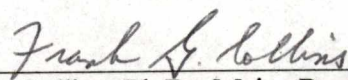
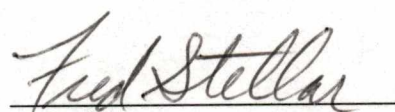


I am submitting herewith a thesis written by Lieutenant Colonel Michael Adrian L'Heureux entitled "Investigation of Blade Stall from a H-46 Helicopter Dynamic Component Upgrade Test." I have examined the final copy of this thesis for form and content and recommend that it be accepted in partial fulfillment of the requirements for the degree Master of Science, with a major in Aviation Systems.

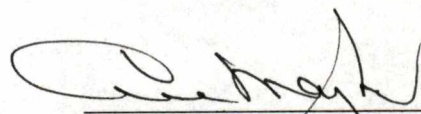

F. Collins, Ph.D., Major Professor

We have read this thesis
and recommend its acceptance.


U. P. Solies


F. Stellar

Accepted for the council:


Interim Vice Provost and
Dean of the Graduate School

**Investigation of Blade Stall from a H-46 Helicopter Dynamic
Component Upgrade Test**

**A Thesis
Presented for the
Master of Science Degree
The University of Tennessee, Knoxville**

Lieutenant Colonel Michael Adrian L'Heureux USMC

August 2001

Copyright © Michael A. L'Heureux 2001

All rights Reserved

Dedication

This thesis is dedicated to my wife

Gerrie L'Heureux

Her hard work, dedication, and tenacity as a wife, mother, student, and professional provide daily example of what is possible for those unafraid to lead with their heart while they think with their mind.

Acknowledgements

When the unexpected happens airborne, quick decisive and correct responses of the crew in their individual actions and working together as a team are paramount to ensure survival. In His beneficence, the Lord has granted me more providential blessing (known secularly as luck) than skill. Certainly a large portion of this blessing was the good fortune to crew with consummate professionals, especially, Major Eric Treworgy, Sergeant Russ Hand, and Mr. Sumner Shaw. Other professionals who assisted with crew duties in the conduct of the project of which this report is based were Major Doug Schueler, Major Jim Evans and Mr. Ed Flores. The maintenance professionals who ensured our aircraft was up and flying well were: Mr. Grubbs, Smith, McNair, Lee, Hughes, and Buss.

I have been extremely fortunate in working with some world class aerospace engineers, especially: Mr. Steve Kracinovich, Jim Busey, An Tran, and Scott Cooley. Their professionalism in quickly analyzing, interpreting, and transmitting essential information were essential to the successful conduct of the test which forms the subject of this thesis. In addition, the assistance of Mr. Rick Funchion in presenting data was invaluable.

I would also like to thank my instructors at both U.S. Naval Test Pilot School, and UTSI for providing me the academic foundation to execute both flight test, and manage the flight test professionals that comprise the Naval Rotary Wing Aircraft Test Squadron.

Abstract

The H-46 Sea Knight has been in service since the early 1960's. At current and forecast usage rates there exists a shortfall of aircraft for the Marine medium lift and the Navy vertical replenishment missions until follow-on aircraft are procured. The Dynamic Component Upgrade Program (DCU) was developed to address life extension of the rotor and drive systems. The testing encompassed a series of engineering and mission aircraft maneuvers conducted during a varied matrix of loading conditions, density altitudes, and flight conditions. During a DCU survey flight data point of sixty degree angle of bank, right level turn, CH-46E bureau number 153355 experienced a rapid excursion beyond peak Cruise Guide Indicator limit. During the recovery sequence, the aircraft experienced a pitch oscillation episode that resulted in exceeding peak static and oscillatory limits on various components in addition to aircraft damage. This thesis surveys pertinent literature to give the reader a basis for understanding the phenomena experienced in flight. It addresses flight test safety aspects and risk reduction for integrated test programs, test methodologies, special concerns for mature platforms, lessons learned, and real time analysis tools used during this flight test. In addition, test results are examined in light of current aircraft usage and configuration to fully understand the ramifications of the test results. The recommendations advocated by this investigation conform to accepted human factor and ergonomic practice, incorporate ideas validated by independent research, and show the utility of this approach to enhance mature aircraft designs increasing safety, reliability, and service life in a cost effective manner.

Preface

The final printing of this thesis ends a personal chronicle for the author that has spanned close to five years. It was the basis for a paper that was presented to the Society of Experimental Test Pilots during the 41st annual symposium in 1997. In the intervening years, in addition to ensuring the incorporation of some of the recommendations of this research, the author has continued working in flight test and flight test operations, was assigned as future plans officer for an overseas Marine Expeditionary Force, and as Chief Test Pilot and Commanding Officer of the Naval Rotary Wing Aircraft Test Squadron. It was during the author's reassignment as Chief Test Pilot and in April of 2000, that he was saddened to hear of a mishap involving a MV-22B conducting operational testing in Arizona. As a result of the mishap, the squadron was tasked to examine aircraft flight characteristics in high rates of descent and slow forward air speeds in the helicopter mode. The side-by-side configuration of the V-22 rotors in the helicopter mode shares similarities to the H-46, that is the subject of this thesis. During V-22 test planning reviews the author has always been cognizant of the control response characteristics of a tandem configuration when one rotor enters a significantly different aerodynamic environment than the other rotor system. Recent events underscore the importance of fully understanding these phenomena.

This thesis introduces the purpose of its subject test program. It relates historic anecdotal information relevant to the phenomena experienced. This thesis develops the theoretical concepts of helicopter blade aerodynamics using "blade element theory" to form the basis with which to allow the reader to understand the effects of the blade stall

Table of Contents

Chapter		Page
	Introduction	1
I	Background.	3
	<u>Setting the Stage</u>	3
II	Review of the Literature	6
	<u>Incipient Blade Stall in Helicopter Applications</u>	6
	<u>Lift</u>	7
	<u>Drag</u>	11
	<u>Stall</u>	12
	<u>Blade Flapping or Feathering</u>	18
	<u>Aerodynamics of Forward Helicopter Flight</u>	22
	<u>Compressible Flow</u>	26
	<u>Retreating Blade Stall</u>	29
	<u>Dynamic Lift and Stall</u>	35
	<u>Negative Damping and Structural Issues</u>	36
	<u>Literature Review Summary</u>	39
III	Methodology	40
	<u>Structural Testing</u>	40
	<u>Test and Test Conditions</u>	42
IV	Results and Evaluation	45
	<u>Project Status Prior to Stall Episode</u>	45
	<u>Critical Parameters and Cruise Guide Indicator</u>	46
	<u>The Flight</u>	49
	<u>Excerpt From Daily</u>	50
	<u>Results</u>	51
	<u>Team Immediate Response</u>	54
V	Lessons Learned	55
	<u>Different Interpretation of Flight Test Procedure</u>	55
	<u>Aircraft Center of Gravity Can Significantly Alter Aircraft Response</u>	56
	<u>Aircraft Dynamics and Stall Response</u>	56
	<u>Difference in Operator Manuals</u>	59
	<u>Pilot Training and Dynamic Stall Response</u>	63
	<u>Flying a Different Aircraft</u>	64

	<u>Simulator Employment</u>	65
	<u>Telemetry Room Training</u>	65
VI	Recommendations	67
	<u>Recommendation Basis-Task Analysis</u>	67
	<u>Witness Marks and Maintenance Requirements</u>	68
	<u>Cruise Guide Indicator Issues</u>	68
	<u>Operators Manuals</u>	71
	<u>Differential Airspeed Hold Response Rate</u>	73
	<u>Operator Training</u>	74
	<u>Increased Inter-service Testing</u>	74
VII	Conclusions	76
	Bibliography	78
	Vita	82

List of Figures

Number	Figure	Page
Figure 1	Boeing CH-46E Sea Knight Helicopter	1
Figure 2	Forces acting on Wing Cross Section	7
Figure 3	C_l vs. Angle of Attack for Various Wing Cross Sections	9
Figure 4	Airfoil Terminology	10
Figure 5	Drag vs. Angle of Attack	12
Figure 6	Flow around an Airfoil	13
Figure 7	Maximum Lift Coefficient at Low Mach Numbers	15
Figure 8	Blade Velocity Distribution in Forward Flight	18
Figure 9	Blade Element in Forward Flight	25
Figure 10	Airfoils in Transonic Flow	28
Figure 11	Drag Rise Boundary and Divergence Region	30
Figure 12	Comparison between Theoretical and Experimental Stalled Areas	34
Figure 13	Lift Characteristics of an Oscillating Airfoil	37
Figure 14	S-N (Stress vs. Number of Cycles) Curve	41
Figure 15	Cruise Guide Indicator	48
Figure 16	Right Side Aft Fairing	52
Figure 17	Test Aircraft	52
Figure 18	Left Side Aft Fairing	52
Figure 19	Aft Drive Arm	53
Figure 20	Front Frame Area, Number 1 Engine	53

List of Figures

Number	Figure	Page
Figure 21	Corrected Blade Stall Test Hazard Analysis Insert	54
Figure 22	Aircraft Blade Stall Sequence	58
Figure 23	Fault Tree Analysis	59

Acronyms and Symbols

ψ	Blade azimuth angle measured from downwind position in direction of rotation (radians)
ρ	Mass density of air (slugs/cubic feet)
α	Wing angle of attack (degrees or radians)
θ	Angle between zero-lift of the airfoil and the chord line in unfeathered position (radians)
ϕ	Inflow angle at blade element (radians)
Ω	Rotational speed (radians per second)
σ	Solidity ratio (dimensionless)
λ	Inflow ratio, total flow through the rotor plane perpendicular to the control axis (dimensionless)
μ	Ratio of forward speed to blade tip rotational speed (dimensionless)
α_r	Blade element angle of attack ($\theta + \phi$) (radians)
a	Speed of sound (feet per second) or slope of section lift coefficient against section angle of attack (dimensionless ratio of radian measure)
A or A_d	Rotor disc area (square feet)
A_b	Effective blade area (square feet)
AFCS	Automatic Flight Control System
AOB	Angle of Bank (degrees)
b	Number of rotor blades
BuNo	Bureau Number
c	Chord (feet)
c.g.	Center of Gravity
c_t	Coefficient of Thrust (referred to blade area) (dimensionless)
C_d	Coefficient of Drag (dimensionless)
C_{do}	Profile Drag coefficient (at zero angle of attack) (dimensionless)
C_f	Aerodynamic Force Coefficient (dimensionless)
CGI	Cruise Guide Indicator
CH	Cargo Helicopter
C_l	Coefficient of Lift (dimensionless)
C_{lmax}	Maximum lift coefficient (dimensionless)
CRT	Component Replacement Times
C_T	Coefficient of Thrust (dimensionless)
D	Drag force (pounds)
DASH	Differential Airspeed Hold
DCU	Dynamic Component Upgrade
ERITS	Empirical Retreating Tip Speed
F_a	Aerodynamic Force (pounds)
FAA	Federal Aviation Agency
g	Acceleration due to gravity (approximately 32 feet per second square)
H	Helicopter

H_d	Density Altitude (feet)
Hz	Cycle
KIAS	Knots Indicated Airspeed
Knot	Unit of velocity (nautical miles per hour)
L	Lift force (pounds)
M	Mach number ratio of speed to speed of sound (dimensionless)
MPH	Miles per hour
NACA	National Agency of Civil Aeronautics
NAS	Naval Air Station
NATC	Naval Air Test Center
NATOPS	Naval Air Training and Operating Procedures Standardization
NAVAIR	Naval Air Systems Command
NAWCAD	Naval Air Warfare Center, Aircraft Division
NRWATS	Naval Rotary Wing Aircraft Test Squadron
PAC	Pilot at the Controls
Pax	NAS Patuxent River MD Complex
PMA	Program Manager, Aviation
q	Dynamic Pressure (pounds per square foot)
R	Blade radius (feet)
RN	Reynolds Number (dimensionless)
RPM	Revolutions per Minute
RTB	Return to Base
RWATS	Rotary Wing Aircraft Test Squadron
S	Surface area
Slugs	Unit of mass
S-N	Stress or Strain vs. Number of Cycles
T	Thrust (pounds)
T_b	Thrust per blade (pounds)
TM	Telemetry
U	Vector sum of rotor velocity components.
U_p	Components of velocity along rotor thrust axis
U_T	Components of velocity perpendicular to thrust axis
V	Free Stream velocity
ν	Ratio of viscosity to fluid density or velocity (feet per second)
VERTREP	Vertical Replenishment
VMC	Visual Meteorological Conditions



Figure 1 Boeing CH-46E Sea Knight Helicopter

Introduction

This thesis investigates the results, conclusions, and recommendations revolving around a specific in-flight incident that occurred during the testing of a Boeing Sea Knight (CH-46E) helicopter (Bureau Number 153355) for the Dynamic Component Upgrade and Fuselage Strain Survey (DCU) testing Program. The Sea Knight, (figure 1) manufactured by Boeing Helicopter, is a tandem-rotor assault helicopter with a maximum gross weight of 24,300 lb. Two T58-16 turbo-shaft engines rated at 1,870 shaft horsepower each at military rating power the aircraft. Power is transferred through a transmission system consisting of a combining gearbox, an aft transmission, a synchronizing shaft, and a forward transmission. Each transmission drives a three bladed 51-foot diameter, fully articulated rotor system.

The H-46 Sea Knight has been in service since the early 1960's. At current and forecast usage rates there exists a shortfall of aircraft for the Marine medium lift and

Navy vertical replenishment missions until follow-on aircraft are procured. The Dynamic Component Upgrade Program (DCU) was developed to address life extension of the rotor and drive systems. The testing encompassed a series of engineering and mission aircraft maneuvers conducted during a varied matrix of loading conditions, density altitudes, and flight conditions. During a DCU survey flight where the aircraft was executing a sixty degree angle of bank, right level turn, at a 22,800 pounds gross weight loading near the forward center of gravity loading limit, the aircraft experienced instrument indications of incipient blade stall. At this time the aircraft rolled out of the right turn to straight and level flight. Prior to lowering the collective, the aircraft exhibited an uncommanded, over 30 degree nose down dive. This precipitated a pitch oscillation episode during the recovery sequence. This episode resulted in exceeding peak static and oscillatory limits on various components in addition to aircraft damage.

This thesis addresses flight test safety aspects and risk reduction for integrated test programs, test methodologies, special concerns for mature platforms, lessons learned, and real time analysis tools used during the flight test. In addition, test results are examined in light of current aircraft usage and configuration to fully understand the ramifications of the test results. The recommendations advocated by this investigation were implemented as operator manual changes and recommendations to the aircraft Program Manager, Aviation (PMA).

I

Background

Setting the Stage

In a helicopter, the preponderance of aerodynamic loads are applied to the aircraft rotor blades which transmit these loads to other dynamic components including rotor hubs, transmissions, and shafting. Generally, dynamic limits on components will dictate aircraft service life limits in a helicopter. A purpose of the fuselage survey portion of the test program was to verify this supposition.

At the time of the aircraft incident that forms the basis of this investigative research, the author was primarily concerned with the flight safety, program schedule, and budget impacts of this event. Eventually, as the causal factors became evident, this incident enabled the test team to develop a clearer understanding of the dynamics affecting helicopter structural capability, and make better recommendations to the PMA.

The PMA, is responsible for determining and budgeting the hierarchy of material upgrades required to ensure aircraft serviceability through planned service life. The PMA contracted Boeing Helicopter to develop a dynamic system upgrade to increase aircraft service life. The PMA also contracted the test team to assist the contractor, test certain flight regimes (those more representative of military service), and provide an independent evaluation of the success of the contractor in meeting the agreed deliverables. In addition to reporting the final results of testing to the PMA, the team supplied the load measurements to the component upgrade manufacturer, Boeing

Helicopter, to assist them in the computation of removal and replacement times of the various components that comprise the helicopter dynamic system.

It was important for this test team, Naval Rotary Wing Aircraft Test Squadron (NRWATS) and Naval Air Warfare Center-Aircraft Division (NAWCAD) DCU to complete the testing on schedule so that appropriate information would allow the computation of cycle life of components that would be installed on currently operating helicopters. These helicopters, due to Navy and Marine Corps operational commitments, were scheduled to reach the maximum level of aircraft operating hour limits. During this period, there were a number of aircraft approaching the maximum limit of 12,500 flight hours. An extension of this limit, to possibly 20,000 flight hours, would be possible if the data the test team collected on fuselage and dynamic component strain supported it. Flight testing, and the compilation of preliminary data accomplished just prior to the incident, had indicated that a service life extension would probably be justified. At the time that the subject flight incident occurred, the Flight Clearance branch of NAWCAD (the naval counterpart to FAA flight certification branch) was still acquiring the data to make a service life determination. Due to the fact that the Flight Clearance Branch would not make a decision that might affect the lives of service members based on incomplete data, team focus was on fixing the aircraft, preventing incident reoccurrence, and completing the testing. Although the team had strong personal desires to understand the phenomena that occurred, this was not a research program; the purpose behind the test effort was directed and highly focussed. The focus was to

complete the test and report the data before fleet squadrons had to start parking aircraft that had reached their maximum operating time limits.

