SUDAN UNIVERSITY OF SIENCE AND TECHNOLOGY

College of Engineering

Department of Aeronautic

STUDY OF HELICOPTER BLADE FLUTTER

Thesis submitted in partial fulfillment of the requirement for the degree of B.SC., in aeronautical engineering

Prepared by:

Sabla Ashraf Ali Mustafa

Tanzel Abdul-Aziz

Supervised by ASS: Prof. Ali AlHussine

بسم الله الرحمن الرحيم

(شهد الله انه لا اله الا هو والملائكة و أولو العلم قائما بالقسط لا اله الا هو العزيز الحكيم)

آل عمران الآية (18)

ABSTERACT

This study is to determine the effect of the flutter phenomena on the helicopter blades.

Analytically the exciting force should not exceed the damping force when it exceeds the damping force the flutter will happen. Also the inter section point between the damping and the exciting forces specified the critical rotational speed.

There is some impendence such as the rigidity of the blade model used which causes possibility of breakdown of the blade because the angle of attack of the blade is already adjusted and has no ability for flapping. But in articulated blade the angle of attack is adjusted to be suitable for various flight conditions.

An experimental facility (wind tunnel) was used to determine the relationship between load distribution at various locations along air foil and possibility of flutter at several angles of attack. It is found that after adding the loads at the leading edge it is observed that the center of gravity will moved forward which lead to delay in flutter comparing with case(without load).

Besides that experimentally the results are not accurate. Also we suffer from proper design of airfoil used because it’s heavy and rigid when it’s made from fiber.in addition the airfoil is longer than the test section.

Acknowledgment

This work cannot reach without the aid of many persons that were surrounding us all the time by direction and support.

Special thanks to our supervisor ASS prof Al –Alhussine.

DEDICATION

We dedicate this project to parents, brothers, sisters, friend’s, loves and everyone for kind deed for us...

Contents

List of figure:

List of table

Abbreviation:

w_1	$velocity\ vector$
w_2	$velocityvector$
w_3	$velocity\ vector$
h	$angular\ moment$
r	$position\ of\ vector\ of\ particle$
m	$mass\ of\ prticle$
v_r	$velocity\ of\ particle$
A, B, C, D, E, F	*Moment of inertia components*
Δy_{ex}	$exiting\ force$
Δy_{dam}	$damping\ force$
c_y	$coefficient$
$\propto$	$angle\ of\ attack$
a	$lift\ curve\ slope$
Δy_{dam}	$twist\ angle$
c_y	$vertical\ velocity$
u	$rotational\ speed$
ρ	$density\ of\ air$
A_b	$blade\ area$
Cl	$lift\ coefficent$
C.G	$center\ of\ gravity$
$NTFC.G$	$national\ trnsonic\ facility$
$NASA$	$national\ aeronutics\ and\ space\ adminstration$
$LAMS$	$Load\ elevation\ and\ mode\ stablisation$
USA	$United\ state\ of\ america$
$NACA$	$national\ advesory\ comitte\ for\ \ aeronautic$

Chapter one

Introduction

1 Chapter one: Introduction

1.1 Introduction

The aero elasticity interest the interface of flexible structure with the surrounded flow, particularly the aircraft structure are flexible due to weight constrain, they are under goes to several type of aero elastic phenomenon

Aeronautical engineer become very interested about the aero elasticity when the aircraft moves throw the air the load act on structure of the helicopter this load leading to disturbance in boundary layers of the helicopter blade and caused distortion of the helicopter flexible structure, this distortion change boundary layer and aerodynamics loads.

There are two loads, aerodynamic and internal elastic loads .These loads must be converge to achieve equilibrium.

In steady airflow the structure statically distortion,

Under different flight condition increase distortion and may be cause structure failure of aircraft.

Aero statically faced several problems one of this problems is the stability of the structure in air when the aerodynamic loads increase rabidly with increasing the velocity of air, the elastic stiffness is independent of velocity of air, then the structure becomes un stable, then may be main rotor of helicopter twist due to elastic instability and this twist cause the flutter.

Flutter is main problem of helicopter structure at the main rotor blade, where induce oscillations, it is described by the interaction of aerodynamic, elastic and inertia forces also it is dynamic elastic instability problem.

When the frequency equal zero, and neglecting the inertia force is static aero elastic instability problem.

It is important for safety of helicopter flight suitable precaution done to avoid self-exiting vibration such as flutter. The understanding of flutter phenomena is very important in design of rotary wing aircraft.

The aero elastic stability of helicopter rotor blade is multifaceted problem for the extreme variation of the aerodynamic environment with in flight envelope of aircraft.

This aero elastic instability is characterized by self-excited undammed oscillations of the blade lifting surface in torsion and bending (elastic flapping). This problem is generally solved by mass balancing the blade about the quarter chord and designing the elastic axis to lie at the quarter chord position.

1.2 Overview

Flutter is defined as an elastic self-excited vibration in which the external source of energy is the air streams where flutter occur the air stream provide energy to the system more rapidly than it is fizzle by damping .Also aero elastic flutter involves the unprofitable interaction of aerodynamic, inertia forces and elastic on structure to produce an unstable oscillation that predominating results in structural failure. Today's aircraft designs the design is free of flutter within the flight envelope .These analytical results are often verified by wind-tunnel flutter models and ground vibration tests. Flight flutter testing provides the final verification of the analytical predictions throughout the flight envelope .In the early years of aviation, no formal flutter testing of full-scale aircraft was carried out. The aircraft was simply flown **to** its maximum speed to demonstrate the aero elastic stability of the vehicle. The first formal flutter test was **car**ried out by Von Schlippe in 1935 in Germany. His approach was to vibrate the aircraft at resonant frequencies at progressively higher speeds and plot amplitude as a function of airspeed. A rise in amplitude would suggest reduced damping with flutter occurring at the asymptote theoretically infinite amplitude this idea was applied successfully to several German aircraft.

In design consideration must be considered that the rotor blade tip design is poorly the flutter should been heard in maneuver flight. The extreme variations of the environment with in flight envelope of aircraft generate several problems' faced aero elastic stability of helicopter rotor blade.

1.3 Aims and objectives

Aims:

The position of center of gravity not returns back by other hand control of center of gravity.

, try to make the structure flexible to be flutter resistance.

Objectives:

- Effect of center of gravity position to lift ,drag and moment coefficients
  - To calculate lift coefficient with respect to angle of attack
  - Read lift force
  - Calculate lift curve slope
  - Calculate twist angle
  - Calculate damping force
  - Calculate the exiting force
  - Shift the Location of CG of blade forward

1.4 Problem statement

1) The helicopter fly at different flight conditions, the aero elastic stability of helicopter rotor blade is multifaceted problem for the extreme variation of the aerodynamic environment with in flight envelope of helicopter. Also the blade rotate at different flight angles through the rotation of plane, and the thrust at one angle differ from other angle, the distribution of thrust around the blade disc is not equal these can be overcome by changing the angle in advance blade and increasing in retreating blade this can done for equalization of thrust around the disc, this un equalization of thrust cause more vibration and oscillation these lead to flutter phenomena

2) Erosion of blade by dust may cause flutter.
3) Bad servicing or repair of the blade may remove blade center of gravity back ward and may cause flutter.

1.5 Proposed Solution

Low speed subsonic wind tunnel is proposed and it is experimental facility used to simulate the nature flow around the aero foil, the method which is used adding balance masses or load at different positions in the air foil, Also try to explain damping and exiting force by using explanation equations.

1.6 Motivation

Because of that the main rotor is the main source of lift and thrust force in helicopter it is very dangerous when the flutter is happening for that reason we decided to study the effect of vibration flutter in the rotor blades .

1.7 Contribution

The study of flutter was done experimentally and analytically by using some different equations to determine some linkage phenomena such as forces (lift, drag, moment, exiting and damping forces), bending and twisting that it have big effect in flutter problem , finally the important primary information should be found .

1.8 Methodology

In our research collective the data from wind tunnel which is used(subsonic wind tunnel) by reading value of forces from digital panel of wind tunnel substitute these data into the equation and explain it in the table then translate the data with figures by using Microsoft excel. The first Chapter about general introduction of flutter definition and it is effect on the blade structure proposed solution and contribution of this study. Followed by the second chapter which contain the history of the helicopter blade flutter and when this study started in real life , tested by scientist and when the ,the first accident ,first tester and Beside that the method used to solve this problem by analytical form. Beside that The third chapter about detailed contribution, detailed analysis, .this experiment under goes to analysing view showing the, advantage and dis advantage and the preferable method to use. Followed by the fourth Chapter about the figures and tables appendix, Then we talked about fifth one which include the appendix of pictures, then the six one about result and discussion. Finally the seventh chapter which include conclusions, recommendations, future work and references.

Chapter two

Literature review

2 Chapter two: literature review

2.1 History and background

Flutter has destroyed countless airplanes since the early days of flying. Before 1930s, little was understood of flutter. The spirit of adventure prevailing in those "barn-storming" days of aviation encouraged the aviators to take great risks with the new sport. Many "cowboy" types even routinely sawed-off portions from the airplane wing-tips in order to get a better aerobatic performance out of their low-powered and flimsy machines. One will never know how many of these intrepid souls succumbed to flutter. In the 1930s, with the availability of better engines and in attempts to set new flying speed records, flutter began to be recognized as the death-trap it was. Consequently, serious engineering effort in analyzing and preventing flutter began in earnest, especially in the design of the faster fighter aircrafts of the 1930s and '40s. One obvious solution was to dramatically increase the structural stiffness, but this was not always possible due to weight considerations. When experiments and analytical models revealed that the flight velocity at which flutter occurs and its characteristic frequency are as much affected by the mass distribution of the structure as its stiffness, mass balancing of the wings, tails and control-surfaces began to be an integral part of aircraft construction. With the passage of time, as the maximum flight velocities of aircrafts increased beyond the speed of sound, it was noticed that flutter was most likely at the transonic speed (close to the speed of sound) due to the unsteady motion of a shock wave on top of the wing. Therefore, a better modeling of the unsteady aerodynamic loads in the transonic regime was required, which resulted in computer codes that had run-time of weeks on the Cray supercomputers. Consequently, the world's largest supercomputer, namely the National Transonic Facility (NTF) of U.S.A., was dedicated to address the unsteady transonic aerodynamic problem. At the same time, experimental facilities at organizations such as NASA-Langley Research Centre were upgraded to study transonic flutter. The valuable data resulting from these studies were applied to most of the airplanes we see flying today, such as Boeing 747-400 and Airbus A-340, which routinely cruise at transonic speeds.

The traditional passive means of avoiding flutter, such as mass balancing and local stiffening, have continued to the present day. These techniques are inefficient (because they add weight to the structure) as well as unsystematic, and they do not always succeed. Consequently, flutter keeps occurring. Recent examples include Taiwan's IDF fighter, which crashed due to flutter of horizontal tail during high dynamic- pressure flight-test in 1992, leading to the cancellation of the project.

Later in the same year, a prototype of the state-of-the-art American fighter, F-22, crashed in a flutter related accident. In September 1997, a U.S. Air Force F-117 "Stealth" fighter crashed due to flutter excited by the vibration from a loose eleven. Every year many small airplanes, usually the home-builds, continue to become casualties of flutter. Flutter models and ground vibration tests. Flight flutter testing provides the final verification of the analytical predictions throughout the flight envelope .In the early years of aviation, no formal flutter testing of full-scale aircraft was carried out. The aircraft was simply flown to its maximum speed to demonstrate the aero elastic stability of the vehicle. The first formal flutter test was carried out by Von Schlippe in 1935 in German. His approach was to vibrate the aircraft at resonant frequencies at progressively higher speeds and plot amplitude as a function of airspeed. A rise in amplitude would suggest reduced damping with flutter occurring at the asymptote of theoretically infinite amplitude as. This idea was applied successfully to several German aircraft until a Junkers JU90 fluttered and crashed during flight

Tests in 1938.Early *test* engineers were faced with inadequate instrumentation, excitation methods, and stability determination techniques. Since then, considerable improvements have been made in flight flutter test techniqu0 V flutter Airspeed _c_7_. Von Schlippe’s flight flutters test method, and response data analysis. Flutter testing, however, is still a hazardous test for several reasons.

First, one still must fly close to actual flutter speeds before imminent instabilities can be detected.

Second, subcritical damping trends cannot be accurately extrapolated to predict stability at higher airspeeds.

Third, the aero elastic stability may change abruptly from a stable condition to one that is unstable with only a few knots' change in airspeed .This paper presents a historical overview of the development of flight flutter testing, including a history of aircraft flutter incidents. The development of excitation systems,

Instrumentation systems and stability determination methods are reviewed as it pertains to flight flutter testing.

The first recorded flutter incident was on a Handley twin engine biplane bomber in 1916. The flutter mechanism consisted of a coupling of the fuselage torsion mode with an anti-symmetric elevator rotation mode.

The elevators on this airplane were independently actuated. The solution to the problem was to interconnect the

Elevators with a torque tube Control surface flutter began to appear during World War I. Wing-aileron flutter was widely encountered during this time. Von Baumhauer and Koning suggested the use of a mass balance about the control surface hinge line as a means of avoiding this type of flutter. Although some mild instances of control surface flutter were Encountered afterward, these were usual wings resulted in more wing flutter incidents. Primary surface flutter began to appear around 1925. Air racers experienced many incidents of flutter from them id- 1920's until the *mid*- 1930's as attempts were made to break speed records. give many examples of these incidents. Another form of flutter dealt with in the 1930's was servo tab flutter.

Collar predicted that this type of flutter would be around for many years. This prediction was correct, for between 1947 and 1956, 11 cases of tab flutter incidences were reported for military aircraft alone. Even today servo tab flutter is still a problem. In1986, the T-46A trainer experienced aileron flutter during test flight that was being flown to find the proper amount of mass balance. These ailerons were free floating and driven by tabs at the trailing edge of the aileron .New aero elastic problems emerged as aircraft could fly at transonic speeds. In 1944, while flight testing the newP-80 airplane, NACA pilots reported an incident of aileron of flutter involving transonic control surface buzz. Prototypes of both the F-100 and F-14 fighters had incidences of rudder buzz.

Today, the transonic flight regime is still flying eliminated. Buzz. From 1947 to 1956, there were 21 incidences Von Schlepped conducted the first formal flight flutter test in Germany in 1935. The objective of his test method was to lessen the risk associated with flutter testing. The usual practice at this time was to fly the airplane to the maximum speed and then to observe the stability of the Structure.

Von Schlep’s technique consisted of exciting the structure using a rotating unbalance weight, measuring the response amplitude, and then recording the response amplitude as a function of airspeed. The forced response amplitude would rapidly increase as the aircraft approached its flutter speed. Therefore, the flutter speed could be estimated from data obtained at subcritical airspeeds .The Germans successfully used this technique until1938 when a Junkers JU90 aircraft fluttered in flight and crashed. Inadequate structural excitation equipment and unsatisfactory response measurement and recording equipment were identified as probable causes for this accident The United States attempted this technique in the 1940's

With flutter tests of a Martin XPBM-I flying boat and a Cessna AT-8 airplane, taken from, the response amplitude data as a function of airspeed. The graph shows that destructive flutter for this airplane was averted by the narrowest of margins during this flight test.

The conventional method of flutter analysis consists of employing:

“An unsteady aerodynamic theory, suitable for simple harmonic motion of the lifting surface by some approximate considerations, this method provides an estimate of the stability present in the system at subcritical speeds. This method is called the k method or the conventional V-g method. While this method is satisfactory for prediction of the flutter boundary flight condition, the estimation of the stability present in the system is not acceptable at speeds below the critical speed. The k method needs to be considerably modified before it can be employed for subcritical damping predictions. A relatively new method called the p-k method has been highlighted study to improve the damping prediction of fixed wings.
An aerodynamic theory is considered to be of the p type if it deals with the motion of the lifting surface that decays in an exponentially damped simple harmonic fashion. In general, all sophisticated type aerodynamic theories require excessive computer time. Hence the p method of aero elastic solution, which can be considered, exact, is numerically time-consuming.
However, if a k type (UN damped simple harmonic motion) aerodynamic theory is applied after suitable modifications to a p type motion, a reasonably accurate and simplified formulation results. This is called the p- k method”.

2.1.1 Active Flutter Suppression

In order to overcome the inadequacy of passive techniques and to fly at a velocity greater than the flutter velocity, a new technique was developed in early 1970s, called "active flutter-suppression". Herein, an onboard automatic control system actuates a control-surface on the wing in response to sensed structural motion, in order to suppress flutter. The first practical demonstration of active flutter suppression was carried out by the U.S.

Air Force in their Load Alleviation and Mode Stabilization (LAMS) program, which resulted in a Boeing B-52 bomber creating history in the skies over the state of Kansas in the U.S.A. in 1973 by flying 10 knots faster than its open-loop flutter velocity. However, despite this early success, active flutter suppression has remained largely experimental, and has still not achieved operational status on any aircraft. This is because of many reasons, the chief among which is the difficulty in designing a control system which is robust to parametric uncertainties. Clearly, the aircraft designers and operators are not willing to take risks.

Active flutter suppression requires an accurate knowledge of the aero elastic modes that cause flutter, and then actively changing their characteristics in such a way that flutter occurs at a higher flight velocity. Although the classical flutter of a high aspect-ratio wing, such as that of a Boeing 747 or an Airbus A-340, is caused by an interaction between the first few bending and torsion aero elastic modes, flutter mechanism of a low aspect-ratio wing, such as that of a fighter airplane is more complicated. To study flutter, as well as active flutter suppression, it is necessary that an accurate aero elastic model based on modeling of the unsteady aerodynamic forces as a transfer-matrix be derived in this study we consider it as a way of solution not as a detailed solution in our study. In the Laplace domain, this matrix post-multiplied by the generalized displacement vector gives the vector containing unsteady aerodynamic forces and moments

. The most common method of obtaining the unsteady aerodynamic transfer-matrix is the use of optimized rational-function approximations for its terms, fitted to the frequency domain data in the harmonic limit. After the transfer-matrix is derived, a linear, time-invariant state-space model for the aero elastic system, including the control-surface actuator, can be obtained. The feedback controller for active flutter suppression can then be designed by standard closedloop techniques, such as engine structure assignment, LQG/LTR, H-infinity, and structured singular-value synthesis. Since there may be errors in the aero elastic modeling procedure, the derived control-law must be robust with respect to modeling uncertainties and sensor noise.

