A STUDY ON CONCEPTUAL STRUCTURAL DESIGN OF FUSELAGE FOR A SMALL SCALE WIG VEHICLE USING COMPOSITE MATERIALS

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Abstract

This study proposed an example on conceptual structural design of the fuselage for a 20 seats composite WIG vehicle using the netting rule and the rule of mixture, and design evaluation and trade off through structural safety and stability analysis using a commercial FEM tool. The fuselage was designed by a semi-monocoque type structure with the skin to carry shear flow, the stringers to carry bending moments and the ring frames to carry local loads under consideration of various design local load cases. The used materials were the carbon/epoxy laminates using the UD prepreg and the sandwich structure with Al alloy core and carbon/epoxy laminates. Through static stress analysis, structural stability analysis, fatigue life estimation and modal analysis, the structural safety and stability, the required 20 years fatigue life and impossibility of resonance were confirmed.

1 Introduction

Recently, Korean government and industries have been interested in a high speed maritime transportation system using WIG (Wing-In-Ground) effect. The upper section of the WIG vehicle has an airplane configuration and its lower section looks like the hull configuration of a high speed ship or boat. Therefore the WIG can take off and land on sea or lake, and it can be operated by adjusting the flight altitude nearby the sea/or lake surface depending on weather conditions. [1] Because the WIG effect is able to satisfy simultaneously the increase of lift and the decrease of drag, the WIG vehicle can not only run much faster than traditional ships or boats due to flying in the air but also increase the payload due to more increased lift than the airplane's one. This is especially useful in ocean area with lots of islands like Korea, it has a great advantage for high speed transportation as an alternative concept instead of other existing high speed maritime transportation systems. Because the WIG craft has a special corrosive environmental condition on its structure surface due to sea or pure water, it should be free from corrosion for long time use.

Currently according to the literature survey, composite materials have not been used yet for the primary as well as secondary structure of the existing WIG vehicle. Therefore, this study proposed an example on conceptual structural design of the fuselage for a 20 seats composite WIG vehicle using the netting rule and the rule of mixture, and design evaluation and trade off through structural safety and stability analysis using a commercial FEM.

2 Design and analysis procedure

The WIG craft to be treated in this study has a similar aerodynamic configuration to the fixed mid-wing airplane. For instances, upper and lower sections of the fuselage have an airplane configuration for reducing the air drag and a boat's hull configuration for reducing the water drag, and the wing is attached to the middle of the fuselage. The tail fins have a 'T' shape that the horizontal tail was vertically attached to the vertical fin. Because the WIG craft has an airplane like behavior after taking-off, therefore the major fuselage design loads become shear forces and bending moments due to the fuselage weight and the payload of 2 tons passengers and cargo with the maximum load factor in flight operation and impact load at landing.

Therefore the fuselage structure can be sized from these calculated major design loads using selected composite materials.

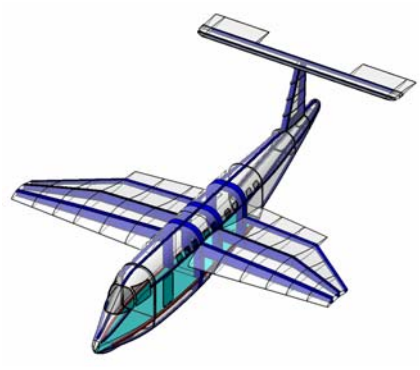

Fig. 1 3-D Model for Whole WIG Vehicle

After an initial design using the netting rule and the rule of mixture, structural analysis was firstly performed to confirm the structural safety and stability using a commercial FEM code PATRAN/NASTRAN. From the structural analysis on the first design configuration, some modifications were drawn due to weak area on buckling and a bit heavier than the target weight. The final structural configuration was fixed through several repeated design modifications and analyses. Figure 1 shows the structural design and analysis procedure of the fuselage applied to this study, and Table 1 illustrates the system specification of the studying WIG craft.