In addition to the delay caused by the in-flight incident, prior DCU program issues had already incurred substantial delays. During a quality review, Boeing had discovered a sub-contractor had applied the wrong material finishing process to the forward upper rotating swashplate. This caused an eight-month delay while this particular part was changed and the component set was re-instrumented.

This background illuminates some of the considerations that set the stage for the schedule pressure and eventual budget pressure associated with conducting this particular flight test. Although the team and author were personally driven to understand the mechanics behind the test experience, this desire would have to be subordinate to the business at hand, completing the loads survey.

II

Review of the Literature

Incipient Blade Stall in Helicopter Applications

To better understand the phenomena of incipient blade stall and how it relates to the helicopter design criteria, this section provides an overview and references selected literature as it relates to helicopter blade stall, and the impact of the phenomena to the operator. In addition, in order to more fully understand the processes by which the damage was incurred, a short analysis of the phenomenon of blade stall as it related to this particular application will be investigated.

Two methods of examining the aerodynamic characteristics of the rotor systems that provide the motive forces for helicopters are; the Momentum Theory, and the Blade Element Theory. Momentum Theory is based on the physics involved with the mass flow of air through the system, and is based on Newton's second and third laws of motion. The Momentum Theory, while an optimal tool for examining the rotary wing environment and forces in a hover, is more limited in usefulness for gaining an understanding of the forces at the blades and related structure in forward flight. The Blade Element Theory is more similar to the analysis used for fixed wing aerodynamic investigation and is the primary concept used to make clear to the reader the effects and forces discovered to exist in certain helicopter flight regimes flown during the conduct of this test.

Rotary Wing flight accurately describes the primary motive force accomplished by the rotor system. A typical rotor blade cross section is essentially very similar to an

airplane wing cross section, and for purposes of analysis, is examined in a similar fashion. Figure 2 shows the forces acting on the wing cross section.

Lift

There are a number of variables that affect the magnitude of lift on a free body (wing). They include:

- 1) Velocity of the air slipstream.
- 2) Air Density.
- 3) Shape of the Body.
- 4) Surface area of the body
- 5) Angle of Attack of the body to the slipstream.
- 6) Air compressibility.
- 7) Air viscosity (Reynolds number).

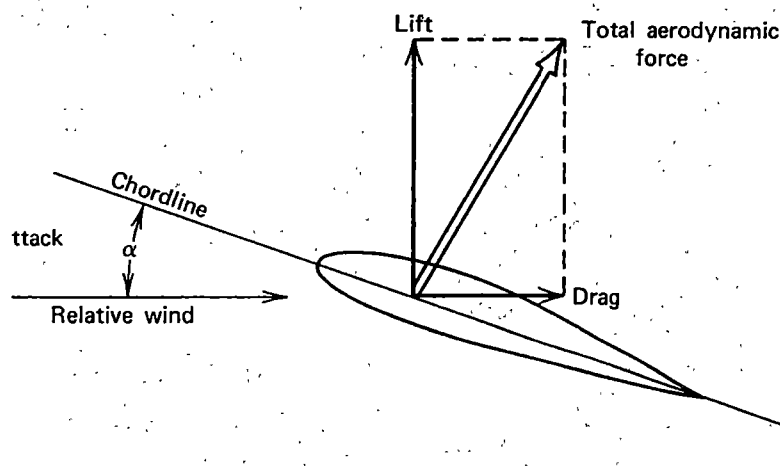


Figure 2 Forces acting on Wing Cross Section

Source: G.H. Saunders, Dynamics of Helicopter Flight (New York, John Wiley & Sons, 1975) 39

The total aerodynamic force in figure 2 can be represented in terms of these variables as,

$$F_a = C_f q S \quad (1)$$

where the total aerodynamic force is F_a expressed in pounds force. The dynamic pressure is q , and is expressed in units of pounds/square foot. Reference area S is expressed in square feet. A dimensionless aerodynamic force coefficient C_f is used to account for the remaining factors of angle of attack of the body to the air slipstream and the effects of compressibility and viscosity. Considering the factors of compressibility and viscosity to be negligible, C_f can be considered to be a function of shape and angle of attack only. From figure 2, lift and drag can be considered components of the sum of the total aerodynamic force:

$$\vec{F}_a = \vec{L} + \vec{D} \quad (2)$$

Where both Lift and Drag are expressed in terms of pounds force. Essentially, both lift and drag depend on the same variables of shape and dynamic pressure. C_l represents the coefficient of lift and C_d the coefficient of drag (dimensionless). Lift may then be expressed as:

$$L = C_l q S \quad (3)$$

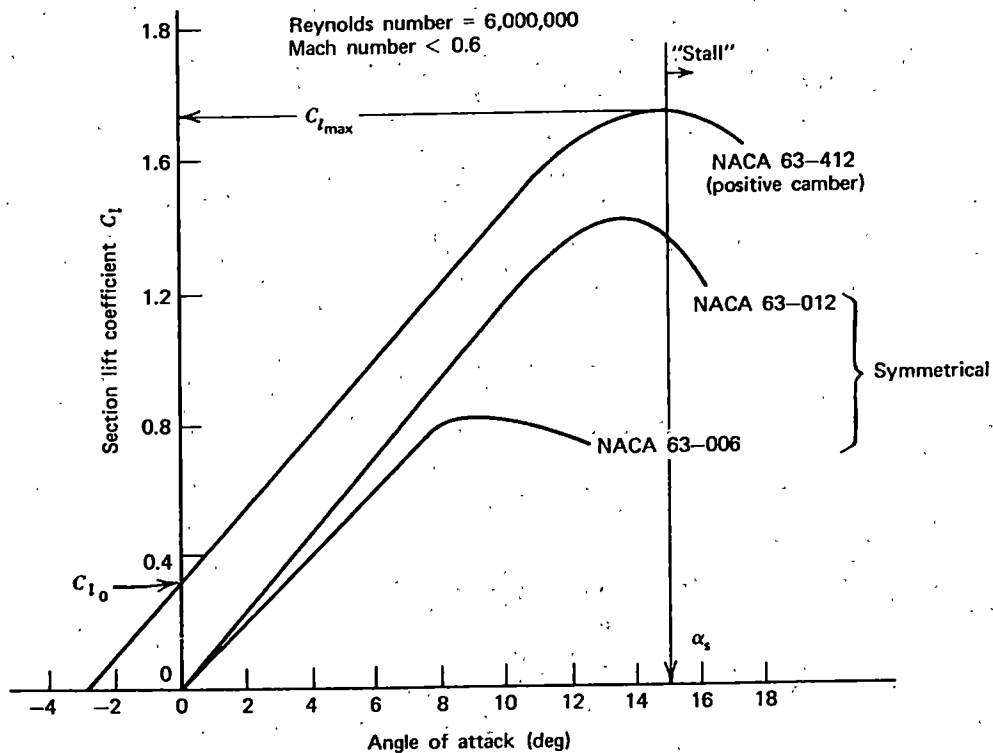


Figure 3 C_l vs. Angle of Attack for Various Wing Cross Sections

Source: G.H. Saunders, Dynamics of Helicopter Flight (New York, John Wiley & Sons, 1975) 42

If the lift coefficient of an airplane wing were plotted against angle of attack of that wing, the graphs in figure 3 would typically result. From the graph of figure 3, the features to note are:

- With positively cambered wings, lift is generated at zero angle of attack.
- The lift coefficient has a linear relationship to angle of attack up to a certain range. The slope is approximately 2π per radian for all thin airfoils. Aspect ratio and viscous effects lower it.
- The curves are relatively linear, becoming non-linear as they approach $C_{l_{\max}}$ (angle of attack for static stall). Beyond this angle, lift decreases.

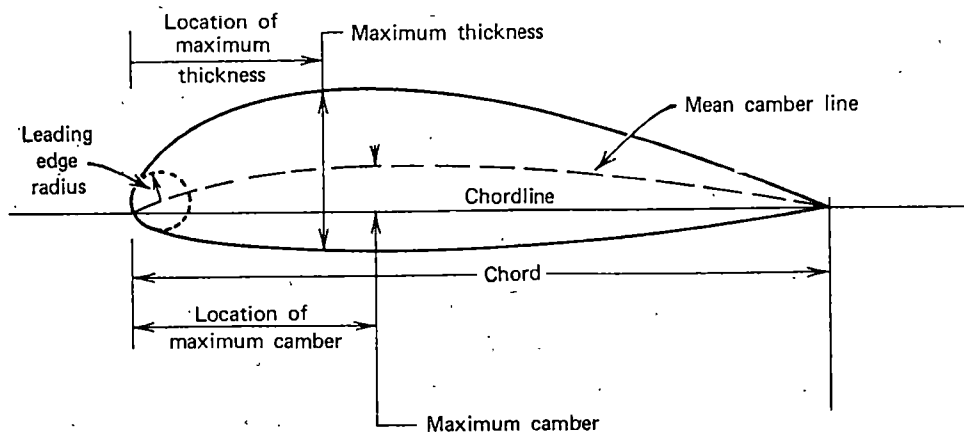


Figure 4 Airfoil Terminology

Source: G.H. Saunders, *Dynamics of Helicopter Flight* (New York, John Wiley & Sons, 1975) 39

The cross sectional shape of the helicopter airfoil is designed to give it the best possible lift, lowest drag, and smallest pitching moments possible within the structural, and aeroelastic limitations of the rotor blade structure. To assist in this purpose, most helicopter blade cross sections (and many other airfoils) have small camber (figure 4 is an illustration of a cambered airfoil with associated terminology). Symmetrical airfoils have upper and lower surfaces the same distance from the chordline. Maximum camber (maximum displacement of mean from the chordline) on helicopter rotor blades seldom exceeds 4% and may be located as far aft as 40% from the leading edge.¹ The airfoil section on the test aircraft which is the subject of this thesis was the NACA 23010 from the blade tip to .5 of blade radius (306 in), and then it was transformed linearly to NACA 23012 at .25 radius, remaining NACA 23012 from .25 radius to the root cutout at approximately .08 radius.² In the NACA classification criterion the first digit indicates the amount of maximum camber in percent chord (18.75 in). In this example the

¹G.H. Saunders, *Dynamics of Helicopter Flight* (New York, John Wiley & Sons, 1975) 40

maximum camber is 2%. The second digit gives the chord-wise position of maximum camber. In this example the maximum camber is located at 3/20 (15%) of the chord.

With the third digit zero, the aft portion of the camber line is a straight line. The last two digits give the maximum thickness in percent chord. The tested blade was slightly thicker toward the base than at the tip.

Drag

Drag is the net aerodynamic force parallel to the relative wind. The sources of the drag force are components of aerodynamic pressure and skin friction that act in the flow direction. As in the development of the equation for lift, drag may be expressed as:

$$D = C_d q S \quad (4)$$

where drag D , is in pounds force, the coefficient of drag, C_d is a dimensionless coefficient, q is dynamic pressure expressed in pounds per square feet, and S the wing reference area expressed in square feet. There are a number of different types of drag;

- 1) Parasite Drag – Associated with non-lifting portions of the fuselage, cockpit, rotor hub, landing gear, etc.
- 2) Profile Drag – Basically the equivalent of parasite drag for the rotor blades or rotating wings themselves
- 3) Induced Drag – Associated with the production of lift, induced drag is a function of the angle of attack.

² Maj. R. Yeend, Mr. S. Kowley, Contractor Structural Demonstration of the H-46 Fiberglass Helicopter Blades, NATC Technical Report RW-12R-78, RWATS, NAS Patuxent River MD. 14 Apr 1978

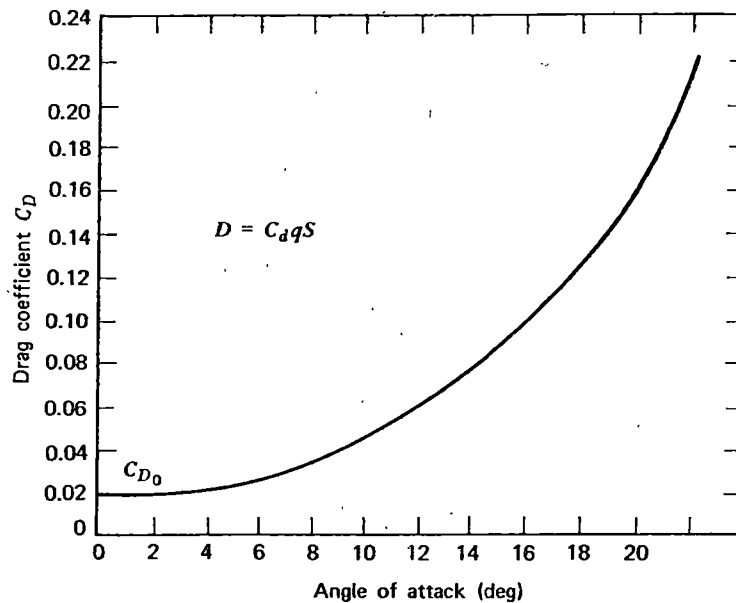


Figure 5 Drag vs. Angle of Attack

Source: G.H. Saunders, Dynamics of Helicopter Flight (New York, Wiley and Sons, 1975) 53

Rotor blade sections produce increased induced drag as a function of local angle of attack. The local angle of attack changes considerably as the rotor blade revolves around the hub due to conditions of helicopter flight. Lift is directly proportional to absolute angle of attack, while induced drag has been found through experiment to be proportional to the square of the angle of attack (or to the square of the lift coefficient). Figure 5 shows a 3-D case of total drag to angle of attack. In this graph C_{D0} is the profile drag, which is the drag where the absolute angle of attack α is zero. At C_{D0} all components of drag are produced by the profile drag of the blade.

Stall

The maximum lift of any airfoil is associated with its stalling behavior. The mechanism that causes this behavior is related to the airflow over the surface of the wing. The airflow near the surface is called the boundary layer, it is the layer where local velocities are reduced by the wing surface to zero at the surface.

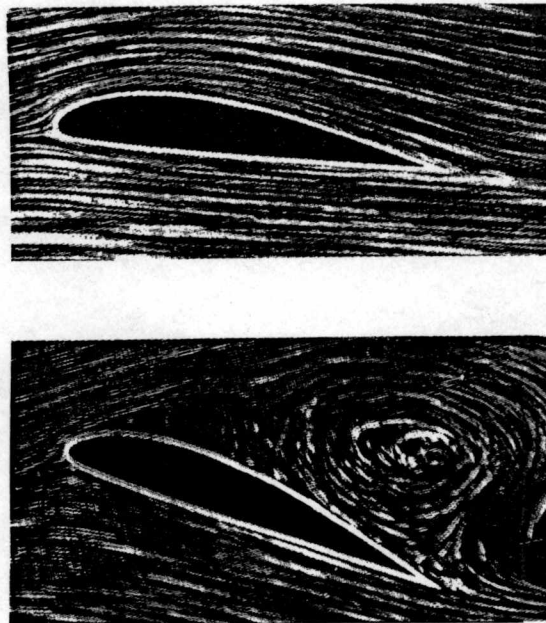


Figure 6 Flow around an Airfoil

Source: H. Schlichting, Boundary-Layer Theory (New York, McGraw Hill, 1979) 37

It is usually a very thin layer which grows in thickness downstream from the stagnation point.

The stagnation point is that point near the wing leading edge where the air slipstream diverges over the top and bottom of the wing and the local velocity is zero. There are three basic types of boundary layers: laminar, turbulent, and transitional. Figure 6, shows the flow of free stream air around a 2 dimensional airfoil with the top portion of figure 6 representing an airfoil with its boundary layer exhibiting attached flow characteristics, the bottom of figure 6 shows the top portion of the airfoil boundary layer exhibiting stalled flow characteristics. The point at which the boundary layer changes from laminar to turbulent is called the transition point (separation can be related to the transition point, but not always). The characteristics of the stall depend on the resultant pressure distribution on the airfoil and whether the boundary layer is laminar or turbulent.

The pressure distribution may cause either gentle or sharp loss of lift, and/or a gentle or sharp change of moment about the airfoil leading edge (different aerodynamic reference points may also be used). Effects of viscosity and compressibility are embedded in the lift and drag coefficients. Viscosity effects are often expressed in terms of the Reynolds number, RN , which is defined as:

$$RN = \frac{Vc}{\nu} \quad (5)$$

Where V = free stream velocity (fps)

c = chord (ft or typical reference length)

ν = kinematic viscosity (ft^2/sec), which is the ratio of the dynamic coefficient of viscosity to the density of the fluid (at some reference condition)

This non-dimensional number is an indicator of the viscous effects. It has been demonstrated that aerodynamic forces and moments depend strongly on the Reynolds number, and that for helicopter rotor blade applications, chord Reynolds numbers typically vary from 1 to 6 million³. Considering the situation illustrated in figure 6, at small incidence angles of approximately 10° , the flow pattern is attached and the airfoil generates lift with minimum profile drag. Increasing the angle of attack to approximately 15° causes flow separation, a marked decrease in lift, and a marked increase in drag. Further experimentation has shown, that the flow separation location varies with Reynolds number,⁴ and depends upon the airfoil profile.