For the past decade, the author has worked on the various aspects of active flutter suppression, namely models for subsonic to supersonic frequency domain aerodynamics, nonlinear optimization for rational function approximations, and design of robust optimal controllers. There was also the satisfaction of the work finding practical application on the F/A-18 E/F fighter of McDonnell Douglas Corp., U.S.A. Since the area is interdisciplinary in nature, it affords an excellent opportunity to study fluid-dynamics, structural-dynamics, and control-systems, all at the same time. Sample the author's present effort on the subject of flutter suppression.

2.1.2 Definitions

- **The center of gravity:**

The center of gravity is defined as the center where all the loads accumulated around it. Main rotor blade of helicopter it has many parts so weight actually distributed through it .And we considered it is average location or in this location there is no rotation .Also the center of gravity it has limits for rear ward and forward movement. These limits were given by the manufacture of safe control.

- **Neutral point:**

Is the point where the differential of coefficient moment about the center of gravity BY the differential angle of attack.

Another definition is the point where the moment about center of gravity independent of angle of attack.

- **Static margin:**

The distance between center of gravity and neutral point this distance called static margin.

The neutral point must be located behind the center of gravity

If the distance between neutral point and the center of gravity is small and the neutral point behind the center of gravity the angle of attack is increased thus the lift is increased also, required for balance in this case the moment arms smaller.

That mean bigger static margin, bigger the moment arm and the less lift require for balance.

The larger static margin is more stable.

2.2 Literature review

0ne of main sources of vibration on an aircraft and helicopter is the power plant in:

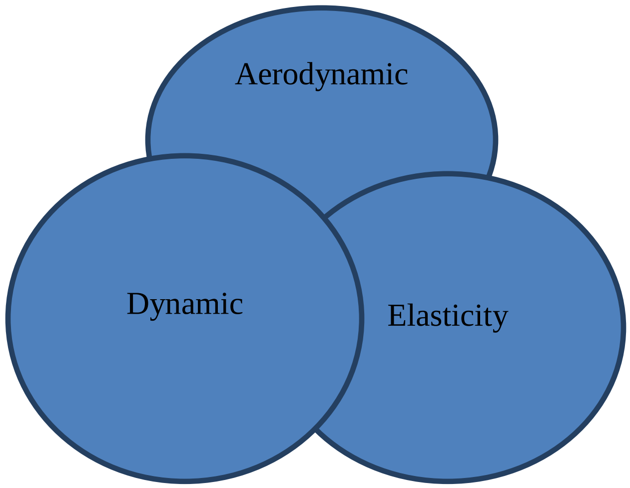
- The engine
- The propeller
- The rotor

The engine and propeller or rotor creates vibrations of two types:

- Mechanical.
- Acoustical.

The reason of propeller Vibration:

Is Static, dynamic and aerodynamic UN balance



- The intersection between aerodynamic and dynamic is the flight mechanic
- The interception between aerodynamic and elasticity is static aero elasticity
- The intersection between dynamic and elasticity is structural dynamic
- The intersection between aerodynamic and dynamic is dynamic aero elasticity

The static Unbalance of propeller appears when:

The center of mass does not coincide with the axis of rotation.

And the value of non-coincidence is determined by the tolerance on static balance of propeller.

a) The dynamic un balance of propeller appear when :
   The axis of rotation of propeller does not coincide with one of principal centric axis it happen when center of mass of separate blade lie in different plane ,normal to the axis of rotation . Thus UN balanced couple with forces concerning Y, Z axis appear

b) Aerodynamic of propeller un balance appear when:

Aerodynamic forces, of separate blade have various values or direction. It happens if:

a. Various blade angle of blade.
b. Center of pressure of blades are located on different distance from the axis Of Rotation.
c. Under big angle of attack there is stalling Phenomena, on blades, which cause oscillation of blades.
d. There is interference in arranged of a propeller with parts of aircraft ,which is behind or in front of propeller (in front of wing or in Front of fuselage).

These things caused by propeller vibration:

1) Decrease of service life and reliability of vibrating parts by causing fatigue material
2) Appearance of backlashes and increase of wear.
3) Problem in instrument while operating and observation of them.
4) Caused passenger and crew un comfort .in this event the vibration can be un acceptable earlier than they achieve structure dangerous value.
5) Pressure disturbance –sealing of pipe lines and system
6) Possibility of resonance .thus the amplitude of movements and the stress in the structure sharply increase.

There are triply connected oscillations –generally when the center of mass and the elastic center lie in axis of rotation of propeller or main rotor.

2.2.1 Conditions of self- exciting vibration:

In helicopter there are three conditions:

1) Ground resonance.
2) Air resonance.
3) Flutter.

Contemporary high performance main rotor systems are made of light weight composite construction. The out board section of the blade operate in the compressible Mach number regime, and are made of camber airfoil section to improve the hover aerodynamic efficiency .furthermore, to augment the stability of the rotor of ground resonance, furthermore ,to augment the stability of the rotor in ground resonance ,air resonance , rotor –pylon aeromechanical stability .

The resonance is characterized by the blade lagging vibrations coupled with the vibration of the aircraft fore and aft, and sideward. While analyzing self-excited vibrations of ground resonance, blade lag vibrations are of prime importance.

2.2.2 Flutter

Flutter is defined as an aero elastic self-exiting vibration where the external source of energy is the air stream. It is very dangerous type of vibration

It involves UN favorable interaction aerodynamic, inertia and elastic force on structure. The aero elastic analysis used to ensure that the design is free of flutter with in flight envelope. it will verified by wind tunnel test.

Flutter is happen in main rotor but never happen in tail rotor because of it is stiffness, why in main because of that the main rotor is the main source of lift and thrust force in helicopter it is very dangerous when the flutter is happening for that reason we decided to study the effect of vibration flutter in the rotor blades.

The study of flutter was done experimentally and analytically by using some different equation to determine some linkage phenomena such as force, bending and twisting that it has big effect in flutter problem finally the important information should be found.

The helicopter rotor blade designing requirements to be free of flutter are contained in federal aviation regulation (F.A.R) “each aerodynamic surface of the rotorcraft must be free from divergence in addition to the requirement of freedom from flutter. The aero elastic stability evaluations required by this regulation include flutter and divergence. Compliance with this regulatory requirement should be shown by analysis and/or flight test, supported by any other means found necessary by the Administrator. The aero elastic evaluation of the rotorcraft should include an investigation of the significant elastic, inertia and aerodynamic forces on all aerodynamic surfaces (including rotor blades) and their supporting structure. The forces associated with the rotations and displacements of the plane of the rotors should be considered."

A rotor blade designed with center of gravity, aerodynamic and elastic axis at quarter chord is heavier than free of that restriction.

Flutter types:

A .Bending (flapping)

B .Bending twist (pitch flap flutter)

In order to solve flutter problem, the free vibration for torsion and bending mode should be determined beside the natural frequencies for rotating blades. To allow analysis of the effect of flap input frequency on the flutter solution the stiffness of the torsion spring for the flap is tuned such that the flap rigid body in coupled natural frequency will equate the flap input frequency.

There are many methods in use to determine the modes of the flutter for example:

Myklested method for bending

Holzer method for torsion mode and frequencies

Holzer method written in transfer matrix:-

$$\begin{bmatrix} 1 & 0 \\ -\mathrm{I}\left(\omega^2_{\alpha-\Omega^2\cos\theta_n}\right) & 1 \end{bmatrix} \begin{pmatrix} \emptyset \\ T \end{pmatrix}_n = \begin{bmatrix} 1 & -l_{n,n+1/\left[(GJ)_{n+1}+\left(CK_a^2\right)_{n+1}\right.} \\ 0 & 1 \end{bmatrix} \begin{pmatrix} \emptyset \\ T \end{pmatrix}_{n+1}$$

Myskled method written in transfer matrix

2.2.3 Main rotor blade flutter

A. Flapping flutter:-

At coupling the axis of stiffness with the center of gravity line of blade section the chord of the blade moves parallel without twist this type of vibration called bending flutter .the figure bellow show this type of flutter. Show figure (5.1)

At bending down of blade end the elastic force which acting on blade center of gravity directed it up words. When the blade is freed, the force accelerates the blade up words. Because of acceleration in inertia force in center of gravity directed dawn. The moment of inertia force equal zero also no twist happen at locating of the two forces in one point.

The elastic force & inertia force decreased beside that the motion velocity increased when the blade element move up.

The velocity reached maximum value when the blade take neutral position the inertia &elastic forces equal zero.

The elastic force changes it is direction causing reduce in blade element velocity and stops in the maximum position at upper, when continuing moving up.

The inertia force becomes max at bigger acceleration in the tip.

At neutral position of the blade section the velocity reached it is maximum value also force and acceleration replaced its direction.

B. Pitch flap flutter:-

The pitch –flap flutter mechanism is that blade flaps, aerodynamics and inertia moments increase tend to twist the blade and develop aerodynamic flapping moment; the inertia effect of pitching on motion of flapping is small. The helicopter blade motion is developed by power full center- fugal moment which increases the flapping & torsion motion stiffness.

Blade flap and twists:-

Velocity component:

$$\omega_{1=\theta+\Omega \sin\dot{\beta}\approx\theta+\dot{\beta}\Omega}$$

$$\omega_{2=-\beta\cos\theta+\Omega\sin\theta\cos\beta\approx\beta+\dot{\theta}\Omega}$$

$$\omega_3 = \beta\sin\theta + \Omega\cos\theta\cos\beta \approx \Omega$$

The angular moment:-

$$h = \sum r.mv_r$$

$$\boldsymbol{m \equiv massofparticle}$$

$$\boldsymbol{r \equiv positionofvectorofparticle}$$

$$\boldsymbol{v}_{\mathbf{r\equiv velocityofparticle}}$$

Component of angular moment:-

$$h_{1=Aw_1-Fw_2-Ew_3}$$

$$h_2 = Bw_2 - Dw_3 - Fw_1$$

$$h_{3=Cw_3-Ew_1-Dw_2}$$

Moment of inertia product:-

$$A = \sum m(y^2 + z^2)$$

$$B = \sum m(x^2 + z^2)$$

$$C = \sum m(x^2 + y^2)$$

$$D = \sum myz$$

$$E = \sum mxz$$

$$F = \sum mxy$$

Neglect D&E the angular moment component become:-

$$h_{1=Aw_1 - Fw_2}$$

$$h_2 = Bw_2 - Fw_1$$

$$h_{3=Cw_3}$$

Differentiating with respect to time:-

$h_{1=A\ddot{\theta}+A\Omega^2\theta+F\ddot{\beta}+F\Omega^2\beta}$

$h_{2=-B\ddot{\beta}-B\Omega^2\theta-F\ddot{\theta}-F\Omega^2\theta}$

$$\frac{d^2\theta}{d\psi^2} + v_1^2 + \frac{F}{A}\left(\beta + \frac{d^2\beta}{d\psi^2}\right) = \frac{L_A}{A\Omega^2}$$

$$\frac{M_A}{B\Omega^2} = \frac{d^2\beta}{d\psi^2} + \lambda_1^2\beta + \frac{F}{B}\left(\frac{d^2\theta}{d\psi^2} + \theta\right)$$

2.2.4 Flutter condition

Exiting force:-

$$\Delta y_{\mathrm{ex}} = \Delta \mathrm{c_y} \mathrm{A_b} \rho \frac{\mathrm{U}^2}{2}$$

$$c_{\mathrm{y=f(\alpha)}}$$

Lift curve slope:-

$$\tan\beta = \frac{\Delta \mathrm{C_y}}{\Delta\alpha} = a$$

$$\Delta C_{\mathrm{y=a\alpha=a}\square}$$

$$\Delta Y_{\mathrm{ex=a}\square\mathrm{A_b}} \rho \frac{U^2}{2}$$

Damping force:-

$$\Delta y_{\mathrm{dam}} = \Delta \mathrm{c_y} \mathrm{A_b} \rho \frac{\mathrm{U}^2}{2}$$

$$\Delta\alpha = \frac{\mathrm{v_y}}{\mathrm{U}}$$

$$\Delta y_{\mathrm{dam}} = \mathrm{aV_y} \mathrm{A_b} \rho \frac{\mathrm{U}}{2}$$

Then the exiting force become:-

$$\Delta y_{\mathrm{ex}} = \Delta \mathrm{c_y} \mathrm{A_b} \rho \frac{\mathrm{U}^2}{2}$$

Damping force become:-

$$\Delta y_{\mathrm{dam}} = \mathrm{a}\mathrm{V}_{\mathrm{y}}\mathrm{A}_{\mathrm{b}}\rho\frac{\mathrm{U}}{2}$$

The flutter happens only when:

Exiting force is bigger than damping force.

The condition found at any blade but the second condition can cause the flutter

2.2.5 Critical speed of flutter:

It is the rotational speed at which the damping forces equal the exiting forces .the critical speed of flutter corresponding in the critical (ROTATIONAL SPEED) of flutter.

The dependence of critical rotational speed of flutter on different condition:-

- The stiffness of the blade
- Location of the blade center of gravity
- Axis of stiffness

Location of center of pressure is relative to axis of stiffness.

The critical rotational speed of flutter depends upon:

The flight speed ; at the increase of it the resultant speed in the azimuth angle 90 also increased so this can cause flutter ,therefore as the increase of flight speed the critical rotational speed of flutter decreased .

The flapping motion and bend of the blade decreased due to centrifugal force.

Therefore at the act of centrifugal force the critical rotational speed of flutter ,increase the flutter of main rotor blades due to less stiffness increased intensively ,as the flutter of aircraft wing ,and it can be discovered and stop it.

The reason of increase rotational speed of flutter:

- Flight speed, at the increase of it resultant speed in the azimuth 90.
- Less stiffness of main rotor blade increase intensively.

The reason of decrease rotational speed of flutter:

- Flapping movement, the amplitude of vertical displacement of blade element increased.
- Less stiffness of blade.
- Bigger coefficient of compensation.
- Increase of flight speed.
- The lines of blade center of gravity move back ward.
- Decrease of construction stiffness.

2.2.6 The flutter of articulated blade:

Flapping flutter in the articulated blades cause by horizontal and axial hinges, relative to this hinges flapping movement go with vibration of blades, the amplitude of this vibration increased. So this vibration is possible also at stiffed blade, without bend –twist.

The amplitude of vertical displacement of blade element at flapping movement is increased .the amplitude added from the displacement of flapping and bending. In this case the critical rotational speed of flutter decreased, for that the flutter caused by the horizontal hinge.

One of cause of decrease the blade stiffness at twist is the axial hinges. the rotation relative the axial hinges possible at elastic deformation of control organ of blade pitch .the less stiffness of these organs , the less critical rotational speed of flutter ,therefore the flutter caused by the axial hinges.

The flapping compensator causes the decrease of flap pitch at flapping up. And the flapping compensator causes the increase of flap pitch at flapping up.

This change like bending twist vibration flutter, for this reason of flutters the flapping compensator, the bigger of the coefficient of compensation, the less critical rotational speed of flutter. For most hubs the coefficient of compensation K=0.5.

The critical rotational speed of flutter decreases till the range of operation, this at the bigger value of compensation coefficient.

Some reason of flutter:

- Horizontal hinges can cause flutter.
- Axial hinges can cause flutter.
- Flapping compensator can cause flutter.
- Critical rotational speed.
- Azimuth angle.
- Upset of mass balance.

Operation reason of flutter:

The flutter excluded, at the construction of aircraft and helicopter. This means at the calculation of critical of flutter, they choose:

- Stiffness.
- Blade centering.

That critical rotational speed of flutter becomes bigger than the maximum permissible rotational speed of the main rotor.

Flutter can happen due to operational reason as follow:

- The upset of the mass balance.
- Decreased of construction stiffness.

The upset of mass balance of the blades, at the mixed of frame construction

In the blade of wooden ribs always affected by the air humidity, if the humidity increase the line of blade center of gravity moves backward, which lead to decrease of flutter rotational speed.

The upset of mass can happen at: bad repair of the blade which causes flutter.

The decrease construction stiffness leads to:

- The decrease of flutter rotational speed, which happen at disturbance of separate force element of the blades.
- Damage at blade skin.

The flutter can happen at the main rotor on ground and flight.

The flutter can be detected by strong vibration of helicopter, and the blade can- not fly in similar manner in the same track, or not having the correct spacing.

Chapter three

The experiment

3 Chapter three: The experiment

Introduction:

The wind tunnel is an experimental devices used to simulate the natural flow of air around the body (aero foil or aircraft).

The experiment was done by subsonic law speed wind tunnel.

3.1 Objectives:

To show the critical point at which the flutter could occurs and the critical angle beside the difference when adding masses

3.2 Procedures:

1-Equal five holes were drilled in air foil

2-Support airfoil or attach it to the wind tunnel at center of gravity.

3-Switch power on

4-Speed adjusted 20M/S

5-Load was positioned at different holes (8, 11, and 13) and the value of lift and drag is written at the following tables at different values of angle of attack.

6-the value of lift and drag force without adding masses was written.

In our airfoil blade of main rotor of helicopter center of gravity calculation we estimate the location of many parts of the airfoil blade such as the location of center of gravity and should be in the acceptable range.

In our experiment:

1. Determine the mass of one loads and alternative this load through many location.
2. Drilled five holes along airfoil span with elected distance.
3. Measured the lift force.
4. Measured drag force.
5. Change the angle of attack to get many values of above forces.

