1 Introduction
Fatigue life of a helicopter rotor blade can be evaluated applying safe life, flaw tolerant safe life and fail safe concepts individually or two or more of those in combination as presented in the airworthiness regulations of rotorcraft, FAR-27/29[1,2] and AC-27/29[3,4]. In here, safe life is to determine the replacement time by evaluating the time when fatigue failure will occur with an assumption that there is no flaw or damage, and fail safe intends to set the inspection interval by evaluating the time when the object will reach the threshold limit under the condition that growth of flaw and damage are allowed. On the other hand, flaw tolerant safe life intends to evaluate the inspection interval under the condition that the flaw and damage already applied do not grow any more, and if the inspection interval exceeds the replacement time, no inspection interval is required to be separately set.

Fatigue evaluations for rotor blades of commercial or military rotorcraft have been carried out using the safe life concept since 1950s. Particularly, in the case of a rotor blade made of a composite material, a highly reliable fatigue life could be predicted by evaluating the cumulative damage using combination of fatigue life curve (S-N curve) and load spectrum. But, there is a limit in adequately evaluating the strength reducing phenomena caused by flaws or defects generated during the manufacturing process or impact damage induced by operational usages, using only the safe life concept. Accordingly, the necessity to consider the effect of flaw and damage when evaluating fatigue of composite components of rotorcraft came to fore after 1980s[5], and the requirements for fatigue evaluation using flaw tolerant safe life and fail safe concepts were reflected on the airworthiness regulations.

As shown in Figure 1, the main components comprising a composite rotor blade include spar, rib, skin, and core. Spar is a part which supports the centrifugal force generated by rotation of the blade and the flap moment and lead-lag moment generated by aerodynamic force, and is manufactured using unidirectional glass fiber (UD glass, 0°) which is flexible and has superior formability. Rib forms a torsion box in combination with a spar to support the torsional moment, and skin is a component which maintains the aerodynamic shape of the blade and is manufactured using carbon fabric (± 45°). Also, a rotor blade has a disadvantage that the internal structure is easily damaged when the blade collides with foreign object damage (FOD) due to the structural characteristic of the blade of lacking strength in the thickness direction. Accordingly, fail safe concept can be applied as a method to evaluate the fatigue life taking such characteristics into account. But, any efficient inspection method to identify growth of flaw and damage has not been presented up to now, and a fast rotating dynamic component such as a helicopter rotor has a problem of requiring much maintenance cost as the inspection interval cannot be but shorter than that of fixed wing aircraft.

Fig.1 Internal structures of composite rotor blades
On the other hand, flaw tolerant safe life, equivalently to the damage tolerance concept, can consider the effect of flaw and damage and, at the same time, efficiently and economically evaluate the fatigue life of a rotor blade made of composite materials, and is also a method applied by advanced helicopter manufacturers [6,7]. In this study, we presented the method to conduct a fatigue test of a rotor blade made of composite materials on the basis of flaw tolerant safe life concept and the procedures to evaluate fatigue life using the test data.

2 Application of Flaw and Damage

2.1 Flaw Tolerant Safe Life

When testing a rotor blade made of a composite material by the evaluation method of flaw tolerant safe life, the flaw or damage already applied should not grow to a defect measurable within the preset replacement time or inspection interval, and should not generate another defect. Also, it should be verified that no structural failure will occur even when an ultimate load is applied for 3 seconds or longer through a residual strength proof test.

2.1.1 Manufacturing Defect

The main causes which can generate defects during manufacturing process of a blade made of a composite material are typically adulteration of foreign substances and damage resulting from an impact applied by a falling fixture or a tool. The forms of the defects generated are shown in Table 1. In order to evaluate the flaw tolerant safe life by applying various types of flaws to the blade, a systematic analysis of the manufacturing process should be carried out first, and the method to detect and evaluate the relevant defects should be also presented at the same time. Also, as presented in AC 20-107B[8], the effect of the strength degradation for each form of the defect can be evaluated through a fatigue test of specimens classified into stages such as coupon, element and component in accordance with the building block concept. Accordingly, in this study, TEFLOM films are inserted when laminating the composite material in order to simulate the defect caused by adulteration of foreign substances or an impact damage during the manufacturing process of the blade. The result of checking the defects actually applied to the blade specimens through X-ray test are shown in Figure 2.