³ G.H. Saunders, Dynamics of Helicopter Flight (New York, John Wiley & Sons, 1975) 49

⁴ H. Schlichting, Boundary-Layer Theory (New York, McGraw Hill, 1979) 39

Abbot and Von Doenhoff in their text, Theory of Wing Sections (New York, Dover, 1959) presented the results of 118 airfoils in a two dimensional wind tunnel at chord Reynolds numbers from 3 to 9 million. This corresponded to the Reynolds numbers existing at the retreating tips of rotor blades with chord lengths from 1 to 3 feet. A summary of the stall characteristics of these airfoils is obtained by plotting the maximum lift coefficient ($C_{l_{max}}$) against the ordinate at the 25% chord station, measured in percentage of the chord length and is depicted in figure 7. A study of Figure 7 shows the differing airfoils grouped as a function of their particular type of stall effects (In the case of the subject aircraft undergoing test, the NACA airfoil was 23010/23012).

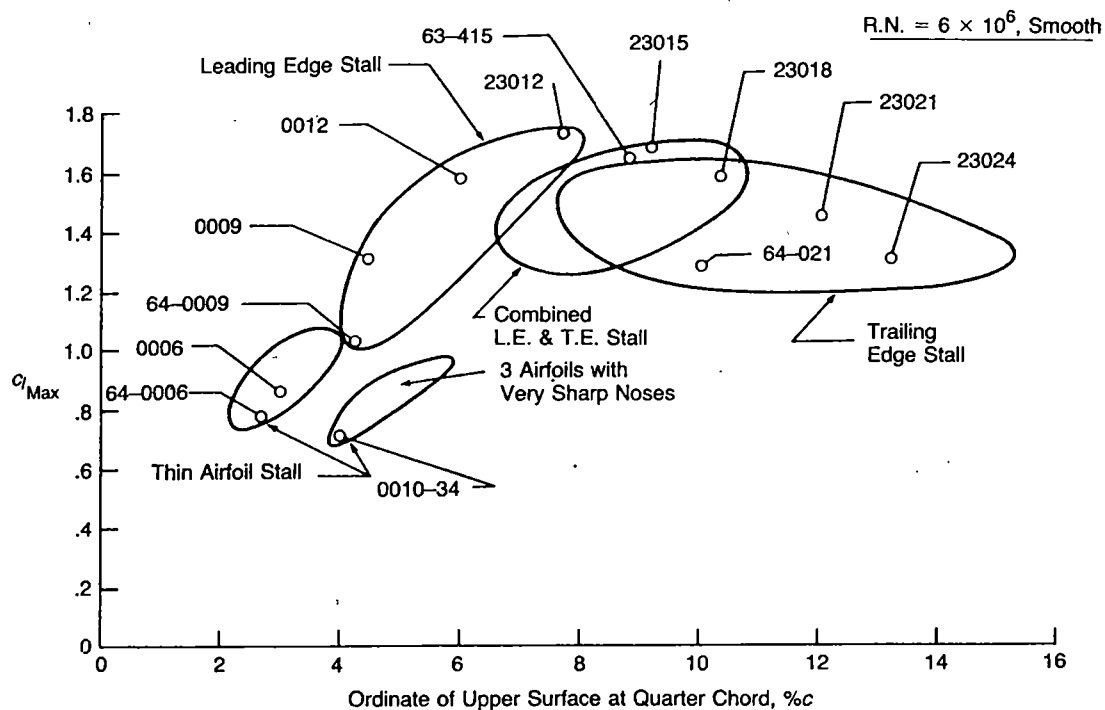


Figure 7 Maximum Lift Coefficient at Low Mach Numbers

Source: R.W. Prouty, Helicopter Performance, Stability, and Control (Malibar Florida: Robert E. Krieger, 1990) 387.

Thin Airfoil Stall At low Reynolds number, stall is caused by the separation of a laminar boundary layer which produces a bubble whose outer (at the shear layer) surface is laminar. At moderate angles of attack, the flow reattaches (most likely because the shear layer has become turbulent) and becomes turbulent before reaching the trailing edge. Thin airfoils have gentle lift (depending on profile) stall but sharp moment stall characteristics.

Leading Edge Stall Caused by separation of the laminar flow at the leading edge that produces a bubble. The separation bubble can act in various ways that produce the same effect:

- 1) The boundary layer can separate when it is laminar and not re-attach.
- 2) The boundary layer can separate when it is laminar and the shear layer can transition and re-attach.
- 3) The boundary layer can become turbulent, separate, and turbulent shear can attach.

As angle of attack is increased, the separation point moves forward rather than aft as similar to the case with the Thin Airfoil Stall. The characteristics of the Leading Edge Stall are sharp change in both lift and moment. In a dynamic helicopter environment the situation is far from being understood.

Trailing Edge Stall Caused by gradual separation of the turbulent boundary layer at the trailing edge and moving forward as angle of attack increases. Trailing Edge Stall is gentle in both lift and moment change⁵.

⁵ R.W. Prouty, Helicopter Performance, Stability, and Control (Malabar Florida: Robert E. Krieger, 1990) 380-382

All airfoils used in helicopters are thin. Stall depends greatly on the pressure gradient, which is controlled by the profile and angle of attack. Leading Edge or Trailing Edge stall are characteristics of various profiles in the static stall case and rarely apply to the dynamic environment that a helicopter blade experiences.⁶

Mr. R. Prouty in his book Helicopter Performance, Stability, and Control discusses the design philosophy used for increasing the helicopter blade performance. In his discussion, he uses as an example the NACA 23012 airfoil, which is the same type of airfoil used in the blades of the helicopter undergoing test which is the subject of this thesis. The following comments are germane to his discussion; “A study of the figure (same as author’s figure 7) suggests several potential methods for increasing maximum lift coefficient. One obvious method is to increase the thickness ratio as in the 0006, 0009, and 0012 series. Another method is to introduce forward camber, or “droop snoot” as in going from the 0012 to the 23012. This improvement comes from modifying the path from the stagnation point to the upper surface...delaying the formation of the laminar stagnation bubble and also decreasing the magnitude of the deceleration required as the air goes toward the trailing edge, thus decreasing the unfavorable pressure gradient.”⁷ Mr. Prouty goes on to say; “The modification resulted in a 40% increase in maximum lift. ...In the airplane industry, airfoils with drooped noses such as NACA 23012 have had a bad reputation for abrupt stall. This characteristic is not significant on rotors, however because stall conditions are entered gradually starting with a small

⁶ F. Collins, Notes to Author, University of Tennessee Space Institute, Tullahoma TN, April 2001

⁷ R.W. Prouty, Helicopter Performance, Stability, and Control (Malabar Florida: Robert E. Krieger, 1990) 387-388

portion on the retreating side and the also because the stall becomes less abrupt as a result of compressibility effects at Mach numbers of .3 or .4⁸.

Blade Flapping or Feathering

Figure 8 shows the distribution of velocity at helicopter rotor blades in forward flight. With the rotor rotating counterclockwise, and the helicopter traveling in a direction toward the top of the page, the blade on the right side in the figure is the advancing blade. The blade on the left side of the figure is the retreating blade. The resultant velocity at each blade station is the sum of the relative wind the blade

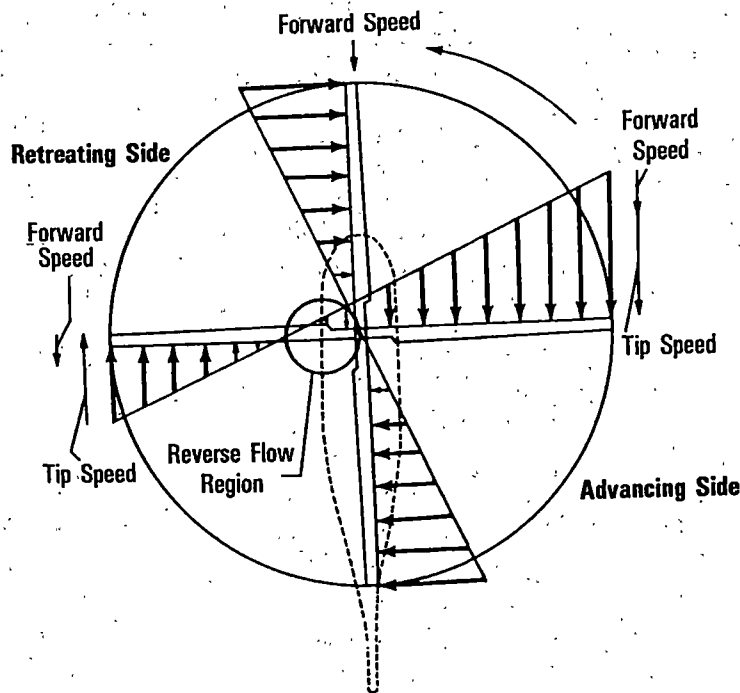


Figure 8 Blade Velocity Distribution in Forward Flight
Source: R.W. Prouty, Helicopter Aerodynamics (PJS Publications Peoria IL, 1985) 17

⁸ R.W. Prouty, Helicopter Performance, Stability, and Control (Malibar Florida: Robert E. Krieger, 1990) 388

experiences due to rotation in still air, and the velocity relative wind due to the forward velocity of the aircraft plus the flapping velocity. The blades on the top and bottom have the same distributions as those in a hover. On the left or retreating side there is a region where the air over the blade flows in reverse.⁹ This particular phenomenon was an early impediment to the helicopters development and necessitated the innovation of the flapping or feathering main rotor. Blade flapping is the up and down blade motion about the horizontal hinge at the blade root or about a corresponding flexible portion of the hub. Feathering is the rotation of the blade about its spanwise axis to change its pitch. The blade may be held fixed in pitch through a linkage connecting it to a swashplate. Tilting the swashplate allows for changing the pitch angle of the rotor blades. The helicopter appears to be quite complicated in its blade motion because both feathering and flapping motions exist with respect to the physical, powered shaft. If, however, the feathering is thought of as a means of tilting the control axis through a given angle, then the system can be reduced to one in which only pure flapping exists with respect to the control axis¹⁰. For brevity, and due to the equivalent nature of both treatments, the author will confine the following discussion to the flapping case only.

The history of the blade flapping innovation, while not entirely pertinent to this discussion, is interesting as it chronicles the human element interplay and impact on technical contributions and scientific advancement.

⁹ R. W. Prouty, *Helicopter Aerodynamics* (Peoria: PJS Publications, 1985) 17

¹⁰ A. Gessow/G.C. Meyers Jr., *Aerodynamics of the Helicopter* (New York: Frederick Ungar Publishing, 1967) 166

The inventor of this mechanism was a Spanish gentleman whose name was Juan de la Cierva y Cordoniu, more commonly known as Juan Cierva. Cierva had been an aircraft designer who had designed and built a three engine bomber for the Spanish Army in 1919. During testing of his design, the pilot had flown too slowly and stalled the airplane close to the ground. Although the pilot lived, Cierva's creation was destroyed and this appeared to motivate the designer to design an aircraft that would be less susceptible to stall dangers at slow airspeeds.

The principle of autorotation was earlier noted by the Russian Boris Yuriev in 1912 who used the large rotor experimentation of Sorokovmovski to demonstrate that during a descent, an unpowered rotor blade would keep turning of its own volition. These phenomena would also allow forward gliding. Yuriev called this rotor gliding. This autorotation or rotor gliding phenomenon would allow for Cierva's eventual invention of the autogiro.¹¹

An autogiro is a cross between a helicopter and an airplane. The autogiro utilizes the propulsive force of an aircraft engine and propeller to move an aircraft fuselage forward through the air. Cierva's autogiro had the fuselage, tail, and landing gear of an airplane. Instead of normal airplane wings, it had small wings and large freewheeling rotating wings (rotor) on a shaft above the fuselage, tilted slightly aft. As the craft moved forward, this aft tilt caused airflow which would power the rotor and create a lifting force

¹¹ C. Gablehouse, Helicopters and Autogiros (New York J.B. Lippincott, 1969) 36

that would lift the aircraft from the ground in a very short distance at speeds significantly slower than early airplanes.

An early problem with Cierva's prototypes (1921-1922) was that as soon as the craft would begin to lift off the ground, one side would have more lift than the other, (the reason for this is apparent from the previous discussion concerning the distributions of velocity in figure 8) causing it to roll over. This caused him to damage two initial prototypes. In his work to discover why this happened; he discovered the effect of dissymmetry of lift on a rotor system moving through an air mass. This is what caused Cierva's aircraft to want to tip over on take off.

During the early development of the autogiro, Cierva tried to obtain structural integrity of his rotors by bracing the blades with wires as the wings of airplanes were braced. His early experimentation was performed with scale models using flexible rattan blades, which did not require structural augmentation due to their relative size. They did not exhibit the rolling tendency that his full-scale aircraft did. After much thought, Cierva realized it was the difference between the rigid wire braced rotor of the full-scale aircraft and the flexible blades on the rattan bladed model that caused the difference in the rolling moments. The flexibility or ability for the blades to flap caused the lift over the rotating blades to equalize when moving through an air mass. This eliminated the lift dissymmetry and enabled the autogiro to fly. Also there was another enhancing characteristic of his design. When the forward airspeed of the craft dropped below the

necessary 15-20 MPH to ensure sufficient lift for level flight, it would enter a stable straight down descent, eliminating the dangers of airplane stall.

Although this dissymmetry of lift was discovered earlier by some one else, Cierva was the first to actually incorporate the flapping rotor design in a successful aircraft. He demonstrated this in 1923¹², and again at Farnborough, UK, in 1926. This flapping main rotor was incorporated by Igor Sirkorsky nearly 20 years later in what was to become the predominate helicopter design for main rotors. There are two standard means available in overcoming the dissymmetry of lift in forward flight:

- 1) By hinging the blades at the roots so that no moments can be transmitted through the hub. Control is then achieved by tilting the resultant rotor lift vector in the direction of intended movement.
- 2) Cyclically feather the blades, decreasing pitch on the advancing side and increasing pitch on the retreating side.

Both methods are fundamentally equivalent and differ only in the axis of reference¹³

Aerodynamics of Forward Helicopter Flight

One of the largest problems for helicopter design and analysis is the complexity of the aerodynamic model with which flight phenomena are based. An attempt is made to offer an extremely simplified treatment of the rotor system in forward flight. The

¹² C. Gablehouse, Helicopters and Autogiros (New York J.B. Lippincott, 1969) 44

¹³ A. Gessow/G.C. Meyers Jr., Aerodynamics of the Helicopter (New York: Frederick Ungar Publishing, 1967) 138

author applies a basic fixed wing model, developing a forward flight rotary wing case from that model, and then, with the developed model, illustrates factors that contribute to blade stall. This model will neglect all 3-D and unsteady effects.

The basis for all the commonly used aerodynamic treatments of a rotor in forward flight is the original analysis of Glauert who undertook the work to investigate the validity of Cierva's claims for his autogiro, which was then being built and tested in England. Glauert's analysis initially contained many simplifying assumptions adopted due to the mathematical complexity of the forward flight model.¹⁴ A slightly more advanced treatment of Glauert's work was derived by Wheatley and is the basis of the following.

For the airplane, the forces of lift previously developed were:

$$L = C_l q S \quad (6)$$

The dynamic pressure is q and is expressed in units of pounds/square foot. Reference area is S is expressed in square feet. A dimensionless lift coefficient C_l is used to account for the remaining factors of angle of attack of the body to the air slipstream and the effects of compressibility and viscosity. C_l can be considered to be a function of shape and angle of attack only. The dynamic pressure q is:

$$q = \frac{1}{2} (\rho v^2) \quad (7)$$

¹⁴ A. Gessow/G.C. Meyers Jr., Aerodynamics of the Helicopter (New York: Frederick Ungar Publishing, 1967) 180

The symbol ρ represents the mass density of air (slugs/ft³). For a rotor system in forward flight, the velocity, v , is the vector addition of velocity from four different sources:

<u>Induced Flow</u>	This occurs as a function of inflow of air through the rotor system, The direction of flow is dependent on the rotor system being in either powered or autorotational flight.
<u>Blade Rotation</u>	This is a product of the rotational velocity and the distance of the blade element from the center of rotation.
<u>Forward Flight</u>	Dependent on the helicopters forward speed.
<u>Blade Flapping</u>	This velocity contribution is from the vertical motion of the blades about a physical or virtual horizontal hinge at the blade root and due to flexure of the rotor blade.

In Figure 9, the velocity U_P represents the components of resultant velocities along the control (thrust axis) of the blade element (induced flow and blade flapping) while U_T represents the components perpendicular to the thrust axis (forward and rotational velocity). C_l is a section property which varies along the length of the blade, which is the same case as v .

