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Development of FBG-MFC hybrid SHM system for aircraft composite structures in collaboration study with Airbus

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Abstract

Fuji Heavy Industries (FHI) has been developing a Structural Health Monitoring (SHM) System for Carbon Fiber Reinforced Plastic (CFRP) structures of aircraft. The SHM system can detect damages such as delamination and debonding in aircraft composite structures and it is thought as one of the most important technologies for new generation commercial aircraft. Japan Airbus SHM Technology for Aircraft Composite (JASTAC) project was launched in 2006 to develop the SHM technology and to discuss practical use of it between Japan and Airbus. FHI has been conducting collaboration study with Airbus about the SHM technology as a member of the JASTAC project. In this paper, outline of the SHM system and results of the joint tests for the SHM system with Airbus are discussed.

Keywords: aerospace, composite, guided waves (lamb waves), Ultrasonic, FBG, Special session Airbus-Japan

1. INTRODUCTION

Recently, development of SHM technology has progressed along with expansion of implementation of composite materials into various commercial aircraft structures[1-2]. Composite materials have plenty of advantages such as high strength, high stiffness and good corrosion resistance and these benefits contribute reducing structural weight and maintenance cost especially for latest large commercial aircraft. On the other hand, there might be characteristic damages in aircraft composite structures such as debonding and delamination. Almost of these damages cannot be detected by visual check from outside of structure and Non-Destructive Inspection (NDI) is only way to detect it. SHM technologies are thought as essential ones of current and future commercial aircraft to reduce maintenance and inspection cost because the SHM technology can detect damages including debonding and delamination and diagnose structural integrity.

Various SHM technologies have been developed all over the world, among those, FHI's SHM technology which is using optical fiber have many advantages. Principal feature of our SHM system is to diagnose structural integrity by using a Lamb wave which is generated by a Macro Fiber Composite (MFC) piezoelectric actuator and is measured by a Fiber Bragg Grating (FBG) optical fiber sensor. The optical fiber is lighter than other conventional electric sensor system and is not affected by electromagnetic interference. Moreover, the FBG sensor can measure wider frequency range of the Lamb wave than usual piezoelectric ultrasonic sensor. In addition, the MFC actuator can propagate the Lamb wave which has high directionality and wide frequency range. The Lamb wave is a type of ultrasonic wave and it can propagate long distance through a thin plate thus the Lamb wave is suitable for

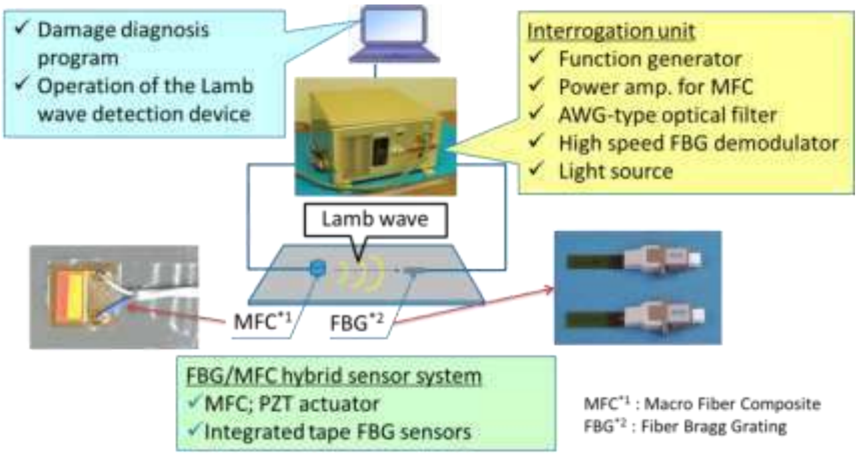

Figure 1 Overview of the developed SHM system.

aircraft composite thin plate structures. In addition, the SHM system can diagnose damage size and location qualitatively or quantitatively by analyzing change of the Lamb wave such as a reflection or a waveform change.