<table>
<thead>
<tr>
<th>Flaw Types</th>
<th>Leading Edge</th>
<th>Thick Area</th>
<th>Thin Area</th>
</tr>
</thead>
<tbody>
<tr>
<td>Scratch Delamination Poor Bonding</td>
<td>Scratch Sharp Dent Void Foam Crack Delamination</td>
<td>Scratch Smooth Dent Sharp Dent Piercing Dent Debonding Delamination</td>
<td></td>
</tr>
</tbody>
</table>

2.1.2 Impact Damage

Impact damage applied to the blade can be generated by a falling tool or fixture during the manufacturing process, or can be generated by a collision with a FO (Foreign Object) around the runway during operation of the aircraft. Accordingly, in order to determine the form and size the defect generated by an impact damage and the part to be applied, the task of identifying the threat factors through analysis of the blade manufacturing process and aircraft operation should be carried out, and the level of the impact energy of each threat factor is required to be determined. The threats of impact damage which can
be generated during operational process of rotorcraft and the relevant impact energy levels are shown in Table 2.

<table>
<thead>
<tr>
<th>Impact Threats</th>
<th>Energy Level (Joule)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Damage induced during aircraft operation</td>
<td></td>
</tr>
<tr>
<td>runway stones</td>
<td>5~136</td>
</tr>
<tr>
<td>tire debris</td>
<td>&quot;</td>
</tr>
<tr>
<td>hailstone</td>
<td>1~5</td>
</tr>
<tr>
<td>Damage induced during maintenance</td>
<td></td>
</tr>
<tr>
<td>stowed baggage</td>
<td>34</td>
</tr>
<tr>
<td>dropped tools</td>
<td>9</td>
</tr>
<tr>
<td>dropped parts</td>
<td>14</td>
</tr>
<tr>
<td>aircraft refueling</td>
<td>8</td>
</tr>
<tr>
<td>nozzles</td>
<td>41</td>
</tr>
<tr>
<td>pneumatic starter-coupling</td>
<td>8.5</td>
</tr>
<tr>
<td>toot traffic impact</td>
<td>22</td>
</tr>
<tr>
<td>boot impact</td>
<td>51</td>
</tr>
<tr>
<td>edge and corner impact</td>
<td></td>
</tr>
<tr>
<td>terrain objects</td>
<td></td>
</tr>
</tbody>
</table>

The defects caused by an impact damage which can be generated during the manufacturing process of a blade made of a composite material and operation of the aircraft are classified into BVID and CVID depending on whether the defect can be identified through a visual inspection, and the boundary values of the impact energy required to generate BVID and CVID can be determined by analyzing the type and size of the damage caused by each level of impact energy through a pre-test before conducting the fatigue test of the actual blade.

- **BVID (Barely Visible Impact Damages):** Damages unidentifiable by a general visual inspection by the inspector
- **CVID (Clearly Visible Impact Damages):** Damages identifiable by a general visual inspection by the inspector

A spherical impactor is generally used to apply impact damage, and an artificial defect can be applied to a blade specimen by applying impact energy through adjustment of the weight or height of the impactor. The impact tester of Instron (Dynatup Model 9250HV) was used to apply impact damage as presented in Figure 3. The maximum applicable weight is 1.47 kg and the applicable impact energy is between 2.5 and 945 J. A pre-test was conducted using blade specimens cut in various sections in order to evaluate whether the defect generated by each energy level can be visually identified or not, and we could confirm that, the higher the impact energy, the more scratch, sharpdent, debonding, delamination and foam crack there are on the part to which the impact energy is applied. The result of inspecting the blade specimens after conducting the pre-test is shown in Figure 4. The result of visually inspecting the blade specimens showed that the impact damage of maximum up to 30 J can be applied to generate BVID, and the same level of impact energy was applied to the blade specimen to be used for fatigue life evaluation.

**Fig.3 Configuration of impact test equipment**
3 Blade Fatigue Test

3.1 Application of the Test Load

3.1.1 Life Test Method

Life test is a method to verify that the requirements for fatigue life are satisfied by confirming that no fatigue failure occurs to a structure made of a composite material even when a load spectrum bigger than one fold of the required life is applied to it. Also, it has an advantage of being able to simulate the load distribution and the behavior of the specimen most closely to the actual flight condition, and the fatigue test can be conducted with the main structural members and the surrounding structures assembled together. Accordingly, it is a suitable method for the case where diverse behaviors of the specimen are required to be checked, and is generally applied to fatigue tests of fixed wing aircraft.

3.1.2 S-N Test Method

S-N test is carried out with the purpose of acquiring the fatigue limit for each composite material used to manufacture the structure. A safe S-N curve can be produced through a statistical analysis of the fatigue limits acquired through fatigue tests, and then the fatigue life is determined by applying the load spectrum to the safe S-N curve to evaluate the cumulative damage. In particular, it has an advantage of being able to maximize utilization of fatigue test results as it can be applied even in the case the load spectrum is changed due to change in the flight limit or operation method of the aircraft. Accordingly, in the case the uncertainty of the dynamic load predicted through analysis is relatively high like rotorcraft, it is thought to be a test method which can be effectively applied. In this study, the shape of the test fixture was determined in a way a fatigue test can be conducted applying the S-N type fatigue test method.