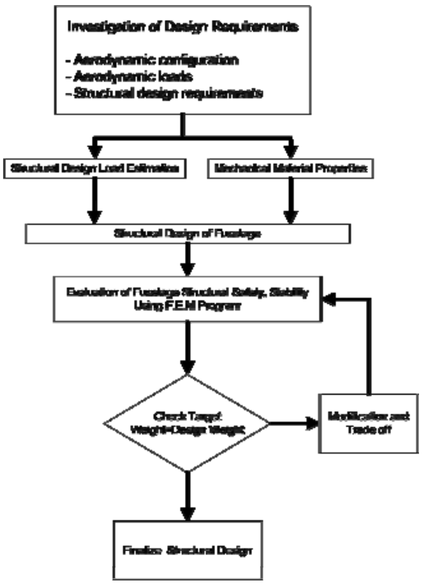

Fig. 2 Structural design procedure of WIG vehicle's fuselage

Table 1 System Specification of Small Scale WIG Vehicle

Length	23.52m
Height	8.15m
Pay Load	18Passengers+2Pilots +luggage
Fuselage Width	2.4m
Gross Weight	8.5ton
Empty Weight	6.5ton
Maximum Speed	170km/h
Operation Condition	Wave Height 2m
Operation Altitude	Flight Height 150m
Engine Power	2×1000 hp
Material	Carbon/Epoxy laminates +Al honeycomb core sandwich

3. Load case definition

It was assumed that the loads applied to the fuselage were divided into the symmetric fuselage load (load case 1), the asymmetric fuselage load (load case 2) and the landing impact loads on the sea/or lake water surface (load case 3-1; the nose splash down case, the load case 3-2; the center body splash down case, the load case 3-3; and the tail splash down case). Where the load case 1 was defined as the fuselage symmetric load with a load factor of 2 at maximum flight speed, load case 2 was defined as the fuselage asymmetric load due to lateral bending and twisting moments of the vertical fin, and load case 3-1, 2 and 3 were defined as the quasi-static impact landing splashdown loads in nose, center (normal landing case) and tail direction, respectively.

3.1. Symmetric Load

According to the previously study results, the maximum total load of the typical fixed wing airplane can be divided into the main wing's lift, the horizontal tail's lift, and the inertia load of the whole vehicle at maximum flight speed with the following relationship. [2]

$$nW = L_{WB} + L_T \qquad (1)$$

Where n = load factor, W = weight, L_{WB} = wing/body lift and L_T = the horizontal tail lift. At L_{WB}, It is assumed that the fuselage generates only 3% among the L_{WB}, and therefore the fuselage lift can be ignored for the conservative design in most design cases. However this design considered this fuselage lift load.

3.2 Asymmetric Load

The asymmetric load of the fuselage is defined as the lateral bending and twisting moment of the vertical fin, and therefore the total twist moment 'T' can be expressed by the following equation.

$$T=L_F h_F + 0.1L_T b \fallingdotseq L_F h_F \quad (2)$$

Where L_F = vertical fin lift, h_F = vertical distance from the aerodynamic center of the vertical fin to the center of gravity of the fuselage, $0.1L_T$ = difference lift between left and right horizontal tail fins, and b = horizontal distance from the aerodynamic center of the horizontal tail to the center of gravity of the fuselage. However the difference lift can be ignored due to small quantity in most design cases. [3]

3.3 Splash down impact load

The splash down impact load can be defined as the quasi-static impact loads in nose, center (normal landing case) and tail direction, respectively. According to the previous study, it was found that the maximum load among three cases of the splash down loads was the center body splash down landing case. However at stress analysis step, all three splash down impact loads were considered. The quasi-static load splash dawn landing load can be expressed by the following equation. [4]

$$A_{impact} = 9 \cdot 10^{-3} M_{\max}^{2/3} \quad (3)$$

$$P^e = \frac{F^e_{impact}}{A_{impact}} \quad (4)$$

$$F^e_{impact} = n^e_{impact} \cdot M_{reduction} \cdot g \quad (5)$$

Where p^e = pressure on impact area, A_{impact} = impact area, F^e = impact load, n^e_{impact} = impact load factor of 1.5.

4. Fuselage design

Because the studying WIG craft has a similar configuration to the fixed wing airplane, the fuselage structure was designed by a semi-monocoque type structure composed of skin, stringers and ring frames.