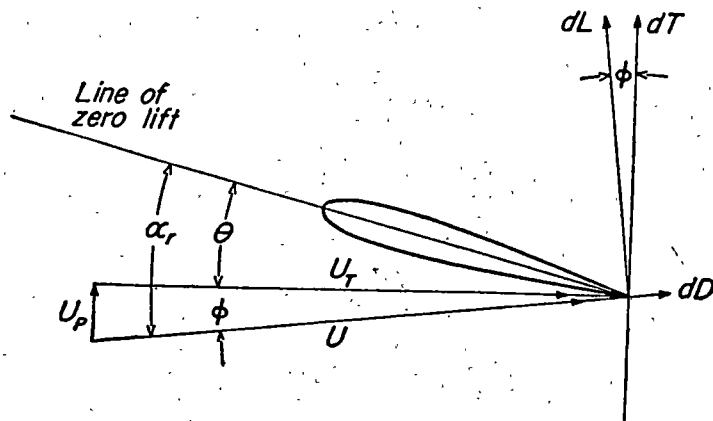


Figure 9 Blade Element in Forward Flight

Source: A. Gessow/G.C. Meyers Jr., Aerodynamics of the Helicopter (New York: Frederick Ungar Publishing, 1967) 187

In the Fixed Wing case we have:

$$L = C_l \frac{1}{2} (\rho v^2) S \quad (8)$$

The differential lift on the blade element of one blade may be written as:

$$dL = \frac{1}{2} (\rho U^2) C_l c dr \quad (9)$$

If the lift coefficient C_l is assumed to be proportional to the angle of attack, an assumption valid for lift coefficients below onset of flow separation, then:

$$C_l = a\alpha_r = a\left(\theta + \frac{U_P}{U_T}\right) \quad (10)$$

From figure 9:

$$dT = dL \cos \phi + dD \sin \phi \quad (11)$$

Where dT is the force component of lift and drag in the direction of the rotor axis.

When ϕ is small,

$$\cos \phi \approx 1 \quad (12)$$

$$\sin \phi \approx 0 \quad (13)$$

and

$$dT \approx dL \quad (14)$$

$$C_t \approx C_l \text{ (which will be used later)} \quad (15)$$

Substituting the equations gives:

$$dT \approx \frac{1}{2} \rho a (\theta U_T^2 + U_P U_T) c dr \quad (16)$$

Hence, the total thrust can be found by integrating the differential thrust around the azimuth and then along the blade span.^{15,16}

It can be found through experimentation that for hovering, and for flight at lower

¹⁵ A. Gessow/G.C. Meyers Jr., Aerodynamics of the Helicopter (New York: Frederick Ungar Publishing, 1967) 187-189

¹⁶ J.B. Wheatley, The Aerodynamics of the Gyroplane Rotating-Wing System NACA Report No. 492 Washington D.C. March 1934

airspeeds and altitudes, the above equation agrees very closely with actual measured values of thrust. The agreement between experimental and calculated thrust results is not as close for higher helicopter velocities, altitudes, and gross weight loading. Some of the assumptions made for the lower speed case are not true. Development of the concepts of the compressible flow assist in discovering some of the reasons for the variance of actual and calculated values at the more extreme operating areas and assist in discovering aeronautical and structural limitations to rotary wing flight.

Compressible Flow

The development of the forgoing expressions has assumed the effects of compressibility to be negligible. At slower airspeeds, this is essentially true. As the true airspeed of an airfoil approaches a velocity at which pressure disturbances are propagated through the air mass, the assumption of compressibility being negligible is not true. In the atmosphere, this velocity is known as the speed of sound. The relationship between the speed of sound and the drag rise which occurs on the airfoil as it approaches this value is evident when viewing figure 10. If the airfoil is traveling at speeds below the speed of sound, pressure disturbances are propagated ahead of the airfoil. In the supersonic case, the air ahead of the airfoil is not influenced due to the fact the airfoil speed is higher than the capability of the medium to propagate pressure disturbances. Drag across a shock wave (wave drag) can be predicted. Helicopter blade design efforts seek to avoid this effect. The term used to describe the relationship of an airfoil to the speed of sound is its Mach number. Mach number is defined as:

$$M = \frac{v}{a} \quad (17)$$

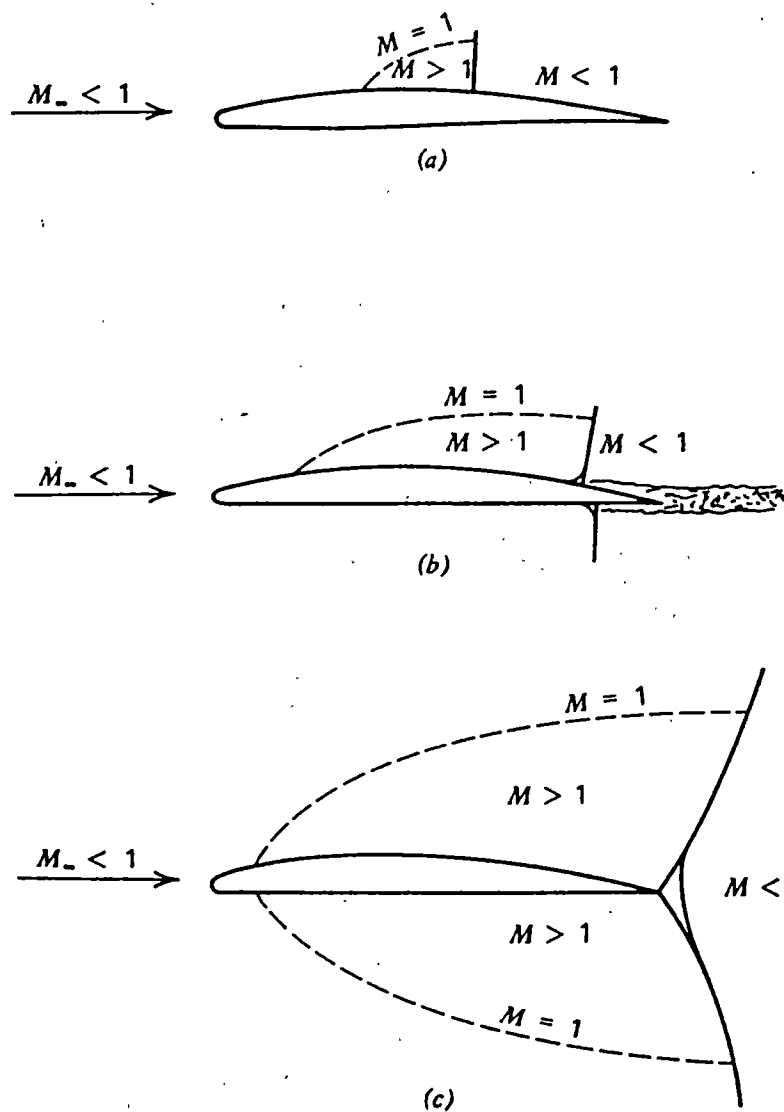


Figure 10 Airfoils in Transonic Flow

Source: B. W. McCormick, Aerodynamics, Aeronautics, and Flight Mechanics (New York, Wiley and Sons, 1995) 232

where M is the Mach number, v is the absolute velocity of air approaching the airfoil, and a is the speed of sound.^{17,18}

In helicopter applications, the first area where airflow approaches the speed of sound is on the outboard portion of the advancing blade tip during high-speed forward flight. The drag rise that results from approaching the speed of sound causes an increase in thrust required for forward flight. As the equation for thrust (16) developed assumed incompressible flow and a small angle of attack, differences in calculated and actual values can be attributed to the compressibility effects of the airfoil, and regions of local stall. An estimate to determining the additional torque required due to compressibility has been calculated to a first approximation by airfoil drag at zero lift (smallest angle of attack). Because tests show results are also a function of thrust coefficient per blade area, empirical corrections to the drag rise Mach number are established through wind tunnel and flight test. A graphical representation of this region (Drag Divergence Region) is illustrated in figure 11 where $(r/R)_{dr}$ is the radius station at which drag rise is first experienced and ψ_1 and ψ_2 are the azimuth angles bounding the drag rise region.¹⁹

Retreating Blade Stall

As can be deduced from the discussion earlier concerning blade flapping, and the purposes for which blade flapping or feathering is a predominant helicopter rotor design,

¹⁷ G.H. Saunders, Dynamics of Helicopter Flight (New York, Wiley and Sons, 1974) 59

¹⁸ C.D.Perkins, R.E. Hage, Airplane Performance Stability and Control (New York, Wiley and Sons) 19

¹⁹ R.W. Prouty, Helicopter Performance, Stability, and Control (Malibar Florida: Robert E. Krieger, 1990) 181-182

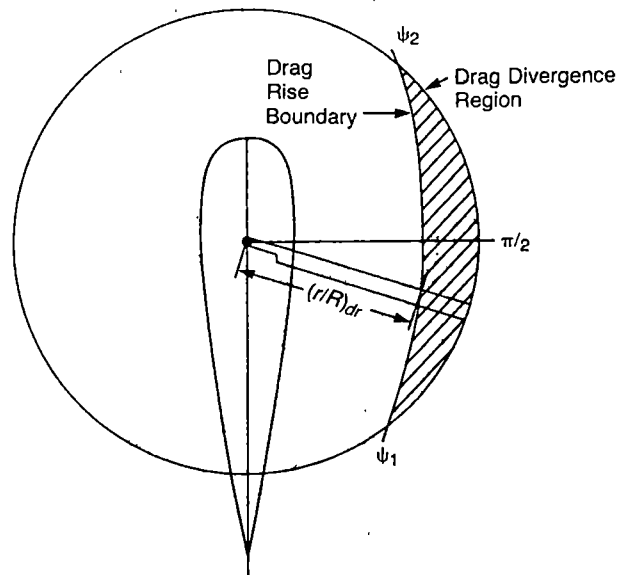


Figure 11 Drag Rise Boundary and Divergence Region

Source: R.W. Prouty, Helicopter Performance, Stability, and Control (Malibar Florida: Robert E. Krieger, 1990) 182

it follows that there is a finite forward helicopter speed limit. Because the retreating blade angle of attack is directly related to the helicopter forward speed, a convenient manner to express forward flight performance is advance ratio. Advance ratio (μ) is the ratio of forward speed to rotor blade tip rotational speed. It is a convenient method for expressing the forward helicopter aircraft speed. Additional factors of increased aircraft weight, decreased air density, and decreased rotor rotational speed are factors that require the retreating blade to increase its angle of attack to the point where it stalls and there is insufficient lift to prevent a helicopter rolling moment toward the retreating side. The factors of weight, density, and rotor speed are expressed through the use of non-dimensional coefficients. Considering the previously developed equation 16, and

including momentum theory and blade flapping considerations the following equations were derived.^{20,21}

$$dT \approx \frac{1}{2} \rho a (\theta U_T^2 + U_P U_T) c dr \quad (16)$$

Integrating equation 16 along the blade radius yields the thrust per blade:

$$T_b = \int_0^R \frac{dT}{dr} dr \quad (18)$$

The total thrust produced by a rotor of b blades is then found as:

$$T = b T_b \quad (19)$$

Define the solidity ratio, or the ratio of the total blade area to the total area of the disc that the blades circumscribe:

$$\sigma = \frac{bc}{\pi R} \quad (20)$$

²⁰ W.Z. Stepniewski/ C.N. Keys, Rotary Wing Aerodynamic, (New York, Dover Publications, 1984) 97-104

²¹ A. Gessow/G.C. Meyers Jr., Aerodynamics of the Helicopter (New York: Frederick Ungar Publishing, 1967) 180-191

Introduce a dimensionless parameter λ , inflow ratio:

$$\lambda = \frac{(V \sin \alpha - v)}{\Omega R} \quad (21)$$

a convenient method to account for rotor induced velocity (total flow through the plane perpendicular to the control axis). Introduce the tip speed ratio μ , the ratio of blade tip rotational speed to helicopter aircraft forward speed:

$$\mu = \frac{V \sin \alpha}{\Omega R} \quad (22)$$

executing substitutions indicated in reference 21 and performing the integration, results in²²:

$$T = \frac{1}{2} \rho a b c \Omega^2 R^3 \left[\frac{\theta}{3} + \frac{\mu^2 \theta}{2} + \frac{\lambda}{2} \right] \quad (23)$$

Reference 23 defines thrust coefficients (referred to total blade area) as:²³

$$c_t = \frac{T}{\rho (b c R) (\Omega R)^2} \quad (24)$$

²² Gessow/G.C. Meyers Jr., Aerodynamics of the Helicopter (New York: Frederick Ungar Publishing, 1967) 185

based on the effective blade area $A_b = bcR$, and on the rotor disc area, $A_d = \pi R^2$:

$$C_T = \frac{T}{\rho(\pi R^2)(\Omega R)^2} = \sigma c_t \quad (25)$$

With this, equation 23 can be written in coefficient form as:^{22,24}

$$C_T = \frac{1}{2} a \sigma \left[\frac{\theta}{3} + \frac{\mu^2 \theta}{2} + \frac{\lambda}{2} \right] \quad (26)$$

It should be noted that this equation is an expression of the thrust of an untwisted, untapered rotor (no variation of pitch angle or chord with radius was assumed before integrating).

To validate the intuitive conclusion that high speed flight will result in retreating blade stall, early experimenters measured in flight stall by mounting a camera on the rotor hub and observing wool tufts mounted on the rotor blade at different span wise stations. The experimenters considered a blade section stalled when the yarn tufts

²³ W.Z. Stepniewski/ C.N. Keys, Rotary Wing Aerodynamic, (New York, Dover Publications, 1984)101

²⁴ P. Solies, Notes to Author, University of Tennessee Space Institute, Tullahoma TN, May 2001

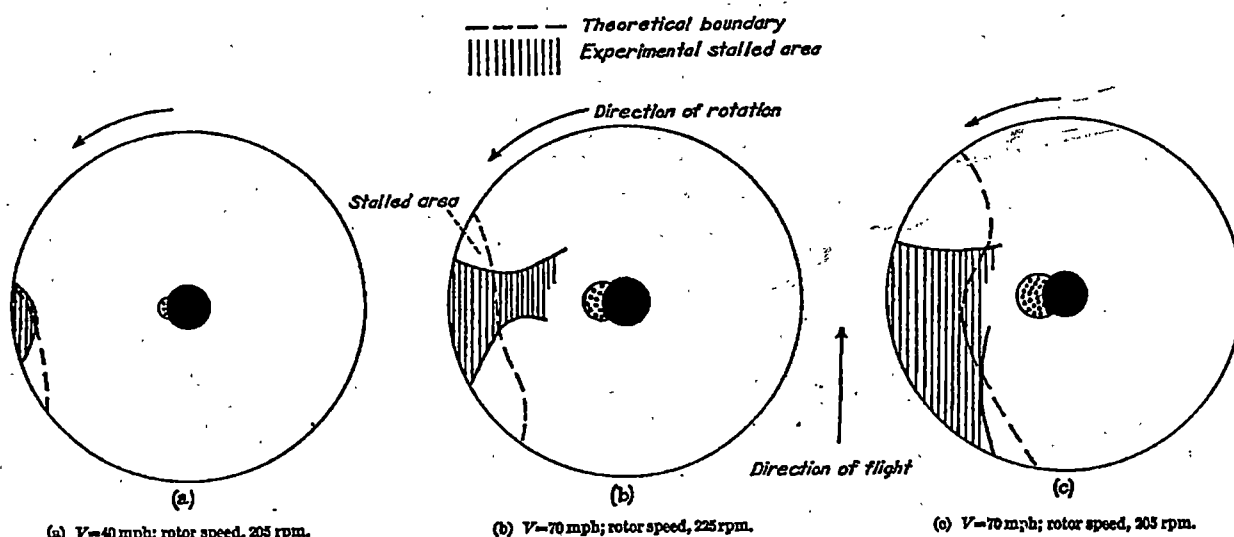


Figure 12 Comparison between Theoretical and Experimental Stalled Areas

Source: F.B. Gustafson, C.G. Meyers, Stalling of Helicopter Blades, NACA Technical Report Number 840, 15 April 1946

indicated reversed or burbled airflow over the rear 30 percent of the blade. Figure 12 shows a representation of the results of the observations, superimposed on the calculated theoretical stall boundaries. A comparison of this figure with figure 8 (page 18) shows the growth of the stalled areas on the retreating blade sections as forward aircraft speed increases, or rotor blade rotation speed is decreased. An excerpt from the report of these test results which produced the comparisons of figure 12 expresses the following: "The angle of attack reached at the retreating tip of the blade depends upon three variables shown ... It depends first upon the advance ratio μ , that is, the ratio of aircraft forward speed to rotor rotational speed. It also depends on the ratio of the thrust coefficient to the solidity C_T/σ (which is a measure of the mean blade lift coefficient) for, as the angles of attack all around the disk increase, the angle of attack of the retreating tip increases.

For a given helicopter, this quantity may be increased by increasing the weight, or by decreasing rotational speed. It will also be increased by operating at low density (high altitudes). For efficient operation this mean lift coefficient should be kept reasonably high. Finally, the angle of attack at the retreating tip depends upon the ratio of parasite drag to lift; that is the degree the thrust vector must tilt forward to overcome the drag of the fuselage.”²⁵

Dynamic Lift and Stall

The author has examined the effects of two different aspects of high speed helicopter forward flight, high drag divergence as the advancing blade approaches the speed of sound, and retreating blade stall as the angle of attack on the retreating blade exceeds the static stall limit. The importance of these effects, and how they can work in tandem on a helicopter is explained by Mr. Prouty as follows;

“Helicopters are different from airplanes. ... When the wing of an airplane stalls, the important effect is the loss of lift (especially if one wing stalls before the other). The accompanying nose-down pitching moment is only a minor effect. On the other hand, when a rotor blade stalls, the important effect *is* the nose-down pitching moment.

