Numerical and Experimental Analysis of Aircraft Wing Subjected to Fatigue Loading

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ABSTACT

This study deals with the aircraft wing analysis (numerical and experimental) which subjected to fatigue loading in order to analyze the aircraft wing numerically by using ANSYS 15.0 software and experimentally by using loading programs which effect on fatigue test specimens at laboratory to estimate life of used metal (aluminum alloy 7075-T651) the wing metal and compare between numerical and experimental work, as well as to formulate an experimental mathematical model which may find safe estimate for metals and most common alloys that are used to build aircraft wing at certain conditions. In experimental work, a (34) specimen of (aluminum alloy 7075-T651) were tested using alternating bending fatigue machine rig. The test results are ; (18) Specimen to establish the (S-N) curve and endurance limit and the other specimens used for variable amplitude tests were represented by loading programs which represents actual flight conditions. Also it has been obtained the safe fatigue curves which are described by mathematical formulas. ANSYS results show convergence with experimental results about cumulative fatigue damage (D), a mathematical model is proposed to estimate the life; this model gives good results in case of actual loading programs. Also, Miner and Marsh rules are applied to the specimens and compared with the proposal mathematical model in order to estimate the life of the wing material under actual flight loading conditions, comparing results show that it is possible to depend on present mathematical model than Miner and Marsh theories because the proposal mathematical model shows safe and good results compared with experimental work results.

Key words: aircraft wing, fatigue loading, life estimation, simulation.

التحليل التجريبي والعددي لجناح طائرة يخضع لحمل الكلال.

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الخلاصة

دراسة هذا البحث تتعلق بالتحليل العددي والتجريبي لجناح طائرة والذي خضع لحمل الكلال ، حيث تم التحليل عددياً بواسطة استخدام برنامج (ANSYS15.0) وتجريبياً عن طريق تسليط برامج تحميل إجهادية على عينات الكلال في المختبر لتقدير عُمر معدن العينة حيث استُخدمت سبيكة الألمنيوم (7075-T651) والتي هي نفس السبيكة المستخدمة في جناح الطائرة في الصناعات الحديثة لهياكل الطائرات والاجنحة ، بعدها تمت المقارنة بين العمل العددي والتجريبي للعينات التي تم الإختبار عليها. كذلك تم اقتراح نموذج رياضي يستطيع إيجاد تقدير آمِن للمعدن المستخدم في بناء هيكل الجناح وغيره من المعادن الاخرى المستخدمة لنفس الغرض. في الجانب العملي تم اختبار (34) عينة لسبيكة الألمنيوم (7075-T651) بإستخدام جهاز اختبار الكلال ذو إجهاد الإنحناء الترددي ، كانت نتائج الإختبار كالتالي ; (18) عينة للحصول على منحنى (الإجهاد - عدد الدورات) وكذلك للحصول على حد الكلال للسبيكة المستخدمة . أما العينات المتبقية فقد أختبرت تحت تأثير إجهادات ببرامج

تحميل تمثل حالة الطيران الحقيقية من إقلاع طيران مستقر ثم هبوط ، كذلك تم الحصول على منحنيات الكلل الأمين التي وصِفت رياضياً بمعادلات . نتائج برنامج ANSYS أظهرت تقارب من نتائج الإختبار العملي فيما يخص الضرر التراكمي للكلال ، النموذج الرياضي المُقترح اعطى نتائج جيدة في حالة برامج التحميل الحقيقية المسلطة ، أستُخدمت نظرية ماينر ونظرية مارش للضرر المتراكم على عينات الكلال للسبيكة المُستخدمة وتم مقارنة النتائج مع النموذج الرياضي المقترح وكانت كالآتي ; حيث يُمكن الإعتماد على النموذج الرياضي المقترح للتقدير اكثر من نظرية ماينر ومارش لإن النموذج المقترح أظهر نتائج آمنة مقارنة مع الإختبار العملي ، أما النظريتين فأظهرتا نتائج آمنة نوعاً ما لكنها أكبر قيمة من الإختبار العملي.

الكلمات الرئيسية: جناح الطائرة، حِمل الكِلال، تقدير العُمر.

1. INTRODUCTION

Aircraft wings subjected to fatigue loading are considered as one of most complex cases when aircraft are exposed to large changes due to loads, moments, air circumstances and many other changes. The aircraft wing is possible to have a normal flight or in the case of difficult conditions. All previous conditions play an important role in determining life (age) of the aircraft parts. Therefore, researchers are trying to get maximum possible and accurate estimation for life of these parts. Most complex problems that were facing research centers is fatigue failure which define it as main fracture mechanism in most structural parts where attributed (80-90 %) of failure in machines and structures due to cracks in metal, **Callister William, 2003.**The reason of fatigue failure in aircraft wing due to subjecting the structural parts of wing to cyclic stresses often lower yield strength of wing material.

Fatigue phenomenon can be defined as crack initiation due to cyclic stresses then crack propagation until it reaches failure represented by quick final fracture, **Forrest,** and **George, 2013**. So, fatigue or failure phenomenon is one of the most important considerations involved in the design, operation and maintenance in many important structural parts like parts of aircraft turbine engines for example, operationally preferably detected fatigue damage in early time, this step is very important, if it is to be wanted to avoid malfunctions and sudden collapse so the detection of fatigue failure is considered important goal of disassembly operations, checking and maintenance periodically.

More than 100 year ago, it was started to use laboratory specimens and replace the full scale tests, for that reason similarity condition is found and need samples which represent similarity models. Models were early developed such as relation between stress range-fatigue life(Goodman scheme), fatigue reducing factor and Miners rule, these relations still give satisfying fatigue estimations for many conditions, with this fatigue failure still repeated, so demand increased for economical and strength designs as well as work to find modern and orderly fatigue research. Besides that, in the case of aircraft work outside of limitations through subjecting to variable flight conditions, all of these factors are used to determine the final life of the metal. Therefore, obviously first interest is to study (S-N) curve as well as designer must come to conclusion which is gain to this(S-N) curve at safe form in order to make design become away from danger because any error in design leads to waste human life and costs huge sums.

2. RELATED PAST RESEARCHES

Al-Alkawi, 1991, studied fatigue phenomenon in aluminum alloy (2024-T35) under the effect of constant and variable amplitude stresses which results from applying bending stresses (Low-High) and (High-Low) until failure occurs, also proposed mathematical model to describe short and long fatigue cracks propagation for the used alloy as Eq. (1):

$$\frac{da}{dN} = 4.6 \times 10^{-34} б^{12} (W - a_{av.})^{-2.4} \tag{1}$$

And the model showed good results in life estimation compared with experimental results.

Emad Natiq, 1994 studied the aerodynamic stresses applied to the metal of aircraft wing (2024-T4) in order to predict the life of this metal. This investigation includes practical work in addition to theoretical work. This research proposed mathematical model which predicted fatigue life of metal in hours as Eq. (2):

$$W = \left(\frac{n_i}{Nf}\right)^{-\propto} . \frac{(б_{max.} - б_i)}{б_i} . n_p . K_t \tag{2}$$

Where:
K_t: Stress concentration factor ($K_t = 1.57$) according to specimen dimensions.
Also, Miner rule is applied to the specimens and compared with the proposed model in order to predict the life of the metal wing under actual flight loading conditions and it showed good results.

Muhammad, 2003 studied and calculated applied forces and moments on aircraft structure under different flight cases. From experimental work life equation under constant cycles was found, as in Eq. (3):

$$б_f = 704 \left(N_f\right)^{-0.1465} \tag{3}$$

Also, it built mathematical model to predict fatigue life for specimens under variable loading, as Eq. (4):

$$\mathrm{N_E} = \frac{2(б_\mathrm{H} - б_\mathrm{L})\,\mathrm{W}}{3.596 \times 10^{-21} \left(б_\mathrm{H}^{7.868} - б_\mathrm{L}^{7.868}\right) \mathrm{K_t}} \tag{4}$$

This model gives safe fatigue predictions when compared with other predictions theories.

Cowell, 2006 demonstrated that a fatigue life methodology can be successfully utilized to predict the service life of aircraft components in a practical manner and to determine if reworked parts are suitable for continued service. This methodology was first demonstrated by investigating the predicted fatigue life of a flat plate with a centrally located hole under constant amplitude and variable amplitude loading. This approach was validated by comparing simulated life predictions using several stress-life and strain-life algorithms with previously published experimental data.

Gope, et al., 2007 reached to mathematical model that gave good estimation to parts life that subjected to fatigue depending on mechanical properties for material, where this model has been applied to aluminum, copper, steel and titanium alloys and it gave safe and acceptable results from where design and parts life, the summary of this model is in this formula, as Eq. (5):

$$Nf = \int_{ao}^{af} \frac{da}{\frac{0.139}{\alpha\, \mu\, E\, бy} (\mu\, \Delta \boldsymbol{K}I - \mu\, \Delta \boldsymbol{K}th)^2 \; \frac{1}{1 - \left(\frac{\boldsymbol{K}max}{\mu\, \Delta \boldsymbol{KIC}}\right)^2}} \tag{5}$$

Al-Alkawi, et al., 2010 designated the fatigue behavior of an aluminum alloy 2024 – T3 under considered constant and variable amplitude of stresses. The applied load adopted is a rotating bending one, the study consists of two parts; experimental and theoretical. The experimental part includes carrying out laboratory tests on two groups of specimens(diameter of 6.74mm) the first group was tested under constant stress amplitude to establish the S-N curve of the specimen's material, while the second group was tested under variable amplitude of stress to assess the effects of the accumulated fatigue damage. Theoretical part includes study of Miners rule for cumulative damage and elastic crack growth theory, also a proposed mathematical model is given in Eq. (6):

$$N_f = \frac{2\times(б_H - б_L)\times D}{9.625\times10^{-33}\left(б_H^{12.363} - б_L^{12.363}\right)\times K_t} \quad (6)$$

This model showed good and safe results as well as two theories with respect to experimental work.