3.3 Experiment calculation method:-

1. Read the lift coefficient value from the digital panel which is connected with the wind tunnel model.
2. Read the drag coefficient value from the digital panel which is connected with wind tunnel model.by changes the angle.
3. I get several values of forces by change the angle of attack sequence from0 to12.
4. The center of gravity about 4cm from the leading edge of the airfoil.
5. The loads were alternative in elected distance from the center of gravity (explain the moment arms).
6. Determined or calculated moment force about the center of gravity.
   (M=lift force*it is distance from the center of gravity)
7. Determined or calculate lift coefficient.
8. Determined or calculate lift curve slope.
9. Determined or calculate exiting force.
10. Determined or calculate damping force.
11. Explain the critical speed.
12. Explain operational mode.

3.4 Methodology of the experimental:

Firstly we divided the airfoil into five distances. The center of gravity located at 4 cm from the leading edge of the airfoil, When the load in 8 positions is tested the following result:

On position8 the loads are about 8cm from the leading of the airfoil, and far about 4 cm from center of gravity.

In this position the failure or the angle which has been the stall occurs in angle 8.After that the rates of lift decrease progressively.

The drags also in this location reach the minimum value at 8 angles here the value of drag increase progressively.

Also the moment at this point has a minimum value equal (46) as shown in figure (27) due to small arm (equal 4 cm) multiplied with lift forces.

Also the lift coefficient increases until 8 angle then decreases progressively as same as the lift diagram show that in figure (4.3)

The result due to the lift coefficient is small scale of lift force.

In this location the static margin (the distance between position 8 and the center of gravity) is smaller than of any positions. In the position 8 is the nearest location to the center of gravity .from this point we have talked about the stability and control.

Whenever the distance between the center of gravity and the position of load, is smaller than the any other, stable and the stall late occur in far from the center of gravity.

Whenever the distance between the center of gravity and the position of load is smaller than other,.

Thus the position 8 is more stable than other and more controllable , When the load in position11 is tested the following result:

The second load located in position11 is far about 7cm from center of gravity and 11 cm from the leading edge from the airfoil in this position the failure or the angle which has been the stall occurs in angle 6.

Then the rates of lift decrease progressively as shown in figure (4.5).

The drag also in this location reaches the minimum value at 6 angles then the value of drag increase progressively.

Also the moment has a medium value due to the arm (equal 7) then the moment value equal(70.5)as shown in figure(28), is bigger than in position8 (equal 4) and smaller than position 13 (equal 9) multiplied by lift force.

Also the lift coefficient increase until 6 angle then decrease progressively as same as the lift diagram show that in figure (4.8).

The result due to the lift coefficient is small scale of lift force.

Relatively the stability this location is more stable than location 13 and less stable than location 8.

Relatively the control also is more controllability than location 13 and less controllability than location 8.

, When the load in position13 is tested the following result:

The third load located in position13 is far about 9cm from center of gravity and 13 cm from the leading edge from the airfoil in this position the failure or the angle which has been the stall occurs in angle 4. Then the angle of the rates of lift decrease progressively.

The drag also in this location reach the minimum value at 4 angle after this angle the value of drag increase progressively .

Also the moment has a maximum value due to the biggest arm from the center of gravity (equal9) multiplied by lift force showing in figure (29) the value of moment about (70.95).

Also the lift coefficient increase until 4 angle then decrease progressively as same as the lift diagram show that in figure (4.12).

Relatively the stability at this location is less stability than location 11 and less stability than location 8.

Relatively the control also is less controllability than location 13 and less controllability than location 8.

When the failure or stall has been observed in the airfoil without load at any position the stall happen at earliest angle before it happen when the load is used regard less about the position of load .

The stalls in this case happen in 2 angles. After this angle the rates of lift decrease progressively.

The drags also in this location reach the minimum value at 2 angle after this angle the value of drag increase progressively.

When there is no moment due to no moment arm from the loads because there is no load

3.5 Damping and exiting force:

Relatively the damping and exiting force Calculation of the exiting force from equation mentioned previously in chapter two, Therefore the exiting force, proportional to, the angle of attack and squire root of rotational speed.

Calculation the damping force from equation also mentioned previously in chapter two, therefore the damping force is proportional to the bending speed Vy and rotational speed U

The lift curve slop should be get by measured the angle of proper distance along the lift curve then determined tan of the angle this is the lift curve slope in the figure (4.1) and figure (4.5) and figure (4.9) and figure (4.13).

Explained the damping and exiting force in location 8, and location 11, and location13.also explained the damping and exiting force without load at any location.

And the figures explained that the damping and exiting force with load 8, and load11, and load13 are (4.20), (4.23) and (4.26) respectively. The exiting force not exceeded the damping force.

But without load the exiting force would exceed the damping force.

This thing shows clearly in the figure (4.17).

3.6 Calculation:-

Given:-

$$\rho = 1.225kg/m^3$$

$$A_b = 15m^2$$

$$u = 220m/s^2$$

$$v_y = 10m/s^2$$

The lift curve slope given from data with respect to the value of lift coefficient.

$$a = \frac{dcl}{d\alpha}$$

Or:-

$a = \tan \alpha$

$a_{atload8=\tan 56=1.4}$

$a_{11} \tan 57.17 = 1.55$

$\mathrm{a}_{13=\tan 60=1.7}$

$a_{\text{without load}=\tan 50=1.19}$

$\Delta\alpha = \frac{v_y}{\mathrm{u}} = \frac{10}{220} = 0.o4545$

$\Delta C_{\mathrm{y=a\alpha=a}\square}$

$\Delta \mathrm{cy}_8 = 1.48 * 0.04545 = 0.0666$

$$\Delta cy_{11} = 1.55 * 0.04545 = 0.0704$$

$$\Delta cy_{13} = 1.7 * 0.04545 = 0.0778$$

$$\Delta cy_{without} = 1.19 * 0.04545 = 0.53$$

Then the exiting force become:-

$$\Delta y_{ex8} = 0.0666 * 15 * 1.225 * \frac{U^2}{2}$$

$$\Delta y_{ex11} = o.o1548 * 15 * 1.225 * \frac{U^2}{2}$$

$$\Delta y_{ex13} = 0.0778 * 15 * 1.225 * \frac{U^2}{2}$$

$$\Delta y_{exwithout} = 0.53 * 15 * 1.225 * \frac{U^2}{2}$$

Damping force become:-

$$\Delta y_{dam8} = 1.48 * 10 * 15 * 1.225 \frac{U}{2}$$

$$\Delta y_{dam11} = 1.55 * 10 * 15 * 1.225 * \frac{U}{2}$$

$$\Delta y_{dam13} = 1.73 * 10 * 15 * 1.225 * \frac{U}{2}$$

$$\Delta y_{damwithout} = 1.19 * 10 * 15 * 1.225 * \frac{U}{2}$$

The value of u is divided from 0to220 as shown in the tables (3.5), (3.6) and (3.8).

Chapter four

Figures and tables

4 Chapter four: Figures and table

4.1 Figures

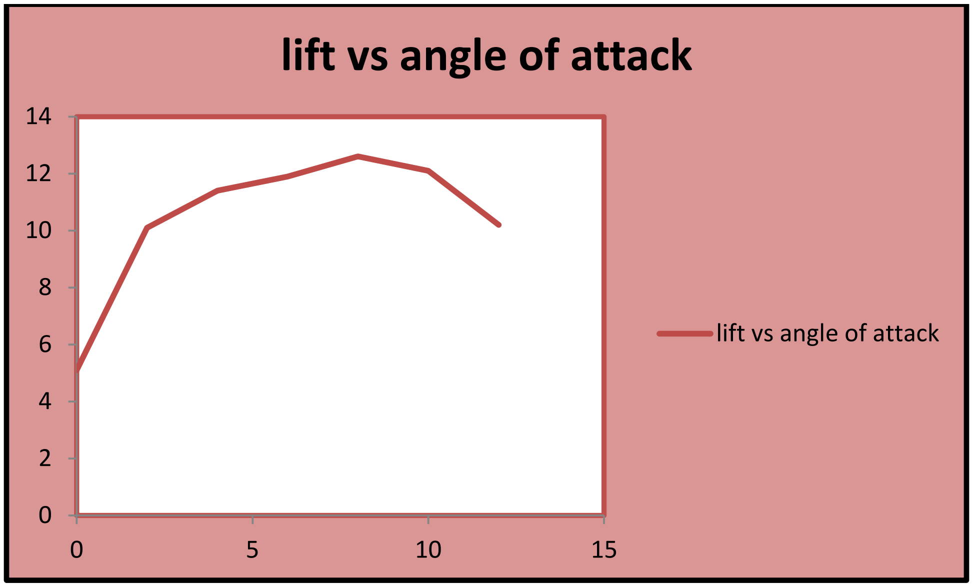


Figure 4-1: lift in position 8

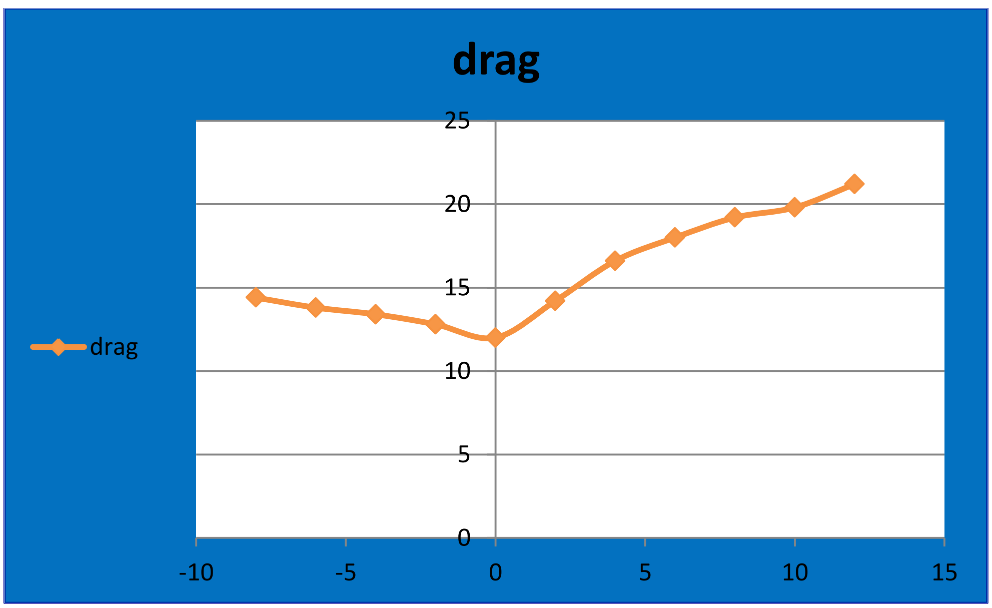


Figure 4-2: drag in position 8

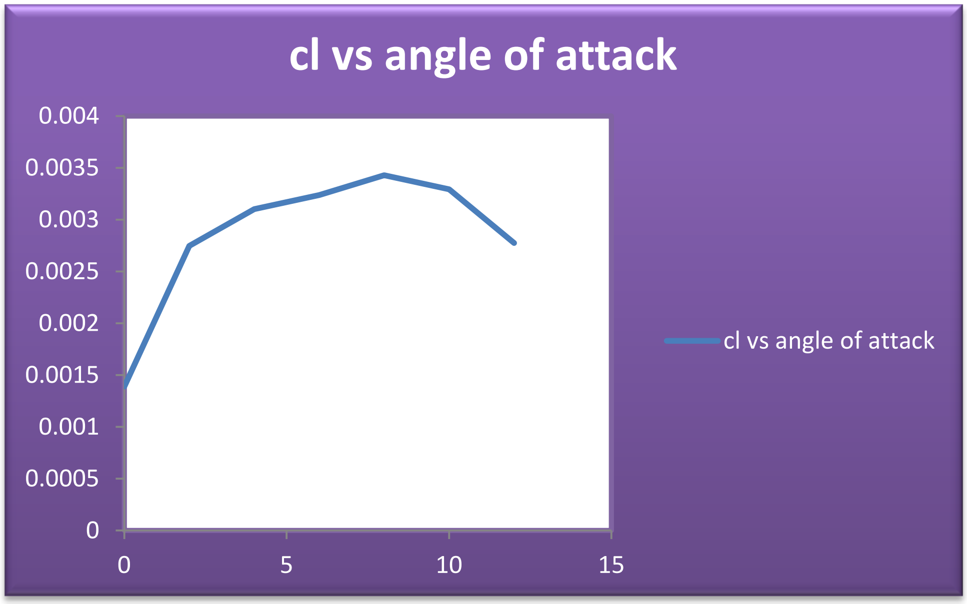


Figure 4-3: cl in position 8.

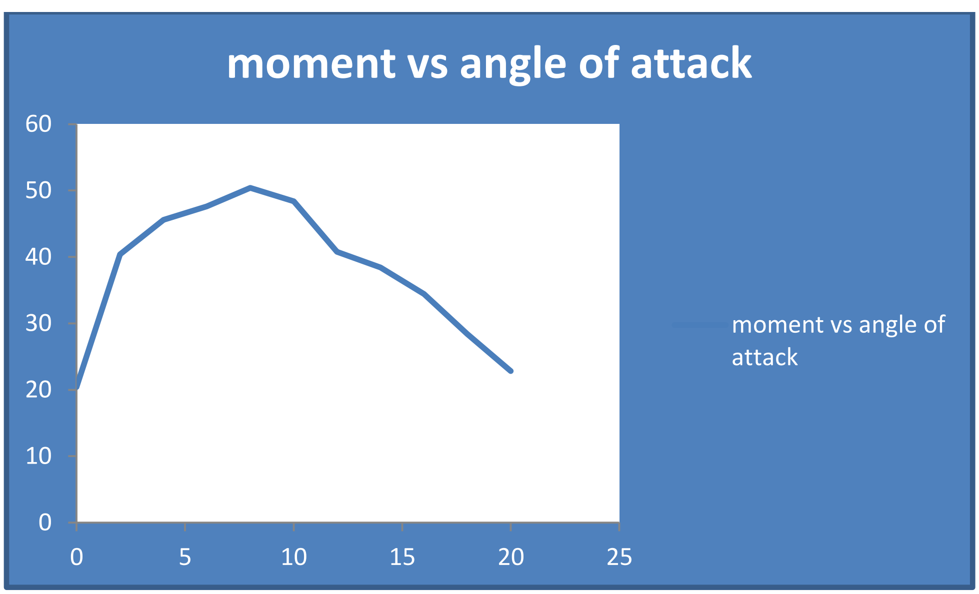


Figure 4-4: moment in position 8.

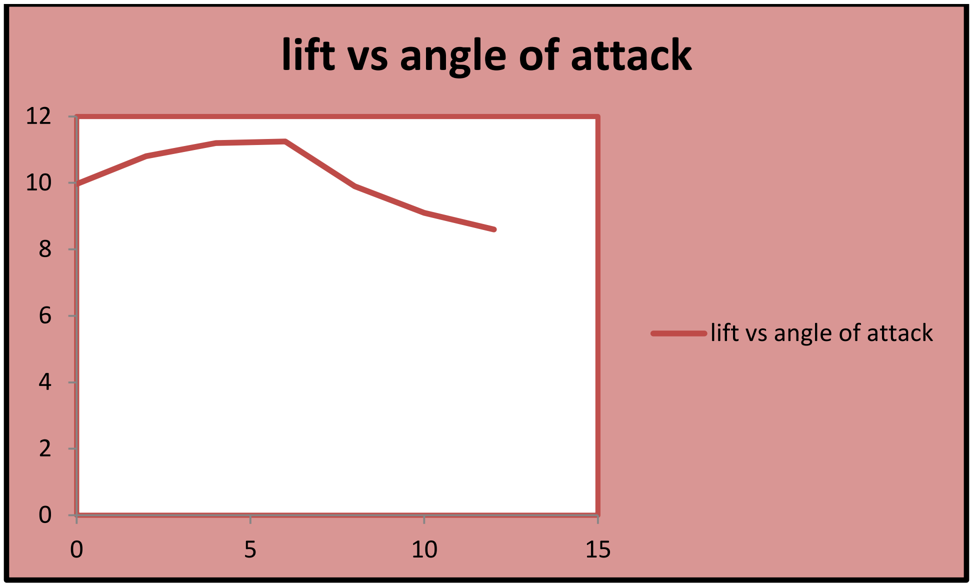


Figure 4-5: lift in position 11.

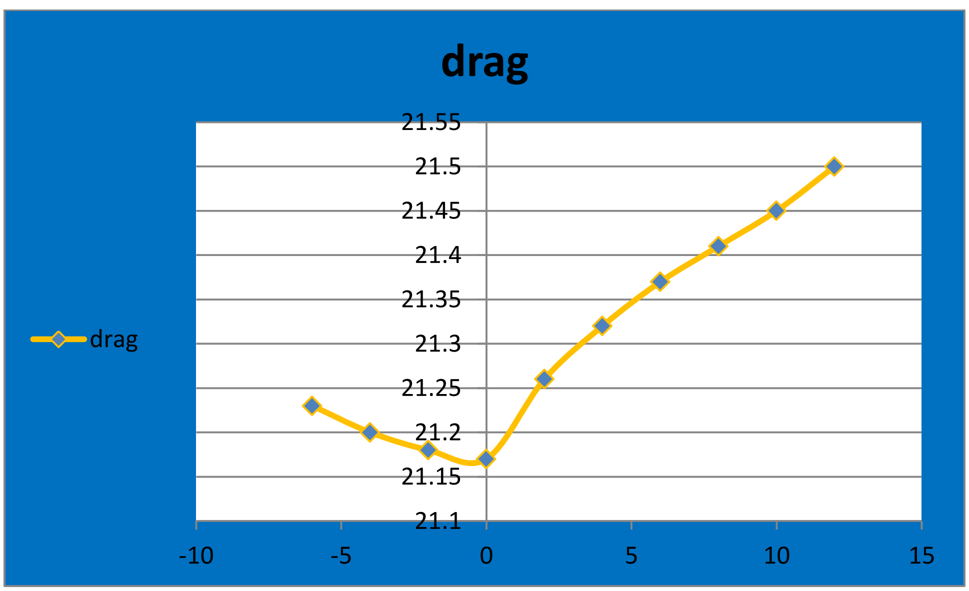


Figure 4-6: Drag in position 11.