The test load for a S-N type fatigue test can be applied being largely divided into two steps. The
first step test load is the test load which induces fatigue failure at $10^6$ cycle based on the safe S-N curve of the composite material used for the part which the structural margin is minimum among the sections of the blade, and its objective is to confirm that the test load is normally applied to the blade specimen and there is no problem in acquiring the test data. And the second step test load is the test load which induces fatigue failure at $10^6$ cycle based on the mean S-N curve of the composite material used for the part which the structural margin is minimum among the sections of the blade. At this time, it is important to equally maintain the ratio of the flap moment and the lead-lag moment applied to the first step and the second step.

3.2 Test Fixture

For the fatigue test of the attachment component, the centrifugal force was applied by using a hydraulic actuator, and the flap moment and lead-lag moment are applied using electric motor and eccentric rotating disk. Also, the application ratio of the flap moment and lead-lag moment were designed to be changed by adjusting the setting angle of the blade specimen attached on the test fixture. Figure 6(a) shows the method to apply the test load to the specimen of the attachment component and the shape of the test fixture.

For the fatigue test of the airfoil component, we employed the method of applying the test load using resonance of the blade specimen, for which we employed the method of vertically exciting one end of the blade specimen using electric motor and eccentric rotating disk. Also, a freely rotating support condition was applied to both ends of the blade in order to allow easily change of the flap direction by resonance. When resonance is generated in the blade specimen through vertical excitation, a situation occurs where the centrifugal force which should be constantly maintained in the spanwise direction of the blade becomes oscillating in accordance with deformation of the blade. As the change in the centrifugal force is insignificant in an actual flight condition, the dynamic change is required to be minimized, for which we connected an accumulator with the cylinder block of the hydraulic actuator in series to let it take the function of a hydraulic damper. Figure 6(b) shows the method to apply the test load to the specimen of airfoil component and the shape of the test fixture.

3.2 Measurement of the Test Load

A full-bridge circuit comprised of 4 strain gauges was used to measure the flap moment and lead-lag moment applied to the blade specimen. In the case of flap moment, the bridge circuit was built by attaching two gauges to the upper side and lower side of the blade respectively along the quarter chord line, and in the case of lead-lag moment, the bridge circuit was built by attaching two gauges to the leading edge and the trailing edge of the blade respectively. In particular, the gauge attached to the trailing edge of the blade was attached on the point where the principal axis of the relevant section intersects the upper side of the blade so that the output signal of the bridge circuit for measurement of the lead-lag moment can be minimized when the flap moment is applied. Figure 7 shows the composition of the bridge circuits for measurement of moments and the locations where the strain gauges are attached on.

In order to measure the moments using the bridge circuits, a load calibration process is required, which can be divided into 3 steps. The first step is the process of tuning the gage locations, and the flap gauges are attached in order to minimize the output signal of the bridge circuit for measurement of the flap moment when only the lead-lag moment is applied.
direction of the flap and lead-lag as shown in Figure 8. And, the third step is the process of calculating the calibration coefficient using the voltage signal measured from the bridge circuit, and the moment value can be calculated from the calibration coefficient $A_{ij}$ and the output of the bridge circuit as shown in Eq.(1).

$$\begin{bmatrix} M_{f_{bp}} \\ M_{l_{bg}} \end{bmatrix} = \begin{bmatrix} A_{11} & A_{12} \\ A_{21} & A_{22} \end{bmatrix}^{-1} \begin{bmatrix} \Delta R/R_{f_{bp}} \\ \Delta R/R_{l_{bg}} \end{bmatrix} \quad (1)$$

4 Fatigue Life Evaluation

4.1 Generation of S-N Curve

In order to evaluate the fatigue life of a blade made of a composite material using S-N test method, the mean and safe S-N curve of each composite material is required. A S-N curve can be divided into low cycle fatigue zone and high cycle fatigue zone in reference to $10^6$ cycles, and particularly, in the case of a rotor blade, the vibration load in the high cycle fatigue zone has an important influence on the fatigue life. Wöhler Equation was applied in order to generate the S-N curve in the high cycle fatigue zone as shown in Figure 9, where $\sigma_{ast}$ means the fatigue limit expressed in stress value at the time the test load cycles of $10^6$ is applied, and $\sigma_{ult}$ means the ultimate stress. Also, a S-N curve generated through a fatigue test represents a mean curve, and a safe curve can be generated using the deviation ($q$) and the safety factor ($k$) as presented in Eq.(2) to (4). In here, the safety factor ($k$) can be expressed as a function of the reliability ($k_r$) and the failure probability ($k_p$) obtained from the statistical analysis of the test data, and the deviation can be expressed as a function of the number of the blade specimens ($n$) and the fatigue limit obtained for each specimen.