3. BUILDING PROPOSED MATHEMATICAL MODEL

To build mathematical model which predicts metal service life, the following steps must be followed:

First: Computing the sum of cumulative damage of metals by calculating the cumulative damage during a one program, by knowing number of iterations for this loading program, range of applied stress for each specimen of loading program and proposed mathematical model for constant stresses which was calculated previously, the mathematical model which predicts fatigue life can be derived as Eq. (7):

$$\sum\left(\frac{n}{N_f}\right)_i = \frac{n_1}{N_{f1}} + \frac{n_2}{N_{f2}} + \cdots \quad (7)$$

Where the value of (n) does not change, so the above equation can be written in the following form:

$$\sum\left(\frac{1}{N_f}\right)_i$$

And more comprehensive form, the above equation can be written as follows:

$$\sum\left(\frac{n}{N_f}\right)_i = \int_{б_L}^{б_H} \frac{1}{N_{fi}}\, dx_i \quad (8)$$

By taking the slope of (S-N) curve as shown in **Fig. 1**, this equation can be concluded as follows:

$$X_i = \frac{N_{fm}(б_i - б_L)}{(б_H - б_L)} \quad (9)$$

By deriving the Eq. (9) and restitution for (dx_i) in Eq. (8) it can be obtained:

$$\sum\left(\frac{n}{N_f}\right)_i = \int_{б_L}^{б_H} \left(\frac{1}{N_{fi}}\right) \frac{N_{fm}\, dб_i}{(б_H - б_L)} \quad (10)$$

Where (α) represents rate of increase of stress and it is controller factor on test variable as shown in **Fig. 1**, it can be obtained from following equations:

$$\alpha = \frac{б_H - б_L}{N_{fm}} \quad (11)$$

So the Eq. (10) can be shortened as follow:

$$\sum \left(\frac{n}{N_f}\right)_i = \frac{1}{\alpha}\int_{б_L}^{б_H} \left(\frac{1}{N_{fi}}\right) dб_i \tag{12}$$

Second: When there is an equation of (S-N) curve for material which is subjected to fatigue tests, it become possible to make integration on Eq. (12) and reach to calculate required cumulative damage for material and for one program from the curve of variable stress (S) with number of cycles to failure (N):

$$б_f = 807.75\,(N_f)^{-0.111} \tag{13}$$

When Eq. (13) is substituted in Eq. (12) and substitute (α) in Eq. (8) can get:

$$\sum \left(\frac{n}{N_f}\right)_i = \frac{N_{fm}}{(б_H - б_L)} \int_{б_L}^{б_H} \frac{dб_i}{\left(\frac{б_f}{807.75}\right)^{-\frac{1}{0.111}}} \tag{14}$$

$$\sum \left(\frac{n}{N_f}\right)_i = \frac{N_{fm}}{(б_H - б_L)\left(\frac{1}{807.75}\right)^{-\frac{1}{0.111}}} \int_{б_L}^{б_H} \frac{dб_i}{(б_f)^{-\frac{1}{0.111}}} \tag{15}$$

According to linear damage rule theory as Eq. (16):

$$(D_i)_{P.M} = \sum_{i=1}^{m} \frac{n_i}{N_i} = \frac{n_1}{N_1} + \frac{n_2}{N_2} + \frac{n_3}{N_3} + \cdots = 1 \tag{16}$$

By substituting Eq. (16) in Eq. (15):

$$D = \frac{N_{fm}}{(б_H - б_L)(0.00124)^{-9}} \int_{б_L}^{б_H} \frac{dб_i}{(б_f)^{-9}}$$

And by making integrals:

$$D = \frac{(7 \times 10^{-27.5})\,N_{fm}}{(б_H - б_L)} \left(б_H^{10} - б_L^{10}\right)$$

$$D\,(б_H - б_L) = (7 \times 10^{-27.5})\,N_{fm}\left(б_H^{10} - б_L^{10}\right)$$

$$N_{fm} = \frac{2 \times (б_H - б_L) \times D}{(7 \times 10^{-27.5})\left(б_H^{10} - б_L^{10}\right)} \tag{17}$$

Eq. (17) is multiplied by (2) to include phases of raise and descent for one program or vice versa.

4. CUMULATIVE FATIGUE DAMAGE

Fatigue damage increases with applied load cycles in a cumulative manner. Cumulative fatigue damage analysis plays a key role in life prediction of components and structures subjected to field load histories.

5. CUMULATIVE FATIGUE DAMAGE THEORIES

There are many theories and assumptions that focused in explanation of fatigue phenomenon which occur in structural parts subjected to variable stresses amplitude, one of these theories:

5.1 The linear damage theory

Miners theory states; result damage (D_i) at any stress level ($б_i$) is proportional linearly with the ratio of applied cycles (n_i) at that level to total number of cycles causing failure (N_i) when same stress is applied, **Miner, 1945**. The failure occurs as soon as damage summation equals to unity ($D_i = 1$), as Eq. (18).

$$(D_i)_{P.M} = \sum_{i=1}^{m} \frac{n_i}{N_i} = \frac{n_1}{N_1} + \frac{n_2}{N_2} + \frac{n_3}{N_3} + \cdots = 1 \tag{18}$$

5.2 Marsh Theory

Simple cumulative damage theories in the forefront (Miners theory) did not take into consideration the stresses under endurance limit, as Marsh theory, **Marsh, 1965.** Assure that these stresses have active role if it is added to the stresses which are greater than endurance limit, number of cycles to failure according to Marsh theory $(N_f)_{marsh}$,can be calculate in eq. (19):

$$(N_f)_{marsh} = \frac{(б_H - б_L)}{(б_H - б_{E.L})} \times \mathrm{N_f} \tag{19}$$

6. CUMULATIVE DAMAGE (D) CALCULATION

To obtain good prediction for parts life, it must propose a mathematical model which calculate cumulative damage (D), this mathematical model mainly depends on number of applied cycles (n_i) at certain stress level, final number of cycles to failure (N_f)and also number of loading programs (n_p) . So, the mathematical model is as in the following form in Eq. (20):

$$D = \sum \frac{n_i}{N_{fi}} \,.\, n_p \tag{20}$$

7. NUMERICAL ANALYSIS

A finite element modeling for aircraft wing is presented to evaluate the cumulative fatigue damage of a wing structure; the finite element method must be used to solve the problem due to the complexity of the wing structure and boundary conditions.

7.1 Elements Used

For the case study analysis in this research, the following elements were used: shell 181 and beam 188**, ANSYS15.0 Help, Mechanical APDL, Element Reference, shell 181 and beam 188.**

7.2 Meshing Control

The next stage in creating a finite element model is dividing the geometry into nodes and elements. This process is called meshing. The ANSYS program can automatically generate the nodes and elements after specifying the element attributes and sizes, where:

1. The element attributes include element type (s), real constants, and material properties.
2. The element size controls the fines of the mesh. The smaller the element size, the finer is the mesh.

Mapped mesh type was used in the present analysis to get best accurate mesh, and the obtained meshes of the aircraft wing model geometry are shown in **Fig. 2**.

7.3 Applying Boundary Conditions

The next step of the analysis involves applying the appropriate boundary conditions, the wing is connected with fuselage through the supports therefore the transient displacement and rotation about X, Y and Z axis for the zone of supports which attached with fuselage have been given zero (U_X, U_Y, U_Z, ROT_X, ROT_Y and ROT_Z are zero). The element type and boundary condition of aircraft wing are shown in the **Fig. 3**, where the zoom of the element type is shown.

7.4 Applying loads and Solution

In this stage, the analysis type must be defined, such as static analysis, modal analysis, harmonic response, transient dynamic, buckling …. etc, then the loads are applied and if needed load increments are specified. For the present study the load will be used for the transient dynamic analysis of the wing when it is subjected to aerodynamic forces so the loading programs (which is clarified in next section) are applied at the lower surface of the aircraft wing (position of aerodynamic center CG), **Fig. 4** (a).

7.5 Wing Specification

The specifications of aerobatic aircraft wing used in this work are shown in Table 1. and **Fig. 4** (b).

8. EXPERIMENTAL WORK

In present work it has been used aluminum alloy (7075-T651) specimen in experimental test at laboratory. This alloy is one of the important aluminum alloys because it is used in wide ranges of aircraft manufacturing, gears, shafts, aerospace applications and other highly stressed structural applications where very high strength and good resistance to fatigue loading. The chemical composition and mechanical properties of the base material are presented in **Table 2** and **Table 3** respectively.

8.1 Fatigue Apparatus and Specimens

The device which was used in fatigue tests is (Alternating Bending Fatigue Machine HSM20), where the fatigue specimens subjected to stress which was bending moment; the fatigue device is shown in **Fig. 5**. All specimens are fixed on fatigue device from end and free from other (Fixed-Free) exactly like aircraft wing, so the loading apply at free edge of specimen then bending moment will be generated, therefore, the surface of specimen will be subjected to tensile and compression stresses as shown in **Fig. 6** and **Fig. 7**. The test on this fatigue device was at (1500 cycle/min.), to know the number of cycles that specimen fail, the device equipped with digital counter (automatic stop when specimen fails).

