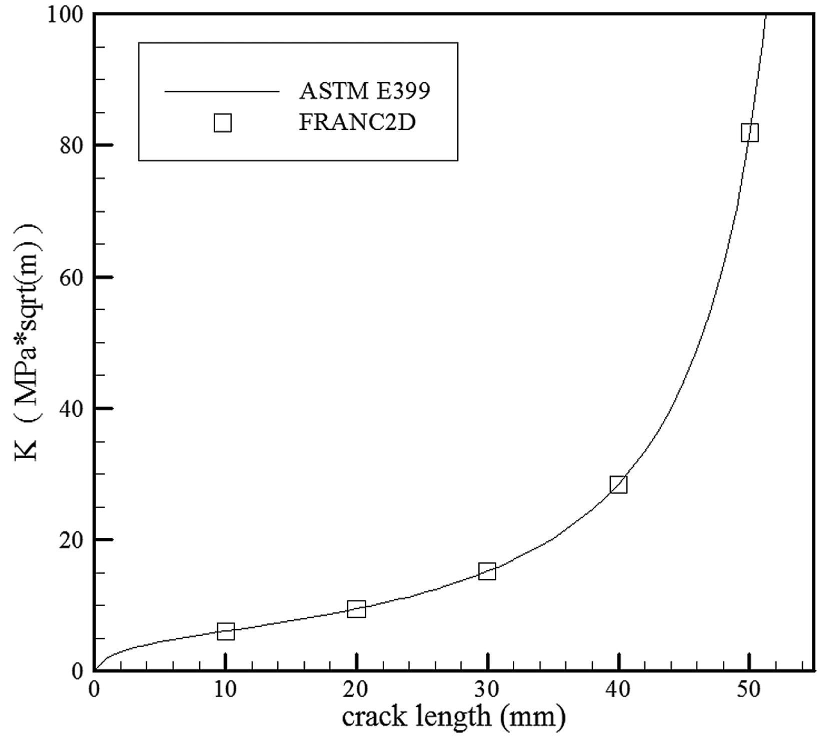

Fig. 3. A comparison of the results provided by FRANC2D and the ASTM standard.

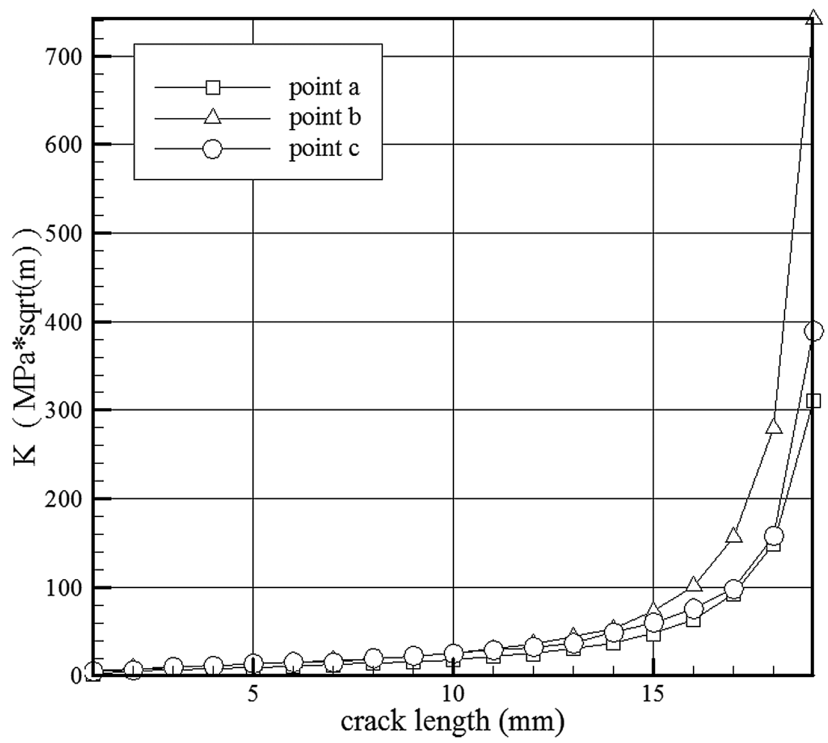

Fig. 4. The relationship between the SIF and the crack length at various positions, under load.