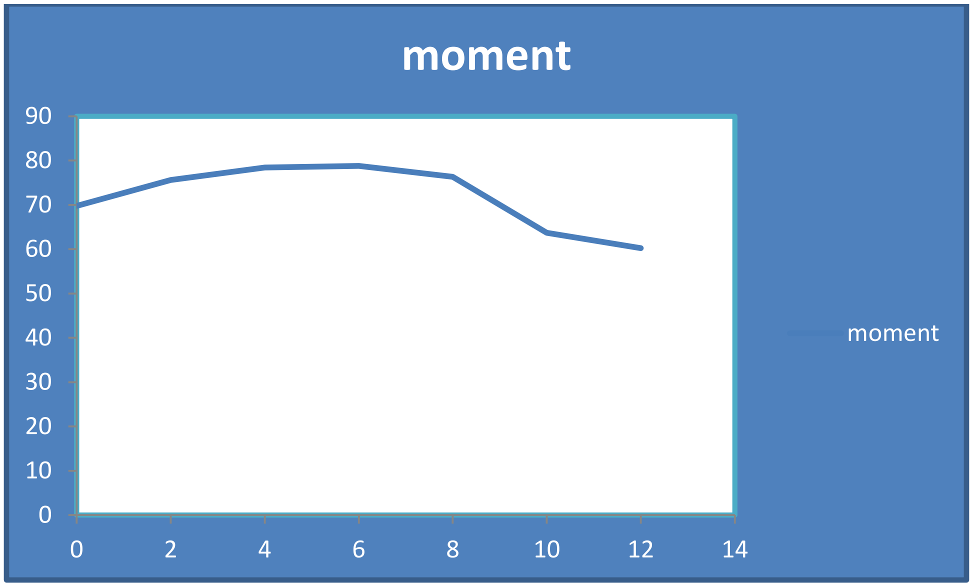


Figure 4-7: moment in position 11.

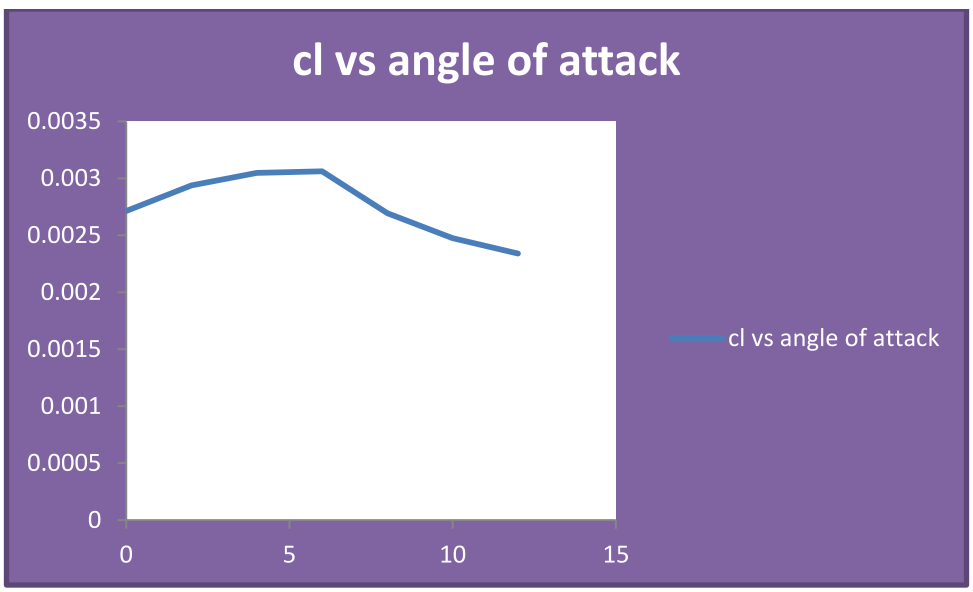


Figure 4-8: cl in position 11.

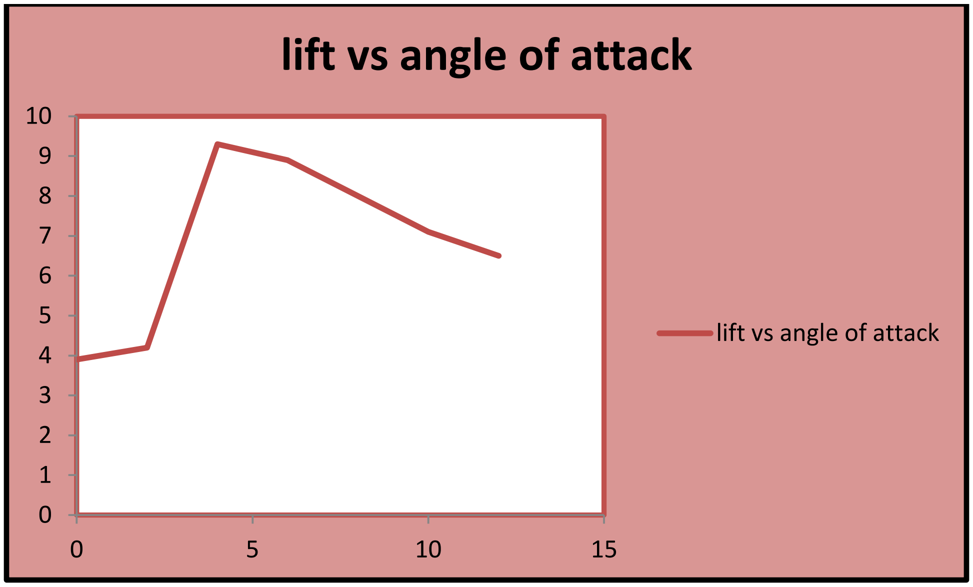


Figure 4-9: lift in position 13.

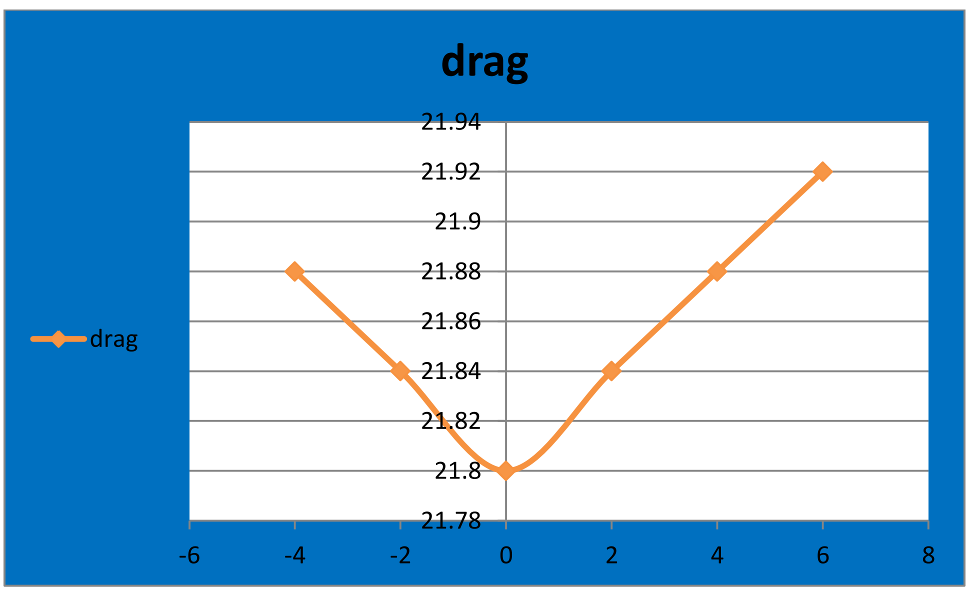


Figure 4-10: drag in position 13

.

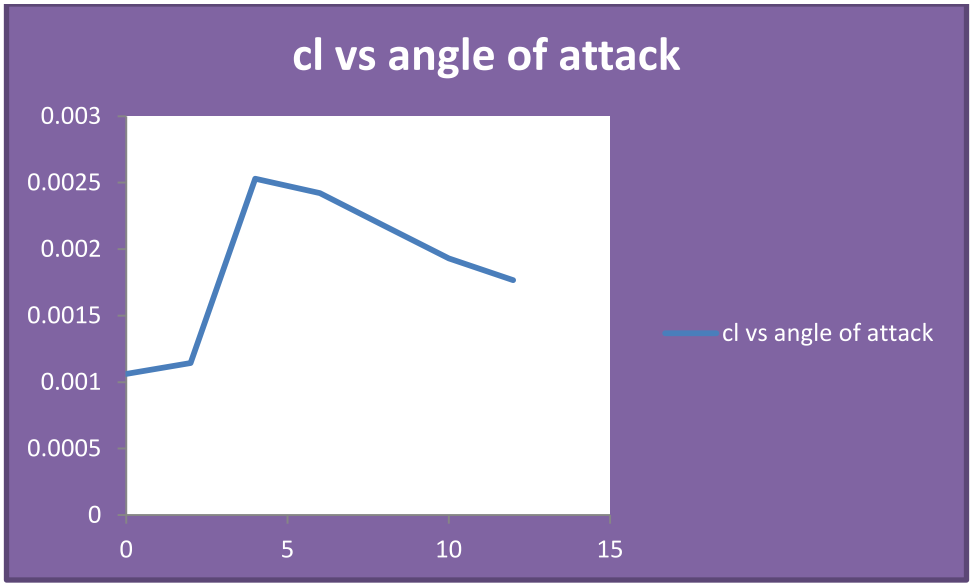


Figure: 4-11cl in position 13.

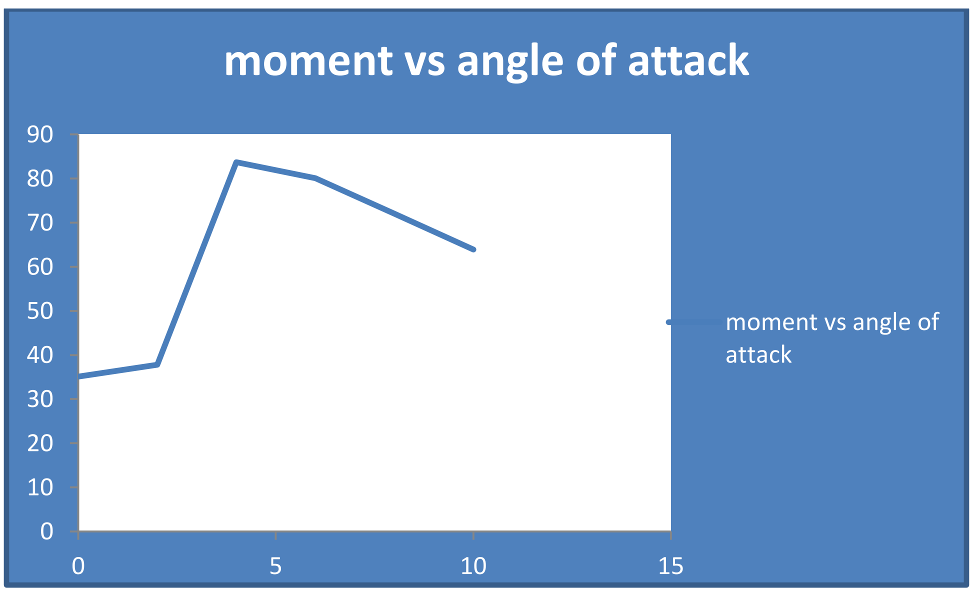


Figure 4-12 : moment in position 13.

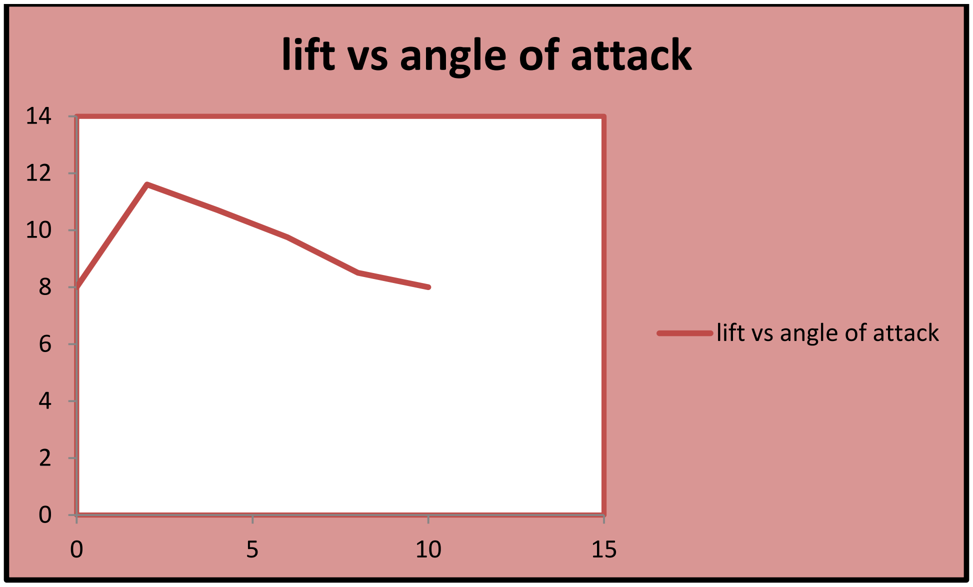


Figure 4-13: lift without load.

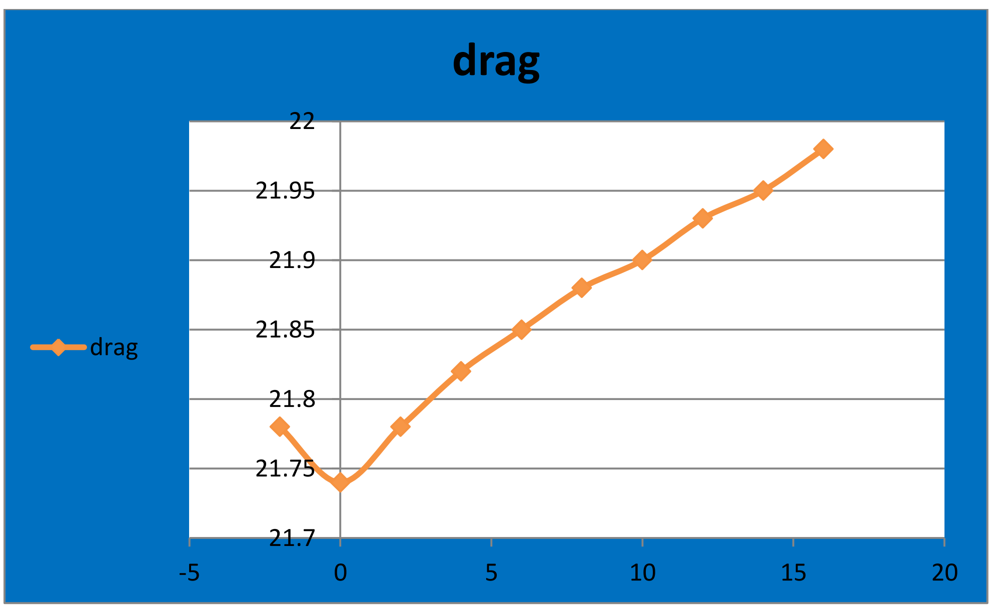


Figure 4-14: drag without load.

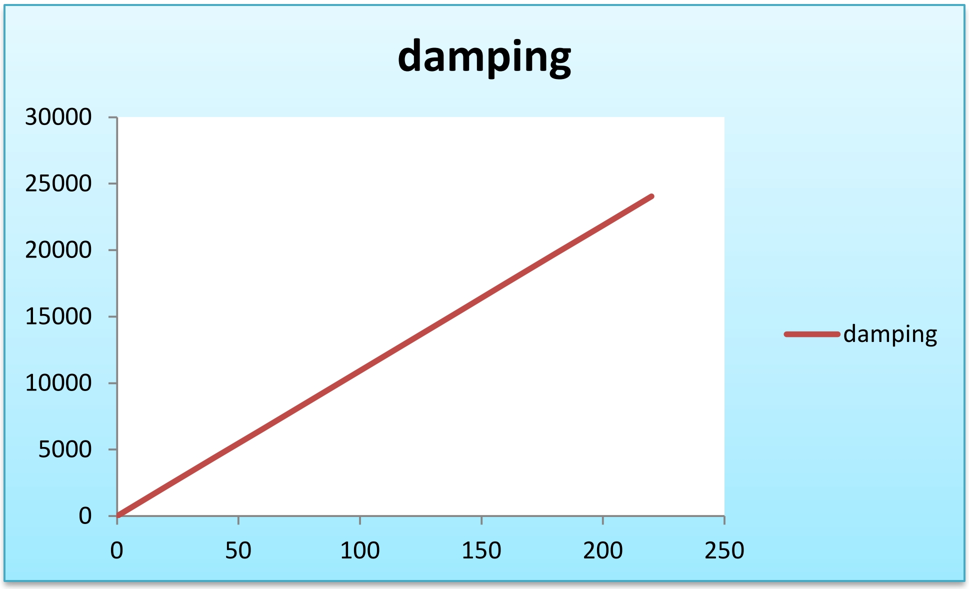


Figure 4-15: damping without loads.