8.2 Experimental Test Programs

Fatigue test occurred by applying bending stresses on a group of standard (24) specimens, divided to (8) groups i.e. (3 specimens for each group), every group has loading program different from other group to make the fatigue test include main aviation cases like take-off, stable flight and landing (L-H-L), the loading programs are shown in **Fig. 8**, **Fig. 9**, **Fig. 10**, **Fig. 11**, **Fig. 12**, **Fig. 13**, **Fig. 14**, **Fig. 15**.

8.3 Achieved Experimental Tests at Laboratory

All specimens are subjected to alternating bending stresses, at first the specimen is fixed from one end and free from other end, then the required stress level is fixed and let fatigue device starts on until specimen fails. Three specimens tested at every stress level, after that stress level must change for other three specimens and so on to get (stress(S)-number of cycles to failure (N_i)) curve. Other specimens have been tested also at variable loading(S) at constant cycles (ni).

Also, other group of specimens are subjected to variable loading(S) with variable cycles (ni) because the applied loads on aircraft wing are variable loads continuously, the behavior of loading was (low-high-low) and the applied loads were aerodynamic stresses taken from actual condition on aircraft wing. **Fig. 16** represents specimens tests theoretically using ANSYS, all of them subjected to the same loading programs in experimental work and it shows converging between theoretical and experimental cumulative damage results.

9. RESULTS AND DISCUSSION

This part introduces results that have been obtained from experimental tests for specimens that are subjected to alternating bending stresses on fatigue machine at varying values according to loading programs (low-high), (high-low), as well as a basic loading program in this research (low-high-low) which represents actual flight condition like; take-off , stable flight and landing. There are two types of tests applied on fatigue specimens; first one is constant amplitude tests through which (S-N) curve has been drawn and endurance limit for Aluminum alloy (7075-T651) is obtained. Second was variable amplitude tests represented by previous loading programs, where these tests applied with constant cycles at every stress level reached(10000 cycle) for take-off and landing, and for stable flight condition, tests have been applied with constant cycles at stress level for one program reached (100000 cycle). As well as, this part includes results that have been obtained from software program (ANSYS15.0 APDL) which show numerical analysis for the aircraft wing represented by cumulative damage (D). The point which joined between fatigue test results and wing model as well as ANSYS program clearly appears in the simulations of test samples subjected to same loading programs at ANSYS. Results between them seem converging in the summary of cumulative damage results.

9.1 Discussion of Fatigue Tests Results

9.1.1 Constant Amplitude Fatigue Tests Results

Fatigue tests are applied on six groups of specimens each group contains three specimens by applying constant value of stress level to each group until failure was occurred, failure cycles for each specimen were calculated to obtain best results and accuracy in (S-N) curve. **Table 4** shows results from this fatigue test, the curve equation between applied stress and number of cycles to failure for (7075-T651) is obtained as follows in Eq. (21):

$$б_f = 807.75 \left(N_f\right)^{-0.111} \tag{21}$$

After that, endurance limit has been found which is estimated about (135 Mpa) at failure cycles reach to (10^7) for aluminum alloy (7075-T651) from (S-N) curve as shown in **Fig. 17**.

9.1.2 Variable Amplitude Fatigue Tests Results

According to loading programs in experimental work, standard fatigue specimens (7075-T651) were tested by applying variable alternating bending stresses with variable applied cycles(n_i) in case of take-off, stable flight and landing, where it was calculated number of cycles to failure experimentally (fatigue life) at every loading program, compared with two theories of cumulative damage (Miners cumulative linear damage rule and Marsh rule). As well as it was derived mathematical model and compared with experimental results as in **Table 5**.

9.2 Safe Fatigue Curves

It has been reached to the equations of safe fatigue curves with different probabilities rates from (55%-90%) and it gave good and economic predicting results compared with experimental work as follow:

(Probability 55%; $б_f = 805.17(N_f)^{-0.111}$), (Probability 70%; $б_f = 804.45(N_f)^{-0.111}$), (Probability 80%; $б_f = 804.5(N_f)^{-0.111}$), (Probability 90%; $б_f = 804.55(N_f)^{-0.111}$).

Table 6 shows lower and upper limits for the number of cycles to failure with different probabilities.

9.3 (ANSYS APDL) RESULTS

The numerical analysis part is represented by ANSYS APDL, analysis results have been obtained which pertaining with aircraft wing that is subjected to fatigue loading. This study has used transient dynamic analysis (applied load with respect to time) in order to obtain fatigue analysis and reach to cumulative damage of the wing structure through the application of variable aerodynamic loads, where each load event has two load numbers; first load number of each event represents the condition of raise, its value is positive, while second load number of the same event represents the condition of descent, its value equal to first load number but in negative sign.

9.3.1 Cumulative Fatigue Damage Usage by (ANSYS APDL)

The cumulative fatigue damage usage value is sum of the partial usage factors (Miners rule), where the aircraft wing which was designed by ANSYS are subjected to different loadings according to loading programs in experimental work part. Cumulative distributed aerodynamic pressures will be represented by forces acting on one node of meshed wing; this node represents position of aerodynamic center (C.G). According to **Fig. 18**, node (6164) represents the position of aerodynamic center of wing its coordinates (X= 0.85555, Y=0, Z= -1.3333) and there are other nodes like wing root (node 227),wing middle (node 2387), wing tip (node 4192). Its cumulative damage calculations are represented in next part of results.

9.3.2 Aircraft Wing Cumulative Fatigue Damage Results

According to loading program in **Fig. 10** for specimen (B2) the cumulative damage for aircraft wing equals to (0.17978) for one program and for all chosen nodes. As shown in **Fig. 19 (a, b, c)**.

In the same manner it can be found the cumulative damage for one program for remaining other loading programs, so according to different previous loading programs that effect on wing during flight the total cumulative damage usage (D_{Total}) can be calculated depending on this eq. (22) as follow:

$$\boxed{\left(\mathbf{D}_{\mathbf{Ansys}}\right)_{\mathbf{Total\ programs}} = \left(\mathbf{D}_{\mathbf{Ansys}}\right)_{\mathbf{One\ Program}} \times \left(\mathbf{No.\ of\ Programs\ to\ Failure}\right)_{\mathbf{Experimental}}} \tag{22}$$

Table 7 shows the point which joined between fatigue test results and wing model as well as ANSYS program. Also **Fig. 20 (a, b)** shows fatigue analysis and the result which are represented by cumulative damage for specimen (B8) as example in ANSYS program, other specimens are in the same manner.

10. CONCLUSIONS

There are two types of tests applied on fatigue specimens; first one is constant amplitude test through which has been drawn (S-N) curve and obtains endurance limit while second is variable amplitude tests represented by above loading programs. From present work it has been reached to the equations of safe fatigue curves with different probabilities rates from (55%-90%) and it gave good and economic predicting results compared with experimental work. According to loading programs, it has been obtained results of cumulative damage with variable cycles (Low-High-Low) which represent actual flight condition. Results that have been obtained from software program (ANSYS15.0 APDL) show numerical analysis for the aircraft wing represented by cumulative damage (D) are which compatible with experimental cumulative damage results. Also, predicting theories used in this research were Miners damage rule and Marsh rule which show good results but not economic compared with experimental and mathematical model predicting results. So in the case of cumulative damage for variable cycles according to loading programs it is preferably to use mathematical model to predict number of cycles to failure because of its proximity from experimental results.

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NOMENCLATURE

D_i : cumulative damage.
$\mathrm{D_{Ansys}}$: cumulative damage by ANSYS software.
$D_{exp.}$: experimental cumulative damage.
D_{Miner} : Miner cumulative damage.
$(N_f)_i$: total number of cycles causes failure at stress level.
n_i : applied cycles at stress level.
$(N_f)_{Miner}$: number of cycles causes failure according to Miner theory.
$(N_f)_{marsh}$: number of cycles causes failure according to Marsh theory.
$N_{f\,(exp.)}$: number of cycles causes failure according to experimental work.
N_{fm} : number of cycles to failure according to mathematical model.
$\mathrm{ROT_X}$, $\mathrm{ROT_Y}$, $\mathrm{ROT_Z}$: rotation at directions (X,Y,Z) respectively.
$\mathrm{U_X}$, $\mathrm{U_Y}$, $\mathrm{U_Z}$: displacement at directions (X,Y,Z) respectively.
X_i : slope of curve.
б$_i$: intermediate stress.
б$_H$: maximum applied stress.
б$_L$: minimum applied stress.
б$_{E.L}$: endurance limit stress.
α : rate of increase of stress.