Figure 1 shows an overview of the SHM system. The FBG sensor has a portion of grating in which the refractive index changes periodically. Center wavelength of the light reflected from the grating varies with the grating period. An arrayed waveguide grating (AWG) is used as a filter to detect changes in center wavelength of the FBG sensor, and a high-speed demodulator can measure a change in center wavelength of the FBG sensor as a voltage change in the SHM interrogation unit. Therefore, the SHM system can measure Lamb waves precisely and can diagnose damages in aircraft composite structures by analyzing changes of the Lamb wave. There are several procedures for analyzing Lamb waves to evaluate some types of structural damage.

In previous study, damage detection capability of the SHM system was evaluated and confirmed that this system could diagnose various damage conditions by using several way of analyzing the Lamb wave in aircraft composite structures[3-5].Figure 2 shows outline of the diagnosis procedures for debonding damage in the edge of skin/stringer bonded structures and generally this kind of region is called “hot-spot” which means structural critical point. Damage length can be evaluated by arrival time of the Lamb wave in this case because a propagating path of the Lamb wave is elongated when debonding occurs in the “hot-spot” region.

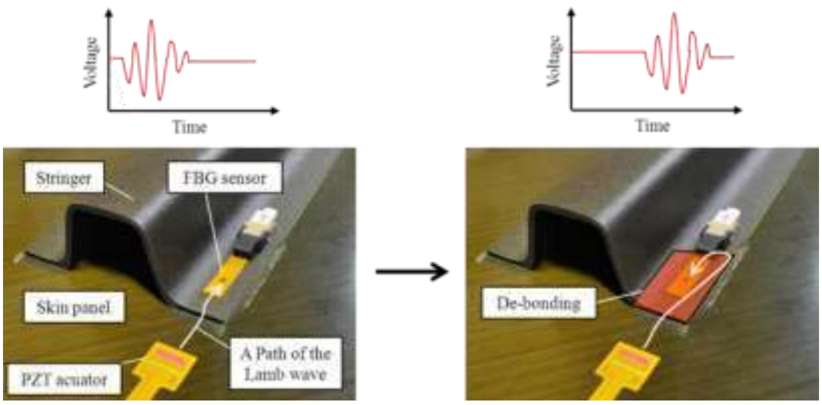

Figure 2 Principle of the "Time delay" analysis method.

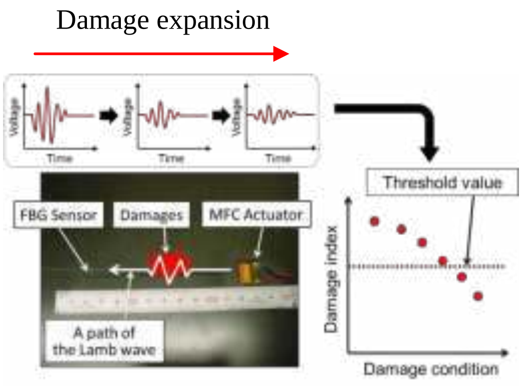

Figure 3 Principle of the "Waveform change" analysis method.

Another way to diagnose structural condition is evaluating difference of waveforms by using various signal processing procedures. Figure 3 shows basic principle of the procedure named "Waveform change" analysis method. Most important feature of the procedure is that adaptive regions are not limited within "hot-spot" but also wider region of aircraft composite structures. The analysis method can diagnose damages between a sensor and an actuator by using the waveform change of the Lamb wave even if the Lamb wave doesn't reflect at the damage region. Cross-correlation analysis is one of the signal processing procedures and it can indicate damage condition as a value of damage index. The damage index is decreased along with growth of damage length or size and when the index value reach a threshold value the SHM system makes an alert of damage detection.

FHI has been joining JASTAC collaboration and evaluating damage detection capability of the SHM system with multiple test opportunities from Airbus. Evaluation of damage detection capability of the system is main task of the project and the capability has been demonstrated by using various test specimens and structures.