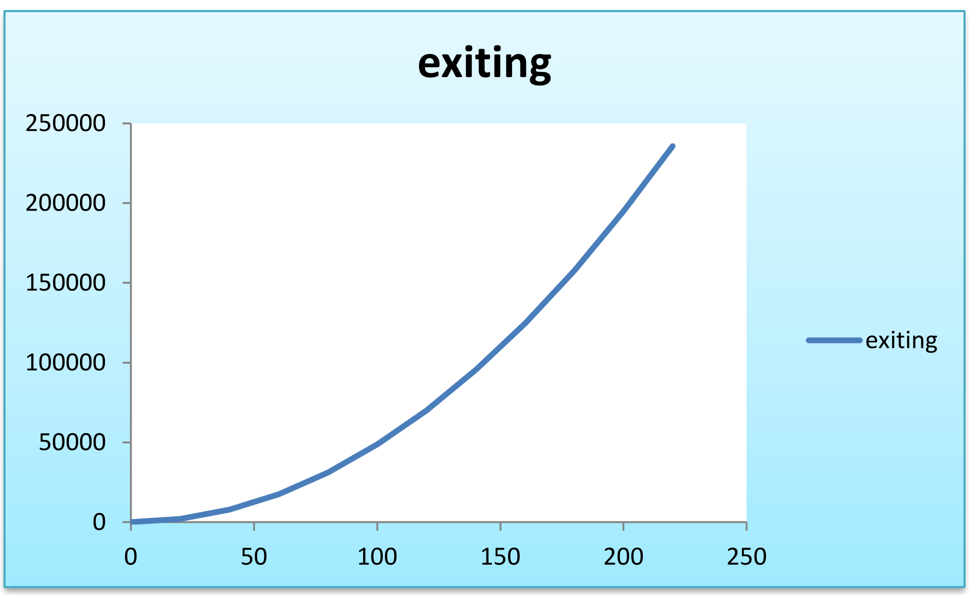


Figure 4-16: damping without load

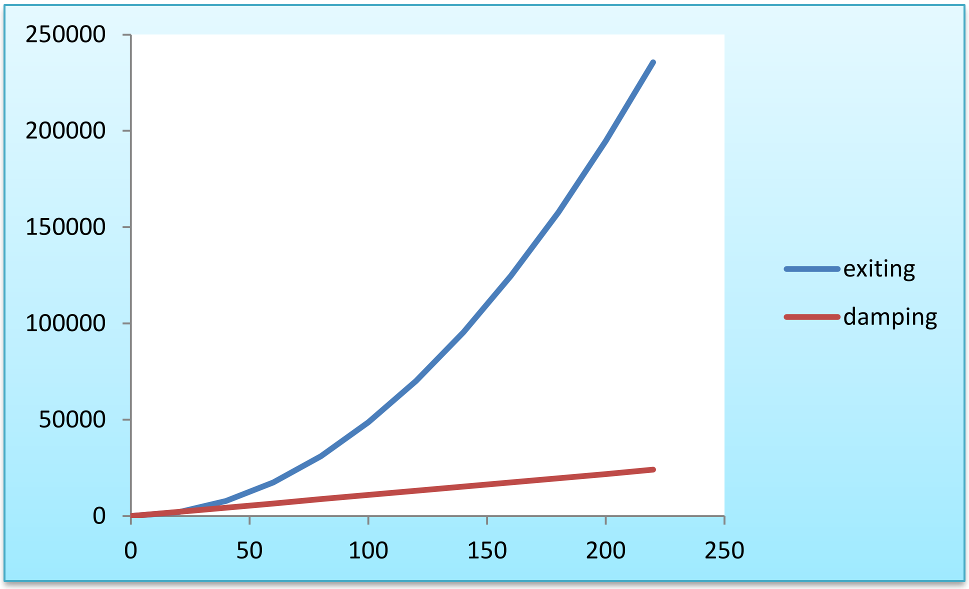


Figure 4-17:Figure4.17: damping and exiting without load.

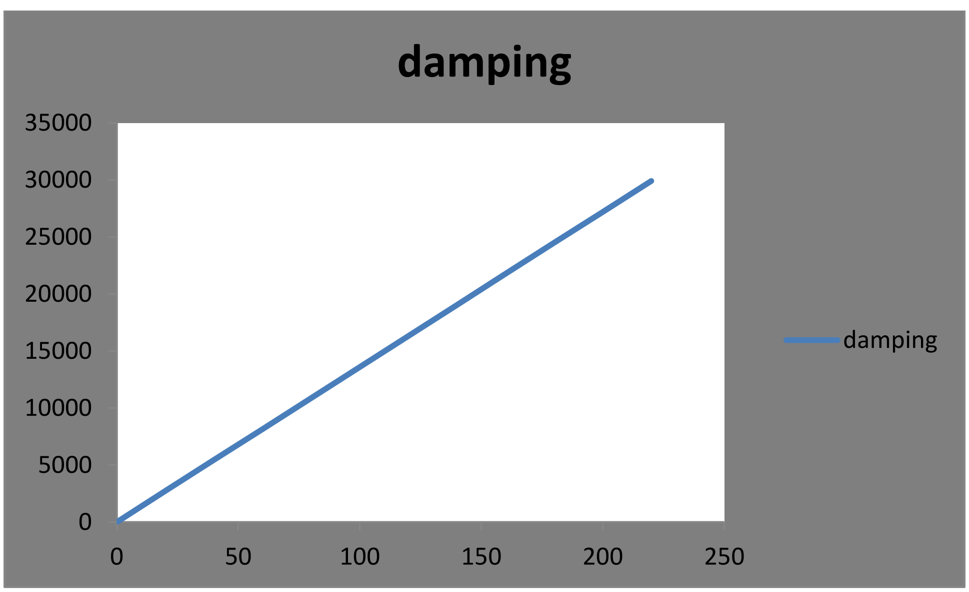


Figure 4-18: damping with load 8.

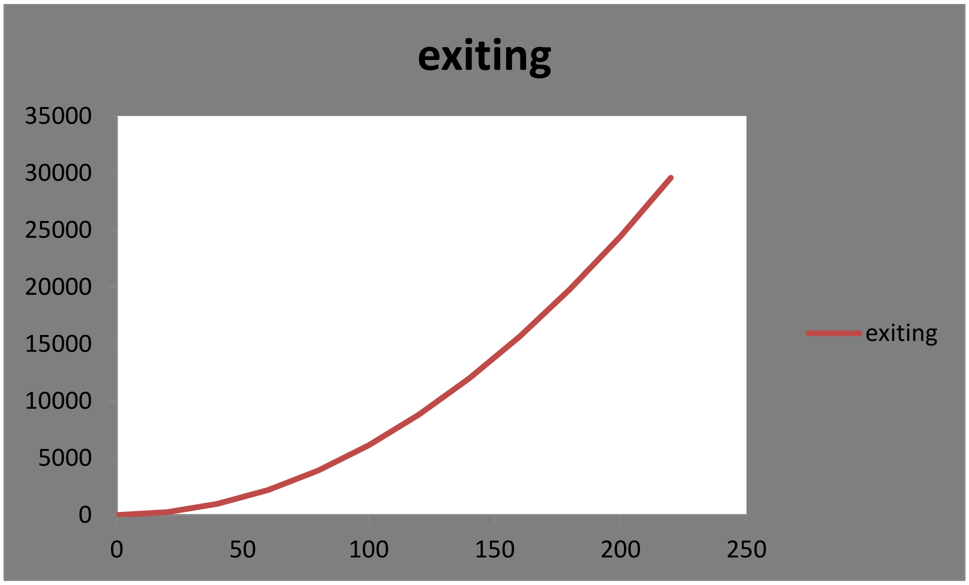


Figure 4-19 : exiting with load 8

.

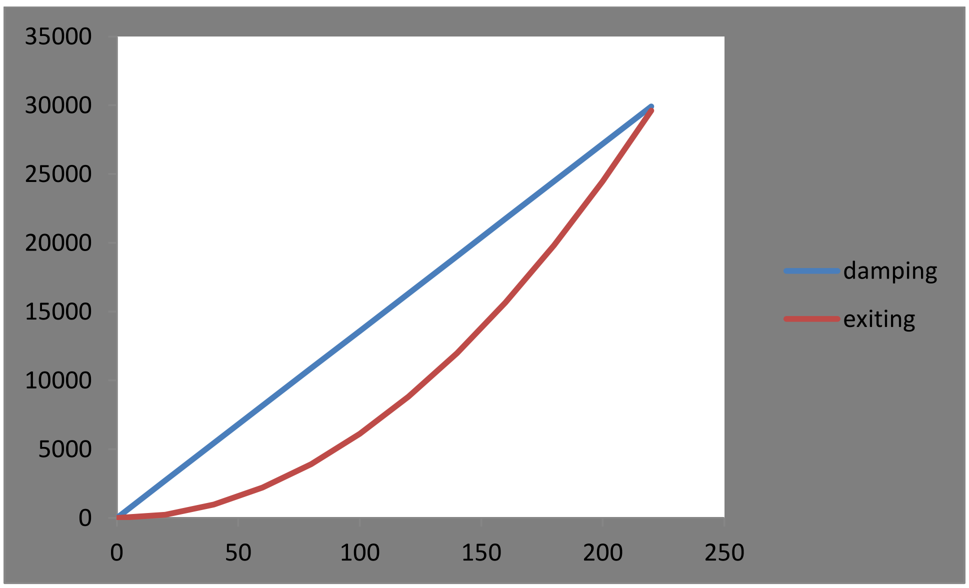


Figure 4-20: damping and exiting with load 8

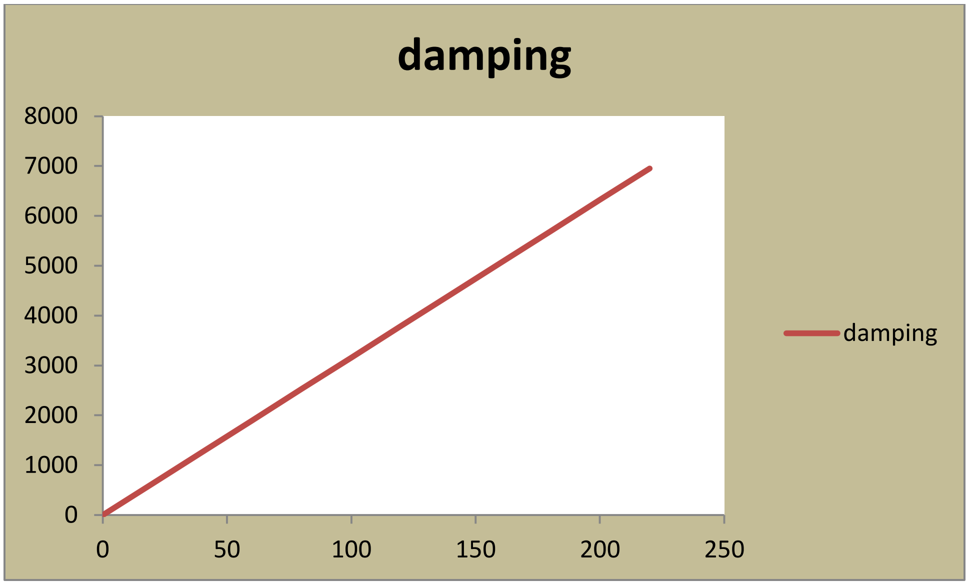


Figure 4-21: damping with load 11.

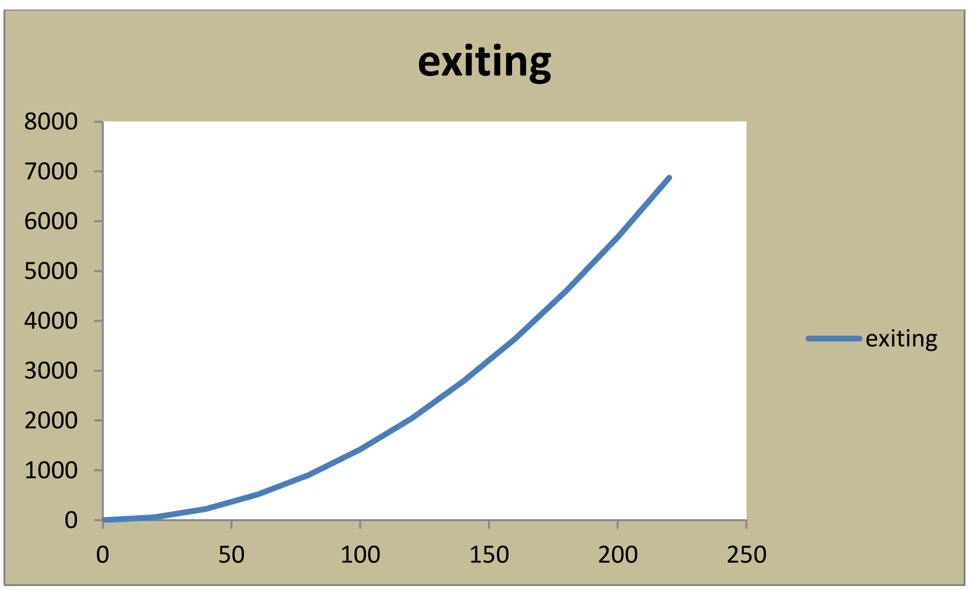


Figure 4-22: exiting with load 11.

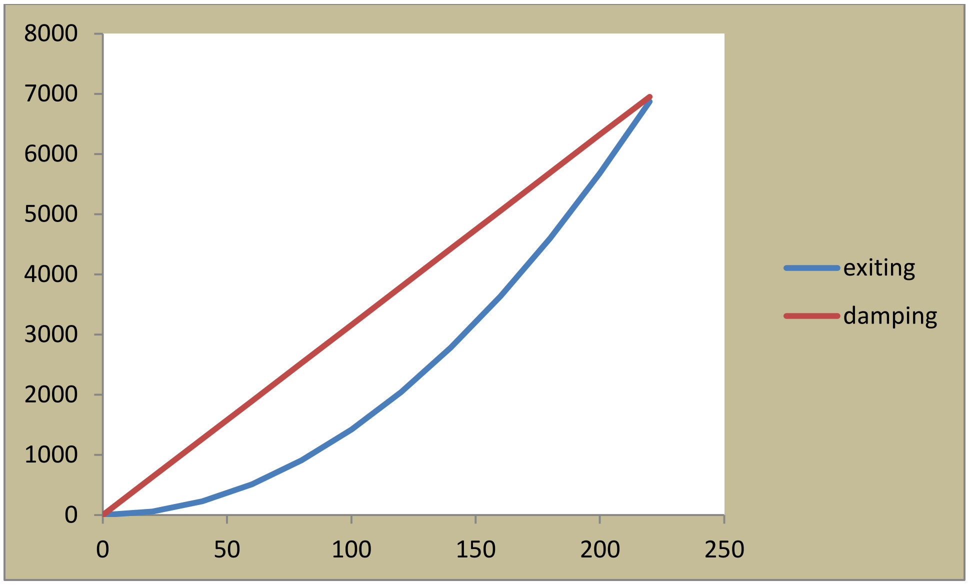


Figure 4-23damping and exiting with load 11.

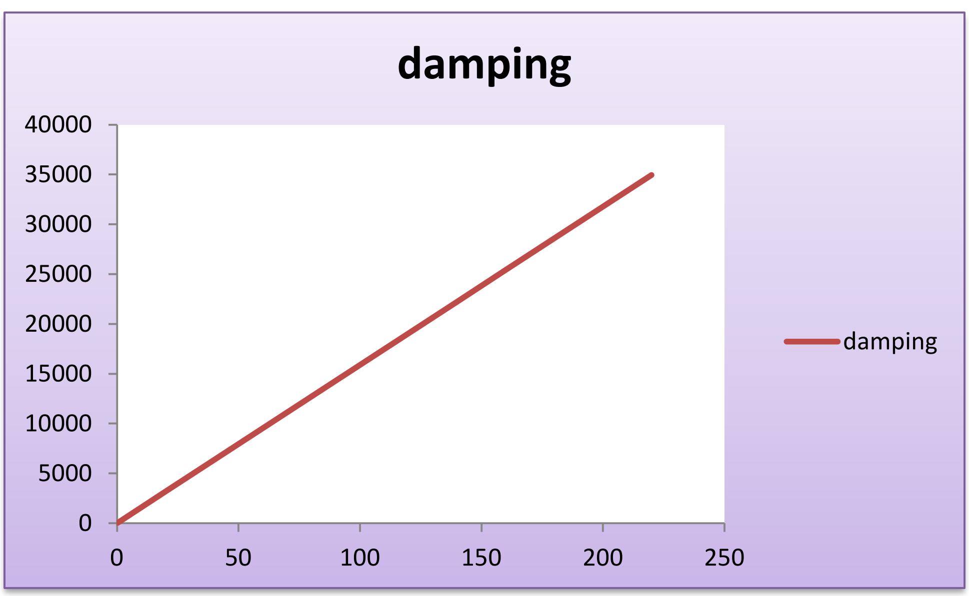


Figure 4-24: damping with load 13

.

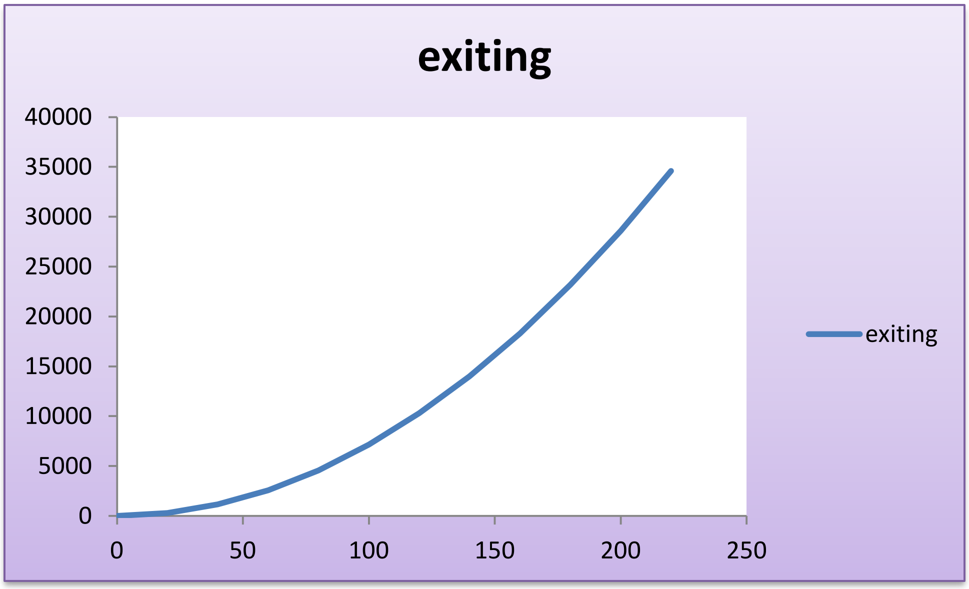


Figure 4-25: exiting with load 13.