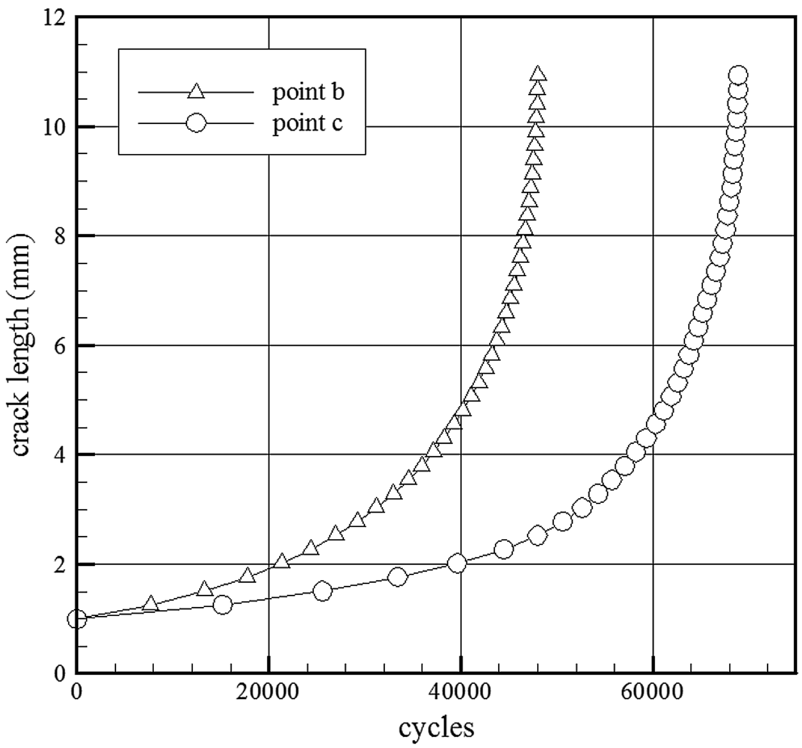

Fig. 5. The relationship between crack length and the life cycles at various positions, under load.

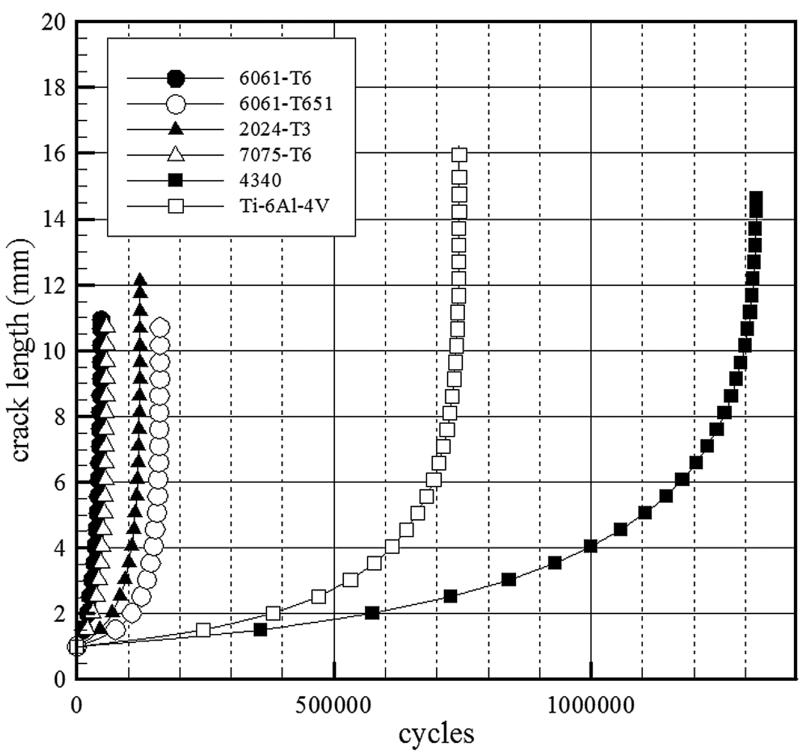

Fig. 6. The relationship between the crack length and the life cycles of different materials, under load.