Moreover, one of the long term objectives of developing this technology is realize "Smart structure" as an element of new generation aircraft composite structures. The optical fiber will be embedded into the "smart structure" to diagnose structural integrity like a nervous system of human body. In order to evaluate the effect of embedding optical fibers into aircraft composite structures against its mechanical properties, material qualification tests which are foundation of a Building Block Approach concept in figure 4 were also conducted in the JASTAC project and the results are shown in this paper.

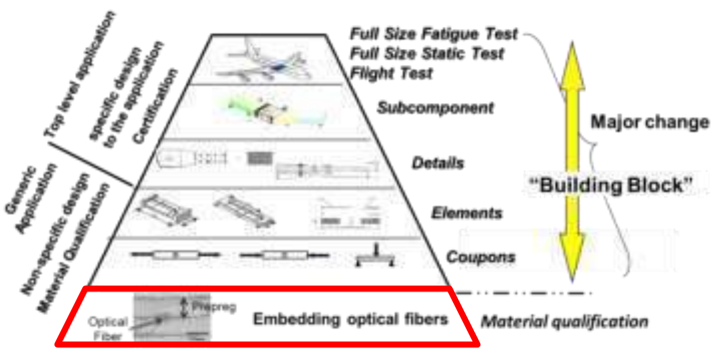

Figure 4 Building Block Approach concept.

2. MATERIAL QUALIFICATION TEST

2.1. Test procedure

The concept of embedding the optical fiber system requires evaluation of adverse effect by the embedding procedures and materials against mechanical properties of aircraft composite structures. Therefore, the material qualification tests have been conducted in the JASTAC project. Main objective of the material qualification test is gathering basic data of effect of the embedding procedures against aircraft composite structures and the test results will be foundation to support the Building Block Approach concept. Thus, the material qualification tests are independent from other evaluation tests of damage detection capability of the SHM system.

Table 1 shows test items for the material qualifications. Tensile test, compression test, shear test, fatigue test, Compression After Impact (CAI) test, G_{IC} and G_{IIC} evaluation test were conducted. These items are chosen in accordance with non-disturbance concept and the items are mandatory for validation of mechanical properties of the structures. Directions and locations of optical fibers were considered as parameter of the material qualification tests in addition to configuration of the specimens or bonding procedures because influence of

Table 1 Test items for the material qualification tests.

No.	Type of tests	RT	HW	LTD
1	Non Hole Tension(0°Lamina)	v		v
2	Non Hole Tension(90°Lamina)	v	v	
3	Non Hole Compression(0°Lamina)	v		
4	Non Hole Compression(90°Lamina)	v		
5	In plane shear strength(±45°)	v		
6	CILS, co-cure	v	v	
	CILS, 2ndary bonding	v	v	
	CILS, Co-bonding	v	v	
7	Non Hole Tension(Quasi-isotropic)	v		
8	Non Hole Compression(Quasi-isotropic)	v		
9	Open Hole Tension(Quasi-isotropic)	v		
10	Open Hole Compressoion(Quasi-isotropic)	v		
11	CAI (Quasi-isotropic)	v		
12	Filled hole tensile strength	v		
13	Filled hole compression strength	v		
14	Bearing Strength(Quasi-isotropic)	v		
15	Double Lap Shear(Quasi-isotropic), 2ndary bonding	v	v	
	Double Lap Shear(Quasi-isotropic), Co-bonding	v	v	
16	ILSS, Co-cure	v		
	ILSS, 2ndary bonding	v		
	ILSS, Co-bonding	v		
17-1	GIc, Adhesive line, Co-cure	v	v	
	GIc, Adhesive line, 2ndary-bonding	v	v	
	GIc, Adhesive line, Co-bonding	v	v	
17-2	GIc, Two layers below adhesive line, Co-cure	v	v	
	GIc, Two layers below adhesive line, 2ndary bonding	v	v	
	GIc, Two layers below adhesive line, Co-bonding	v	v	
18	GIIc, Static, Co-cure	v	v	
	GIIc, Static, 2ndary bonding	v	v	
	GIIc, Static, Co-bonding	v	v	
19	Fatigue(Quasi-isotropic), Tension—Tension	v		
20	Fatigue(Quasi-isotropic), Tension-Compression	v		
21	Fatigue(Double lap shear), Tension—Tension	v		