Similarly, when the wing of an airplane flies at a high enough Mach number to compress the air in front of it, the important effect is the rapid increase in drag. The accompanying nose-down pitching moment is only a minor effect. On the other hand, when a rotor blade experiences compressibility, the important effect *is*, again, the nose-down pitching moment. The primary reason for the importance of pitching moments in the case of the

²⁵ F.B. Gustafson, C.G. Meyers, Stalling of Helicopter Blades, NACA Technical Report Number 840, 15 April 1946, 108

helicopter is the torsional flexibility of the long skinny rotor blades and the ease with which they can be twisted into odd shapes.”²⁶

Stall pitching effects and blade torsional flexibility can also contribute to another phenomena that puzzled early helicopter aerodynamicists. They found that in wind tunnel tests, it was possible to generate higher lift coefficients than could be expected based on the airfoil's static stall angle of attack. The reason behind this phenomena was that if the angle of attack was changed quickly, or, at least quicker than the time it took for the separation point to travel along the airfoil to the point of stall, it could generate higher than expected lift (critically dependent on airfoil section and type of stall). This is the same phenomena that allows high performance fighter aircraft to momentarily achieve up to 20% more lift in a rapid pull up. The same phenomena allows helicopter blades to achieve lift overshoots well beyond static stall limits and is the reason that rotors in flight do not lose thrust due to lift stall.²⁷ Figure 13 shows an example of the lift characteristics of an oscillating airfoil that can cause a dramatic increase in lift.

Negative Damping and Structural Issues

Another result of the rapid oscillation (of the helicopter blade) from stall back to nose-down and no stall conditions arise from the characteristics of the helicopter blades individual torsional elasticity and the moment in this cycle of constant pitch movement

²⁶ R.W. Prouty, Helicopter Aerodynamics (Peoria: PJS Publications, 1985) 24-26

²⁷ R.W. Prouty, Helicopter Aerodynamics (Peoria: PJS Publications, 1985) 26

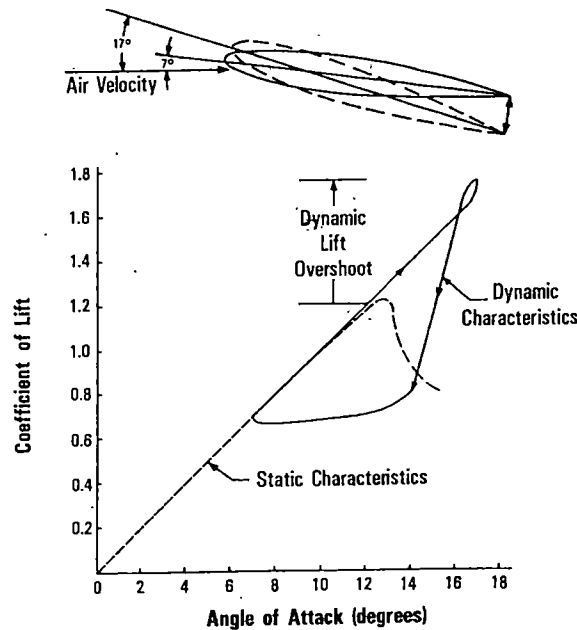


Figure 13 Lift Characteristics of an Oscillating Airfoil
Source: R.W. Prouty, Helicopter Aerodynamics (Peoria: PJS Publications, 1985) 26

where stall occurs. If the nose-down moment occurs when the blade is pitching down, and the nose-up moment when the blade is twisting up, this will cause a negative damping effect where the blade will extract energy from the air and cause the blade torsional effect to amplify. This effect may cause excessive structural loads to be transmitted through the blades to the pitch links and supporting structure.

Mr. Stepniewski discusses this during level flight: "Level-flight rotor limits are encountered because of the development of stall on the retreating blade ($240^\circ < \psi < 300^\circ$). As the blade enters and leaves this region, the section pitching moment decreases abruptly due to moment stall of the airfoil section, and thus induces the aeroelastic phenomena known as stall flutter."²⁸ Mr. Prouty discusses the same situation: "For example, it is common to generate high oscillating loads in the control system when the

blade goes into stall on the retreating side. The resultant change in aerodynamic pitching moment can twist the blade nose down to an unstalled condition, only to have it spring back into stall. Depending on the torsional natural frequency of the blade and control system, each blade may go through several cycles of this oscillation-sometimes called *stall flutter* while passing through the retreating side.”²⁹

Literature Review Summary

The preceding literature review focussed primarily on rudimentary helicopter blade theory analysis, contrasting the similarities and differences from airplane analysis by developing airplane lift, drag, and stall concepts, and then applying them to the helicopter blade. This included developing the concept of blade flapping as a method to equalize rotor system lift. The literature review also described the problems of supersonic compressible flow, retreating blade stall and the outcomes of the pitching moments that result from these effects. It is this author's conjecture that the primary effect that resulted in the damage experienced by the H-46 aircraft test, which is the subject of this thesis, was due to the aft rotor system experiencing negative damping effects or stall flutter. This phenomenon reduced the effectiveness of the helicopter flight controls in the pitch axis, causing the pilot to apply excessive forward stick input. At some point, the aircraft stopped experiencing aft rotor system stall and full effectiveness was returned to the aft rotor controls, which resulted in an abrupt aircraft dive. The

²⁸ W.Z. Stepniewski/ C.N. Keys, Rotary Wing Aerodynamic, (New York, Dover Publications, 1984) 103

²⁹ R.W. Prouty, Helicopter Performance, Stability, and Control (Malibar Florida: Robert E. Krieger, 1990) 345

recovery from this dive and a resultant pitch oscillation caused some subsequent aircraft damage. Although the aircraft was not instrumented to record torsional blade aerodynamic loads at specific azimuth angles, the author believes the load record and physical damage evidence support this conclusion. This is explored further in chapter IV in the section on results.

III

Methodology

Structural Testing

The accepted industry practice for the introduction of new aircraft rotor blades, rotor hubs, control links, power output shafting and transmissions, is an in-flight structural load survey. This verifies that the calculated design loads that predicated the design of these parts were, in fact, valid. This takes on an especially significant importance in helicopter applications. This importance is due to the way the rapid cyclical loads are transmitted in a helicopter. What may be an acceptable load during one or two cycles of use becomes an unacceptable load when subjected to the rapid number of cycles that occur due to the feathering, flapping, and the rotation of each helicopter blade. These cycles are what cause fatigue of the components and result in fatigue failure, that is, loading below the yield and ultimate limit of the component material but above the stress limit which ultimately results in material failure. The determination of these cyclical loads that a helicopter experiences, is critical in determining the component replacement times (CRT) for dynamic components. The combination of cyclical loads, and the requirement to replace key components when calculated load limits are reached, has significant impact on helicopter availability rates and their cost of maintenance. A typical analysis is based on what is termed an S-N Curve methodology. The S-N curve is illustrated in figure 14.

The S-N Curve methodology and the way in which it determines the method for flight test verification was summed up succinctly by Mr. Kracinovich in his lecture on structural testing as follows;

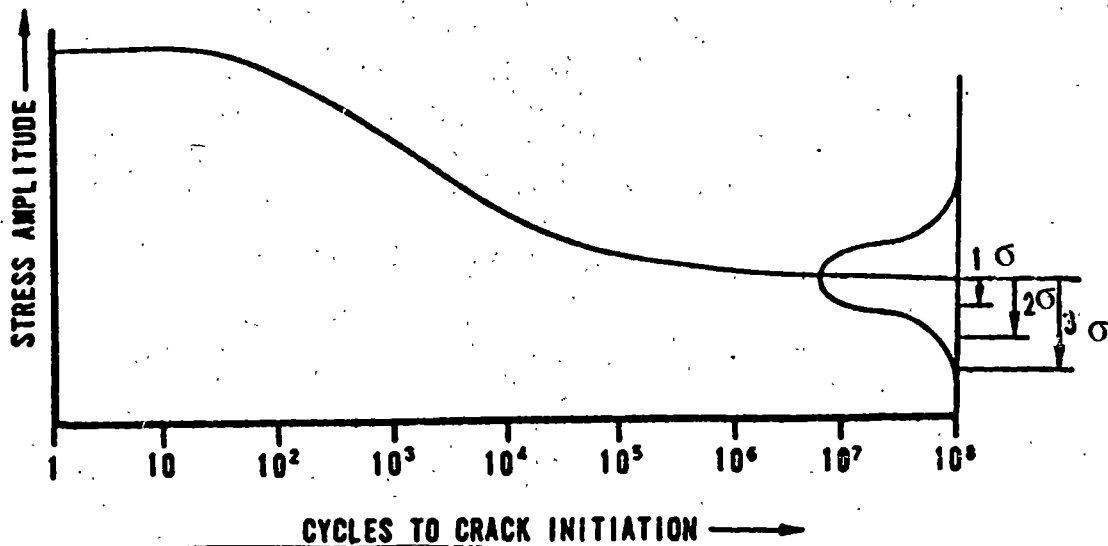


Figure 14 S-N (Stress vs. Number of Cycles) Curve

Source: S. Kraciovich, Structural Flight Testing Lecture U.S. Naval Test Pilot School, NAS Patuxent River MD, October 1995

- "1) S-N curves exist for most aerospace metals.
- 2) These (published) material S-N curves are used mainly for analysis, not for component replacement times in dynamic components.
- 3) A few (2 or 3) individual components are tested in a fixture for expected flight loading, based on analysis.
- 4) Initial flight test limits are established (Oscillatory limits) based on initial laboratory testing with an appropriate safety factor applied.
- 5) Laboratory testing continues based upon loads experienced during flight test and at a large enough spread of points to "flesh out" the S-N curve.
- 6) The final S-N curve used is a compromise between the practical number of specimens to be tested, experience with the material and from the definition of "run out", the

level of actual load experience, and the maximum load capability of the testing machine.

(Quote continued)

-Moral #2-Always assume the model is wrong unless proved otherwise.

-Corollary to Moral #2- Good flight testers ALWAYS assume that nothing works as expected the first time (note: bad flight testers have a way of weeding themselves out of the business, especially pilots)"³⁰

Test and Test Conditions

The prerequisite to bring the H-46 helicopter test article, which is the subject of this thesis, to the NAS Patuxent River complex was essential to the test team. This requirement allowed the government customer the means to ensure the loads applied to the aircraft were truly representative of fleet service. This capability is available at the Patuxent River complex due to the line of sight telemetry antennae, the tactical low-level terrain flight routes, slope-landing areas, and confined area landing sites. This capability was not available at the Boeing Wilmington Delaware facility. In addition to the mission representative maneuvers, the test plan included a survey of air to air tactical maneuvering. This effort was to collect loads data significant to supporting emergent pilot training requirements for aerial combat maneuvering flight sorties.

In total, the testing included approximately 98 flight hours, of which 38 were flown at the Wilmington, Delaware flight test facility, and 60 flown at the NAS Patuxent River MD complex. The following tasks were part of the testing:

- 1) Rotor Brake release and rotor acceleration
- 2) Steady state rotor operation

- 3) Ground Taxi
- 4) Normal/Maximum performance/Running/Shipboard/Jump takeoff
- 5) Normal/Steep Approach
- 6) Normal/No Hover/Running Landings
- 7) Shipboard/Confined Area Landings
- 8) Sloped Landings
- 9) Forward/Sideward/Rearward Flight
- 10) Level Flight/Dives/Climbs/Turns
- 11) Climbing/Descending Turns and Forward Flight
- 12) Service Ceiling definition
- 13) Normal/Maximum rate acceleration/decelerations
- 14) Autorotations
- 15) Vertical Replenishment/Low Level Terrain Following/Evasive Maneuvering
- 16) Engineering control inputs

These maneuvers, performed at Low, Medium and High load conditions (21,800-24,300 pounds gross weight with mid and extreme forward and aft center of gravity points) comprised the planned sum of military required test points and contract data requirements considered sufficient for determining that Boeing had satisfied the

³⁰ S. Kracinovich, Structural Flight Testing Lecture U.S. Naval Test Pilot School, NAS Patuxent River MD, October, 1995.

requirements of their contract. These points also verified the suitability of the new dynamic components for mission use within the fleet.

IV

Results and Evaluation

The results of the tests indicated that the loads experienced were generally consistent with the pre-DCU rotor systems, and that within the scope of the tests, the DCU system was a satisfactory replacement as an upgrade for the legacy rotor system. Detailed comparisons were presented via Boeing fatigue reports and were consistent with (and included) government data.

An ancillary objective of the test was to obtain data and loads generation during evasive maneuvering flight, and to allow aircraft operation at an increased operating envelope where determined appropriate as a result of the testing. This included operation at up to 60 degrees angle of bank turn, higher rearward and sideward flight envelopes, and the performance of evasive flight maneuvers. The pursuit of these goals led to the discovery that operation near 60 degrees angle of bank under certain loading conditions could cause severe blade stall, resulting in unpredictable helicopter response and cause severe structural loads that might lead to catastrophic failure. The event that led to this discovery occurred midway through the testing program and at the start of the last half testing at the Patuxent River complex. The following discussion sets the stage for the event, describes the aircraft response, the team reaction to it, and the recommendations that followed.

Project Status Prior to Stall Episode

Approximately thirty-eight flight hours of testing had been previously accomplished at the Boeing Wilmington Test Facility. Plans to ferry the aircraft to the

NAS Patuxent River Air Station (Pax) were accelerated to permit a planned shutdown of the Wilmington Facility. Shortly before aircraft transfer, the team held a data review with NAVAIR and Boeing engineers and flight test representatives. This review included a Boeing pilot debrief and summary review of the Boeing-flown test matrix. The aircraft departed Wilmington and arrived at NAS Patuxent River in March of 1996. While the government was performing routine maintenance acceptance inspections at NAS Patuxent, Boeing discovered some quality issues with the forward swashplate caused by a vendor which halted testing for about 4 months. During this time Boeing conducted a thorough review of all DCU parts and requested Pax halt planned evaluation flights in the interim. To keep the test program on track, the program office substituted a pre-DCU stationary forward swashplate in the forward swashplate assembly. The forward swashplate assembly (1/2 DCU and 1/2 pre-DCU) was delivered from Naval Depot Cherry Point NC in June, 1996. The test team received NAVAIR flight clearance for the assembly in July of 1996. Post maintenance functional check flights and instrumentation re-installation and calibration were accomplished during August and September of 1996.

Critical Parameters, The Cruise Guide Indicator

The flight included monitoring of defined "critical parameter" data by real time telemetry processing. The critical parameters were composed primarily of the mechanical links, blade and fuselage aerostructure, and dynamic components where aerodynamic flight loads were transmitted to mechanical structural loads. In-flight verification of the loads on these items was the purpose of this test. One critical parameter, that also defined

the specific loading, which indicated blade stall in the H-46, is the Cruise Guide Indicator. The Cruise Guide Indicator (CGI) provides an indication of dynamic loading of the rotor and structural components during flight. The CGI displays stress loading measured by strain gauges mounted on the forward and aft longitudinal rigid links. The links are located between the rotor system and the upper dual boost actuators. System components are: the longitudinal rigid links, a signal conditioner, a signal processor, and a cockpit cruise guide indicator gauge. This instrument was added to the aircraft during its initial design phase after positive pilot comments for its utility. The purpose for including it into the pilot instrument panel was to obtain real time indication of damaging loads. Retreating blade stall can be encountered when operating the helicopter at high airspeeds, high gross weight, high altitudes, and in turbulence, or when maneuvering in a combination of the above conditions. Below stall, lift is proportional to blade angle of attack and the relative wind velocity over the blade. On the retreating blade, the resultant velocity is equal to the rotational velocity, minus the forward speed of the helicopter. Therefore, as the airspeed is increased, a corresponding increase in the retreating blade angle is required to maintain lift. When the retreating blade angle of attack reaches the stall point, blade stall occurs. These phenomena may also be exasperated by drag rise on the advancing blade during high-speed flight. The net result during a stall sequence is that extremely large aerodynamic loads are transmitted to various rotating and stationary components as a structural load. Constant application of loads of this magnitude result in material failure that would cause loss of aircraft control. Retreating blade stall is the primary factor limiting forward speed for all conventionally designed helicopters. Most helicopters have an elaborate system of charts that define the allowable maximum

operating speed. The reason these charts are so elaborate is due to the fact that forward airspeed limits are so dependent on aircraft weight, operating altitude, and angle of bank or maneuvering. The H-46 CGI employs two strain gauges on the forward and aft rotor longitudinal rigid pitch links that measure structural loads directly. The airspeed limit in the operator manual is 145 KIAS or CGI limit, whichever is lower. The only time that pilots need to utilize extensive pre-flight planning charts for determining blade stall conditions is if the CGI is inoperative. There is only one cockpit CGI gauge located on the pilot (right) side of the instrument panel. The gauge is presented in figure 15. Steady state operation in the green arc is allowed with occasional spikes in the yellow (avoid) area. Steady operation in the avoid area is not allowed and requires decreasing forward airspeed or reducing maneuver severity to resume operation in the green. Operation in the red area is prohibited, and is an indication of developed stall requiring immediate increase of rotor rpm, reduction of collective pitch, reduction of airspeed, or, if maneuvering, a decrease in

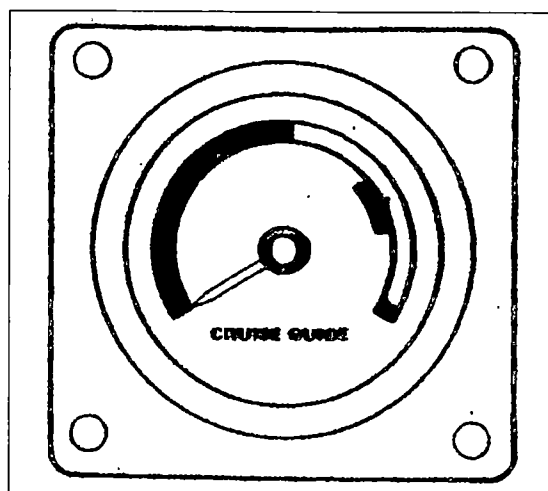


Figure 15 Cruise Guide Indicator

Source: NATOPS FLIGHT MANUAL, Navy Model CH-46E (SR&M) Helicopter, March 1992, I-2-71

the severity of the maneuver.¹ The operator manual does not correlate gauge operating area with aircraft loads.