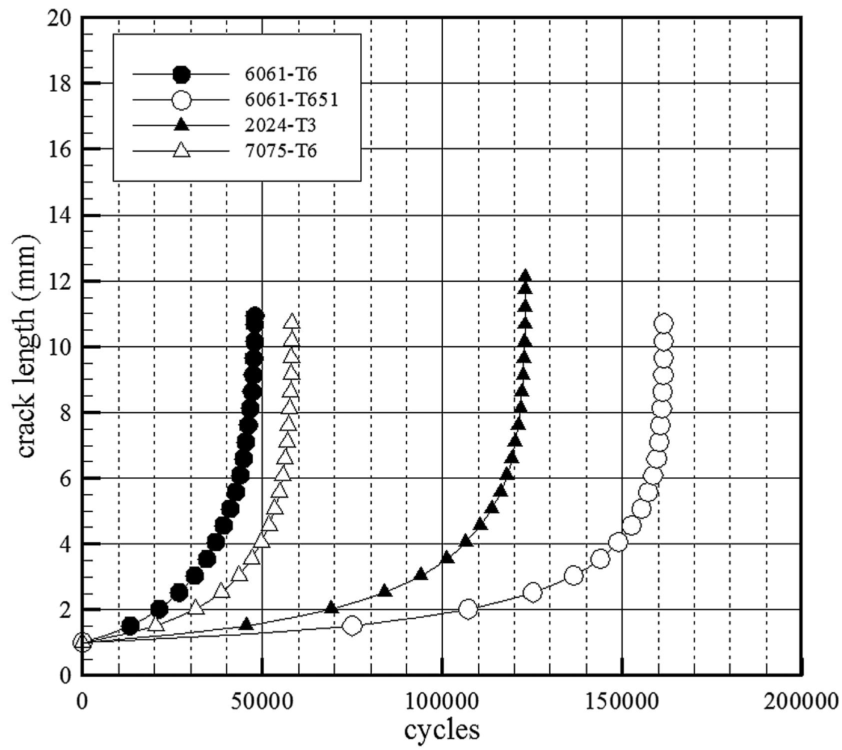

Fig. 7. The relationship between the crack length and the life cycles of various aluminum alloys, under load.