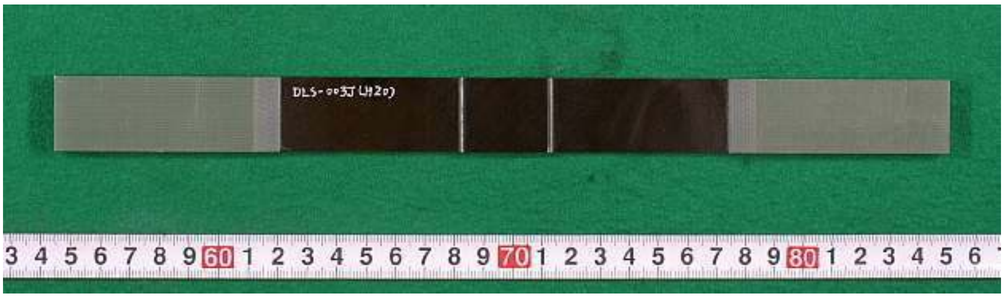

Figure 5 Overview of the specimen for the material qualification test.

optical fibers to mechanical properties are main objective of the tests. Small diameter optical fibers were developed and chosen for embedding to reduce adverse effect against mechanical properties of aircraft composite structures. Figure 5 shows typical specimen for material qualification tests. Optical fibers were embedded into specimens and its appearance is completely same as usual specimen without optical fibers. Important test items were selected in accordance with the non-disturbance concept because test numbers of complete material qualification for real certification procedure will be enormous. In the paper, the results of tensile test, fatigue test and G_{IC} evaluation test will be shown and discussed. These results are typical mechanical properties of aircraft composite structures and the material qualification tests were conducted under various environmental conditions.

2.2. Results and Discussion

Figure 6-8 show a part of results of material qualification tests. There is no significant difference of tensile strength between with and without optical fibers in figure 6. Moreover, variability of strength value which is described by error bar length is not so large in each test conditions. These results show that embedding optical fibers doesn't affect against tensile strength of aircraft composite structures. Figure 7 shows typical results of fatigue test and these are same as tensile test. Variability of cycle number to breaking was same level regardless of with and without the optical fiber in each fatigue load conditions. Diameter of the small type optical fiber is about 50 μm and this value is smaller than half the value of diameter of normal optical fibers. Small diameter optical fibers don't introduce delamination, large void, resin-rich region and fiber waviness into composite structures during layup and curing procedures. Thus, these good results of tensile and fatigue test are thought as advantageous effect of small diameter optical fiber. Almost of other test items show that embedding small diameter optical fibers don't affect the strength and stiffness of structures.

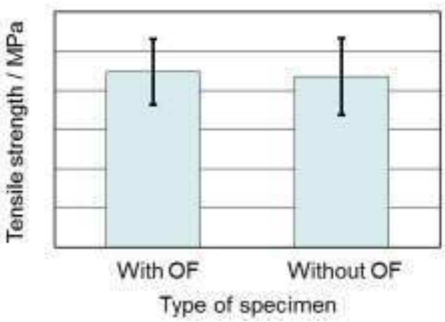

Figure 6 Results of Non-Hole Tensile test (Quasi-isotropic).

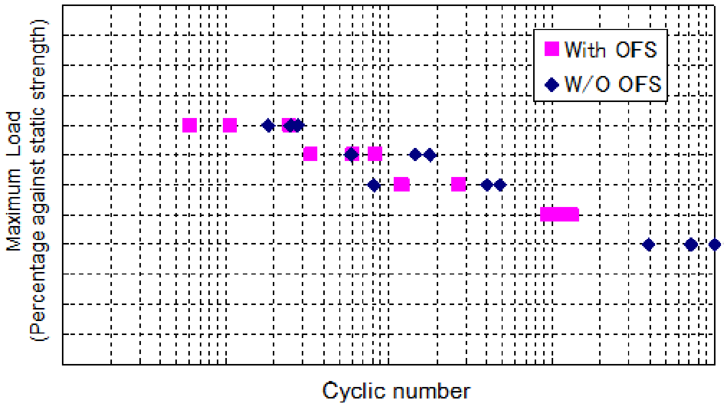

Figure 7 Results of fatigue test.