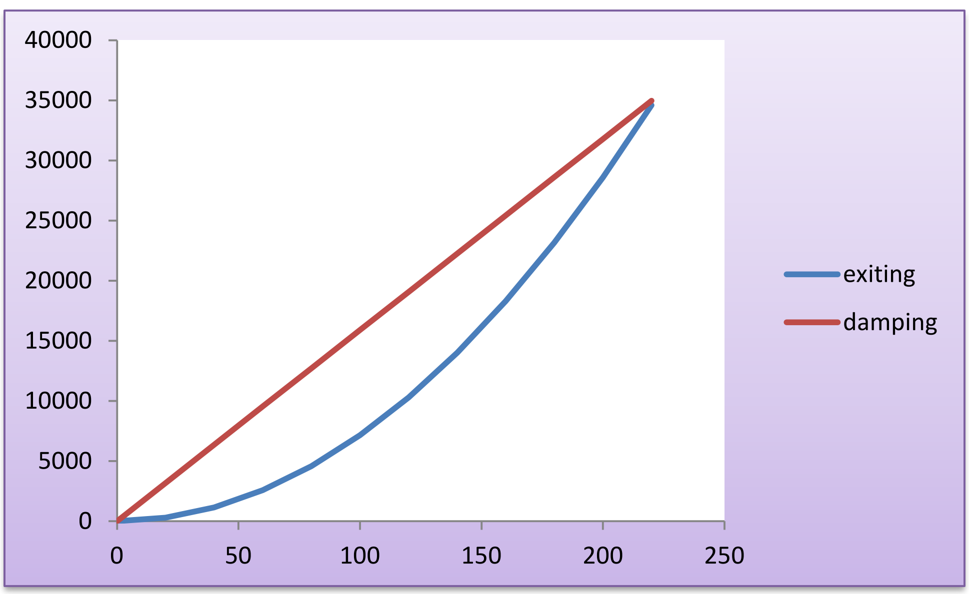


Figure 4-26: damping and exiting with load 13.

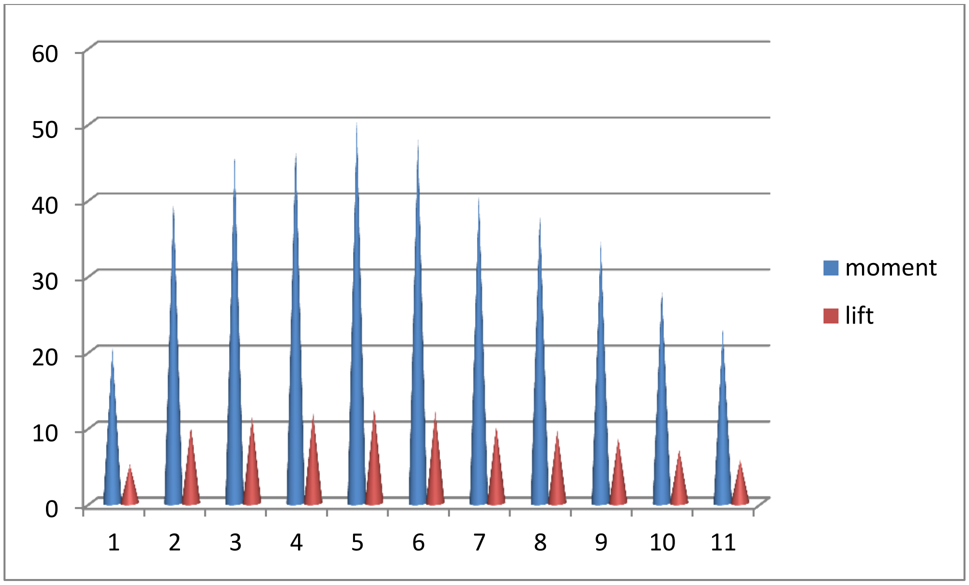


Figure 4-27 : relation between moment and lift in position 8.

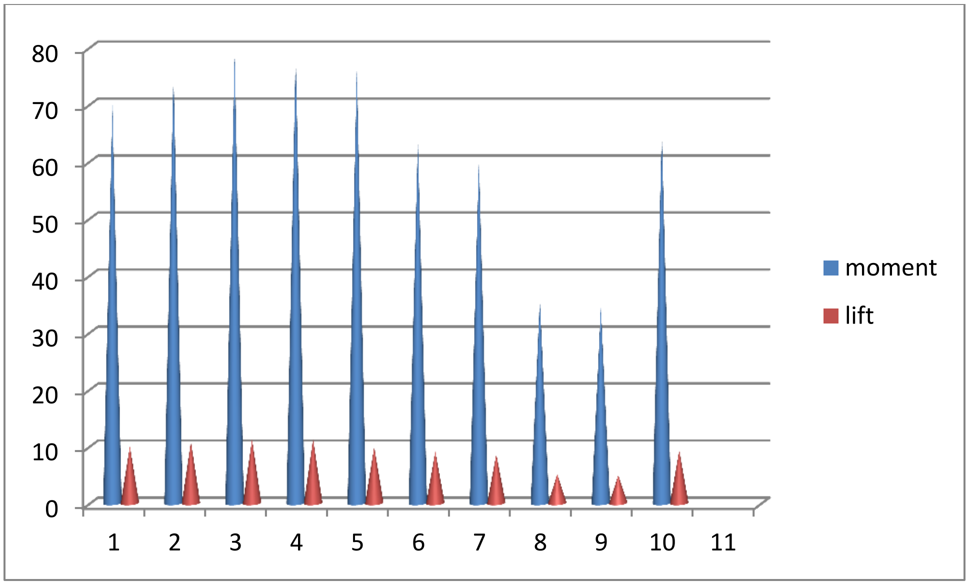


Figure 4-28: relation between moment and lift in position 11.

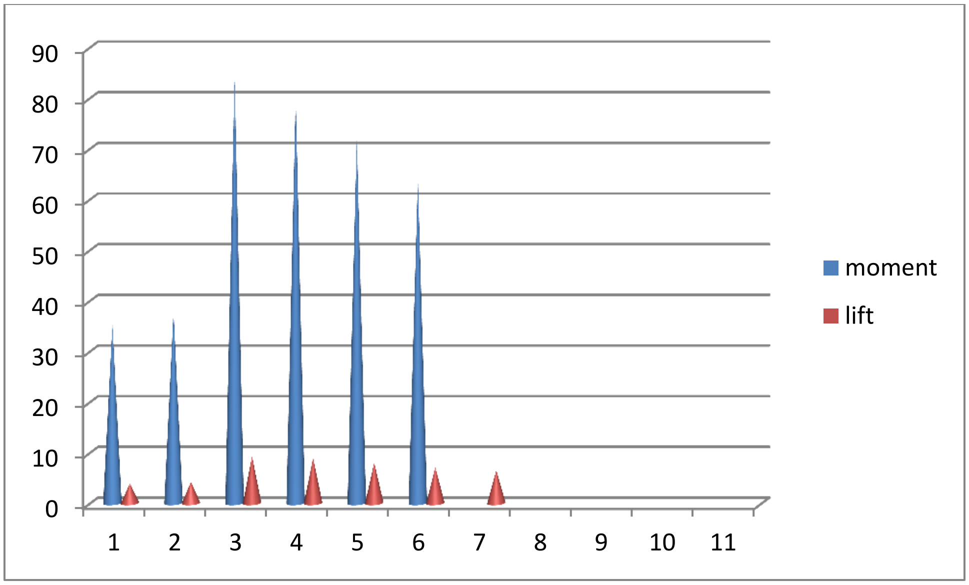


Figure 4-29: relation between moment and lift in position 13.

4.2 Tables:

Table 1; forces and moment at position 8

Angle	lift	moment	cl
0	5.1	20.4	0.001387
2	10.1	40.4	0.002747
4	11.4	45.6	0.003101
6	11.9	47.6	0.003237
8	12.6	50.4	0.003427
10	12.1	48.4	0.003291
12	10.2	40.8	0.002774

Table 1.1

angles	drag
-8	14.4
-6	13.8
-4	13.4
-2	12.8
0	12
2	14.2
4	16.6
6	18
8	19.2
10	19.8
12	21.2

Table 2

Table 2: forces and moment at position 11

angle	lift	moment	cl
0	9.97	69.79	0.002712
2	10.8	75.6	0.002938
4	11.2	78.4	0.003046
6	11.25	78.75	0.00306
8	9.9	76.3	0.002693
10	9.1	63.7	0.002475
12	8.6	60.2	0.002339

Table 2.2

angls	drag
-6	21.23
-4	21.2
-2	21.18
0	21.17
2	21.26
4	21.32
6	21.37
8	21.41
10	21.45
12	21.5

Table 3

Table 3: forces and moment at position 13

angle	lift	moment	Cl
0	3.9	35.1	0.001061
2	4.2	37.8	0.001142
4	9.3	83.7	0.00253
6	8.9	80.1	0.002421
8	8	72	0.002176
10	7.1	63.9	0.001931
12	6.5		0.001768

table 3.3

angles	drag
-4	21.88
-2	21.84
0	21.8
2	21.84
4	21.88
6	21.92

Table 4

Table 4: forces at without load

angle	lift
0	8
2	11.6
4	10.7
6	9.75
8	8.5
10	8
12	7.8

table 4.4

angles	drag
-2	21.78
0	21.74
2	21.78
4	21.82
6	21.85
8	21.88
10	21.9
12	21.93
14	21.95
16	21.98

U	damping	exiting	U^2
0	0	0	0
20	2719.5	244.72	400
40	5439	978.88	1600
60	8158.5	2202.48	3600
80	10878	3915.52	6400
100	13597.5	6118	10000
120	16317	8809.92	14400
140	19036.5	11991.28	19600
160	21756	15662.08	25600
180	24475.5	19822.32	32400
200	27195	24472	40000
220	29914.5	29611.12	48400

Table 5

Table 6

U	damping	exiting	U^2
0	0	0	0
20	632	56.8	400
40	1264	227.2	1600
60	1896	511.2	3600
80	2528	908.8	6400
100	3160	1420	10000
120	3792	2044.8	14400
140	4424	2783.2	19600
160	5056	3635.2	25600
180	5688	4600.8	32400
200	6320	5680	40000
220	6952	6872.8	48400

U	damping	exiting	U^2
0	0	0	0
20	3178	285.912	400
40	6356	1143.648	1600
60	9534	2573.208	3600
80	12712	4574.592	6400
100	15890	7147.8	10000
120	19068	10292.83	14400
140	22246	14009.69	19600
160	25424	18298.37	25600
180	28602	23158.87	32400
200	31780	28591.2	40000
220	34958	34595.35	48400

Table 7

Table 8

U	damping	exiting	U^2
0	0	0	0
20	2186.625	1947.748	400
40	4373.25	7790.992	1600
60	6559.875	17529.73	3600
80	8746.5	31163.97	6400
100	10933.13	48693.7	10000
120	13119.75	70118.93	14400
140	15306.38	95439.65	19600
160	17493	124655.9	25600
180	19679.63	157767.6	32400
200	21866.25	194774.8	40000
220	24052.88	235677.5	48400