4.1 Mechanical properties of selected materials

The major material was selected as the sandwich composed of Carbon/Epoxy face sheet and Al honeycomb core. For easy manufacturing the UD and fabric prepregs produced by HexWeb were applied. Table 2 and 3 shows mechanical properties of the Carbon/Epoxy UD prepreg and the Al honeycomb core.

Table 2 Mechanical Properties of Aluminum Honeycomb Core

Material / Property	Al Honeycomb Core1	Al Honeycomb Core2
Compressive Strength (Mpa)	0.69	4
Plate Shear Strength (Mpa)	0.41	1.58
Compressive Modulus (Gpa)	206.84	1185.89
Plate Shear Modulus (Gpa)	20	186.15
Densitty (kgf/mm3)	25.63E-9	66.88E-9

Table 3 Mechanical Properties of Carbon/Epoxy Prepreg

Material / Property	Carbon/Epoxy UD	Carbon/Epoxy Fabric
Longitudinal Modulus (Gpa)	145	63.4
Transverse Modulus (Gpa)	10	58
Axial Shear Modulus (Gpa)	4.8	56.1
Poisson's Ratio	0.25	0.17
Longitudinal Tensile Strength (Mpa)	1240	635
Longitudinal Compressive Strength (Mpa)	1240	572.9
Transverse Tensile Strength (Mpa)	41	411.7
Transverse Compressive Strength (Mpa)	170	304.4
In-Plane Shear Strength (Mpa)	80	114.5
Density(kg/mm3)	1.58E-6	1.58E-6
Thickness(1Ply)(mm)	0.14	0.2

4.2 Skin and stringer design

Based on some assumptions that the skin mainly endures shear and buckling loads and the stringers mainly endues mainly bending moments, they are initially sized. The selected materials was a Carbon/Epoxy materials which is acceptable for most aerospace applications. (See Table 3) Equation 6 and 7 express the relationships between allowable shear and bending stresses and their loads. Therefore the 'T' cross section shape stringers can be sized by the above loads. However because mostly the skin may be very weak for bucking loads due to thin thickness, the skin can be initially designed based on the buckling strengths using equation 8 and 9 . [5]

$$\tau_{allowable} = \frac{4}{3}\frac{V_{\max}}{2\pi Rt} \tag{6}$$

$$\sigma_{allowable} = \frac{M_{\max}}{4A_s y_i} \tag{7}$$

$$\sigma_{cr} = K_c \frac{\pi^2 E}{12(1-\nu^2)}\left(\frac{t}{b}\right)^2 \tag{8}$$

$$\tau_{cr} = K_s \frac{\pi^2 E}{12(1-\nu^2)}\left(\frac{t}{b}\right)^2 \tag{9}$$

Where the buckling coefficients K_c and K_s can be found from buckling load diagrams of the curved shell using $Z = \frac{b^2}{rt}(1-\nu^2)^{\frac{1}{2}}$, $\frac{r}{t}$ = ratio of radius to thickness and $\frac{a}{b}$ = ratio of length to width, respectively. [5] Consequently, the safety condition of the designed feature can be expressed by equation 10 and 11 using R_c ,*and* R_s = shear and normal stress ratios.

$$R_c = \frac{\sigma_b}{\sigma_{cr}},\ R_s = \frac{(\tau_t + \tau_q)}{\tau_{cr}} \tag{10}$$

$$R_c + R_s^{\,2} \le 1;\ Safety\ Condition \tag{11}$$

$Where\ \sigma_b = M_b y_A / I_y\ \tau_t = T/(2\pi r^3 t),\ \tau_q = (q_l + q_r),$

$q_l = circumferential\ shear\ flow\ and$

$q_r = longitudinal\ shear\ flow.$

4.3 Ring frame design

The ring frame for attaching the main wing was designed by the bending moment transferred from the main wing and vertical shear load due to the local. From the following equation 12, [6] the local bending moment and the axial load of the ring frame can be calculated, and then they can size the ring frame with 'U' cross section shape. Other ring frames can be designed by the same manner as the main wing ring frame.

$$M_0 \cdot \sum \frac{(y_1' - y')}{I}\Delta s - P_0 \cdot \sum \frac{(y_1' - y')^2}{I}\Delta s - \sum \frac{M_q}{I}(y_1' - y')\Delta s = \tag{12}$$

Where M_q = bending moment due to shear flow q induced from the local vertical load.