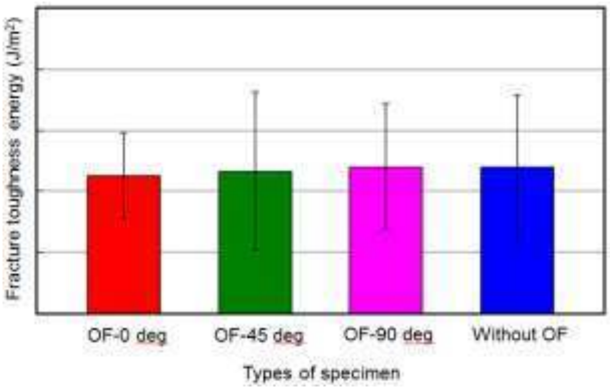

Figure 8 Results of G_{IC} evaluation test.

Figure 8 shows results of G_{IC} evaluation test. There are four conditions about direction of the optical fiber (0deg, 45deg, 90deg, without optical fibers) and no significant differences of average G_{IC} value and variability were observed between each test conditions. Thus, embedding optical fibers don’t make large decrease for G_{IC} value.

The results of the material qualification tests shows that embedded small diameter optical fibers will not make adverse effect against mechanical properties of aircraft composite structures. Adequate data could be gathered about "smart structure" as basic knowledge of the long term objective of the development of the SHM technology.

3. JOINT TEST WITH AIRBUS HTP DEVELOPMENT STRUCTURE

3.1. Test procedure

Evaluation of damage detection capability of the SHM system with actual aircraft composite structures is important task for realization of this new generation technology. Various evaluation tests have been conducted in the JASTAC project and Airbus provided several sub-component and component level structures for evaluation test of this system. The largest test structure was joint test with Airbus Horizontal Tail Plane (HTP) development structure and results of the joint test are discussed in this paper.

Figure 9 shows overview of the Airbus HTP development structure. This test opportunity was provided by Airbus and verification of the SHM system was one of the main objectives of the verification test. The test structure consisted of large flat panel and T-shaped stringers.

The flat panel was manufactured by automated layup procedure and the stringers were bonded with the panel by co-bonding process. The FBG optical fiber sensors and the MFC actuators were bonded into the edge of the skin/stringer bonded structure called "hot-spot". Location of the "hot-spot" and configuration of installed sensor system are also shown in figure 9. Several pairs of the FBG sensor and the MFC actuator were bonded on the panel and the stringer at "hot-spot" by using epoxy type adhesive for aircraft composite structures. Thin Teflon sheet (20*30mm) was inserted into the bond line of a "hot-spot" as an artificial defect and Barely Visible Impact Damage (BVID) was introduced by using a mobile impactor under a web of another "hot-spot". Initial Lamb wave is measured before start of test procedure as a standard of signal processing to diagnose structural condition. Then, tensile load was applied to the test structure and debonding and delamination was occurred at the “hot-spot” by the loading. The load was applied up to a certain value then it was eliminated. After unloading, the Lamb wave was measured and actual damage area was determined by using conventional Non-Destructive Inspection (NDI) system. Loading and unloading processes were repeated several times up to 150 kN, 250 kN, 270 kN and 290 kN.

Measured Lamb waves were analyzed by using cross-correlation analysis and relationship between the damage index and the actual damage area was compared to evaluate damage detection capability of the SHM system. All test procedures were conducted at test site of Airbus at room temperature and dry condition.

