Static testing for composite wing of a two-seater seaplane

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Abstract. The paper studied the strength and deformation characteristics of a composite wing of a two-seater seaplane to get the certificate of airworthiness which static testing is indispensable. The primary wing structure component includes upper and lower skins, leading edge, trailing edge, a root rib and main spar. The main purpose of static testing is to examine the bending strength of the wing. The testing results show good agreement with the FE analysis results, and the bending strength of wing is strong enough to support the limit loads (the maximum loads to the expected in service) without detrimental and permanent deformation, and without failure for at least 3s under the ultimate loads (limit loads multiplied by prescribed factors of safety), which meets the requirement of ASTM F2245-16c (Standard Specification for Design and Performance of a Light Sport Airplane).

1. Introduction

A seaplane is a powered fixed-wing aircraft capable of taking off and landing on water. Seaplanes that can also take off and land on airfields are in a subclass called amphibious aircraft. Seaplanes and amphibians are usually divided into two categories based on their technological characteristics: floatplanes and flying boats. A floatplane is a type of seaplane, with one or more slender pontoons mounted under the fuselage to provide buoyancy. By contrast, a flying boat uses its fuselage for buoyancy. For two-seater seaplane which maximum takeoff weight (MTOW) is less than 650 kilograms can be classified as a light sport airplane that meets certain regulations set by a Civil Aviation Authority of Thailand (CAAT).

To ensure flight safety and receive certificate of airworthiness for a light sport airplane, the static structural testing of the aircraft’s wing must be performed and proved of safety. According to ASTM F2245-16c (Standard Specification for Design and Performance of a Light Sport Airplane), a structure should be designed to be able to withstand limit load (the maximum loads to be expected in service) without permanent damage or deformation upon unloading, and withstand ultimate load (limit loads multiplied by prescribed factors of safety) for at least 3 seconds without collapse. Therefore, the static test must verify the strength and stiffness of the wing structure under the design of load provision. Smith et al.[1] described the static testing of an ultralight airplane and replaced with masses to fulfill center-of-gravity requirements before testing. Wong et al. [2] presented the design of the test rig and the results of the static test of the Lockheed P-3 Orion Wing Leading Edge center section structure. Sullivan et al.[3] presented the results of the strength and stiffness characteristics of a carbon composite wing of an ultralight unmanned aerial vehicle. Sullivan et al.[4] described vibration testing of a full-scale carbon composite ultralight unmanned aerial vehicle. Kang et al. [5] performed static testing on the wing of a two-seater aircraft powered by Li-ion battery electric propulsion compliance with ASTM F2245-11.

This paper described the static testing and finite element (FE) analysis results of the seaplane wing. The static load is specified regarding the limit loads and the ultimate loads, which is the limit loads multiplied by prescribed factors of safety. The ultimate load safety factor is 1.5. The property of material

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[1] Smith et al.
[2] Wong et al.
[3] Sullivan et al.
[4] Sullivan et al.
[5] Kang et al.
system, the result of finite element (FE) analysis, the static loading method and the testing results are described in the following section.

Figure 1. Seaplane (Amphibious)

2. Wing Structure
Composite materials have been widely utilized in airplane due to their high specific strength as compared to conventional isotropic materials. The two-seater seaplane main structure was made from all-composite which much lighter than traditional metal. The seaplane structural components were intentionally designed complying to ASTM F2245-16c which is the standard specification for design and performance of a light sport airplane. According to ASTM F2245-16c structure compliance section, the seaplane structure must be able to support limit loads (the maximum loads to be expected in service) without detrimental and permanent deformation, and ultimate loads without failure for at least 3 seconds. In order to verify the strength and deformation characteristics of the wing structural assemblies, the wing structural static testing was performed on a test rig following steps of applying limit loads prescribed in ASTM F2245-16c (positive limit loads not less than 4G and negative limit loads not greater than -2G).

The two-seater seaplane was designed with maximum takeoff weight (MTOW) less than 650 kilograms (1G), maximum empty airplane weight is 325 kilograms and each wing weights about 50 kilograms. The wing span of this plane is 9.73m, aspect ratio is 9.26, and the mean aerodynamic chord is 0.487m.