The Flight

The first flight at Patuxent was flown on 11 October 1996. This consisted of 1.3 flight hours in the Pax local flying area under day VMC conditions. The tests were comprised of engineering flying qualities including qualitative evaluations. The briefed sequence of events included: vertical take off to hover, maximum continuous power climbs at 70 KIAS to the density altitude of 2000 ft., acceleration and steady state flight at 98 KIAS and 2000 ft. H_d, longitudinal doublets to 2 inches cyclic displacement at 98 KIAS, followed by constant altitude and airspeed turns from 15° to 60° angle of bank. During a sixty degree angle of bank right level turn at 22,800 pounds loaded near the forward center of gravity limit the test aircraft, CH-46E Buno 153355, experienced incipient blade stall as indicated by increased airframe vibration and a departure of the cruise guide indicator (CGI) from the edge of the green band area to the red prohibited region. During the recovery sequence, the aircraft experienced a pitch oscillation episode that resulted in exceeding the peak static and oscillatory limits on various components in addition to aircraft damage.

¹ NATOPS FLIGHT MANUAL, Navy Model CH-46E (SR&M) Helicopter, March 1992, I-2-71

Excerpt from Flight Daily

The sequence of events was subsequently described in the flight daily:

“Constant altitude/airspeed turns were uneventful until a 60° right turn was executed. After stabilizing in the turn with 100% torque set, the pilot at the controls (PAC) noticed the CGI progress from the green zone into the red prohibited zone where the needle pegged. The PAC initiated immediate left lateral cyclic to return the aircraft to a level attitude and reduce the severity of the maneuver. Upon reaching wings level attitude, the aircraft suddenly, and without any conscious pilot input started to nose down causing the crew to experience a less than one “g” condition. At the same time as the aircraft’s nose dipped, a shudder in the airframe was felt by the entire crew. The aircrew was concerned with aircraft integrity at this point (especially after the project engineer radioed from the TM room “what was that!”). The PAC commenced an approach to a nearby field, the project engineer radioed that the aircraft appeared normal, but a few parameters were not working and that it would be prudent to return to base (RTB). After RTB and landing at Pax, a post flight revealed the fairing on the aft rotor was gouged.”²

² Maj. E. Treworgy, DCU PROJECT Flight Report for Flight X-57. Attack Assault Branch NRWATS, NAS Patuxent River MD 1996.

Results

The test plan specified that any excursion of specified oscillatory, warning, static, or peak limits would require a cessation of testing with all data sent to Boeing for engineering investigation and recommendation prior to resuming flight test. A quick review of the data by the Pax test team while the aircraft was in flight revealed that empirical retreating blade tip speeds (ERITS) data, CGI, and forward/aft collective pitch link loads indicated blade stall. In addition, the following excursions were noted and confirmed by further Boeing analysis: forward blade lead/lag angles, forward/aft rotor shaft torque's, forward/aft longitudinal rigid link loads, aft collective pitch link loads, and aft drive scissors loads exceeded upper oscillatory load limits. Analysis also showed that the aft longitudinal link load exceeded peak static limits. A detailed aircraft inspection revealed that in addition to the aft fairing scoring mentioned in the flight daily (figures 16 and 18), there was: evidence of scoring on the aft (rotating scissors), drive arm (figure 19), and a crack on the front frame of the number one engine, figure 20 (previously repaired with a weld).

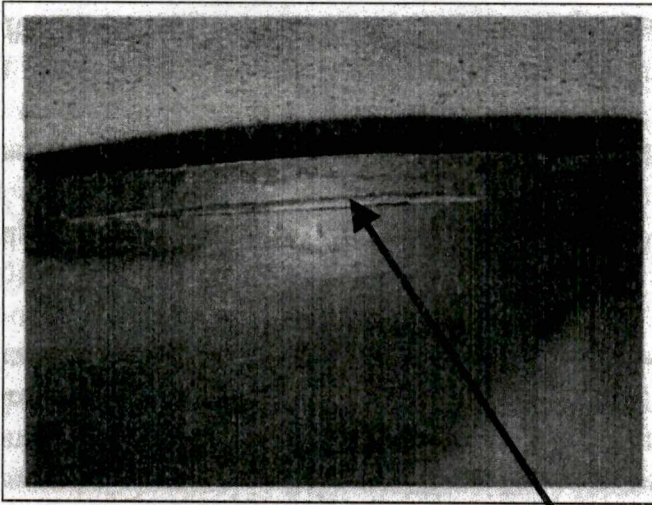


Figure 16 Right Side Aft Fairing



Figure 17 Test Aircraft

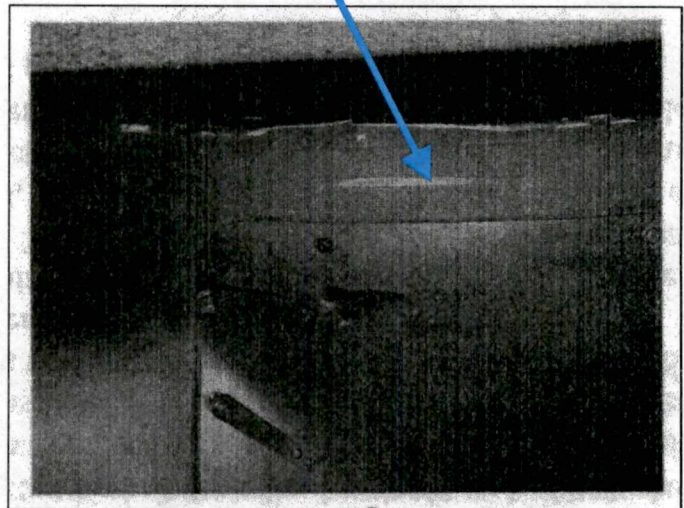


Figure 18 Left side Aft Fairing

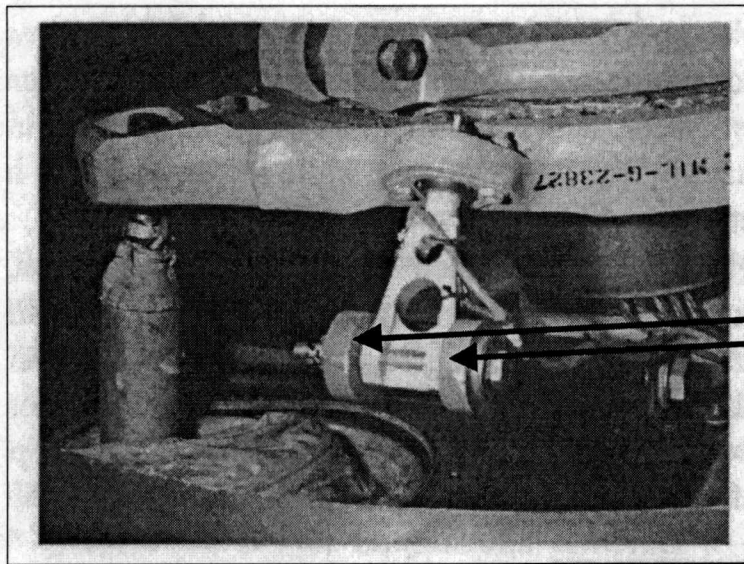


Figure 19 Aft Drive Arm

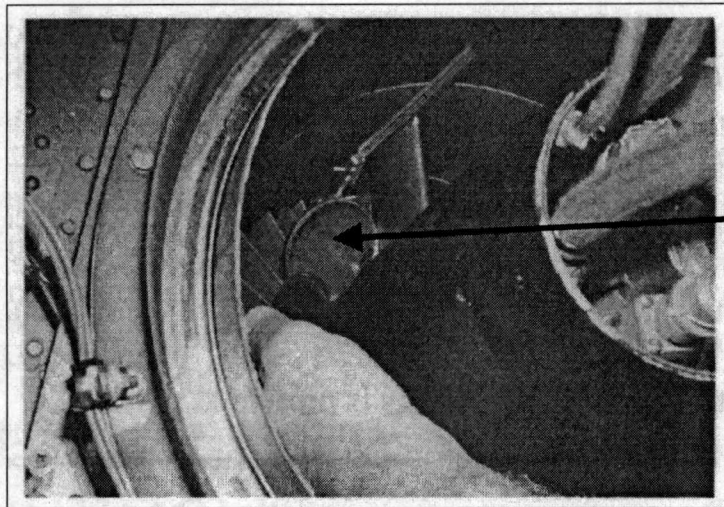


Figure 20 Front Frame Area, Number 1 Engine

Team Immediate Response

In addition to sending the airborne data tapes to Boeing, the team performed a detailed non-destructive inspection of all components involved in peak and oscillatory limit excursions. The aft longitudinal link, aft drive arm, and number one engine were removed and replaced. The team amended the test plan to remove 60° angle of bank points (current NATOPS limit is 45°) and added blade stall precautionary/recovery procedures to the hazard condition table (figure 21).

Hazard Condition	Cause	Effect	Hazard Level	Risk Assessment	Precautionary Measures	Corrective Action (as required)
Blade Stall/Accelerated Blade Stall	Increased load factors at high airspeed and density altitude flight to include high degree AOB level turns, Symmetric pullouts, rolling pullouts, and climbing turns	Attitude rate changes, limit excursions, with possible structural damage	IIIB	Moderate	Pilots to brief stall indications and aircraft response with recovery techniques	Pilot will reduce load by reducing airspeed, angle of bank severity of maneuver, and lowering collective

Figure 21 Corrected Blade Stall Test Hazard Analysis Insert

Different Interpretation of Flight Test Procedure

When interviewing the Boeing pilots who had conducted the first period of testing at the Wilmington Facility, it was discovered that their interpretation of the flight procedures for the level turn portion of the test plan was different than that of the Patuxent River test team. At each test point during level flight turns, the Boeing team would roll back to a wings level condition. In other words, the team would roll into a 10° angle of bank turn, collect the required data, and then roll back to straight and level. From straight and level, the Boeing team would commence to the next test point. The Patuxent team executed their points differently, from the ten degrees angle of bank turn, the team would commence the next test point of 20 degrees angle of bank. This variation of procedure would make the Patuxent team much more susceptible to flying into its own air wake disturbance. Based on steady-state CGI indications just prior to the stall event, the aircraft was flying with the CGI indicating loads close to the avoid area. Flying the aircraft into its own air wake disturbance would cause circumstances resulting in a more severe stall as was experienced by the Pax test crew. The team felt this was the most likely cause for encountering stall conditions in what should have been a more benign stall environment (indicated by the Boeing data). In subsequent testing the team ensured that in addition to limiting angles of bank to 45° for level turns, each level turn data point entry would commence from wings level.

Aircraft C.G. Differences Can Significantly Alter Aircraft Response

Boeing had previously flown the test points the team had flown at the Wilmington test facility at the same weight and density altitude, but with a loading near the aft center of gravity limit. Boeing reached aircraft stall at 60° angle of bank and airspeed of 140 KIAS. The Pax test team achieved stall at 98 KIAS, significantly below the Boeing achieved airspeeds. The most significant difference between aircraft configurations was the location of the aircraft center of gravity, which was varied to the two extreme end points of the manufacturer-approved band at the planned weight.

Aircraft Dynamics and Stall Response

The aircrew had previously experienced aircraft blade stall only during straight and level flight. Aircraft blade stall characteristics in straight and level flight are relatively benign. The onset of incipient blade stall is indicated by an increase in one per revolution vibrations (approx. 3 Hz) and the progression of the needle on the CGI gauge from the green through the yellow avoid region to the red prohibited region. During the conduct of the test flight in the right turn, the aircraft exhibited no abnormal control responses, but the CGI and airframe vibrations indicated stall. The pilots initially recovered from the stalled condition by rolling the wings level which took approximately 3 seconds (the operator manual directs reducing maneuver severity). The pilot delayed reducing collective pitch for approximately of 2-3 seconds at this point. In a constant-altitude constant-angle-of-bank steady-state turn, the turn rate is directly related to pitch

rate, and the cyclic control position was slightly aft of neutral trimmed position in the turn. As the aircraft rolled to the wings level orientation, the pilot translated the cyclic to maintain a level fuselage attitude. It is the author's conjecture the stalled state of the aft rotor in comparison to the forward rotor reduced longitudinal effectiveness, causing additional forward cyclic to be applied. In addition, in a tandem design, forward cyclic increases the aft rotor system collective pitch, which increases stall. When the aircraft returned to wings level flight, and the aft rotor system was no longer in the disturbed airflow of the forward rotor system (or rotor wake), the stall condition on the aft rotor abruptly ended and longitudinal control effectiveness abruptly returned. This effect caused a violent nose down aircraft pitch. The aircraft went from a positive 1.8 "g" flight to a positive 0.3 "g". Some ensuing damage probably occurred during the recovery, which included an episode of pitch oscillation. The pitch oscillation phenomenon is a documented deficiency that has been known to occur during rapid longitudinal cyclic inputs and is exasperated in part due to the slow response rate of the differential air speed hold (DASH). A graphical representation of this sequence of events is presented in figure 22. The graphical representation of events shown in Figure 22 were deduced from the data collected with the real time telemetry, and from aircrew interviews of the event. By examining the data, which, in addition to vehicle state parameters and rates, included control positions and structural loads, a fault tree of the initial stall event was developed.

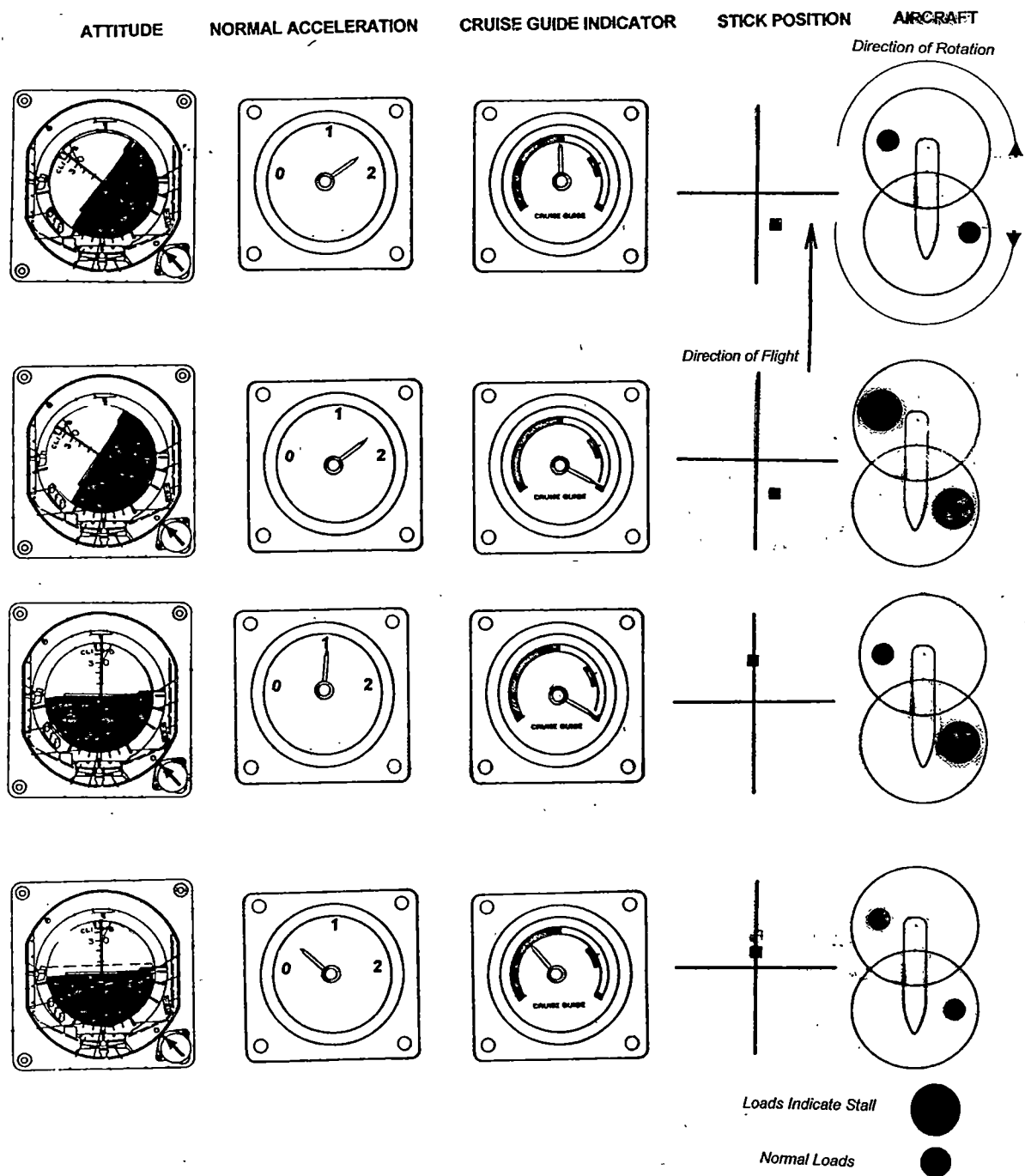


Figure 22 Aircraft Blade Stall Sequence

The fault tree diagram is presented in figure 23.