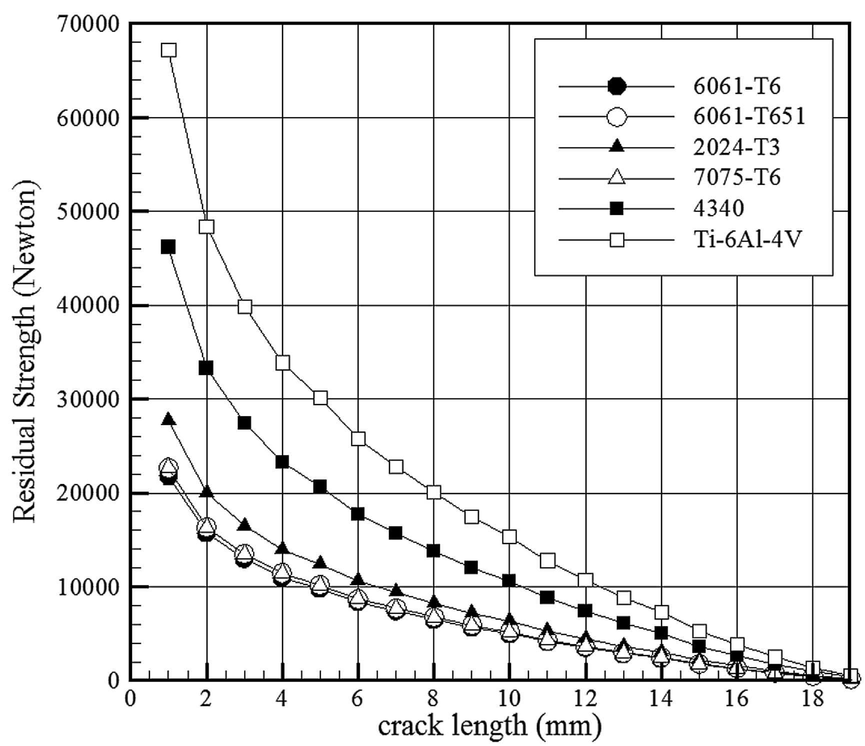

Fig. 8. The relationship between the residual strength and the crack length of various materials.

Table 2. The critical crack lengths and sustainable life cycles of landing gear constructed from various materials under load.

	Critical crack length (mm)	Life cycles
6061-T6	10.942	4.80×10^4
6061-T651	10.998	1.62×10^5
7075-T6	10.990	5.82×10^4
2024-T3	12.107	1.23×10^5
AISI 4340	14.637	1.32×10^6
Ti-6Al-4V	16.246	7.44×10^5

Table 3. Maximum static loads sustainable for landing gear constructed from various materials.

	6061-T6	6061-T651	7075-T6	2024-T3	4340	Ti-6Al-4V
Max. Load (Newton)	12,610	11,072	23,067	16,300	73,815	35,369

subsequent sections of this study focus on a comparison of the relationship between the life cycle and crack growth for various materials at this location.

Figure 6 shows the relationship between the propagation of the crack length and load life cycle, when landing gears constructed from different materials, with an initial crack length of 1 mm, experience a load of 450 kg. Table 2 lists the critical crack lengths and sustainable life cycles for landing gears constructed from different materials, under load cycles. Table 2 shows that the critical crack lengths of aluminum alloy 6061-T6, aluminum alloy 6061-T651 and aluminum alloy 7075-T6 are all approximately 10.9 mm. The critical crack length of aluminum alloy 2024-T3, AISI 4340 and Ti-6Al-4V is 12.1, 14.6 and 16.2 mm, respectively.

Figure 6 also shows that of the materials tested, Ti-6Al-4V exhibits the maximum critical crack length and AISI 4340 has the maximum life cycle. If AISI 4340 is used for the construction of landing gear, it can be subject to 1.3×10^6 cycles before reaching a critical SIF. Figure 7 shows the relationship between the crack growth propagation for the four different aluminum alloys and load life cycles. Of the four aluminum alloys, 6061-T6 has the shortest life cycle. However, the life cycle of 6061-T651, which is 6061-T6 processed with a stress-relieving heat treatment, is four times higher and it has the longest fatigue life cycle of the four aluminum alloy materials. Of the four aluminum alloys, 2024-T3 has the maximum critical crack length and a comparatively excellent fatigue life cycle.

Figure 8 shows the relationship between the residual strength at the junction between landing gear constructed from various materials and the fuselage (point b) and crack length (the initial crack length is 1 mm). The figure shows that the residual strength of Ti-6Al-4V and 4340 is higher than that of the aluminum alloys, at various stages of crack propagation. Of the four aluminum alloys, the residual strength of 2024-T3 is the least influenced by the propagation of crack length. Figure 9 shows the relationship between the residual strength at the junction between landing gear constructed from various materials and the fuselage (point b) and load cycles. Ti-6Al-4V has the highest residual strength in the initial cycles, but this decreases rapidly after 650,000 cycles. Although 4340 has a lower initial residual strength than Ti-6Al-4V, its residual strength decreases at a slower rate than that of titanium alloy, and its performance is significantly superior under long-term load cycles. Of the four aluminum alloys, 2024-T3 has the optimum residual strength although its long-term performance is less robust than that of 6061-T651 as the number of load cycles increases.