The wing structure is an integral type (for stress calculations the main spar is the only element responsible to resist the aerodynamic loads). Each wing has a mono I-beam spar, 10 ribs and a sandwich structure skin shell. A composite shell leading edge and a composite fairing for the aileron and flaps complete the wing structure. All the structural wing elements are made of composite and held together by adhesive. The wing loads are transferred to the fuselage by means of a wing strut attached to the spigot post and by the spar attached to the root rib.

As shown in Figure 2, the primary wing structure includes the main spar, ribs, the upper and lower skins, leading edge, trailing edge, and these structures are glued together by adhesive. However, the ailerons and flaps are removed in finite element (FE) analysis and static testing.

Figure 2. Wing Structure
The skins are made of a sandwich material. The spars and root rib are made of carbon composite. In order to provide the wing bending stiffness and form the carry-through structure, the root-rib/wing box interconnected by two steel screws bear the shear load and torsion.

The wing structure components are made of carbon fabric, glass fiber, foam, and parabeam 3D glass which the material properties are shown in Table 1. The skins are made of sandwich construction with a 2 mm foam core. The laminated ply pattern in the wing region is given in Table 2. Two M10 screws are used to mounted the root rib with the wing box, while another two M10 screws are used to mounted the strut with the wing and the fuselage.

### Table 1. Material Properties of the Seaplane Wing

| Material property | Carbon Fibre | Glass Fibre | Parabeam 3D Glass | Foam core [isotropic] |
|-------------------|--------------|-------------|-------------------|----------------------|
| Density [kg/m³]   | 1.431        | 1.539       | 0.341             | 1.8                  |
| E11 [GPa]         | 36.59        | 11.58       | 0.255             | 0.085                |
| E22 [GPa]         | 36.59        | 11.58       | 0.255             |                      |
| G12 [GPa]         | 2.405        | 2.63        | 0.07              | 0.032                |
| Poisson           | 0.038        | 0.038       | 0.001             | 0.3                  |
| Max Stress [MPa]  | 394.95       | 159.85      | 5.31              |                      |
| Max Shear [MPa]   | 49.26        | 37.95       | 2.24              |                      |

### Table 2. Laminated Ply Pattern

| ITEMS                  | NO. of plies | ORDERING | THICKNESS (mm) |
|------------------------|--------------|----------|----------------|
| SKIN                   | 4            | B-X-F-Y  | 2.9            |
| SPAR - UPPER FLANK     | 24           | A        | 7.2            |
| SPAR - LOWER FLANK     | 12           | A        | 3.6            |
| SPAR - WEB             | 25           | A-P-B    | 11.2           |
| RIB #1                 | 7            | B-X-Y-B-X-Y-B | 2.1  |
| RIB #2                 | 7            | B-X-Y-B-X-Y-B | 2.1  |
| RIB #3                 | 7            | B-X-Y-B-X-Y-B | 2.1  |
| RIB #4                 | 7            | B-X-Y-B-X-Y-B | 2.1  |
| RIB #5                 | 7            | B-X-Y-B-X-Y-B | 2.1  |
| RIB #6                 | 7            | B-X-Y-B-X-Y-B | 2.1  |
| RIB #7                 | 6            | B-X-Y-X-Y-B | 1.8            |
| RIB #8                 | 5            | B-X-Y-X-B  | 1.5            |
| RIB #9                 | 7            | B-X-Y-B-X-Y-B | 2.1  |
| RIB #10                | 4            | B-X-Y-B   | 1.2            |
| spigot post            | 38           | B        | 11.4           |
| Float                  | 6            | B-X-Y-X-Y-B | 1.8  |

Carbon fabric 0/90; B: Carbon fabric +/-45; X: Glass fibre 0/90; Y: Glass fibre +/-45; F: Foam; P: Parabeam