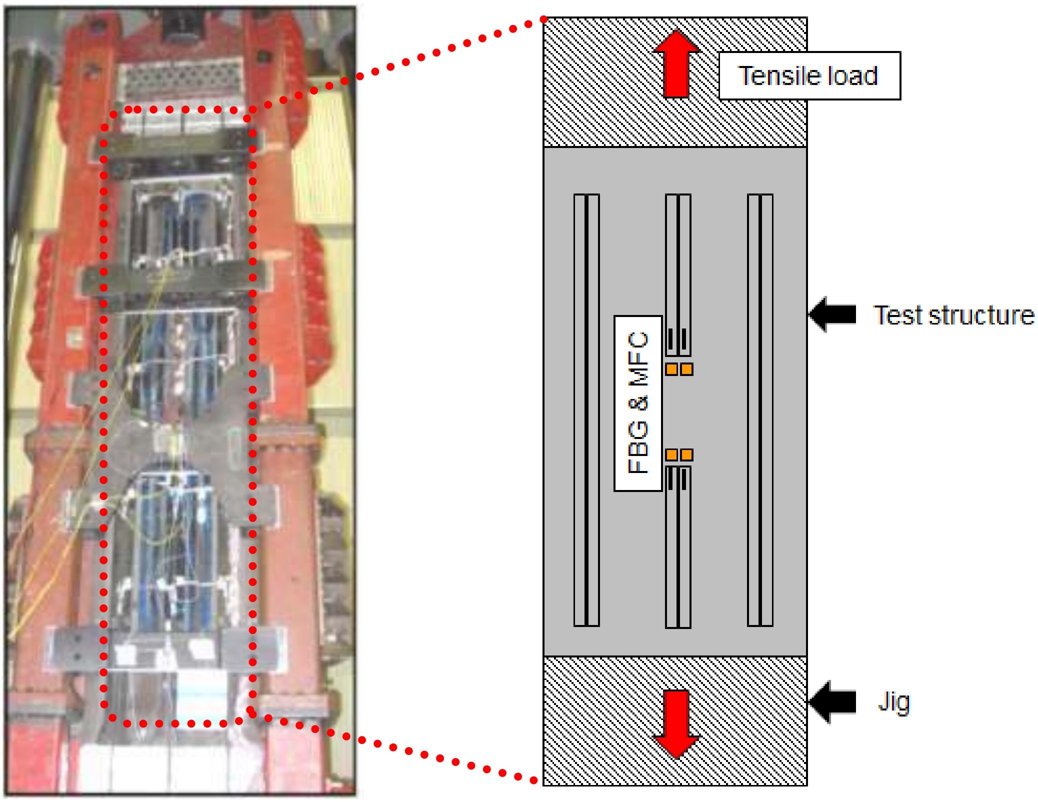

Figure 9 Overview of the Airbus HTP development structure.

3.2. Results and Discussion

Figure 10 shows typical Lamb waves measured at BVID introduced “hot-spot”. Blue curves in each figures show the Lamb wave measured at initial condition and red curves are Lamb waves measured after applying each loading.

Although slight change of the waveform is observed after applying 150kN load as shown in figure 10 (a), waveform change expanded along with increase of applied load from 250 kN to 290 kN as shown in figure 10 (b)-(d). Damage area in this “hot-spot” was not large after applying 150 kN and major damage was observed after applying 250 kN. Trend of the waveform change is well fit with damage expansion along with increase of the loading. Figure 11 also shows measured Lamb waves at "hot-spot" with artificial damage by thin Teflon sheet insertion. Significant waveform change is not observed up to 270 kN but amplitude of the waveform decreased rapidly after applying 290 kN. The drastic change might be caused by cutting a path of the Lamb wave by debonding between skin and stringer.

Figure 12 shows overview of damage condition after applying tensile load 290 kN near the "hot-spot" with BVID observed from skin panel side. Damage region was determined by the NDI system and indicated directly on the panel. The damage index was calculated by analyzing measured Lamb waves with the SHM system. Cross-correlation analysis was applied for calculating these data to show change of waveform as damage index value. Figure 13 shows that the damage index decreases gradually along with expansion of damage area. Damage detection capability of the SHM system is evaluated by using sub-component size test structure and the results show that the system can diagnose structural integrity by calculating with measured Lamb waves.