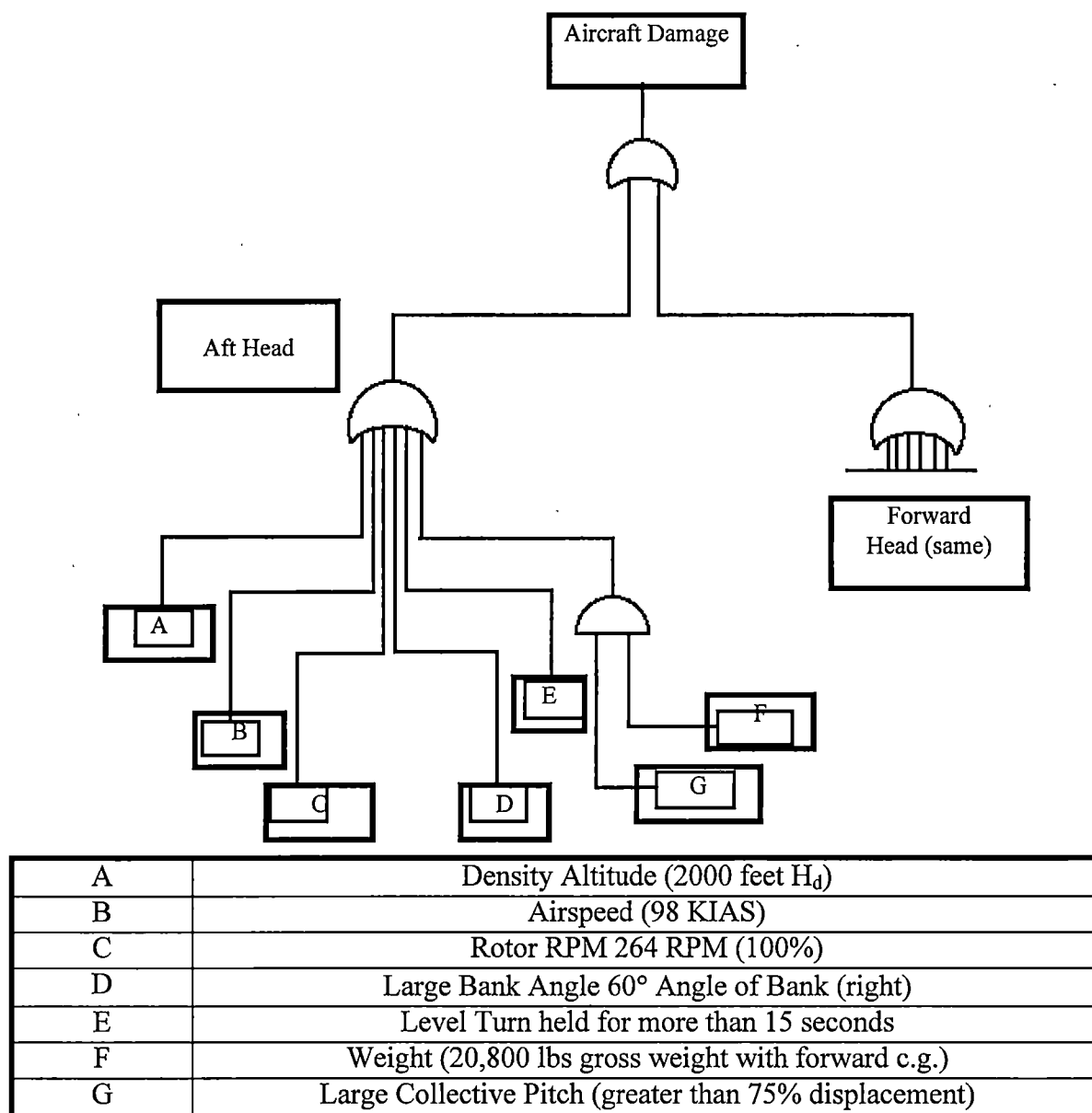


Figure 23 Fault Tree Analysis

Differences in Operators Manuals

During a post flight debrief of the aircraft blade stall response, the aircrew were discussing the pitch characteristics with the test team. It was during this discussion that it was discovered that the Navy and Marine Corps operator manuals had significant

differences when discussing aircraft blade stall characteristics. Although the aircraft are different models: the Navy designation being H-46D and the Marine designation H-46E, the model differences relate mainly to engines and mission equipment and have nothing to do with aircraft flying qualities. The phenomenon the pilots had experienced was more fully described, along with suggested recovery techniques, in the Navy operator manual. The Marine operator manual discusses blade stall with the following paragraph;

“Retreating blade stall can be encountered when operating the helicopter at high airspeeds, at high gross weight, at high altitudes, in turbulence, or when maneuvering in a combination of the above conditions. Lift is proportional to blade angle of attack and relative wind velocity over the blade. On the retreating blade, the resultant velocity is equal to the rotational velocity, minus the forward speed of the helicopter. Therefore, as airspeed is increased, a corresponding increase in retreating blade angle is required to maintain lift. When the retreating blade angle of attack reaches the stall point, blade stall occurs.

Blade stall is indicated by a progressive increase of 1-per-revolution and 3-per-revolution vibrations resulting in a CGI (cruise guide indication) in the red band. The severity of the vibrations will increase as the helicopter is flown further into blade stall; however, no loss of control will result within the airspeed envelope presented in chapter 4. Prolonged operation in blade stall should be avoided because of high stress loads imposed on the dynamic components.

Recovery from blade stall is accomplished by executing one or more of the following: increase rotor rpm, reduce collective pitch, reduce airspeed, or, if maneuvering, decrease the severity of the maneuver.”³³

Compare this description with the description from the Navy operator manual, which for the purposes of this discussion is essentially the same aircraft:

“Blade stall is a loss of lift that normally occurs on the aft rotor in tandem designs. The aft head stalls first because of the disturbed rotor wash from the forward head and the fact that the aft head supports more than half the weight of the aircraft. Blade stall is more likely to occur at high airspeeds, gross weights or altitudes, in turbulence, or when maneuvering under a combination of the above conditions. CGI indications directly reflect the severity of blade stall and the resulting flight control strain loads. The stall generates a vibration that is felt by the pilot through the airframe and controls as a progressive one-per-revolution and three-per-revolution.

Since the aft rotor system stalls first and drops, the pilot may perceive this as a pitchup maneuver. Attempting to push the nose over with cyclic will be less effective than normal and may increase the pitchup since increased blade pitch on the aft rotor will aggravate the stall. A temporary reduction of 1 to 2 inches of collective will quickly alleviate the stall and provide improved longitudinal response. No loss of control will occur within the airspeed limitation and load factor (g limit and angle of bank) envelope of chapter 4.

Figure 4-11 is only valid for straight and level flight. For flight conditions other than straight and level, reduce the maximum airspeed accordingly. Prolonged operation in blade stall must be avoided because of the high loads imposed on the dynamic components. If blade stall occurs, eliminate it by doing one or more of the following:

1. Decrease severity of maneuver.
2. Lower collective.
3. Reduce airspeed.
4. Increase rotor speed.
5. If possible, descend to a lower

³³ NATOPS FLIGHT MANUAL, Navy Model CH-46E Helicopter, Nov 1994

altitude at approximately 500 fpm.”³⁴

The aircrew had been briefed on blade stall, and blade stall recovery techniques using the recovery procedures of the Marine operator manual, which was the model aircraft undergoing test. The initial roll out of the turn to straight and level flight was performed to decrease the maneuver severity. In the Marine blade stall description paragraph it could be argued that this activity should rectify the condition independently of other actions. Even though the Navy operators manual stresses the importance of lowering the collective, it lists decreasing maneuver severity first in the list of recommendations, giving it a primacy over the other actions. The problem encountered in flight was that by performing these actions in a linear versus a simultaneous fashion the pilot became so involved in reacting to the pitch oscillation, it delayed him in lowering the collective. The collective was only lowered after the oscillations had subsided and the damage had occurred. The collective pitch setting also exasperated the conditions experienced. The test plan covered background, purposes, and flight procedures to be followed during the test and included a hazard analysis table. This table examines failure modes to include test, ground, and flight procedures to be followed to reduce the likelihood of each specified failure and procedures to follow in the event the failure should occur. As the aircraft response was uncharacteristic, it was not covered in the hazard analysis table; this hazard was subsequently added to the precautionary/recovery procedures portion of the hazard condition table as was illustrated in Figure 21 (page 54). Although this change allowed for the continuation of testing, the conditions for this type of damage, both immediate, and long term, were not necessarily

³⁴ NATOPS FLIGHT MANUAL, Navy Model CH-46D Helicopter, Dec 94

eliminated. An examination of the fault tree diagram (figure 23, page 59) reveals numerous other paths to failure mode. More importantly, even if visible failure mode does not occur, data indicated that it was possible to put loads on dynamic components that can lead to reaching the calculated fatigue life in a single cycle.

Pilot Training and Dynamic Stall Response

Discussing the flight during a briefing with the Boeing test pilot brought to light the trend of a relative Marine pilot unfamiliarity with the conditions and precautions that cause and/or prevent high CGI indications. It was his observation during the H-46 transition training at the Marine Fleet Refresher Squadron that this gauge was frequently ignored. He was surprised at the inattention given to this flight instrument and commented on that fact to the Instructor Pilots with whom he flew. This is probably due to the limited requirement for the H-46 to fly at maximum gross weight condition because of restrictions placed on the aircraft which, in turn, has limited the amount of pilot experience (carrying heavy loads) in the last five years. In addition, during the first design and fielding of the gauge, it was inaccurate and not considered representative of the actual flight conditions. If one of the strain gauges was inoperative, a fairly frequent instance due to the poor reliability of this item, it was disconnected and an annotation was placed in the aircraft discrepancy book. A problem with this procedure was that the cockpit gauge response would be erratic (after it was disconnected). This erratic behavior tended to mimic the needle sensitivity during normal operation, especially when the needle pegged. In addition, the small red line of the prohibited area makes it hard to

distinguish from the avoid area. Eventually pilots became desensitized to the gauge altogether. Later design changes and current test results point to the exceptional accuracy of the current gauge for initial dynamic load information, but doubtful accuracy for steady state indications between 1 and 4 seconds. It is not uncommon to find this gauge ignored.

Flying a Different Aircraft

During the last ten years the H-46 has gone through some significant changes. These changes have included: fiberglass composite rotor blade upgrades, a survivability/reparability/maintainability upgrade (including a new AFCS), and an increased stub wing fuel capacity upgrade. These upgrades greatly contributed to improving the mission capability of the aircraft over the years, but incrementally the changes have created an aircraft that may be substantially different than when initially introduced. A re-designed AFCS allows increased aircraft reliability and maintainability, but it also magnifies the effects of the slow rate of the DASH actuator. This required the flying pilot's full attention during recovery from a pitch oscillation episode and delayed the initiation of the proper blade-stall recovery technique. The addition of increased capacity fuel cells caused heavier aft weight distribution, even during configurations that place the operating c. g. toward the forward limit.

Simulator Employment

For the test team, one limiting factor was the lack of an available helicopter simulator. The only CH-46E simulators available were both inoperative for extensive upgrade modifications. An H-46D simulator was available on the West Coast, but risk and hazards analyzed during the test planning sequence did not appear high enough to justify its employment. Another issue was how well the simulator modeled the aircraft. When a simulation was flown, the CGI did not come close to modeling the actual stall response.

An important aspect of the simulator, in addition to providing the opportunity to practice flying difficult data points, is skill sustainment. The test team had another H-46 test aircraft so that sustainment was not as important of an issue.

Telemetry Room Training

The test team enjoyed the advantage of having a skilled and highly experienced engineering test team staff the real time telemetry processing room. In addition to the personnel training and assignments the lead engineer accomplished prior to the first flight, a detailed plan of required flight parameters, with pre-determined limit levels and spelled-out actions to be taken on achieving a pre-determined level, were established in the test plan. In addition, the TM room equipment had an automated limit checker warning system to assist in tracking the critical parameters. Later in the program, the test

team was pressured by the program office to discontinue the use of real time telemetry due to high associated costs. It was obvious to the team that without the benefit of a dedicated telemetry team, after the initial excitement of the aircraft response, a likely scenario was that the aircraft team might have continued testing. After the incident there was no noticeable degradation in aircraft control, characteristics, or warning indications. Momentary or even pegged excursions into the prohibited region by the CGI require immediate recovery from those flight conditions, but no special inspections or maintenance provisions.

The test was designed to put low risk points up front to allow the test team to reach maximum cohesiveness prior to the highest risk test points. The reality was that sometimes high-risk points are not discovered until they are reached, as was the case for the subject test team.

VI

Recommendations

Recommendation Basis-Task Analysis

To make full advantage of the data that was collected during these evaluations a concerted effort was made by the test team to examine the conditions of the flight that existed during the blade stall incident. This was compared to the known mission tasks required of fleet users of this particular aircraft. A high level of personal understanding of the tasks required by fleet users existed due to the prior experience of members of the team. Effort was directed to the task of avoiding the conditions of blade stall and/or executing the proper recovery techniques should the conditions occur. The recommendations closely follow the conclusions made from the task analysis. The analysis included the following steps:

1. Specification of system criterion.
2. Determining system functions.
3. Relating function to operator control.
4. Determining display information, necessary for operator activation.
5. Determine indications of control activation feedback adequacy.
6. Determining environmental effects/constraints.
7. Determining time loadings/frequency of task repetitions.
8. Determine special skill requirements.
9. Determine critical skills.
10. Link stimulus to response/Response to stimulus.³⁵

The task analysis would allow the team to ensure that, in general, the basis from which recommendations were advocated and forwarded to the PMA had taken into account:

³⁵ K. B. De Greene, Systems Psychology, (McGraw Hill, New York 1970) 109

- a) System design/evaluation-The collection of information concerning the allocation of functions between human and system components and to suggest additional task aids.
- b) Training design/evaluation- A redesign of current training programs to improve pilot recovery response.
- c) Interface design/evaluation- The collection of information on a specific task to assist in the design of the man machine interface.³⁶

Witness Marks and Maintenance Requirements

The author had previously noted during fleet tours witness marks similar to the one in figures 16 and 18. Until analysis of the test results, the author was unaware of the implication of these marks. Although the team did not have enough data to infer specific damage from specific levels of witness marks, it recommended documenting fleet post-flight stall indication witness marks and further testing and engineering analysis to determine if inspection or replacement criteria were warranted. The team also recommended implementing a cost effective Health Usage Monitoring System (HUMS) by utilizing an interface to provide the CGI signal to the Control Display and Navigation Unit via the current aircraft 1553 data bus.

Cruise Guide Indicator Issues

The CGI gauge is not always monitored due to its location and operator interface issues. Some problems with the gauge include:

Needle Sensitivity- The indicating needle sometimes varies significantly throughout its range; this behavior is very similar to the needle indication that occurred when the strain gauge was disconnected and tends to de-sensitize the operator.

Prohibited Area- The red (prohibited area) of the gauge is difficult to distinguish as being red, it is easy to misinterpret as an end scale line. During numerous flights, the gauge was monitored and there was a noted time delay after the needle pegged. The needle would remain in the red (pegged/prohibited) area for approximately 4 seconds. Even when the proper recovery procedures are applied immediately, the gauge remained pegged for up to 4 seconds. Due to this delay, the operator is unsure of the immediate effectiveness of the recovery procedures. In addition, the operator may not scan this instrument while flying a particularly demanding flight phase that requires full outside attention. If this occurs it is possible that serious structural damage, either immediate or affecting planned service life, may follow yet not be discovered. This could result in premature material failure and possible death or injury. The team felt it was important to redesign this gauge so that it provides a better indication to the actual extent and duration of excursion in the prohibited region.