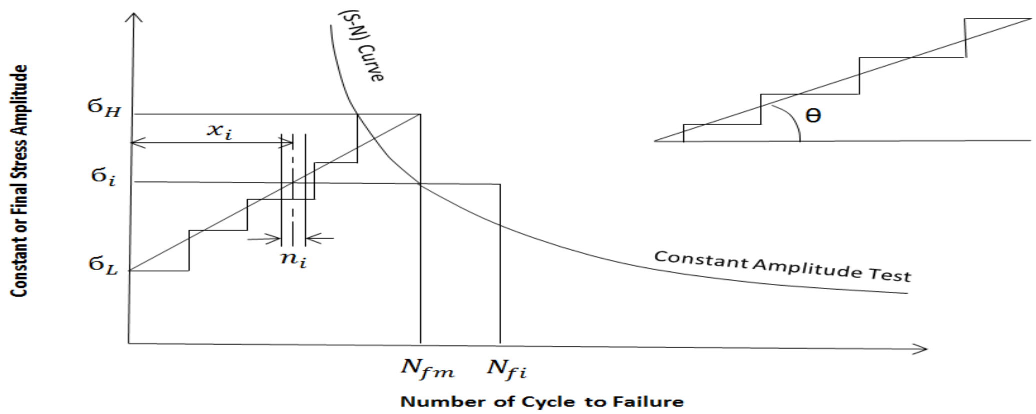

Figure 1. Diagram of stress amplitude (increasing and decreasing).

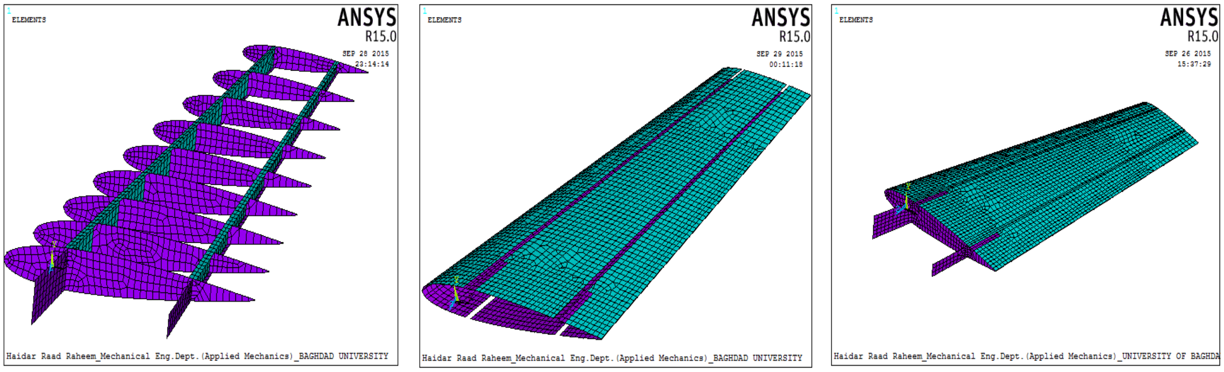

Figure 2. Aerobatic airplane wing details with meshing.

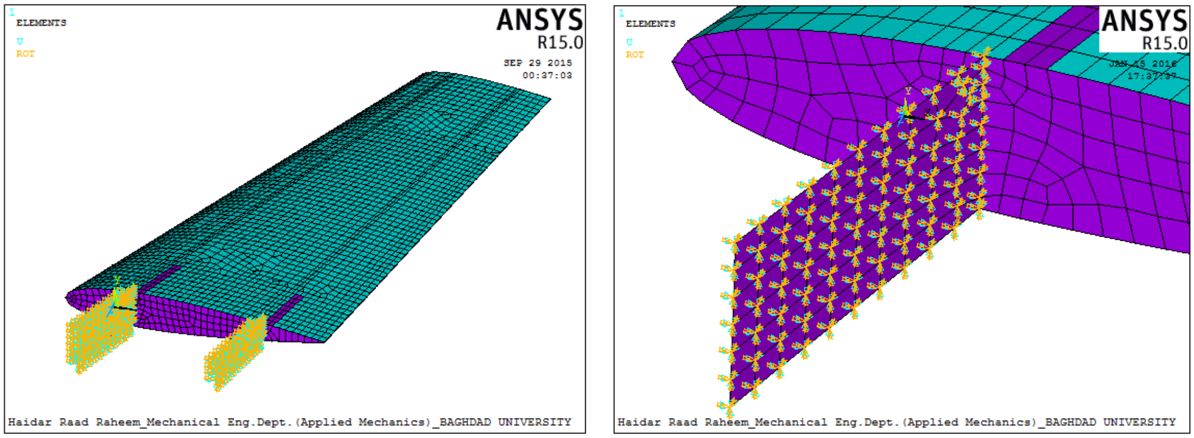

Figure 3. Element type with boundary conditions and zoom of element type.

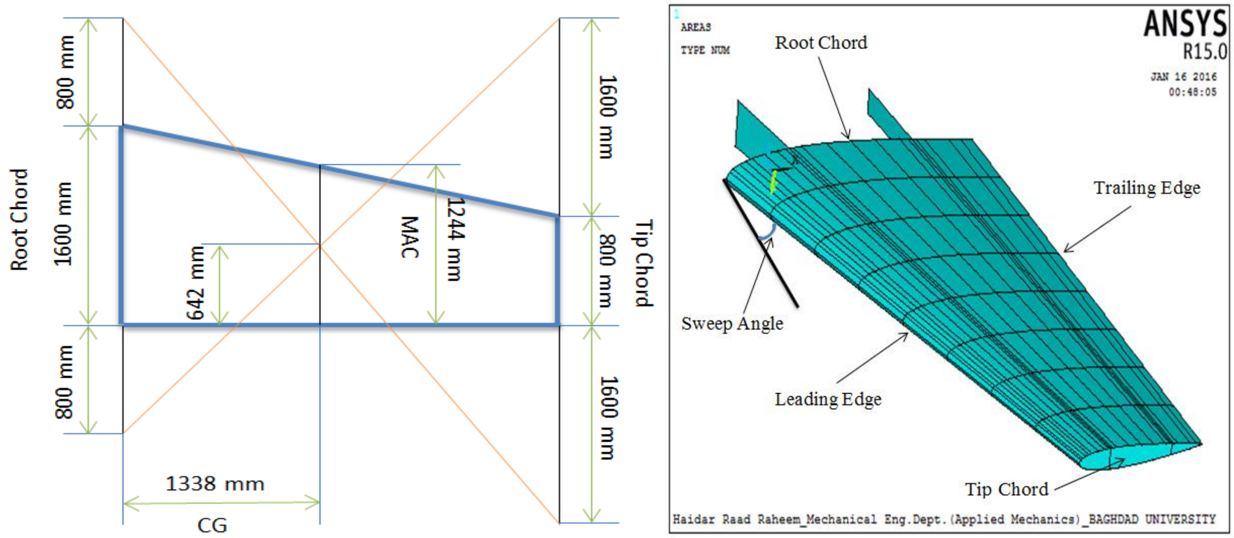

Figure 4. (a) Graphic way to find position of aerodynamic center (CG), (b) Wing specification.

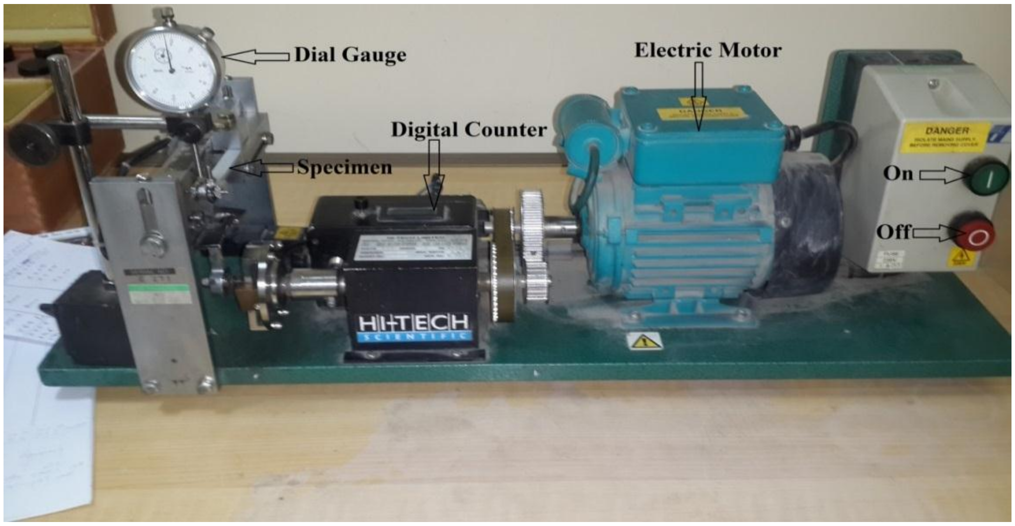

Figure 5. Alternating bending fatigue machine.

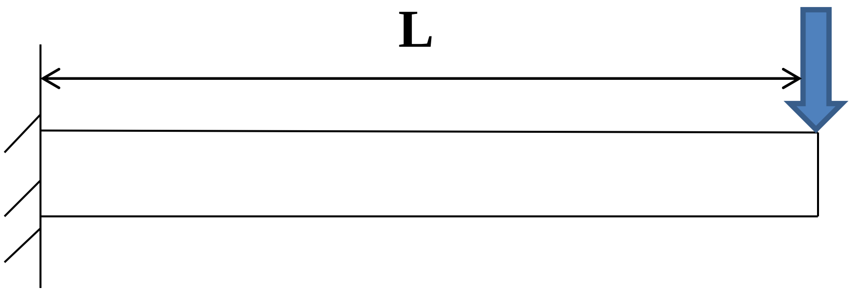

Figure 6. Fatigue test specimen subjected to alternating force.