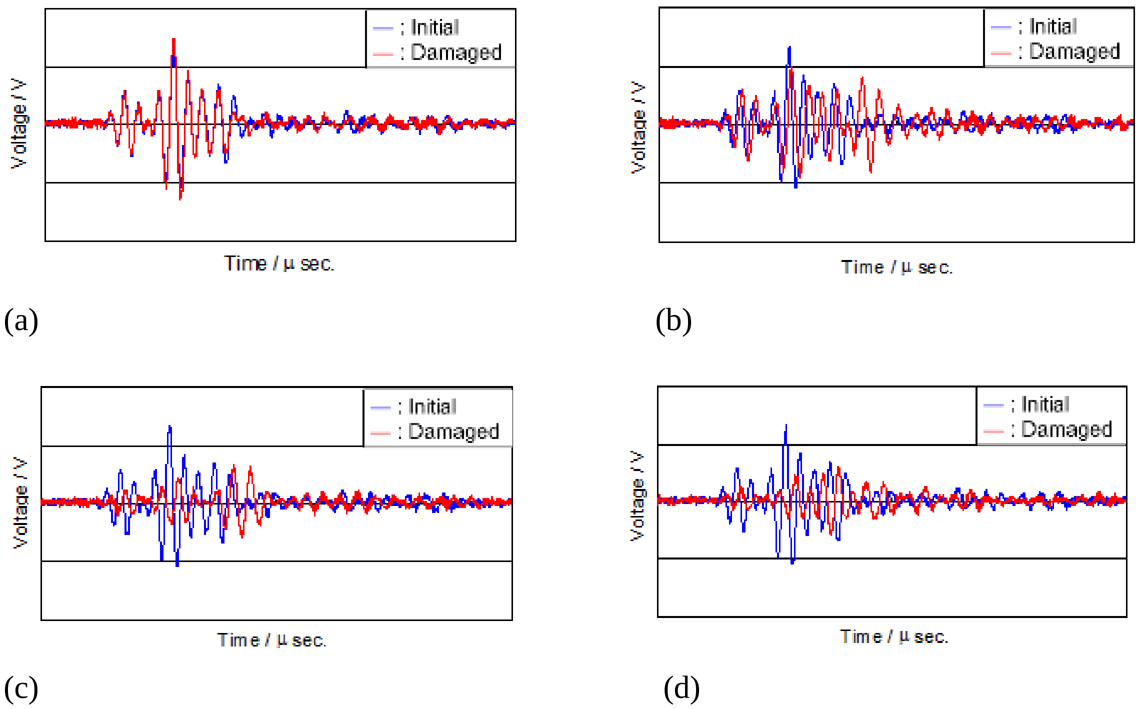

Figure 10 Measured Lamb waves at “hot-spot” with BVID.
(a)Aft. 150 kN (b)250kN (c)270 kN (d)290 kN