4.4 Floor design

For the floor, the sandwich structure using Carbon/Epoxy composite face sheet and Al honeycomb core was adapted with the following two assumptions. [7] For instances, the face sheet can endure the longitudinal bending moment of the floor and the sandwich core can endure the shear stress. The floor was designed by the maximum concentrated payload due to passengers, and its whole thickness keeps constant for easy manufacturing. For the face sheet and core design the buckling also was considered the fuselage skin design.

4.5 First design result

The skin thickness was designed as 7 mm by shear flow loads in consideration of the buckling loads. The stringers were designed to carry major fuselage bending loads, and its thickness was designed as a constant thickness of 5 mm with a 'T' cross section shape from the nose to the rear of the fuselage. The ring frames were designed to carry local loads and were divided into three different types depending on fuselage locations with a 'U' cross section shape. The ring frame be installed to the main wings were designed to carry the shear force and the bending moment of the main wings and its thickness was designed at 10 mm. The ring frame to be installed to the tail stabilizers were designed using the same method as the main wing ring frame, and the other front and rear ring frames were designed to carry the distributed basic weight and the passenger payload under the load factor of 2 with its thickness being designed at 4mm and 4.5mm, respectively.

5. Design modifications

5.1 Design modification and final design feature

In order to investigate the structural safety, the initial design feature of the fuselage was modeled as surfaces from the 3-D CATIA drawings after confirmation of detail configurations such as joint parts, spars, ring frames and so on. In the numerical structural analysis of the WIG fuselage, the static stress analysis, the structural stability analysis and the modal analysis were performed using the commercial FEM code PATRAN/NASTRAN. The applied failure theory for safety evaluation was the well-known Tsai-Wu failure criteria. Total number of elements for FEM mesh generation were 35546 including 19080 for the fuselage skin mesh.

According to the structural analysis results of the first design feature, it was found that the ring frame for main wing attachment and the ring frame for tail stabilizer attachment were very weak against the buckling loads, and therefore the frames were repeatedly modified by the increase of its lay-up thickness until satisfaction on safety against buckling. The skin of the fuselage used the Al honeycomb core for reduction of the sandwich weight. However because it was generally weak against buckling, and therefore the proper Al honeycomb core thickness was sized in consideration of buckling using the same method as the skin design. Because the ring frame for tail stabilizer attachment also was very weak against buckling, the modified Al honeycomb core thickness became much thicker than the first designed one. Table 4 shows flow of the structural design modifications and Table 5 shows the final structural design result of the fuselage. Figure 3 shows final design feature of skin, stringers and Ring Frames.

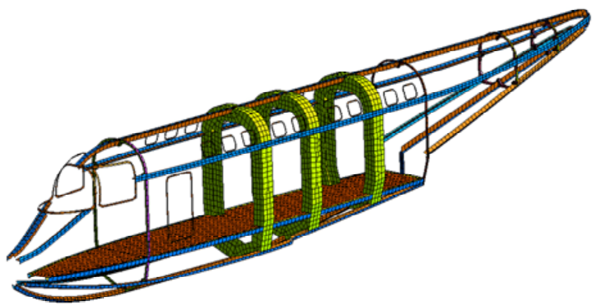

Fig. 3 Final Design Feature of Skin, Stringers and Ring Frames

Table 4 Flow of Design Modification

	Skin	Stringer	Frame				Floor		weight
			Front	Wing Join	Rear	Tail Join	Skin	Core	
1st	7t	5t	4t	10t	4.5t	5t	2.8t	24t	3299kg
2nd	4.8t	9t	4.2t	3.78t (10t)	6.6t	1.96t (30t)	1t	24t	2526kg
3rd	1t	6.2t	4.4t (5t)	3.78t (20t)	2.8t (20t)	36.84t (40t)	1t	10t	1205kg
4th	1.6t	6.2t	2.8t (5t)	2.24t (20t)	2.8t (20t)	36.84t (50t)	1t	10t	1250kg
5th	1.6t	6.2t (20t)	2.8t (5t)	2.24t (20t)	2.8t (22t)	8.4t (40t')	1t	10t	1190kg
6th	1.6t (7t)	6.2t (20t)	4.4t (5t)	2.24t (20t)	4.8t (22t)	6.2t (40t')	1t	30t	1208kg