Chapter five

Appendix of picture

5 Chapter five: Appendix of picture

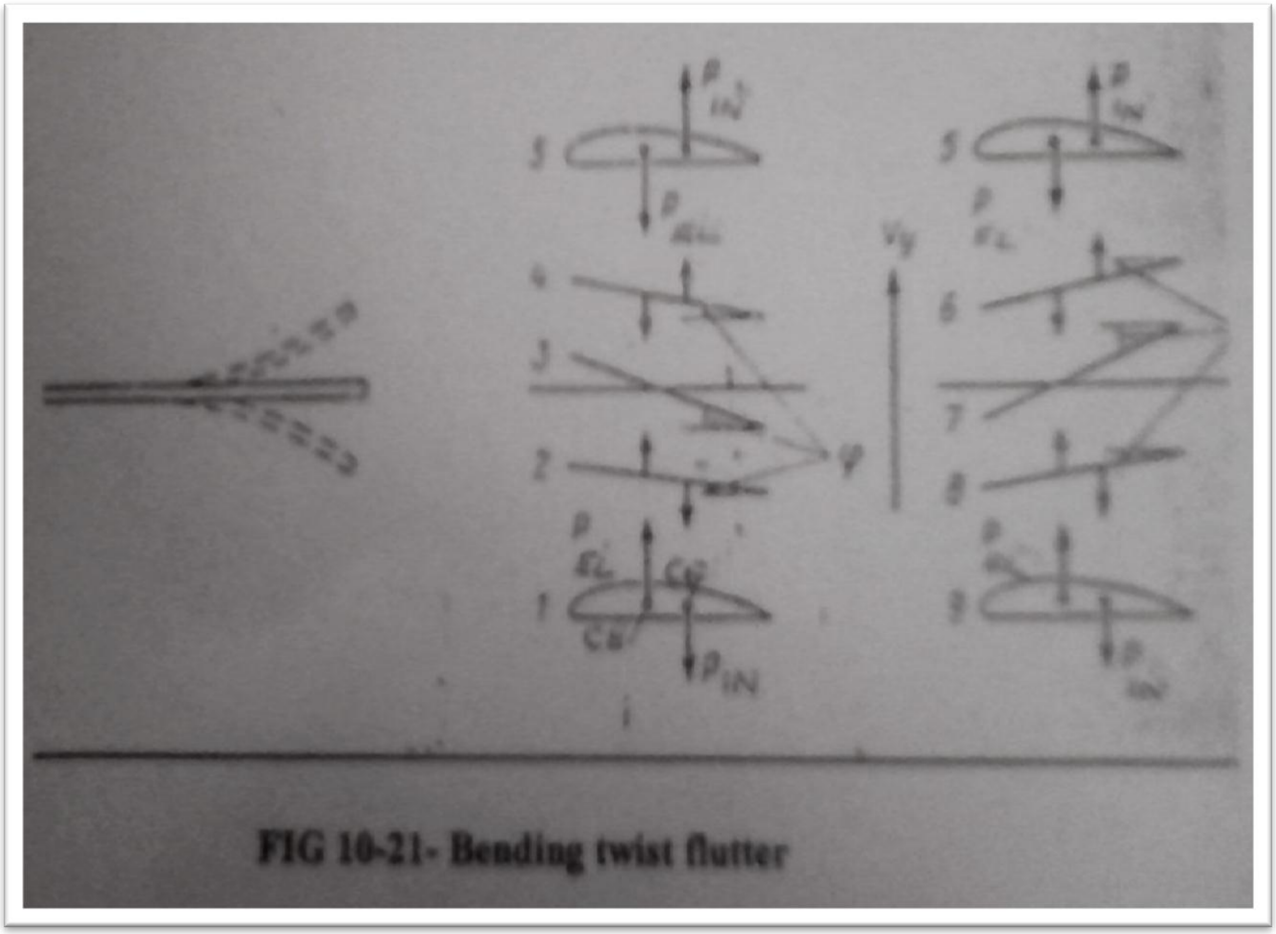


FIG 10-21- Bending twist flutter

Figure 5-1 bending twist flutter

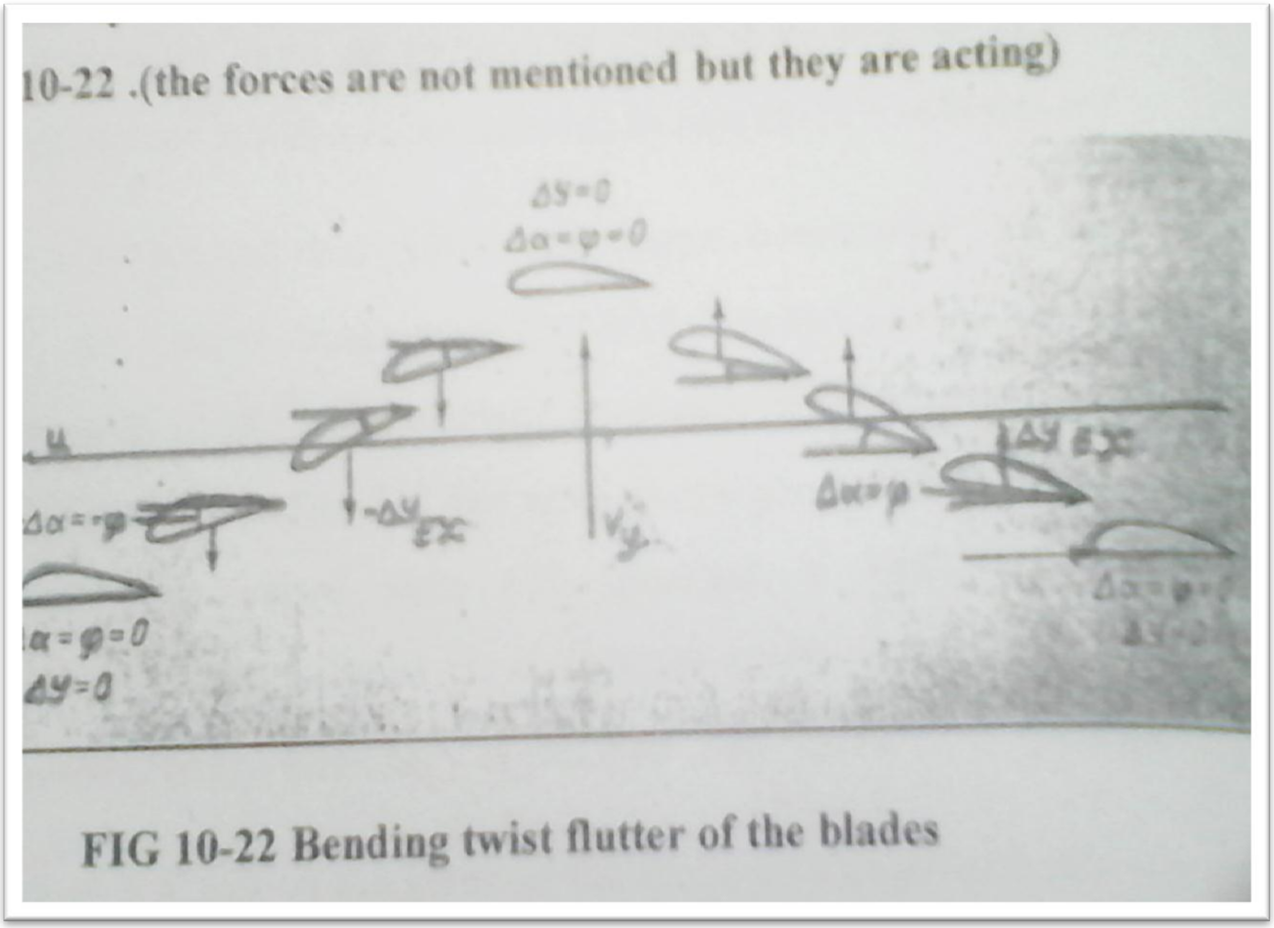


Figure 5-2: Bending twist flutter of blade

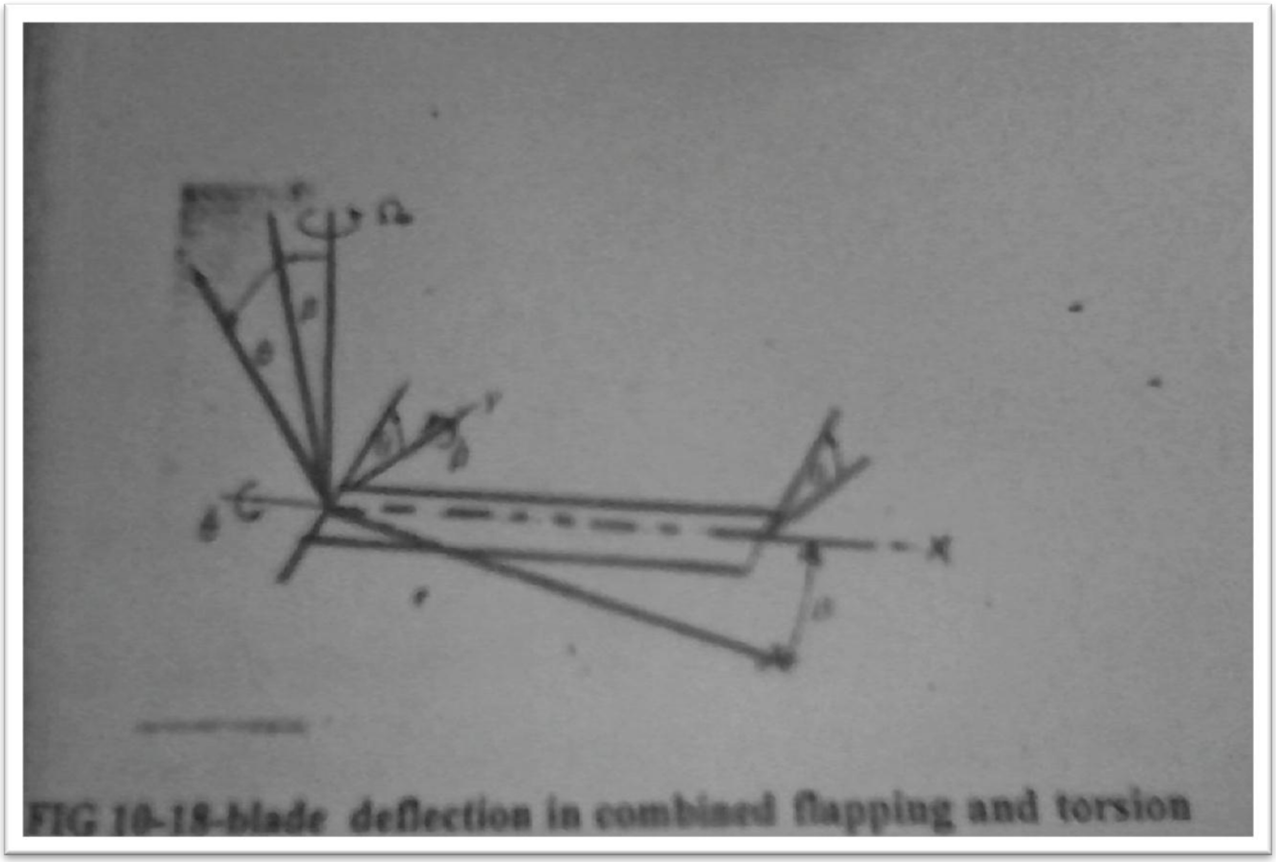


Figure 5-3: Blade deflection in flapping and torsion

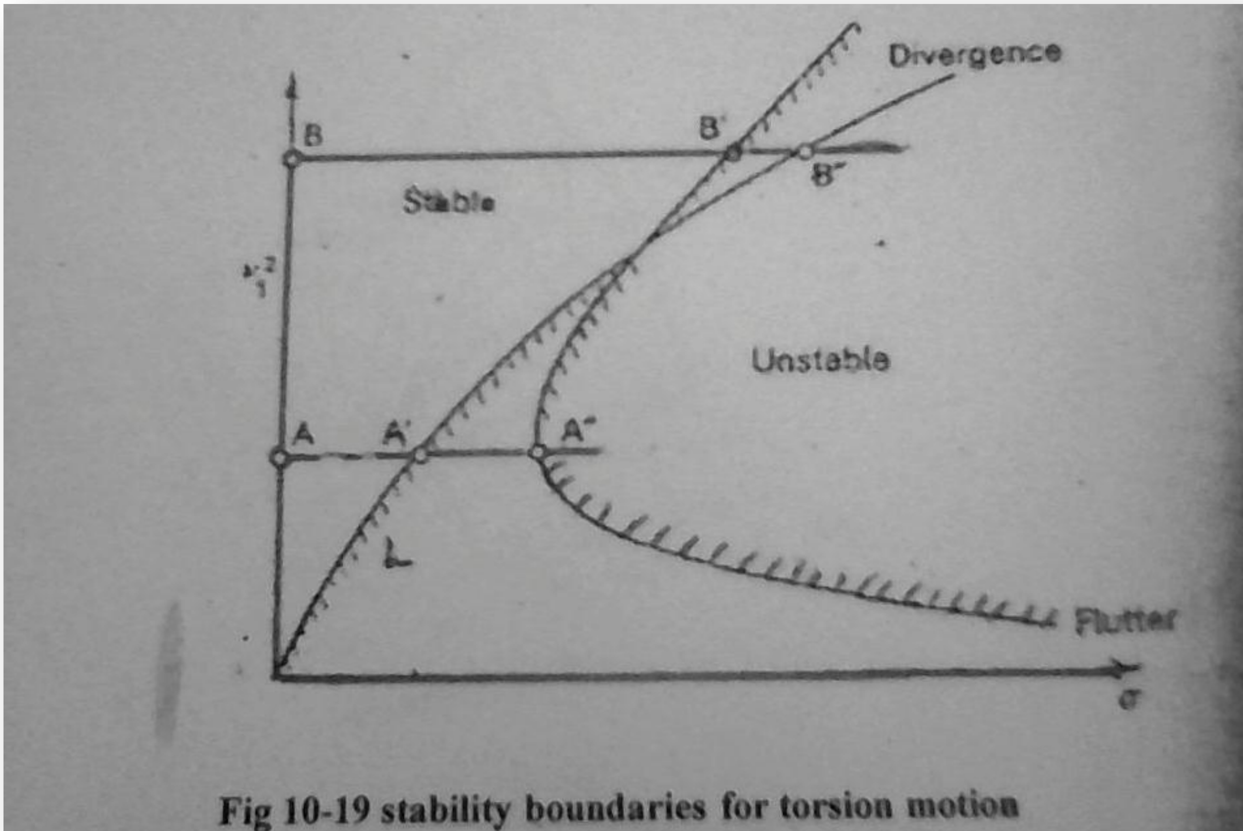


Fig 10-19 stability boundaries for torsion motion

Figure 5-4: Stability boundaries for torsion motion

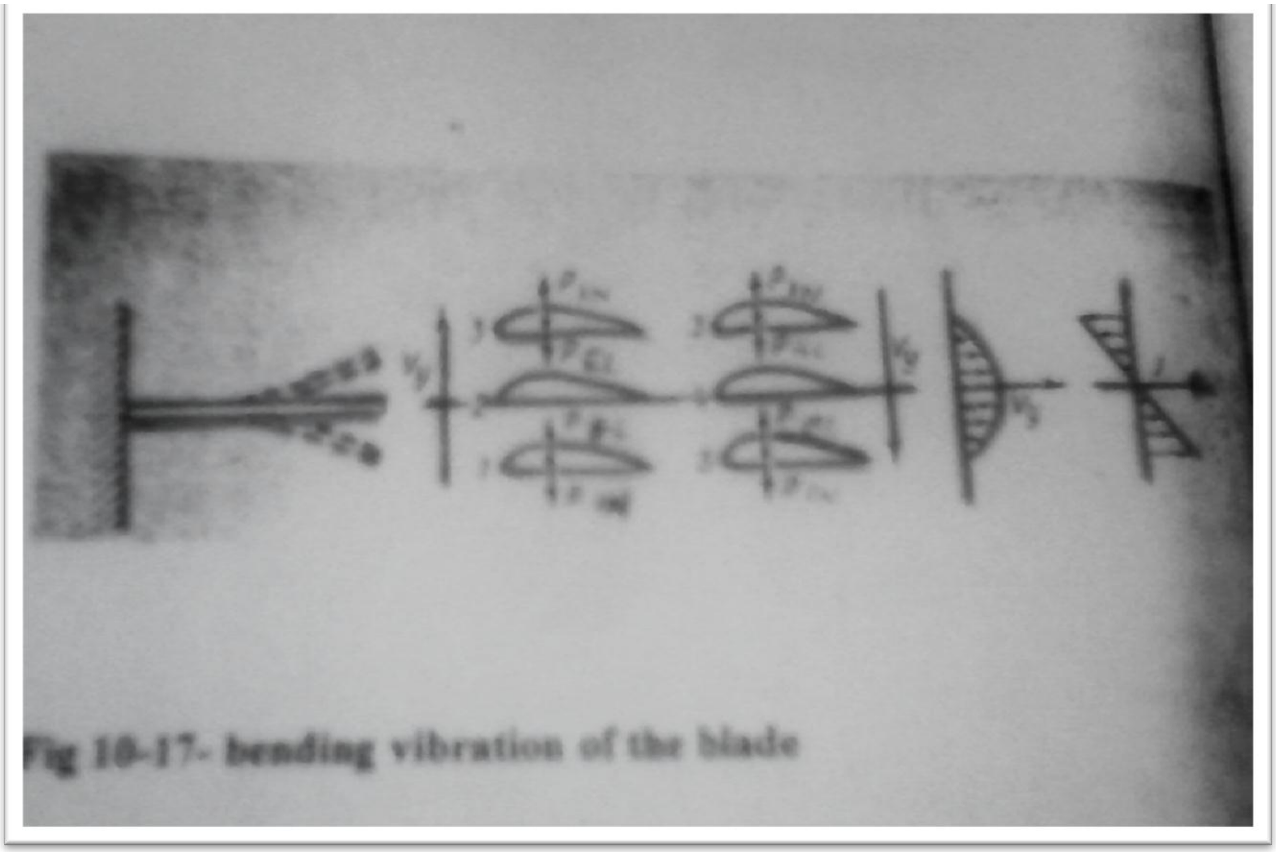


Figure 5-5: Bending vibration of the blade

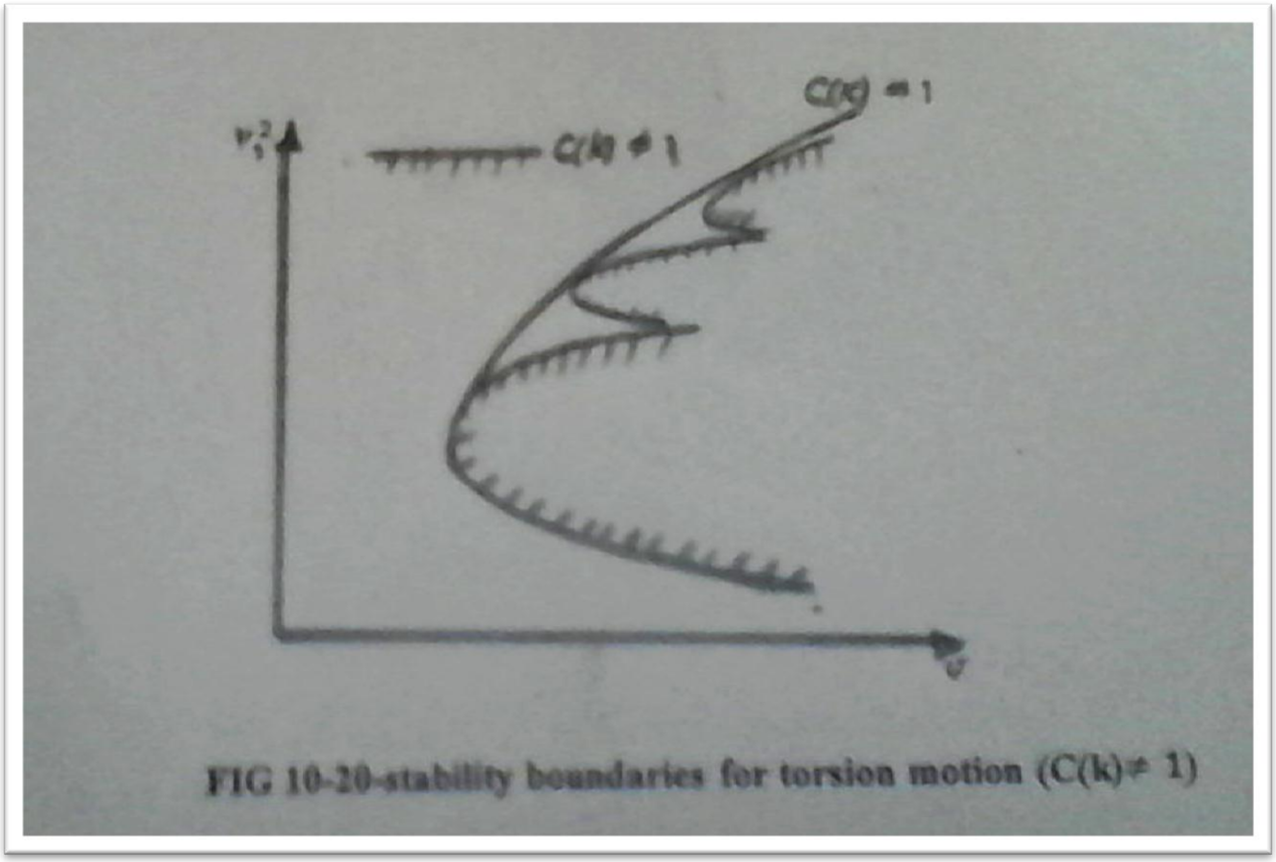


FIG 10-20-stability boundaries for torsion motion (C(k)≠ 1)

Figure 5-6: Stability boundaries for torsion motion (ck≠1)