3. Experimental

A test frame was used to serve as the support structure for static strength testing, as shown in Figure 3. It was made from 2 inches carbon steel square tube in “T” shape mounted firmly on the shop floor. A 400 x 150 mm “I” beam, 8 meters long, mounted with the shop column over the test frame with four chain hoists were used to create bending and pitching moment on the wing. The left wing of the seaplane with the wing strut was installed on the test frame as the normal configuration (zero angle of attack). Both positive/negative distribution load were simulated on the wing following ASTM F2245-16c recommendation. The -2G negative distribution load was applied on the wing by sand bags and the +4G positive load (limit load) up to +6G positive load (ultimate load) by the four chain hoists.
3.1. Loading Method

There were four loading locations on the wing where each location was embraced by a wooden shroud hooked up with the hoist chain and the pulling force on each shroud was 30% location of mean cord. The distance of each loading location away from the fuselage centerline was shown in Fig. 4. The wing loading and the total number of load cases were shown in Table 3. Starting with negative loads, the sand bags were placed on each assigned location until reaching -2G (limit load) following with positive loads by pulling each chain hoist through a load cell until reaching +4G positive load (limit load) and +6G positive load (ultimate load) respectively.

### Table 3. Test load applied on the wing

| Applied Load [Ngf] | -2G [-50%] | -1G [-25%] | 1G [25%] | 2G [50%] | 3G [75%] | 4G [100%] | 5G [125%] | 6G [150%] |
|--------------------|------------|------------|----------|----------|----------|----------|----------|----------|
| **Negative Load**  |            |            |          |          |          |          |          |          |
| Rib#2              | -238.60    | -108.60    | 199.25   | 217.66   | 303.09   | 391.83   | 488.25   | 581.71   |
| Rib#4              | -238.60    | -108.60    | 155.65   | 214.12   | 296.17   | 393.71   | 489.83   | 585.53   |
| Rib#6              | -118.60    | -53.60     | 47.73    | 93.77    | 127.16   | 163.30   | 205.86   | 245.86   |
| Rib#9              | -118.60    | -53.60     | 47.68    | 99.97    | 127.57   | 165.92   | 203.26   | 239.97   |
| **Total**          | -716.40    | -324.60    | 310.10   | 624.91   | 854.00   | 1115.25  | 1383.22  | 1646.07  |

3.2. Measurement Instrumentation

In order to measuring the strength of the wing, 4 strain gauges (rosettes type) were placed on difference locations as shown in Figure 4. The general-purpose strain gauges (KFG-10-120-d17-11L5M2S) were used to measure the strain having a nominal gauge length of 10 mm, resistance is 120±0.8 Ω and gauge factor is 2.07±1.0%. All strain gauges have been tested before bonding to desired location to complete the static testing. To obtain the wing deformation, 4 laser sensors (KEYENCE CMOS Multi-Function Analog Laser Sensor – IL series) were used to measure and record data simultaneously.

In order to collect the testing data, a computer data acquisition system (the NI PXIe-8840 PXI Express/CompactPCI Express embedded controller, PXIe-4330/4331 strain gauge module, and PXIe-4302 data acquisition module) was used to measure and record all strains and deformations.
Figure 4. Location of four strain gauges installed on the wing spar

Figure 5. Strain gauges (rosettes type) glued to the wing skin

4. Finite Element Analysis

4.1. The model of finite element

In order to predict the wing structural deformation and strains under the critical loading conditions, ANSYS is used to establish finite element model. The model is established according to the actual wing structure. The continuum elements are used to the adhesive layer between structural elements (spar, root-rib/skin/spar, and skin/skin interconnections). All structural elements adopt Tri/Quad element, as shown in Fig.6. The FE model contains 8,963 elements and 9,963 nodes.

Figure 6. FE model of wing structure

4.2. Boundary Conditions and Loading

The boundary conditions agree with the static structural testing as identified with the side, up, and front directions defined as the X, Y and Z coordinates in Figure 6, respectively. The fixed nodes were assigned at the root of the main and minor spar, while the displacements of nodes are allowed (in Y
direction) for the spigot (underneath the wing). The resultant force in the form of uniform distributed load was assigned at each load location, which was applied to the upper skin.

4.3. Static Analyses Method
To predict wing deformation as well as laminate strains under incremental loading to design ultimate load, geometric linear static FE analyses were performed. In order to identify the potential failure zones of the wing structure, the deformation and strain distributions were used.