Table 5 Final Design Results Using Carbon/Epoxy/Al-Core Sandwich Structure

Part		Thickness	Ply	Material	Orientation
Skin		7t	35ply	Fabric	7[±45°3,0°/90°,±45°]
Stringer		5t	36ply	UD	3[0°3,45°,0°3,90°,0°3,-45°]
Frame	Front	4t	30ply	UD	3[90°,0°3.90°]s
	Wing join	10t	72ply	UD	4[0°3,45°,0°3,-45°,90°]s
	Rear	4.5t	33ply	UD	3[0°3,90°,0°3,45°,0°2,-45°]
	Tail join	5t	36ply	UD	3[0°3,45°,0°3,90°,0°3,-45°]
Floor	Skin	2.8t	14ply	Fabric	2[0°/90°,±45°,0°/90°,±45°,0°/90°±45°,0°/90°]
	Core	24t	-	Al honey comb	-
	Bond	5t	25ply	Fabric	5[0°/90°,±45°,0°/90°,±45°,0°/90°]

6. Safety evaluation of final design feature

6.1 Static stress analysis result

It was found that the weight of the final design fuselage structure was 1208kg which is a bit heavier than the target weight of 1171kg. As shown in Table 6 and Figure 4, the maximum stress occurred on the ring frame for main wing attachment.

Table 6 Structural Analysis Results

Case of analysis / Analysis result		Symmetric Load (Load Case 1)	Asymmetric Load (Load Case 2)	Splash Down (Load Case 3-1)	Splash Down (Load Case 3-2)	Splash Down (Load Case 3-3)
Max. stress [Mpa]	Ten.	260	253	205	205	194
	Com.	5.1	2.6	3.04	3.04	2.91
Max. disp. [mm]		7.95	16.2	17.8	29	20

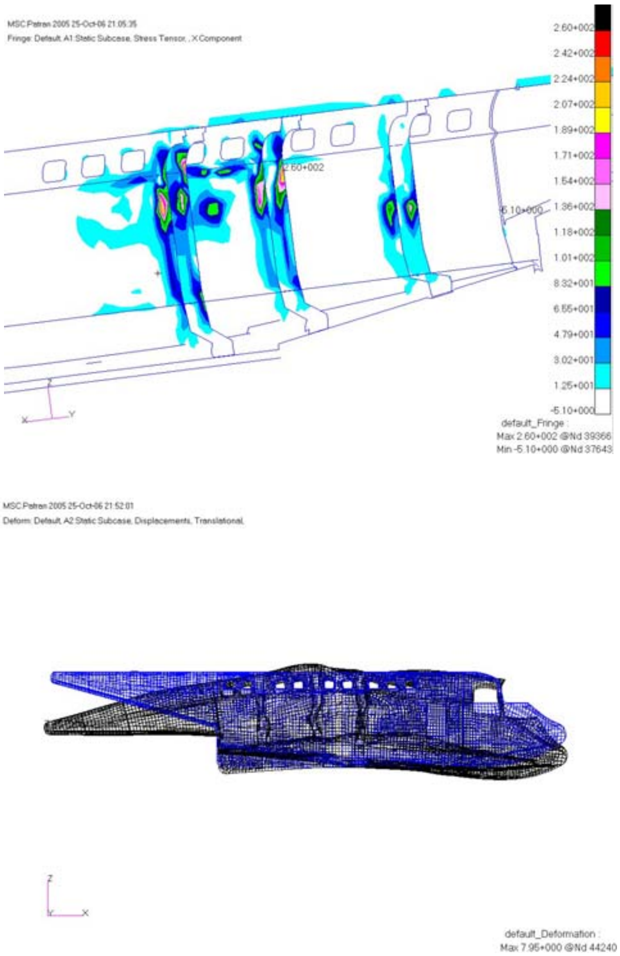

Fig. 4 Stress Contour and Deformed Shape of Fuselage in Load Case 1

6.2 Buckling analysis result

For checking the structural stability of the final design fuselage feature, all the load cases such as symmetric, asymmetric and splash down landing impact load cases were considered. Table 7 and Figure 5 show the buckling analysis results.