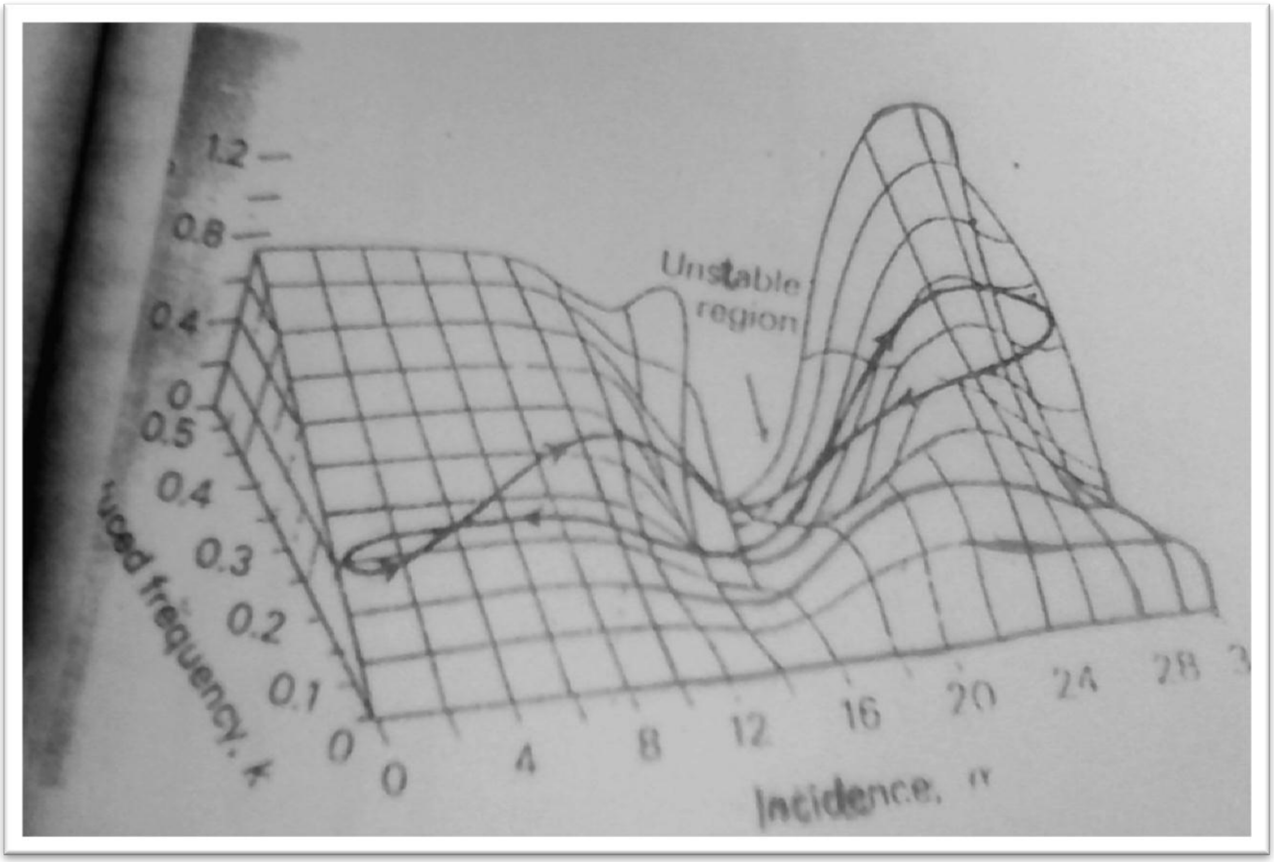


Figure 5-7: Contours of blade torsion damping

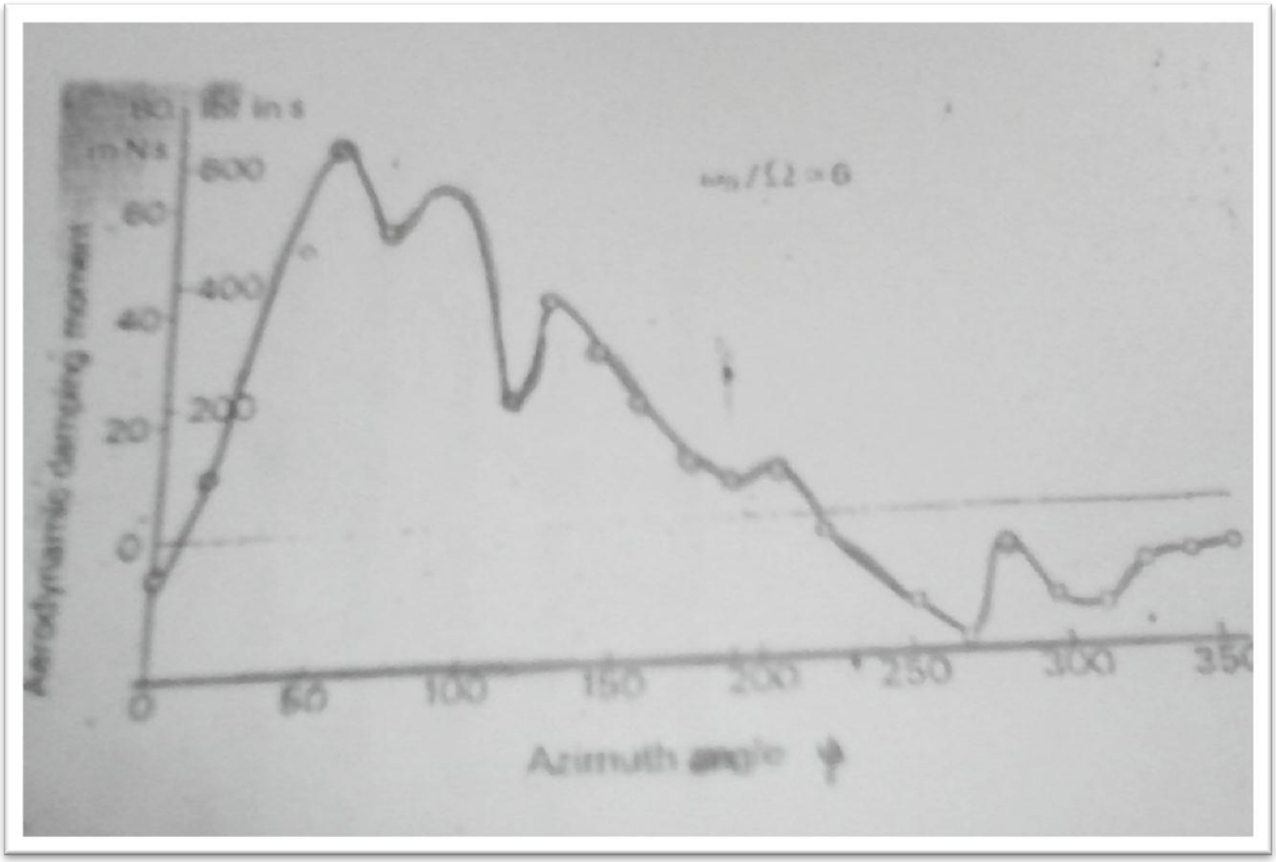


Figure 5-8: Aerodynamic torsion damping vs. azimuth

Figure 5-9: the position of load (11)

Figure 5-10: explain the angle adjustment device

Figure 5-11: the load in position (13) and influence of air to the filaments at certain angle.

Figure 5-12: the influence of air to the filaments at certain angle in position (8)

Figure 5-13: the load in position (8)

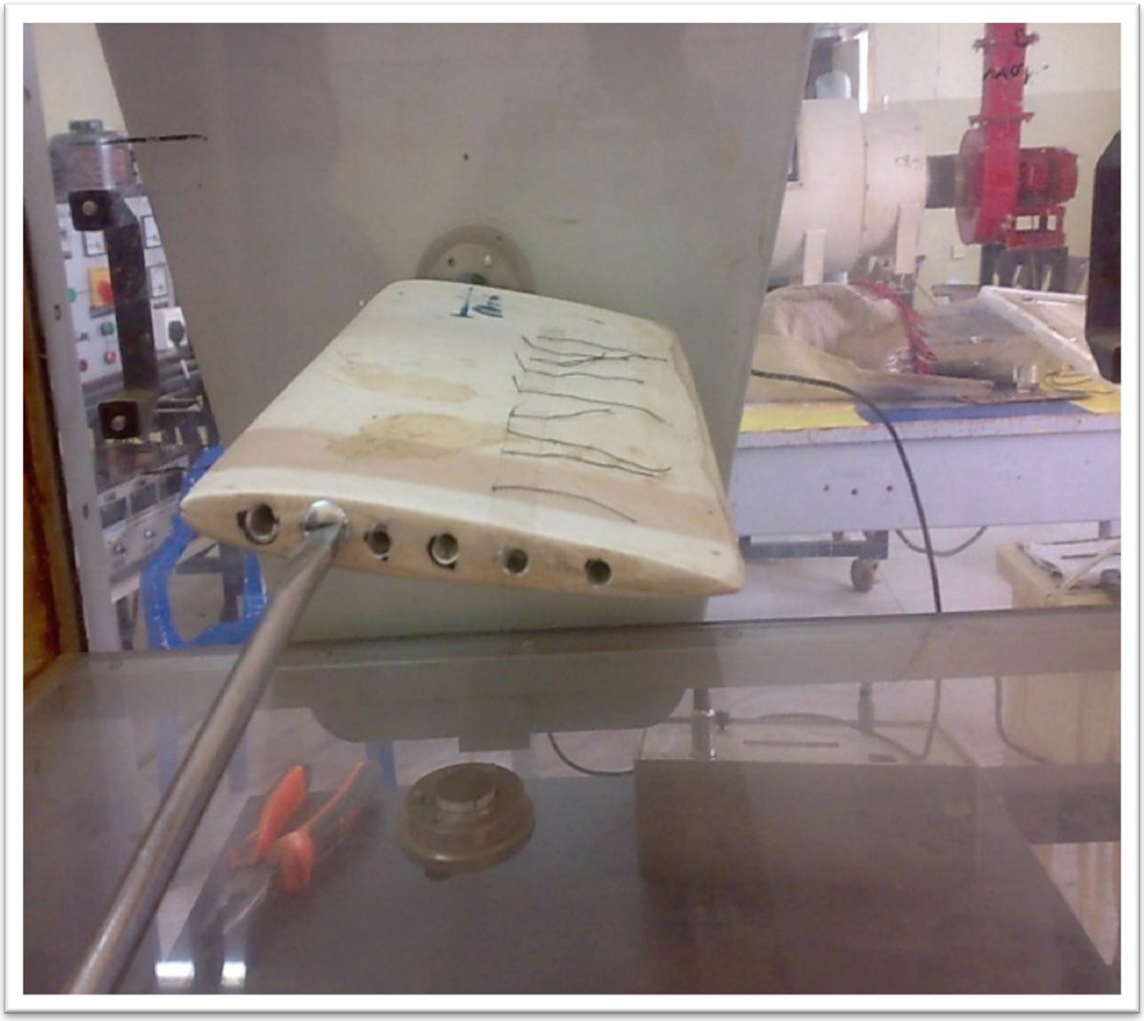


Figure 5-14: the shape of airfoil through the fixing

Figure 5-15: the load in position (8) by other side

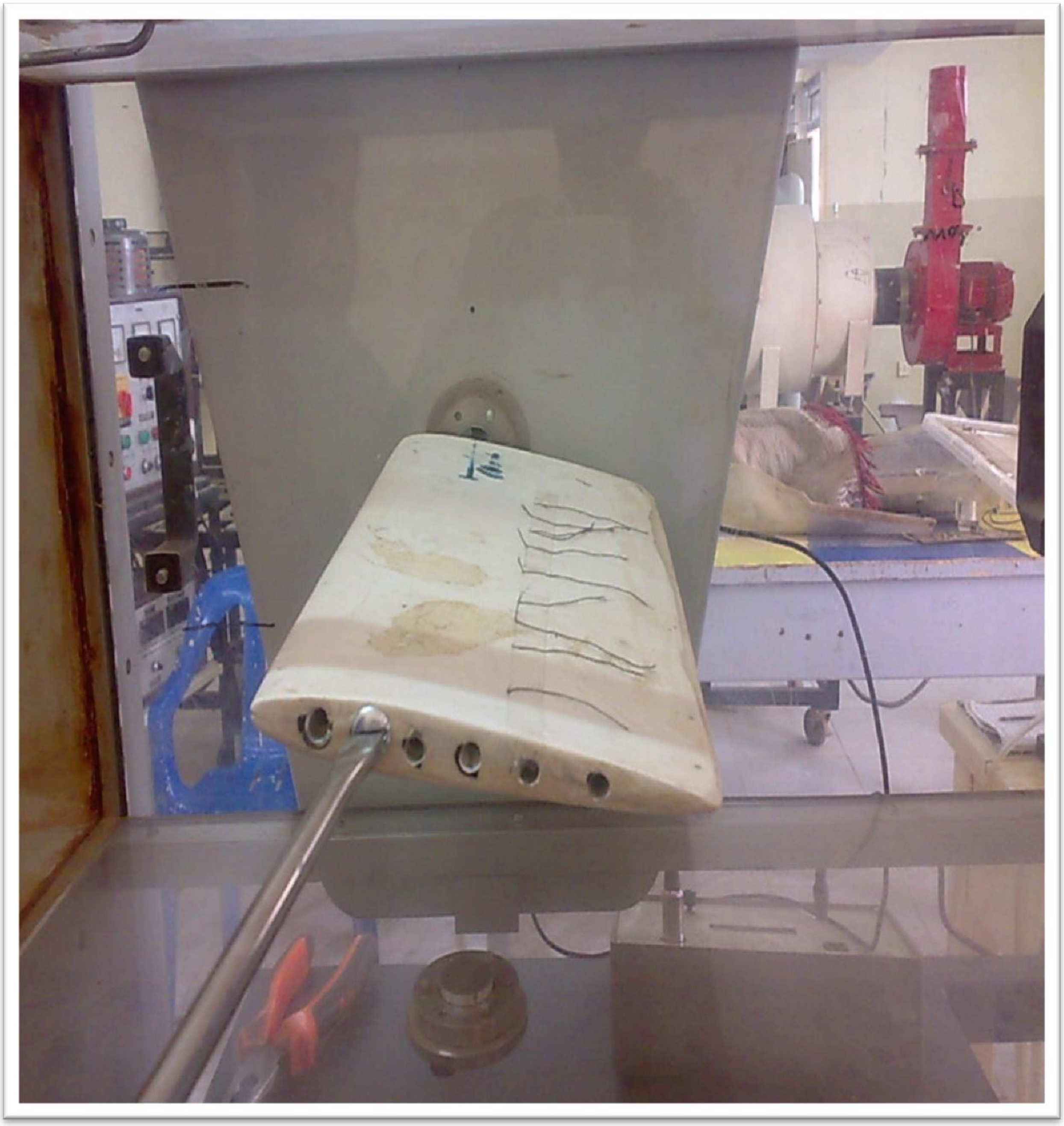


Figure 5-16: the airfoil when fixing without influence of air to the filament

Figure 5-17: helicopter (mi8)

Chapter six

Results and discussions

6 Chapter six: Result and discussion

$$\Delta y_{ex} = \Delta y \rho A_b \frac{u^2}{2}$$

$$\Delta y_{ex8} = 0.6118u^2$$

$$\Delta y_{ex11} = 0.66329u^2$$

$$\Delta y_{ex13} = 0.71478u^2$$

$$\Delta y_{exwithoutload} = 4.86937u^2$$

The result of the above values of exiting force and without load is the biggest one.

$$\Delta y_{dam} = a v_y A_b \rho \frac{u}{2}$$

$$\Delta y_{dam8} = 135.975u$$

$$\Delta y_{dam11} = 146.985u$$

$$\Delta y_{dam13} = 158.9u$$

$$\Delta y_{dam} without = 109.33u$$

The damping force at blade without load is less than in leading and trailing edge.

6.1 Experiment result and discussion:-

The values of lift and drag forces was written in the tables all the readings were taken from low speed subsonic wind tunnel

Also the moment values calculated by multiplied lift force with arm (the distance from the C.G), the moment increase proportional to increase in distance from center of gravity show in figure (27,28,26) .

The stall in lift force:

At load in location 8, at angle 8

At load in location 11, at angle6

At load in location 13, at angle 4

At without state at angle 2

Note:-

$$c_l = a \propto$$

Or

$$c_l = \frac{2 * g}{\rho * v^2 * A_b}$$

When;

G is the weight

V the speed of air flow with in wind tunnel

ρ *is the density of air equal* 1.225

From these results found that the flutter at without load occurs earliest than in with load.

At positioning load near the leading edge the flutter occur later than farther loads from leading edge.

The results show that the stability is extremely sensitive to position of center of mass therefore Forward movement of the center of mass increases the stability.

Chapter seven

Conclusion, recommendations and future work

7 Chapter seven: Conclusion, recommendations and future work

7.1 Conclusions:-

According to the figures in chapter four we found out when using mass balance method to avoid flutter phenomena the result was not good enough because there was more loads. The low speed subsonic wind tunnel which is used in fact has some errors due to calibration. So the lift and the drag force have and errors which resulted in an accurate reading. We reduced the errors by repeating the experiment more time using average of reading. Besides the exciting force should not exceed the damping force to avoid flutter phenomena.

Also the interaction between damping and exciting forces should be explained the critical rotational speed (to avoid operation in this speed), shown in figures (17, 20, 23, 26) when designing the rotary wing aircrafts it is important to consider flutter phenomena .the tip design should be proper to avoid hearing flutter during maneuver the extreme variation of the environment within flight envelope of aircraft generates several problems facing aero elastic stability of helicopter rotor blade

The effects of the center of gravity position to lift , drag and moment coefficient is that the flutter occurs earlier at further position from the center of gravity .also the drag decrease until the flutter point then suddenly increase .relatively the moment increase to the rise in distance from the (C.G).

The solution of flutter problem for rotary wings air craft inherently more complicated than fixed wing counterpart.

7.2 Recommendation:-

Most of the objectives of this study have been achieved because:

- Flutter is very dangerous vibration, to avoid this blade stiffness at bending and twisting should be small.
- Moreover the fabrication of the blade should be from composite material more than the metallic material.
- Also the wind tunnel should be more accurate, developed calibration, include the moment and the blade should be articulated.
- The center of gravity be in the same line with the stiffness axis
- To avoid the flutter problems :

  By flight measurement, to the pilot using: warning (signals, horn or caution light).

  By redesign the blades according to the recommendation of our research and other researches.
- The rotor hub should be design heavier to withstand the increases of center fugal tensile forces, and additional weight..
- The repair material should be from good matter and must not accumulate on apposition of failure.
- that the airfoil must be suitable to the test section of the wind tunnel,

7.3 Future work:

Our project is good and has generated a new idea; the future helicopter will be able to satisfy the demands without flutter problems.

The influence of aero elastic such as the flutter becomes more necessary to develop or improve new helicopter

Safe and economic helicopter will satisfy the demands of the future design.

To dominate the effects of flutter the airfoil design should be thin, flexible structure or semi rigid made from titanium.

The development of the performance of the aircraft will be achieved by using collateral flexibility of structure to stand the loads without vibration, broken and damping high oscillation .this reverses the rigidity of the structure. The blade must be stiff enough to avoid the effects of flutter. .

Relatively the experimental aspect, the airfoil would not have been rigid but articulated to simulate actual blade.

8 References

1. Vibration and it is translation in gas turbine engine propeller and rotor craft
   a. By: ASS. PROF: Ali Elhussien Ahmed
2. NASA technical memorandum 7420(Historical over view of flight flutter) testing by Michael w. kehoe .October 1995.
3. AFLUTTER MODEL FOR ROTOR BLADES WITH FLAPES (MARK A. Couch*, E Roberts wood**U.S NAVY/Comseventhflt**NAVAL post grade school.
4. Jones, k, d and platzer M.F"Time-domain analysis of low speed airfoil flutter" AIAA journal vol.34 NO5, PD.1027-1033.
5. Tunner .MA "A computational investigation of wake – induced airfoil flutter in compressible flow and active flutter control "Master's these Naval post grade school march 1994".
6. APRELIMINARY THEORTICAL STUDY OF HELICOPTER –BLADE FLUTTER INVOLVING DEPENDENCE UPON CONING ANGLE AND PITCH SETTING
   a. By Robert p. Coleman

7. viswanathan – sath –y – p -197705 –p h d - 116547