Figure 7. Predicted Stress-Strain-Deformation on various loading

5. Presentation of Results
Table 4 shows strain value under different load factor ranging from -2G to 6G (ultimate load). The strain value (Y- direction) occurs the most at the rib#2 closed to the root rib indicated that maximum stress occurs also in the neighborhood of main spar near the root rib. The finite element model shows that the maximum stress on 6G (ultimate load) condition is 707.72 MPa while on 4G (limit load) condition is 536.07 MPa. The comparison between the actual and predicted wing deformation on -2G, 4G, and 6G respectively shown on Figure 8 indicates small error at the tip of the wing. This concludes that the FE model is good enough to represent the wing structural testing.

Under the ultimate load, there are no other failures to appear on the wing, while the maximum deformations are 103.57 mm (measurement) and 104.66 mm (predicted). There is no detrimental or permanent deformation on the structure when unloading both positive and negative side. It is concluded
that the bending strength of wing is strong enough to support ultimate loads without failure for at least 3s.

### Table 4. Strain obtained from the measurement under different load factor

| Load Factor | Rib#2X @ -2G [μm/m] | Rib#4X @ -2G [μm/m] | Rib#6X @ -2G [μm/m] | Rib#9X @ -2G [μm/m] | Rib#2X @ 4G [μm/m] | Rib#4X @ 4G [μm/m] | Rib#6X @ 4G [μm/m] | Rib#9X @ 4G [μm/m] | Rib#2X @ 6G [μm/m] | Rib#4X @ 6G [μm/m] | Rib#6X @ 6G [μm/m] | Rib#9X @ 6G [μm/m] |
|-------------|---------------------|---------------------|---------------------|---------------------|---------------------|---------------------|---------------------|---------------------|---------------------|---------------------|---------------------|---------------------|
| 4G          | 62.70               | 45.94               | 200.99              | 201.00              | 124.15              | -2.07               | -439.17             | -40.54              | 220.59              | 7.66                | -122.38             | 148.87              |
| 5G          | 32.36               | 366.13              | 256.67              | 566.71              | 17.88               | 6.28                | -26.90              | -50.05              | -219.97             | 9.32                | -32.00              | -116.58             |
| 6G          | 38.35               | 279.63              | 157.32              | 181.26              | 76.73               | 2.37                | 303.57              | 17.84               | -194.51             | 8.83                | 44.66               | 88.27               |
| 10G         | 29.73               | 195.60              | 106.18              | 105.20              | 69.01               | 7.02                | 231.78              | 8.95                | 48.80               | 7.18                | 32.07               | 52.09               |
| 15G         | 16.93               | 114.89              | 62.28               | 67.03               | 50.92               | 8.96                | -259.63             | 5.33                | -62.24              | 3.32                | -18.22              | -26.60              |
| 20G         | 9.82                | -54.76              | -24.78              | 30.20               | -4.99               | 7.97                | -80.62              | -1.52               | -53.06              | 3.42                | -5.55               | -15.89              |
| 25G         | 2.14                | -91.47              | -4.60               | -11.86              | -19.20              | -41.72              | -91.51              | -6.03               | -58.06              | -6.36               | -10.30              | -15.29              |
| 30G         | 22.60               | 303.02              | 14.98               | 14.63               | 8.35                | 90.58               | 204.63              | 14.06               | 77.98               | 8.15                | 43.20               | 15.98               |

**Figure 8.** Comparison between Experimental and Predicted Wing Deformation

### 6. Conclusions

Static testing for composite wing of a two-seater seaplane was presented. A test frame was designed to serve as the support structure for the wing static testing. The deformation of wing and measured strain data were collected by computer. After unloading from the limit loads (the maximum loads to the expected in service), the ultimate loads (limit loads multiplied by prescribed factors of safety), and the negative load, there were no sign of detrimental or any form of failure happen to the tested wing. It has been concluded that the wing can support the limit loads and ultimate loads without appearing of permanent deformation and structural failure, therefore the wings can withstand ultimate loads condition without failure and comply with the requirement of the ASTM F2245-16c.

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