Table 7 Buckling Analysis Results for various load cases

Case of analysis / Analysis result	Symmetric Load (Load Case 1)	Asymmetric load (Load Case 2)	Splash Down (Load Case 3-1)	Splash Down (Load Case 3-2)	Splash Down (Load Case 3-3)
Buckling Load Factor	1.88	1.93	4.14	1.25	4.19

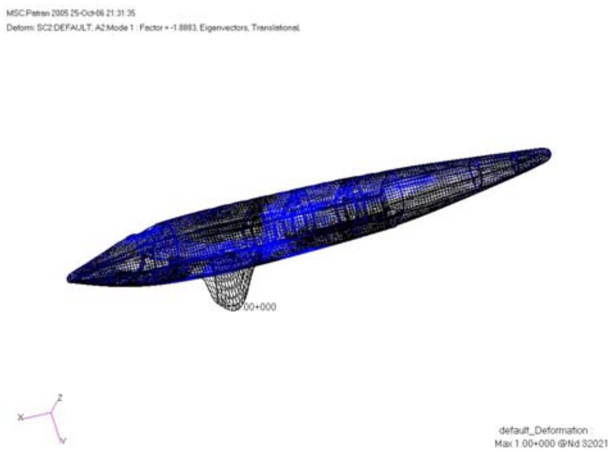

Fig. 5 First Buckling Mode Shape and Load Factor in Load Case 1

6.3 Safety evaluation

According to safety evaluation results using Tsai-Wu failure criterion, it was found that the final design fuselage feature was safe against some critical load cases such as symmetric, asymmetric and splash down landing impact load cases. Table 8 and Figure 6 show the safety evaluation results.

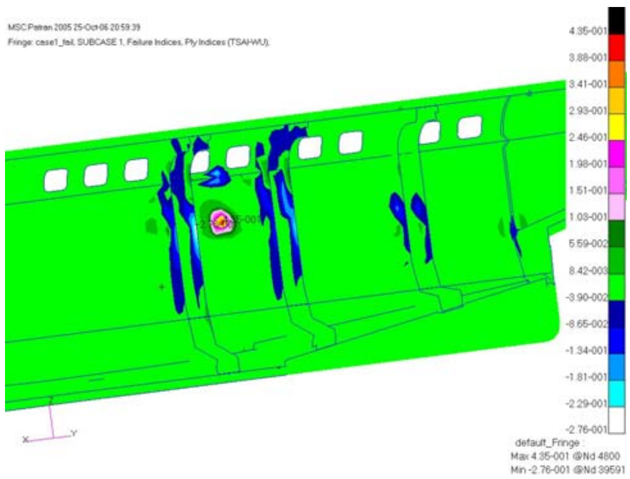

Fig. 6 Safety Factor Distribution by Tsai-Wu Failure Criterion in Load Case 1

Table 8 Safety Evaluation Result Based on Tsai-Wu Failure Criterion

Case of analysis / Analysis result	Symmetric Load (Load Case 1)	Asymmetric load (Load Case 2)	Splash Down (Load Case 3-1)	Splash Down (Load Case 3-2)	Splash Down (Load Case 3-3)
Failure Criterion	0.43	0.67	0.35	0.35	0.34

6.4 Modal analysis result

Through the Campbell diagram, the possibility of resonance due to engines and propellers was investigated, and the first flapwise bending mode with 0.594Hz was not able to make a resonance with the engine rotational speed of 2378rpm as well as propeller rotation with three blades. Figure 7 shows the Campbell diagram, and Table 9 shows the lower three mode shapes and frequencies, respectively.