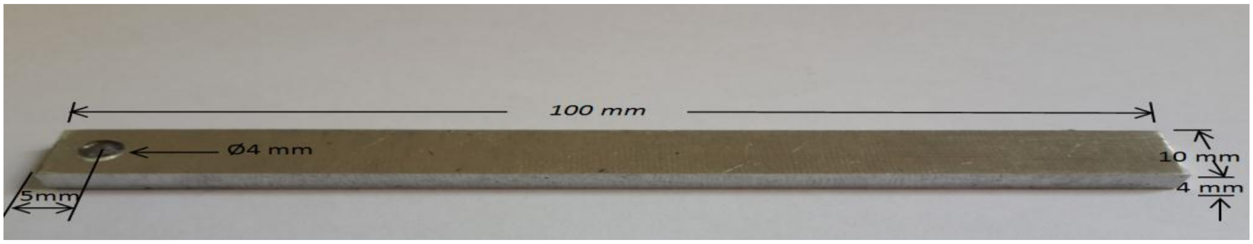

Figure 7. Aluminum alloy (7075-T651) fatigue test specimen.

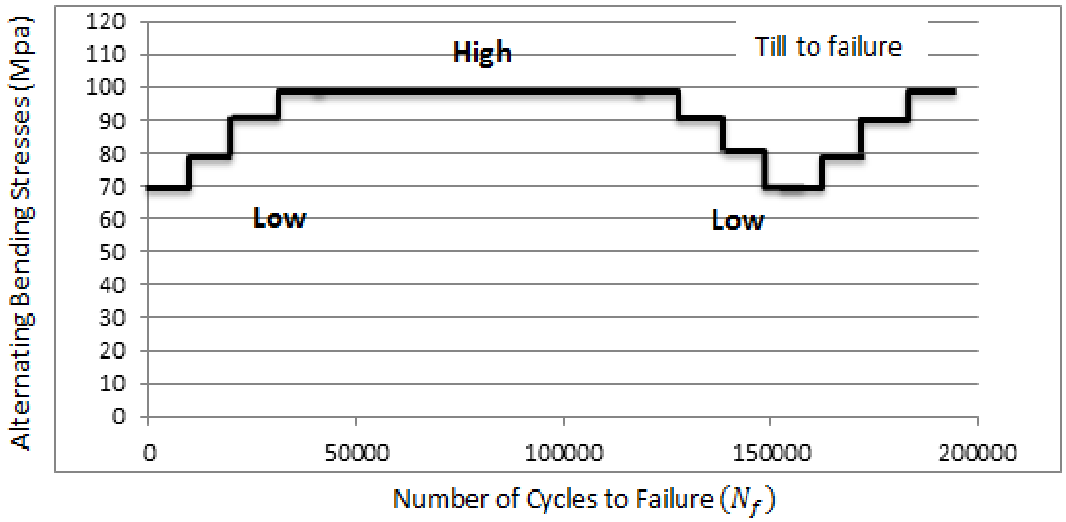

Figure 8. Loading program for specimen (B1)

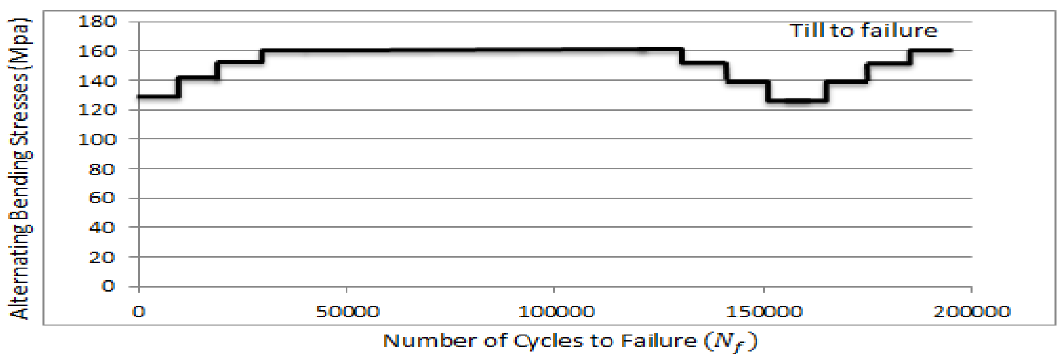

Figure 9. Loading program for specimen (B2)

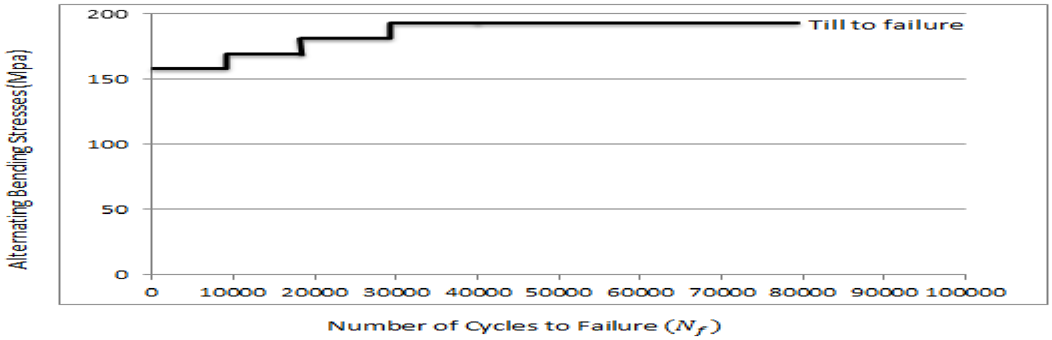

Figure 10. Loading program for specimen (B3).

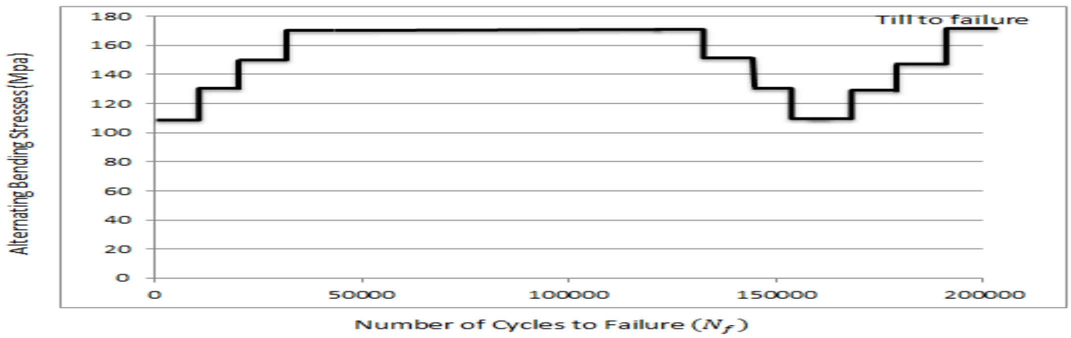

Figure 11. Loading program for specimen (B4)

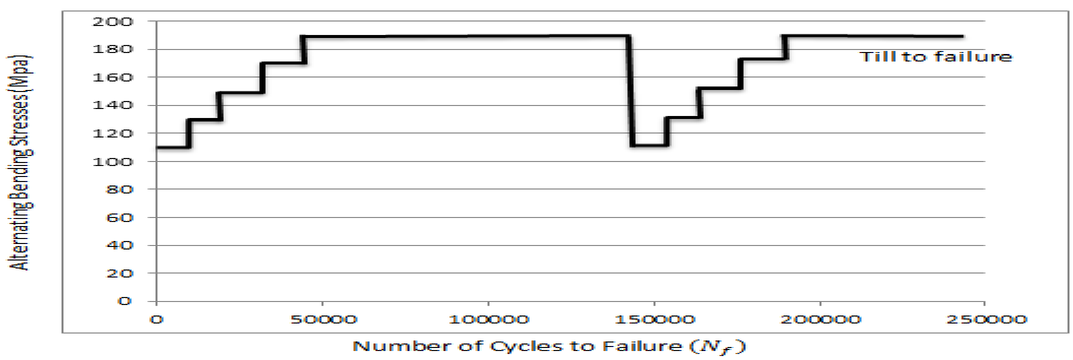

Figure 12. Loading program for specimen (B5)

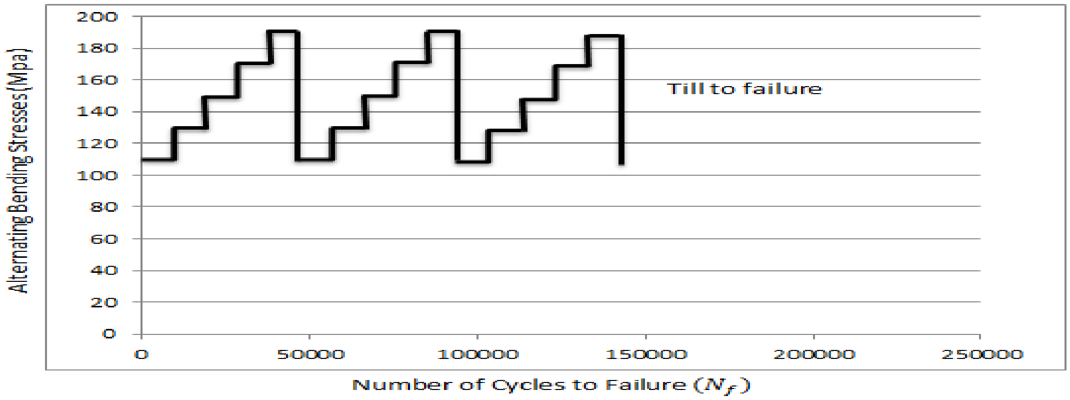

Figure 13. Loading program for specimen (B6)

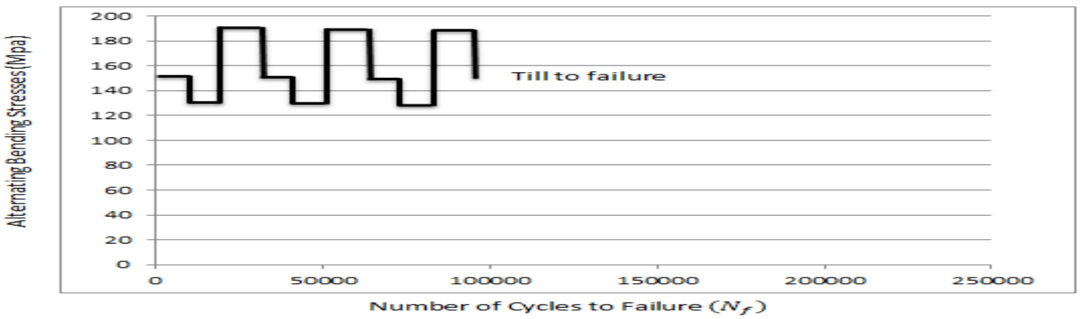

Figure 14. Loading program for specimen (B7).