In order to ensure structural safety, the design of landing gear must meet the requirements for both static strength analysis and residual strength analysis. Table 3 shows the maximum static loads sustainable by CH 701 light aircraft landing gear constructed from different materials. Figure 10 shows the relationship

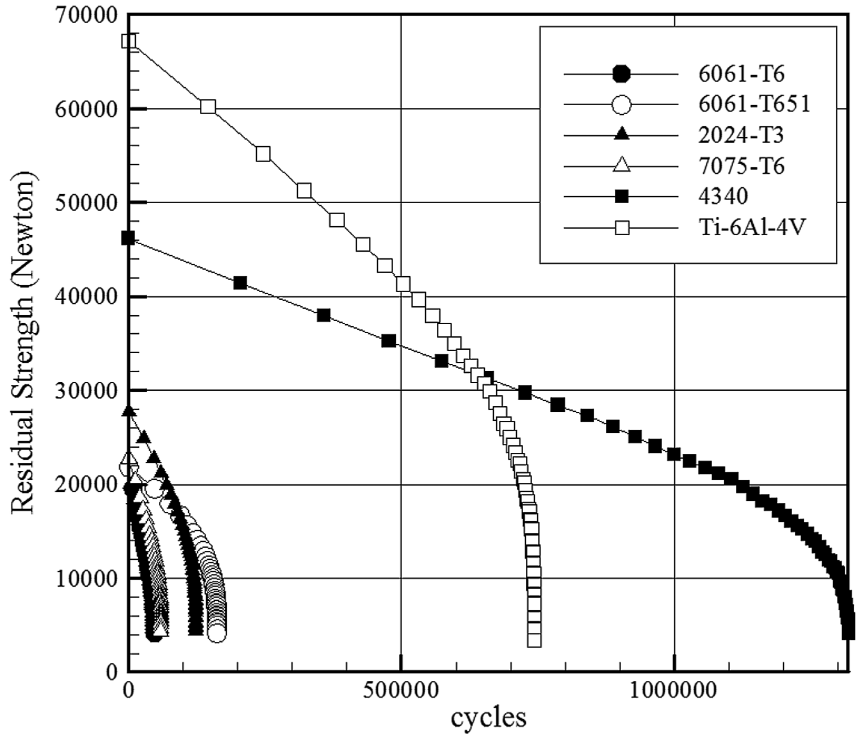

Fig. 9. The relationship between the residual strength and the load cycle number of various materials.

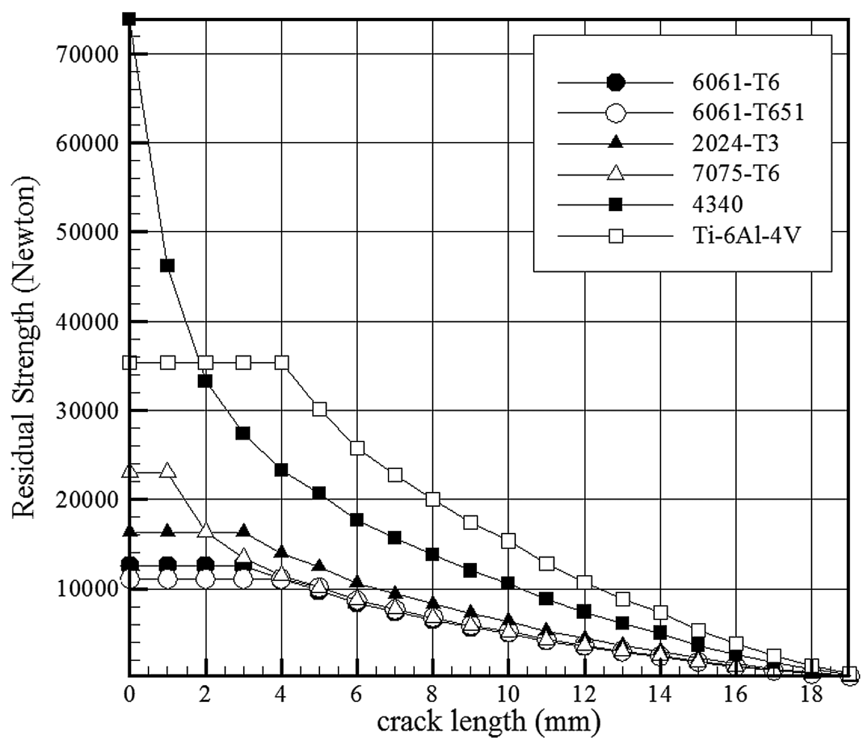

Fig. 10. The relationship between the overall residual strength and the crack length of various materials, considering the influence of static strength.