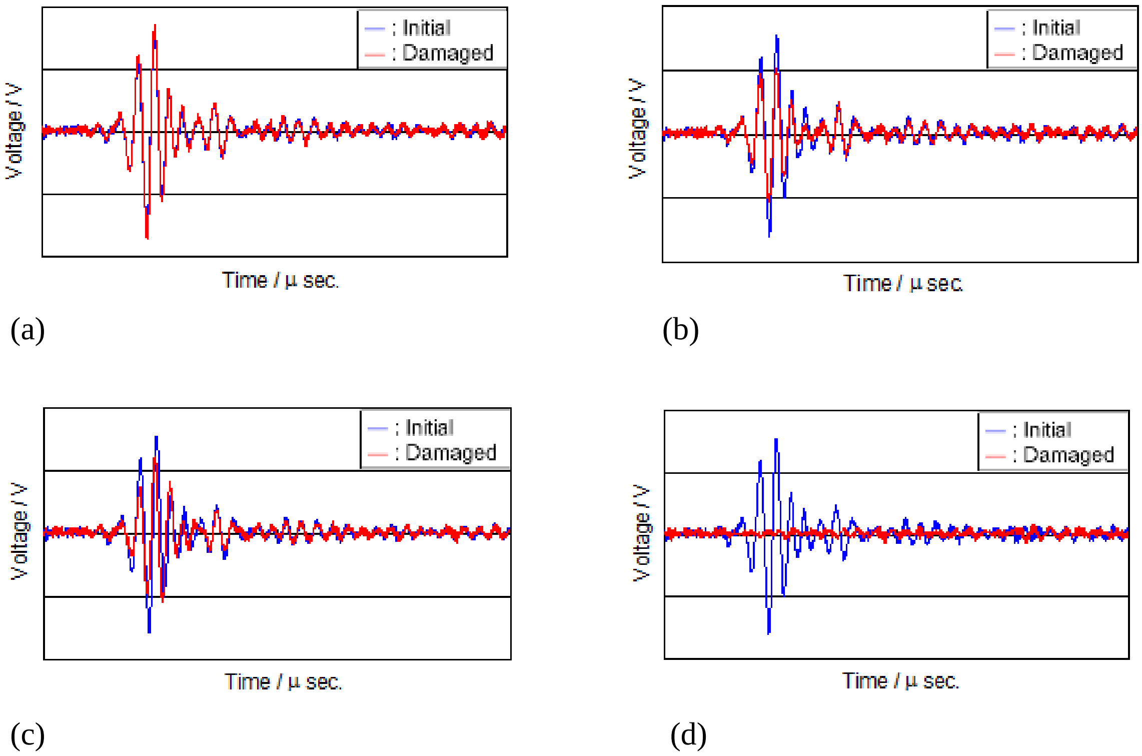

Figure 11 Measured Lamb waves at “hot-spot” with artificial defect.
(a)Aft. 150 kN (b)250 kN (c)270 kN (d)290 kN

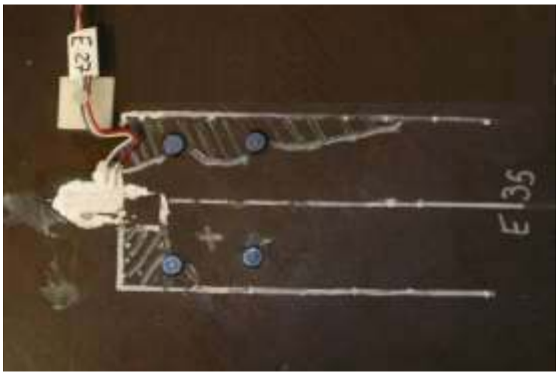

Figure 12 Overview of damage condition near the "hot-spot" after applying load 290 kN.

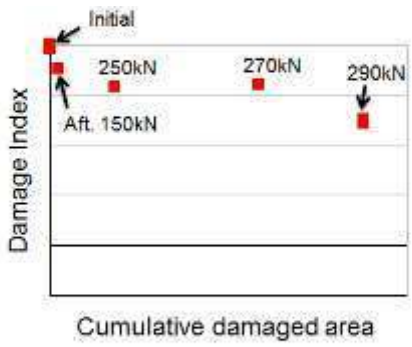

Figure 13 Results of cross-correlation analysis

4. CONCLUSION

FHI has been developing the SHM System for aircraft composite structures and damage capability of the SHM system was evaluated among the joint development scheme between Japan and Airbus called JASTAC project.

The material qualification test was conducted to support the Building Block Approach concept of step-by-step development of the fiber embedded structure. The material qualification test show that embedding optical fibers don't make adverse effect against mechanical properties of aircraft composite structures. These results will be fundamental data of certification procedure for "smart structure" which is one of the long term and ultimate objective of the development of the SHM system.

Joint test with Airbus HTP development structure was conducted to evaluate damage detection capability of the SHM system and the results show the SHM system can diagnose structural integrity by analyzing measured Lamb waves.

These results are necessary for development of the SHM system in accordance with the Building Block Approach concept and it could be considered as a great achievement of the JASTAC project.

5. ACKNOWLEDGEMENT

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