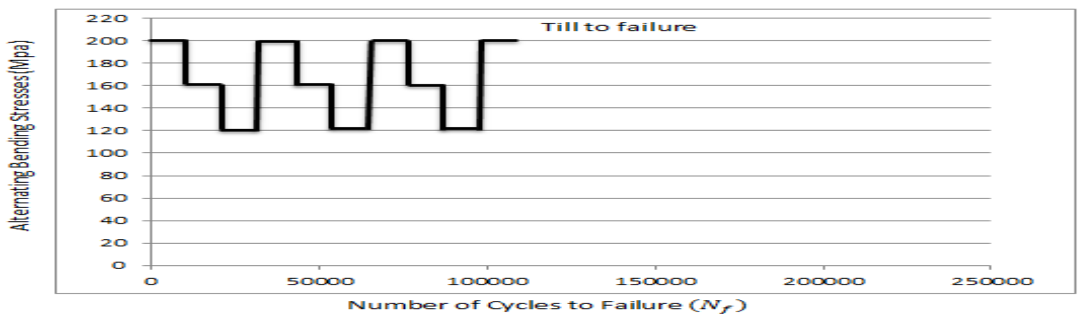

Figure 15. Loading program for specimen (B8)

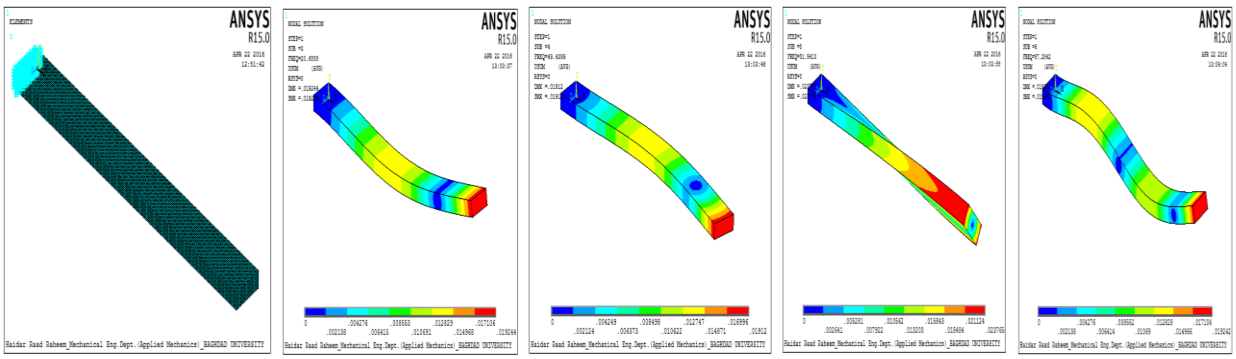

Figure 16. Laboratory Specimens subjecting to loading programs using ANSYS

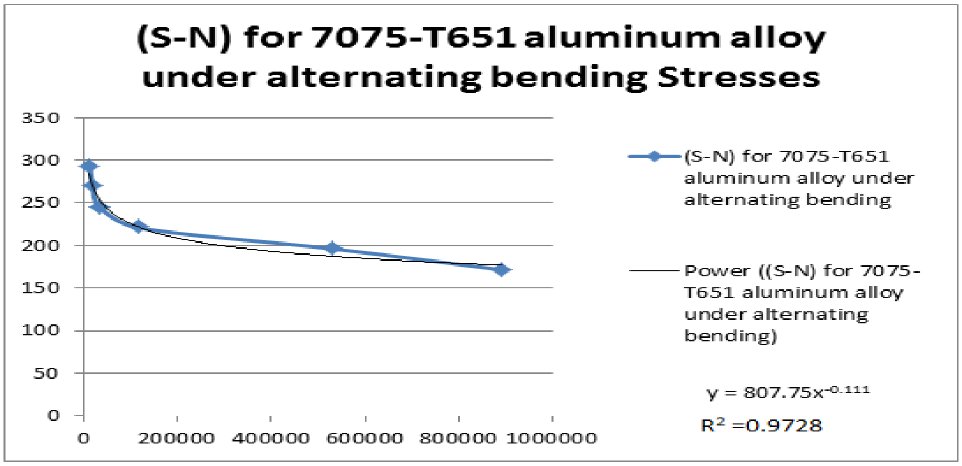

Figure 17. (S-N) Curve for aluminum alloy (7075-T651)

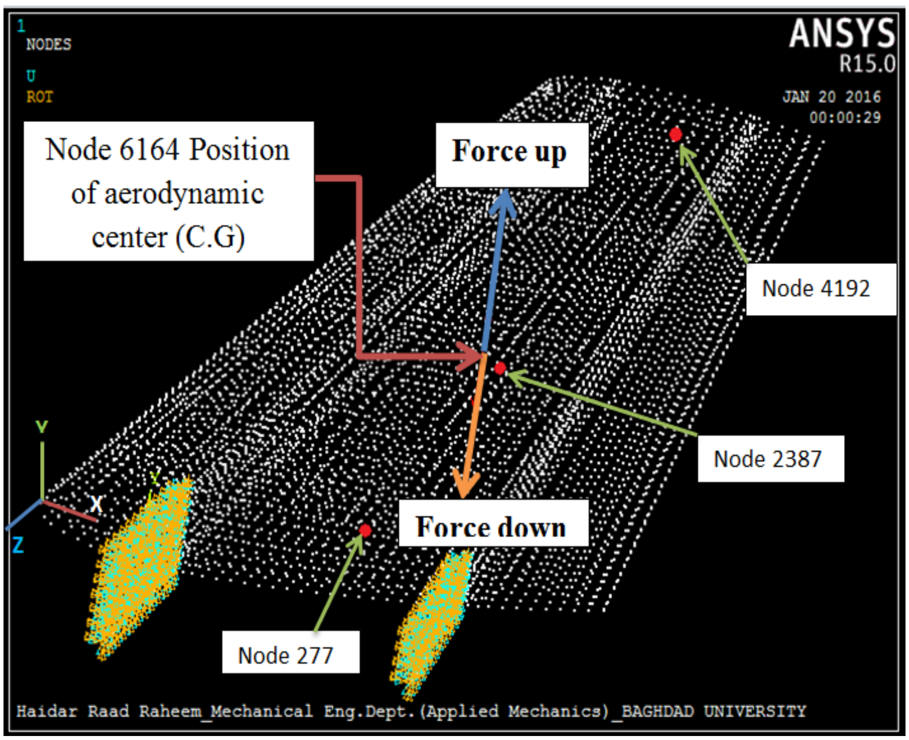

Figure 18. Applied loads on position of aerodynamic center (C.G).