between the overall residual strength and crack growth, when landing gear constructed from different materials is analyzed for both static strength and SIF. The flat area on the left side of the figure is the result of static strength analysis, where the static strength sustainable by the landing gear is primarily restricted by the yield stress of the material. The residual stress at the right side of the curve changes in response to the crack length, which is primarily influenced by the SIF of the material. As shown in Fig. 10, steel alloy 4340 has the highest static strength, but its structural strength declines rapidly, once a crack develops (crack length 0 to 2 mm). When the crack length reaches 5 mm, the decline slows. The overall residual strength of titanium alloy Ti-6Al-4V is barely affected by crack propagation until the crack length exceeds 4 mm, after which the overall residual strength declines slowly, although the performance is the best of the samples tested. Although the static strength of titanium alloy Ti-6Al-4V is half that of alloy steel 4340, its residual strength is superior to that of alloy steel 4340, when the crack length exceeds 2 mm. Aluminum alloy 7075-T6 has the optimal static strength of the four aluminum alloys, but its strength declines comparatively rapidly, once a crack develops. When the crack length reaches 3 mm, the residual strength of aluminum alloy 7075-T6 is similar to that of 6061-T6 and 6061-T651. Although aluminum alloy 2024-T3 has a weaker static strength than 7075-T6, its residual strength declines at a slower rate, when cracks grow.

5. CONCLUSIONS

This study uses the finite element analysis software FRANC2D and the crack propagation life cycle simulation software AFGROW to perform simulation analyses of the structural damage tolerance of light aircraft landing gear. In this work, the Zenith CH701 landing gear was used for modeling and FRANC2D was used to analyze the relationship between SIF and crack growth. In order to examine the various potential materials for light aircraft landing gear, the life cycles and residual strengths of four aluminum alloys, alloy steel and titanium alloy with initial cracks were compared. Based on the results, the following conclusions are drawn:

1. This study compares the effect of SIF for three initial crack positions under load. It is found that when the initial cracks are located at the junction between the landing gear and the fuselage, the maximal crack SIF value is produced. This result confirms those reported by previous fatigue load studies, where no cracks were present [18]. When the aluminum alloy 6061-T6 landing gear has an initial crack of 1 mm, the crack propagation life cycle of the landing gear and fuselage joints is the shortest and it is only capable of withstanding 48,000 load cycles. When the initial crack is in the center of the landing gear, the crack propagation life cycle is longer and the landing gear can withstand approximately 68,900 load cycles, which is 1.4 times more. When the initial crack is located in the crook of the landing gear, the life cycle is not affected by the boundary condition, because the increase in SIF is less than the crack propagation threshold.

2. In order to compare the structural damage tolerance of various landing gear materials, the junction between the landing gear and fuselage was selected for an examination of the critical crack length of various materials. Titanium alloy Ti-6Al-4V performs optimally, with a critical crack length in excess of 16 mm, followed by alloy steel 4340, with a critical crack length of 14.6 mm. Of the four aluminum alloys, 2024-T3 is the best, with a critical crack length of 12.1 mm; the remaining three materials 6061-T6, 6061-T651, and 7075-T6 had a critical crack length of 11 mm.