Table 9 Summary of Modal Analysis Results

Case of analysis / Analysis result	Mode 1	Mode 2	Mode 3
Frequency (Hz)	0.59	0.6	0.61

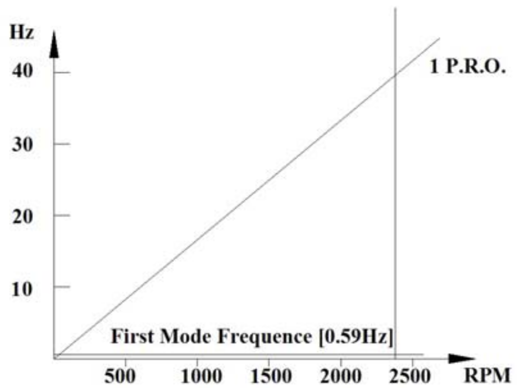

Fig. 7 Campbell Diagram for Final Design Fuselage

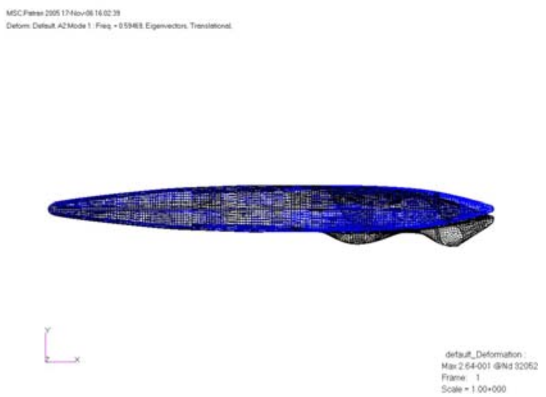

Fig. 8 First Mode Shapes and Natural Frequencies of the Final Design Fuselage

7. Fatigue life estimation

The fatigue life of the final design feature was estimated using the S-N diagram based on the reference Carbon/Epoxy materials, and confirmed the required system fatigue life of 20 years. If the safety factor of 3 may be considered, it becomes WIG craft will operate with the following assumption such as 12 times 1 hour flight per a day, total number of flights during 20 years can be calculated as 87699 times. If the safety factor of 3 may be considered, the total number of flights can be modified as 262,800 times. Moreover it was assumed that the fatigue strength was reduced by 85% due to operation in sea water environment.

Because the maximum compressive stress during operation was 295MPa on the ring frame for the main wing attachment and the maximum compressive strength of the selected composite material was 1250MPa, the stress ratio becomes 0.23. Figure 9 shows the estimated fatigue life result based on stress ratio and the S-N diagram of the selected Carbon/Epoxy composite material. In this estimation, it was confirmed that the final design fuselage feature may have enough fatigue life during the required 20 years operating period.

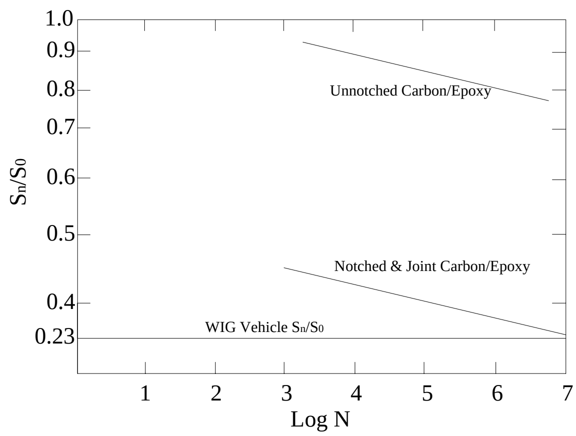

Fig. 9 S-N Curves for Various Carbon/Epoxy

8. Conclusion

In this study a composite fuselage structure, which can satisfy the target weight as well as the structural safety and stability, was designed. The fuselage was designed by a semi-monocoque type structure with the skin to carry shear flow, the stringers to carry bending moments and the ring frames to carry local loads under consideration of various design load cases. The used materials were the carbon/epoxy laminates using the UD prepreg and the sandwich structure with Al alloy core and

Carbon/Epoxy face sheet. Through the static stress analysis, the structural stability analysis, fatigue life estimation and the modal analysis, the structural safety and stability, the required 20 years fatigue life and impossibility of resonance for the final design fuselage feature were confirmed

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