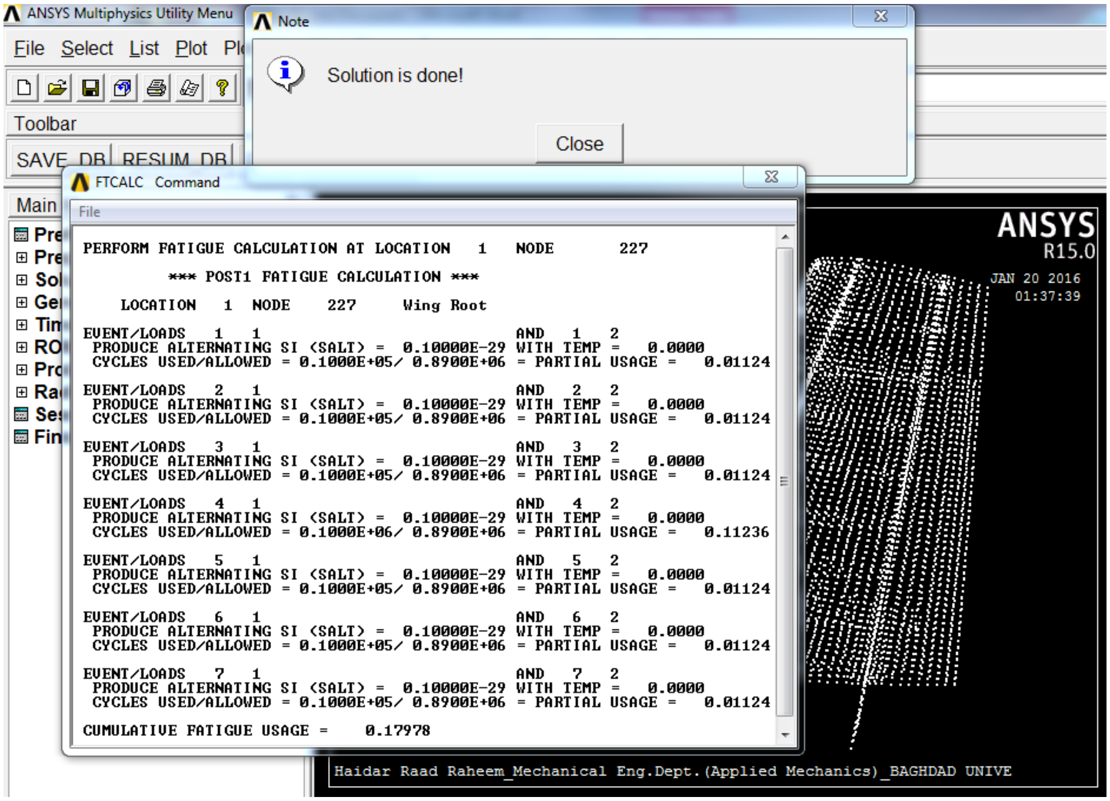

Figure 19 (a). Cumulative fatigue damage for wing root node.

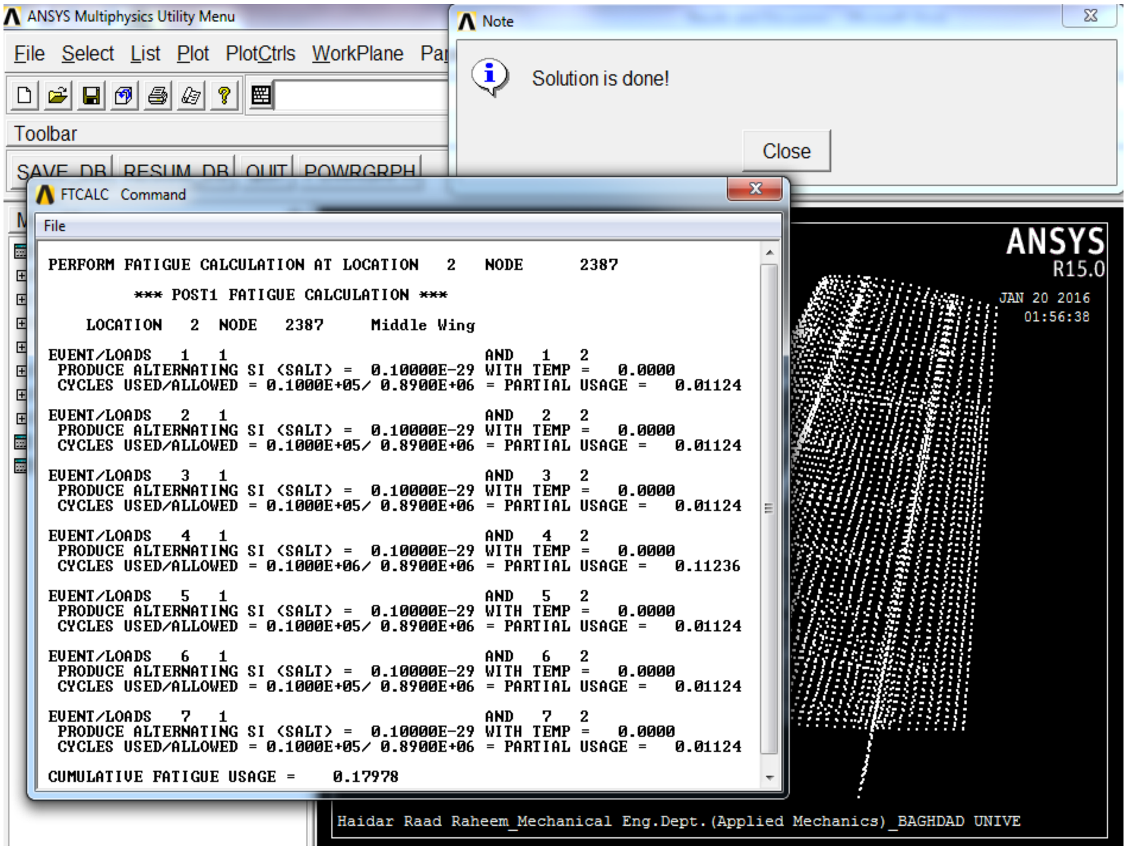

Figure 19 (b). Cumulative fatigue damage for wing middle node.

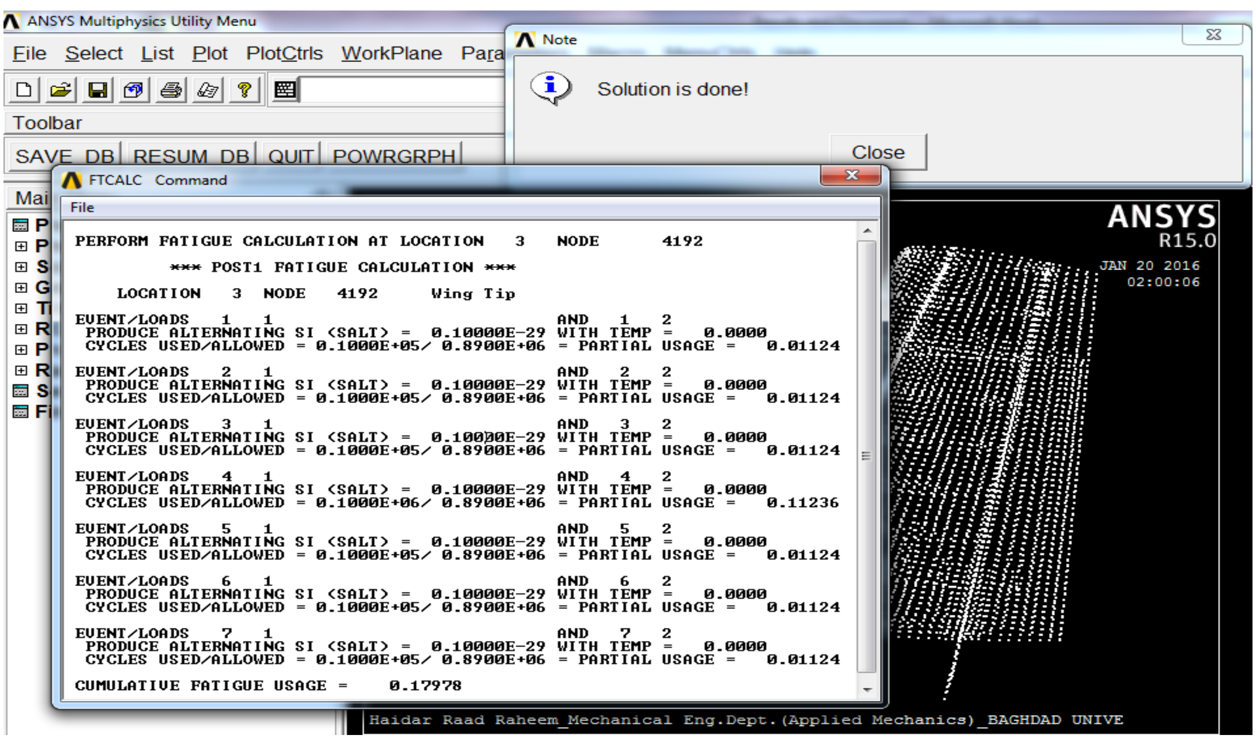

Figure 19 (c). Cumulative fatigue damage for wing tip node.