Blade Stall- This phenomenon can be the result of numerous factors, as can be inferred from an examination of figure 23 (Page 59). Although the team considered the

³⁶ R. B. Stammers, A. Sheperd, Evaluation of Human Work, Task Analysis, (Taylor and Francis, London 1995) 151

mechanization of the CGI deficient from a human factor standpoint, the CGI allowed for the monitoring of a very complex system. The combination of all the factors required for blade stall without the benefit of the CGI indication would make it very difficult for the pilot to monitor and correct for this situation. This would make the CGI an excellent candidate for further automation.³⁷ One of the primary considerations the team considered was the incorporation of an aural alarm in addition to the gauge. This would cue the operator when a blade stall incident was imminent, and provide the cues necessary to implement the correct recovery procedures. In addition, the team felt that alarm mechanization should be triggered so that it would reset when the correct recovery procedures were applied. The primary recovery procedure was determined to be the lowering of the collective pitch. Current research in similar type warning displays points to the efficacy of the team recommendation,

“In virtually all conditions investigated there was a large improvement in the accuracy of monitoring performance for the two-state alarm system over no alarm system. ... This result is consistent with Thompson’s (1980) suggestion that information rich caution and warning annunciator provide relatively greater advantages as the total workload is increased.”³⁸

The ability to record the excursions for each flight would assist in determining if an appropriate service life adjustment was required. Currently, the incorporation of the military standard 1553 data bus and Communication, Navigation, Control System in the helicopter provide a cost effective means for realizing this system. This would become a relatively simple but effective health usage monitoring system. In addition, the placement

³⁷R. E. Curry E. L. Wiener, “Flight-Deck Automation: Promises and Problems” Ergonomics (1980), vol 23, no. 10, 995-1011

³⁸ B. H. Kantowitz, S. C. Kantowitz, R. D. Sorkin, “Likelihood Alarm Displays” Human Factors (1988) 30(4) 445-459

of an aural warning would enhance the ability of the pilot to perceive he was incurring damage, and assist in preventing it in the first place.

Operators Manuals

The deficiency of the Marine current operator manual for the CH-46D/E has been discussed. Although the Navy manual more clearly describes the phenomena the test team experienced, it did not address the overriding importance of lowering collective pitch when experiencing incipient blade stall and both manuals were changed to reflect the primacy of this procedure. One method for determining the relative difficulty of written text based on the number of characters, words, sentences, and word and sentence lengths is the Automated Readability Index. The ARI uses a formula to assign a reading grade level to the analyzed text. Examining both paragraphs for readability, utilizing the Automated Readability Index, gives the Marine manual a reading grade level of 16, and the Navy manual a reading grade level of 10. This is primarily due to the additional procedural instructions of the Navy manual. This instructional set is advantageous in that it provides clear, active voice recovery procedures. The introductory paragraphs in both manuals are written to the level of the operator, with jargon appropriate to the training level of the pilot. The importance and primacy of lowering collective, although alluded to in the Navy manual, is not established in the procedural steps that follow. The team also felt that a subject-verb-object syntax to be better suited to the actions required.³⁹The

³⁹R. W. Bailey, Human Performance Engineering, (Prentice Hall PTR, Upper Saddle River New Jersey, 1996) 420-452

procedure recommended by the team (which has subsequently been adopted for both manuals) reads:

“Blade stall is a loss of lift that normally occurs on the aft rotor in tandem designs. The aft head stalls first because of the rotor wash from the forward head and the fact that the aft head supports more than half the weight of the aircraft. Blade stall is more likely to occur at high airspeeds, gross weights or altitudes, in turbulence, or when maneuvering under a combination of the above conditions. CGI indications directly reflect the severity of blade stall and the resulting flight control strain loads. The stall generates a vibration that is felt by the pilot through the airframe and controls as a progressive one-per-revolution and three-per-revolution.

Since the aft rotor system stalls first and drops, the pilot may perceive this as a pitchup maneuver, or a change in longitudinal control response. Attempting to push the nose over with cyclic will be less effective than normal and may increase the pitchup since increased blade pitch on the aft rotor will aggravate the stall. A temporary reduction of 1 to 2 inches of collective will quickly alleviate the stall and provide improved longitudinal response. No loss of control will occur within the airspeed limitation and load factor (g limit and angle of bank) envelope of chapter 4.

Figure 4-11 is only valid for straight and level flight. For flight conditions other than straight and level, reduce the maximum airspeed accordingly. Prolonged operation in blade stall must be avoided because of the high loads imposed on the dynamic components. As blade stall is directly related to retreating blade angle of attack, the most effective way to decrease retreating blade angle of attack is lowering collective. If blade stall occurs, eliminate it by doing the following:

1. Collective, decrease as altitude and flight conditions allow
2. Maneuver severity, reduce.
3. Airspeed, reduce.

4. Rotor speed, increase.
5. Descend, as conditions allow at 500 fpm.

Differential Airspeed Hold Actuator Response Rate

Airspeed hold, pitch attitude stability and a positive stick gradient (airspeed proportional to forward stick position) are provided by the Differential Airspeed Hold feature of the CH-46E AFCS. A signal representing pitch attitude is developed by each vertical gyro; another signal representing airspeed is developed by the longitudinal stick position transducer. The signals are algebraically summed and amplified. Airspeed or pitch attitude changes without a corresponding stick position change will cause an error. The error signal is applied to the DASH actuator. The actuator will either extend to lower the nose and increase airspeed or retract and raise the nose to decrease airspeed. The slow Differential Airspeed Hold (DASH) actuator rate exacerbates pilot-induced pitch oscillations when longitudinal, or pitch inputs are either made at rates greater than 1 Hz, or at amplitudes are larger than two inches. This effectively changes the response rates of aircraft pitch control. The resultant pilot-induced oscillations most probably contributed to some of the damage incurred during the flight. The pilot delayed lowering the collective during the stall incident primarily due to his reaction to a stall-induced pitch oscillation episode. The low DASH actuator rates are a known deficiency and have been the subject of numerous program efforts to improve.

Operator Training

As referred to by the Boeing Test pilot undergoing transition training in the CH-46 aircraft, there exists unfamiliarity among operators of the dangers and consequences of either intentional or unintentional entry into incipient blade stall conditions. The team subsequently set up operator training sessions at both Navy and Marine training squadrons to brief the instructor pilots on the results and ramifications of incipient blade stall. The team felt that in addition to operator manual changes and CGI re-designs this would allow both:

“The activity to be synchronized with the behavior of the environment to meet the proper purpose and activity to be optimized to form a smooth and efficient pattern within the boundary of the task.”⁴⁰

Increased Inter-service Testing

During the interview with the Boeing Test pilot who had CH-47 experience, a number of issues came to light which were due to simple ignorance on the part of pilots on the Pax River team. The program had called for joint military and Boeing pilot participation; this should have preempted these issues. Unfortunately, schedule and distance constraints between the two teams prevented this from happening.

Increased inter-service officer pilot exchange tours, or possibly a service testing detachment at Patuxent River (as is currently being done for Joint Shipboard Interoperability testing) would prevent the stovepipe of critical information.

⁴⁰ J. Rasmussen, Information Processing and Human Machine Interaction, (Not Holland, New York, 1986) 113

VII

Conclusions

The H-46 DCU flight test program was vital to maintaining Marine medium lift and Navy VERTREP capabilities. Although the damage experienced in the test resulted in significant program delays and increased costs it also highlighted the effectiveness of current NAWC-AD and RWATS flight test safety methodologies. These methodologies were based on flight test experience, at times very dearly purchased, that assisted the team in minimizing its exposure to risk. The short-term effects of the delays caused by entering helicopter blade stall caused program delays. The incident was fortuitous in that it highlighted problems with the current mechanization of the CGI and the procedures used during aircraft operation.

This author has had the opportunity to be associated with numerous flight test programs over the past seven years. It has been his observation that due to the expensive nature of test programs for acquisition category 1 (highest cost level programs), they are frequently success oriented. It is not atypical that unforeseen developmental problems are discovered at this late stage of program maturity. These problems frequently exacerbate the difficulty of meeting cost, schedule, and performance goals. The goal of the flight test team is to provide detailed information of the nature of the developmental problem affecting the program, and provide achievable alternatives to assist the program manager in problem resolution. The test team succeeds when the program succeeds in delivering a product that meets or exceeds operational requirements. It is the author's belief that the data gathered during the conduct of the DCU program, in addition to

assisting the program manager in the fielding of this upgrade, will be invaluable to assisting fleet users in preventing aircraft mishaps and enhancing aircraft life cycles.

Bibliography

G.H. Saunders, Dynamics of Helicopter Flight (New York, John Wiley & Sons, 1975) 40

Maj. R. Yeend, Mr. S. Kowley, Contractor Structural Demonstration of the H-46 Fiberglass Helicopter Blades, NATC Technical Report RW-12R-78, RWATS, NAS Patuxent River MD. 14 Apr 1978

G.H. Saunders, Dynamics of Helicopter Flight (New York, John Wiley & Sons, 1975) 49

H. Schlichting, Boundary-Layer Theory (New York, McGraw Hill, 1979) 39

R.W. Prouty, Helicopter Performance, Stability, and Control (Malibar Florida: Robert E. Krieger, 1990) 380-382

F. Collins, Notes to Author, University of Tennessee Space Institute, Tullahoma TN, April 2001

R.W. Prouty, Helicopter Performance, Stability, and Control (Malibar Florida: Robert E. Krieger, 1990) 387-388

R.W. Prouty, Helicopter Performance, Stability, and Control (Malibar Florida: Robert E. Krieger, 1990) 388

R.W. Prouty, Helicopter Aerodynamics (Peoria: PJS Publications, 1985) 17

A.Gessow/G.C. Meyers Jr., Aerodynamics of the Helicopter (New York: Frederick Ungar Publishing, 1967) 166

C. Gablehouse, Helicopters and Autogiros (New York J.B. Lippincott, 1969) 36

C. Gablehouse, Helicopters and Autogiros (New York J.B. Lippincott, 1969) 44

A. Gessow/G.C. Meyers Jr., Aerodynamics of the Helicopter (New York: Frederick Ungar Publishing, 1967) 138

A. Gessow/G.C. Meyers Jr., Aerodynamics of the Helicopter (New York: Frederick Ungar Publishing, 1967) 180

A. Gessow/G.C. Meyers Jr., Aerodynamics of the Helicopter (New York: Frederick Ungar Publishing, 1967) 187-189

J.B. Wheatley, The Aerodynamics of the Gyroplane Rotating-Wing System NACA Report No. 492 Washington D.C. March 1934

G.H. Saunders, Dynamics of Helicopter Flight (New York, Wiley and Sons) 1974, 59

C.D.Perkins, R.E. Hage, Airplane Performance Stability and Control (New York, Wiley and Sons) 19

R.W. Prouty, Helicopter Performance, Stability, and Control (Malibar Florida: Robert E. Krieger, 1990) 181-182

W.Z. Stepniewski/ C.N. Keys, Rotary Wing Aerodynamic, (New York, Dover Publications, 1984) 97-104

A. Gessow/G.C. Meyers Jr., Aerodynamics of the Helicopter (New York: Frederick Ungar Publishing, 1967) 180-191

A.Gessow/G.C. Meyers Jr., Aerodynamics of the Helicopter (New York: Frederick Ungar Publishing, 1967) 185

W.Z. Stepniewski/ C.N. Keys, Rotary Wing Aerodynamic, (New York, Dover Publications, 1984) 101

P. Solies, Notes to Author, University of Tennessee Space Institute, Tullahoma TN, May 2001

F.B. Gustafson, C.G. Meyers, Stalling of Helicopter Blades, NACA Technical Report Number 840, 15 April 1946 108

R.W. Prouty, Helicopter Aerodynamics (Peoria: PJS Publications, 1985) 24-26

R.W. Prouty, Helicopter Aerodynamics (Peoria: PJS Publications, 1985) 26

W.Z. Stepniewski/ C.N. Keys, Rotary Wing Aerodynamic, (New York, Dover Publications, 1984) 103

R.W. Prouty, Helicopter Performance, Stability, and Control (Malibar Florida: Robert E. Krieger, 1990) 345

S. Kracinovich, Structural Flight Testing Lecture U.S. Naval Test Pilot School, NAS Patuxent River MD, October, 1995

NATOPS FLIGHT MANUAL, Navy Model CH-46E (SR&M) Helicopter, March 1992, I-2-71

Maj. E. Treworgy, DCU PROJECT Flight Report for Flight X-57, Attack Assault Branch NRWATS, NAS Patuxent River MD 1996

NATOPS FLIGHT MANUAL, Navy Model CH-46E Helicopter, Nov 1994

NATOPS FLIGHT MANUAL, Navy Model CH-46D Helicopter, Dec 94

K. B. De Greene, Systems Psychology, (McGraw Hill, New York 1970) 109

R. B. Stammers, A. Sheperd, Evaluation of Human Work, Task Analysis, (Taylor and Francis, London 1995) 151

R. E. Curry E. L. Wiener, "Flight-Deck Automation: Promises and Problems"
Ergonomics, (1980), vol 23, no. 10, 995-1011

B. H. Kantowitz, S. C. Kantowitz, R. D. Sorkin , "Likelihood Alarm Displays" Human Factors, (1988) 30(4) 445-459

R. W. Bailey, Human Performance Engineering, (Prentice Hall PTR, Upper Saddle River New Jersey, 1996) 420-452

J. Rasmussen, Information Processing and Human Machine Interaction, (Not Holland, New York, 1986) 113

Vita

Lieutenant Colonel L'Heureux was born in Corona California, growing up on the banks of Lake St. Clair, on Harsens Island Michigan. He enlisted in the Marine Corps in 1972, attended the University of Michigan on the G.I. Bill, graduating with a Bachelor of Science degree in Electrical Engineering in 1979. He was commissioned a Second Lieutenant in 1980. Upon completion of The Basic School, in October 1980, he was ordered to NAS Pensacola, receiving his wings in March of 1982 and reported to Marine Medium Helicopter Training Squadron 301, MCAS Tustin California for training in the CH-46E Sea Knight Helicopter. While undergoing fleet replacement training he was promoted to First Lieutenant.

Subsequently assigned to Marine Medium Helicopter Squadron-164, he was assigned duties as Tool Control Officer, Individual Maintenance Material Readiness List Officer, Responsible Officer, and Assistant Logistics Officer. He deployed with the squadron to Marine Aircraft Group 36, Okinawa Japan from September 1983 returning to the United States in March of 1984, during this assignment, he was promoted to Captain. Subsequently re-assigned within Marine Aircraft Group 16 in a cadre status, he was a plank owner in the commissioning of Marine Medium Helicopter Squadron 166. Serving as the Logistics Officer, and Quality Assurance Officer, he completed a deployment to the Western Pacific aboard the USS Tripoli assigned to the 13th Marine Amphibious Unit.

In February 1988 he was ordered to Helicopter Training Squadron 18, NAS Whiting Field, Milton Florida. Assigned as Helicopter Flight Instructor, Logs and

Records Officer, Flight Operations Officer, and Standardization Department Head, he served with the squadron until February of 1991. While assigned to HT-18 he was ordered to Fort Benning, Georgia where he completed the U. S. Army Airborne Basic course.

He was ordered to the 1st Air Naval Gunfire Liaison Company in February 1991, completing Landing Force Training Command, Pacific Forward Air Controller school prior to reporting to Camp Pendleton, California. He served as a Supporting Arms Liaison Team Leader, and was subsequently assigned as Brigade Platoon Commander of the 2nd Platoon. Promoted to Major in July of 1992, he was ordered to Marine Aircraft Group 16 in August of the same year.

Assigned to Marine Medium Helicopter Training Squadron 301, he served as Flight Instructor, Administrative Officer, and was then assigned as Executive Officer, decommissioning the squadron in September 1993. He was subsequently assigned to HMM-166 (Composite) and served as Director of Safety-Standardization. He deployed with HMM-166 and the 15th Marine Expeditionary Unit aboard the USS Tripoli in January of 1994. During this deployment he was further assigned as the Executive Officer for Task Force Super Stallion which provided support for Joint Task Force Support Hope Rwandan humanitarian relief operations.

Selected for United States Naval Test Pilot School, he reported to NAS Patuxent River in December of 1994, and served as class leader for USNTPS class 108.

Graduating in December 1995. He was assigned to the Naval Rotary Wing Aircraft Test Squadron and served as H-46 Platform Coordinator, Attack Assault Department Head and Operations Officer. He was promoted to Lieutenant Colonel in July of 1997.

Ordered to the Third Marine Expeditionary Force in July of 1998 Lieutenant Colonel L'Heureux served as the Northwest Asian Plans Officer from July of 1998 to July of 1999. He was subsequently ordered to the Marine Aviation Detachment, NAS Patuxent River MD, for duty as the Chief Test Pilot and then Commanding Officer, Naval Rotary Wing Aircraft Test Squadron, where he currently serves.

Lieutenant Colonel L'Heureux has accumulated over 4600 flight hours in 31 different model aircraft. His assignments included participation in operations Restore Hope, Support Hope, and Southern Watch. His personal decorations include the Navy Marine Corps Commendation and Achievement Medals.

He is married to the former Gerrie Anne Simpson of Singapore. They have four sons, Rejean, Gabriel, Raphael, and Yves-Jerome.