3. Of the six materials used for landing gear, alloy steel 4340 has the optimal performance in withstanding load cycles and is capable of withstanding more than 1,320,000 cycles, whereas the life cycle of titanium alloy Ti-6Al-4V is approximately 743,000 cycles. 6061-T651 provided the best performance of the four aluminum alloys, withstanding approximately 160,000 load cycles. By contrast, 6061-T6

has the shortest life cycle, at only 1/4 of that of 6061-T651. The load life cycles of aluminum alloys 2024-T3 and 7075-T6 are 123,000 and 58,000 cycles, respectively.

4. The crack propagation life cycles of titanium alloy and alloy steel are significantly superior to those of the aluminum alloys. If 6061-T6 is used as a benchmark, the crack propagation life cycle of alloy steel 4340 is 27 times that of the aluminum alloys and the crack propagation life cycle of titanium alloy is 15 times that of the aluminum alloys. Aluminum alloy 6061-T651 is similar 6061-T6, but with a different heat treatment. Additionally, although the critical crack length of 6061-T651 is not significantly higher than that of 6061-T6, its crack propagation life cycle is increased to 163,000 cycles, which is more than three times that of 6061-T6.

REFERENCES

1. "Aircraft Accident Report No. 9/78", Accidents Investigation Branch, Department of Trade, United Kingdom, 8 February, 1979.
2. "Aircraft Accident Report-Aloha Airlines Flight 243", NTSB/AAR-89/03, April 1988.
3. "Damage-Tolerance and Fatigue Evaluation of Structure", FAR Part 25.571.
4. "Aging Airplane Program: Widespread Fatigue Damage", Docket No. FAA-2006-24281, FAA, Federal Register, Vol. 75, No. 219, 15 November 2010.
5. "Fatigue, Fail-Safe, and Damage Tolerance Evaluation of Metallic Structure for Normal, Utility, Acrobatic, and Commuter Category Airplanes", FAA AC 23-13A, 29 September 2005.
6. *2010 General Aviation Statistical Databook & Industry Outlook*, General Aviation Manufacturers Association.
7. "Standard Specification for Design and Performance of a Light Sport Airplane", ASTM F2245, June 2011.
8. "Certification Specifications for Light Sport Aeroplanes CS-LSA", Annex to ED Decision 2011/005/R, European Aviation Safety Agency, 27 June 2011.
9. "USAF Damage Tolerance Design Handbook: Guidelines for the Analysis and Design of Damage Tolerant Aircraft Structures", AFFDL-TR-79-3021, Revision A, March 1979.
10. Cornell Fracture Group, http://www.cfg.cornell.edu/index.htm, September 2011.
11. Alam, M.S. and Wahab, M.A., "Modeling the fatigue crack growth and propagation life of a joint of two elastic materials using interface elements", *International Journal of Pressure Vessels and Piping*, Vol. 82, Issue 2, pp. 105–113, February 2005.
12. Trego, A. and Cope, D., "Evaluation of damage tolerance analysis tools for lap joints", *Boeing Phantom Works*, January 2002.
13. Vega, M. and Quiroz, L., "A framework for damage tolerance assessment", Concepción University Department of Mechanical Engineering, August 2001.
14. "Fatigue Crack Growth Database for Damage Tolerance Analysis", DOT/FAA/AR-05/15, August 2005.
15. Whitney, E.J. and White, K., "Fatigue and damage tolerance evaluation of Cessna 210 wing carry-through structure", in *Proceedings of Aircraft Airworthiness and Sustainment Conference*, Brisbane, 2010.
16. Reddick, H.K. Jr., "Safe-life and damage-tolerance design approaches for helicopter structures", US Army Research and Technology Laboratories, Virginia, 1983.
17. "Standard Test Method for Linear-Elastic Plane-Strain Fracture Toughness KIC of Metallic Materials", ASTM E399-09, 2009.
18. Chen, P., Sheen, Q., Tan, H. and Sun, T., "Fatigue analysis of light aircraft landing gear", *Advanced Materials Research*, Vols. 550–553, pp. 3092–3098, 2012.
19. Chen, P., Chang, S. and Yu, T., "Analysis of the dynamic and static load of light sport aircraft composites landing gear", *Advanced Materials Research*, Vols. 476–478, pp. 406–412, 2012.