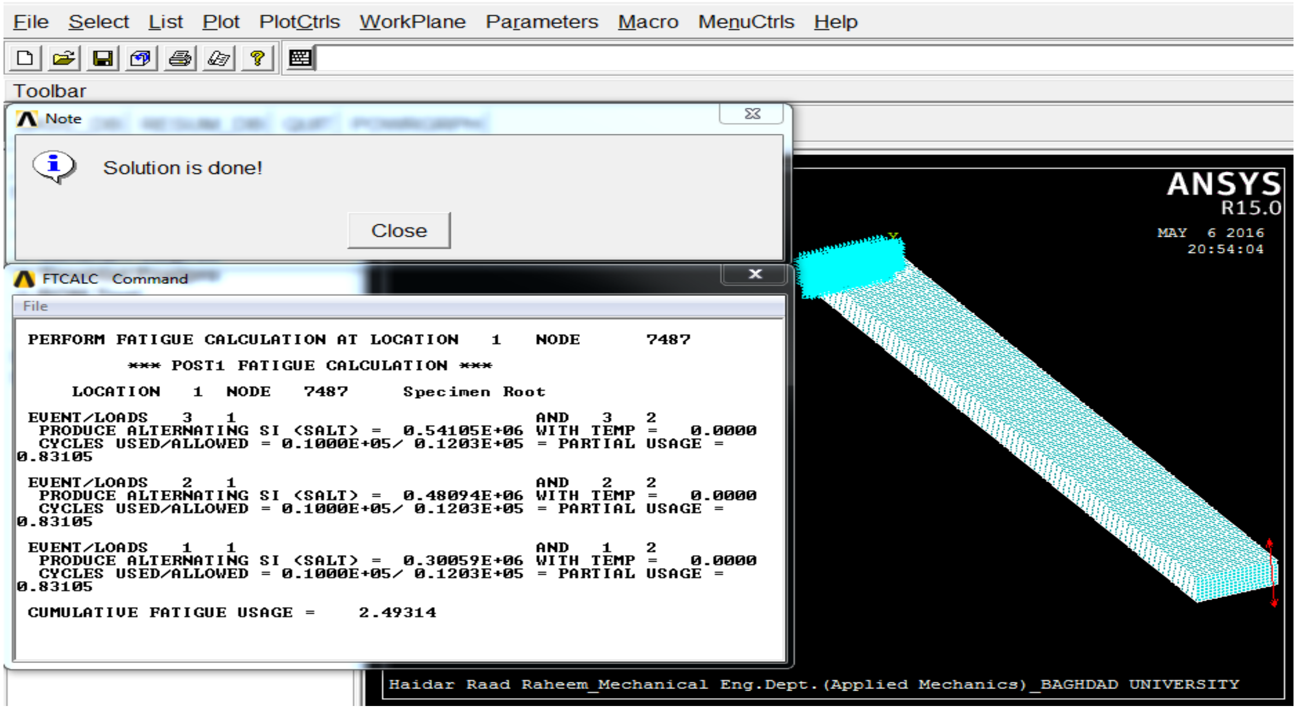

Figure 20 (a). Cumulative fatigue damage for specimen B8 root node.

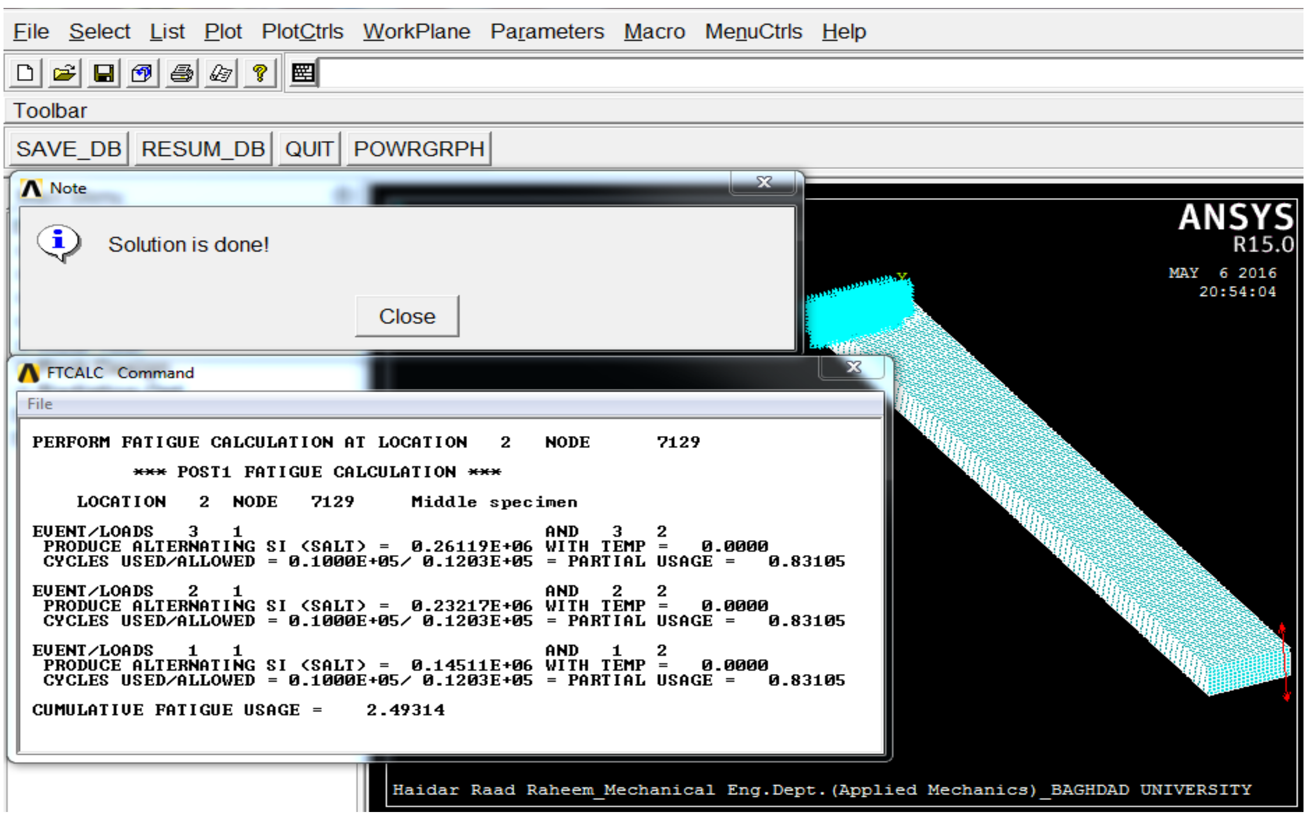

Figure 20 (b). Cumulative fatigue damage for specimen B8 middle node and equal to tip node.

Table 1. Data of the selected wing.

Single wing span (m)	3
Wing area (m^2)	3.6
Aspect ratio	2.5
Taper ratio	0.5
Mean aerodynamic chord (m)	1.244
Root chord (m)	1.6
Tip chord (m)	0.8
Position of aerodynamic center	X= 1.338 and Y=0.642
Wing section profile	Symmetric NACA0015
Sweep angle along leading edge	15°
Wing shape	Trapezoidal in X and Y

Table 2. Chemical compositions for (7075-T651) aluminum alloy, %.

Al. alloy	**Si**	**Fe**	**Cu**	**Mn**	**Mg**	**Cr**	**Zn**	**Al**
7075-T651 Standard	0.4	0.5	1.2-2	0.3	2.1-2.9	0.18-0.28	5.1-6.1	Rem.
7075-T651 Experimental	0.41	0.44	1.4	0.29	2.6	0.26	6	Rem.

Table 3. Standard and experimental mechanical properties.

Al. alloy	**Ultimate strength (MPa)**	**Yield strength (MPa)**	**Elongation %**	**Modulus of Elasticity (GPa)**
7075-T651 Standard	572	503	11%	71.7
7075-T651 Experimental	542	490	7.5%	71

Table 4. Fatigue test results under constant cycles for aluminum alloy (7075-T651).

Specimen Number	Applied Stress (Mpa)	Number of Cycles to Failure (N_f)
A1	294	9300
A2		12800
A3		14000
A4	269.5	15000
A5		18000
A6		22000
A7	245	29000
A8		33500
A9		38000
A10	220.5	105000
A11		116000
A12		132000
A13	196	500000
A14		535000
A15		560000
A16	171.5	790000
A17		880000
A18		1000000

Table 5. Fatigue life according to experimental, mathematical results and cumulative.

Specimen No.	No. of Programs	No. of cycles for one program	$D_{(exp.)}$	$N_{f\,(exp.)}$	$N_{f\,(Model)}$	D_{Miner}	$N_{f\,(Miner)}$	$N_{f\,(Marsh)}$
B1	-	-	-	Unfailed	Unfailed	1	Unfiled	Unfiled
B2	7.55	160000	0.426	1208000	1200727	1	2835680	1932800
B3	1	640000	1.183	640000	637837	1	542372	350000
B4	4.5	160000	0.25	728000	681820	1	2912000	1248000
B5	6	140000	0.7	840000	770370	1	1200000	1221818
B6	25	50000	0.95	1250000	1125926	1	1315790	1818182
B7	45	30000	1.28	1380000	1163640	1	1078125	1505455
B8	27	30000	0.9	810000	810000	1	900000	997000

Table 6. Lower and upper limits for the number of cycles with different failure probabilities.

Specimen No.	Applied load (Mpa)	Range of (N_f) 90%	Range of (N_f) 80%	Range of (N_f) 70%	Range of (N_f) 55%
A1	294	6661→ 16639	8675→ 14624	9760 → 13540	11000 → 12380
A2					
A3					
A4	269.5	12043→ 24957	14940→ 22059	16500 → 20500	18168 → 18832
A5					
A6					
A7	245	25013→ 41987	28726→ 38275	30724→ 36277	33000→ 34140
A8					
A9					
A10	220.5	93728→143272	104230→132071	110957→126043	117406→119594
A11					
A12					
A13	196	471493→588507	496357→563643	509740→550261	524054→535946
A14					
A15					
A16	171.5	701298→1088702	788217→1001783	834955→955005	885039→904960
A17					
A18					

Table 7. Summary of cumulative damage for (AA 7075-T651) aircraft wing.

Figure no. of loading program	(D_{Ansys}) One Program wing structure	Experimental no. of programs to failure for specimens	(D_{Ansys}) Total programs wing structure	(D_{Ansys}) Total programs specimens
Figure (9) B2	0.17978	7.55	1.357339	1.3296
Figure (10) B3	0.71910	1	0.71910	5.3
Figure (11) B4	0.16327	4.5	0.734715	1.3
Figure (12) B5	0.14141	6	0.848846	1.163
Figure (13) B6	0.05	25	1.25	4.155
Figure (14) B7	0.03371	45	1.51695	2.493
Figure (15) B8	0.03333	27	0.89991	2.49