$$b g (\sigma_{ast}) = b g (\sigma_{ast_{min}}) - k \cdot q \quad (2)$$

$$k = \left[ k_r + k_p \frac{1}{\sqrt{n}} \left( 1 - \frac{k_p^2}{A} + \frac{k_r^2}{A} \right) \right]^{-1} \cdot \frac{1}{\sqrt{n}} \quad (3)$$

$$q = \sqrt{\sum (bg \sigma_{ast} - bg \sigma_{ast_{min}})^2/(n - 1)} \quad (4)$$

Fatigue Life = Required Life / Total Damage \quad (5)
HIGH CYCLE FATIGUE LIFE EVALUATION OF DAMAGED COMPOSITE ROTOR BLADES

Fig. 9 Mean and safe S-N curves

- \( A = 2(n - 1) \)
- \( k_r \): constant corresponding to risk of \( 10^{-6} \)
- \( k_p \): constant corresponding to confidence of 90%
- \( n \): number of test specimen

4.2 Cumulative Damage Evaluation

The cumulative damage can be calculated by applying a load spectrum equivalent to one fold of the blade life to the S-N curve generated using Eq.(2) to (4), and the fatigue life of a blade made of a composite material can be finally evaluated using Eq.(5). At this time, as the S-N curve is the result obtained through an evaluation conducted considering only the dynamic load, and it is impossible to directly apply the load spectrum which is divided into static load and dynamic load to the S-N curve. In order to apply the load spectrum which is divided into static load \( (\sigma_{\text{st}}) \) and dynamic load \( (\sigma_{\text{dyn}}) \) directly to the S-N curve, the process of converting it into equivalent load as shown in Eq.(6) and (7) is required.

\[
\sigma_{\text{eq}} = \sigma_{\text{dyn}} \quad (R > 0.9) \quad (6)
\]
\[
\sigma_{\text{eq}} = K \cdot \sigma_{\text{dyn}} \quad (R \leq 0.9) \quad (7)
\]

\[K = (1 + 1.5 \sigma_{\text{safe}} / \sigma_{\text{rm}}) (1/R - 1/R_0)\]

- \( R \): stress ratio, \( \sigma_{\text{dyn}} / \sigma_{\text{safe}} \)
- \( R_0 = 0.9 \)

The load spectrum applied for evaluation of fatigue life of a blade and the result of calculating the equivalent load using Eq.(6) and (7) are shown in Table 3. Also, the S-N curve of the glass fiber (UD glass roving spar, Hexcel TVR-380) used to manufacture the spar of the blade, and the results of evaluating the cumulative damage using the load spectrum are shown in Figure 10.

<table>
<thead>
<tr>
<th>Load No.</th>
<th>Static Load (MPa)</th>
<th>Dynamic Load (MPa)</th>
<th>Equivalent Load (MPa)</th>
<th>Realized Cycles</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>108.8</td>
<td>19.8</td>
<td>50.1</td>
<td>75919</td>
</tr>
<tr>
<td>2</td>
<td>107.0</td>
<td>18.2</td>
<td>45.9</td>
<td>683271</td>
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<td>3</td>
<td>67.6</td>
<td>15.9</td>
<td>37.9</td>
<td>97920</td>
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<tr>
<td>4</td>
<td>66.5</td>
<td>19.3</td>
<td>46.0</td>
<td>456960</td>
</tr>
<tr>
<td>5</td>
<td>66.2</td>
<td>23.7</td>
<td>56.4</td>
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<td>13.8</td>
<td>32.8</td>
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<tr>
<td>82</td>
<td>61.4</td>
<td>3.1</td>
<td>7.5</td>
<td>1128332</td>
</tr>
</tbody>
</table>

Fig. 10 Cumulative damage calculation results, glass roving spar

5 Conclusions

The concept of flaw tolerant safe life was applied to evaluate the fatigue life of a helicopter rotor blade
made of composite materials, and the test load was applied in accordance with S-N test method. TEFLON films were inserted when laminating the composite materials used to make the blade specimens which can be used for the fatigue test in order to simulate the defects possibly generated during the manufacturing process. Also, impact tests were conducted to confirm the possible limit in visually identifying the damages caused by an impact of a foreign object during operation of the aircraft, and the boundary value of the maximum impact energy for generation of BVID was determined to be 30 J. Fatigue tests were performed using blade specimens which both manufacturing defects and impact damages are applied. The cumulative damage was calculated using the S-N curve and the load spectrum generated through analysis of the test data, and the fatigue life of the blade made of composite materials could be evaluated.

Acknowledgements
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References
[1] FAR Part-27, Airworthiness standards: Normal category