USNTPS-FTM-No. 107

U.S. NAVAL TEST PILOT SCHOOL
FLIGHT TEST MANUAL

ROTARY WING STABILITY AND CONTROL

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NAVAL AIR WARFARE CENTER
PATUXENT RIVER, MARYLAND

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This Flight Test Manual, published under the authority of the Commanding Officer, U.S. Naval Test Pilot School, is intended primarily as a text for the pilots, engineers and flight officers attending the school. Additionally, it is intended to serve as a reference document for those engaged in flight testing. Corrections and update recommendations to this manual are welcome and may be submitted to:

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22783 Cedar Point Road
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31 December 1995

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INTRODUCTION

CHAPTER ONE

INTRODUCTION

1.1 OBJECTIVE

The Rotary Wing Stability and Control Flight Test Manual (FTM) is intended to serve as a practical and easy to use reference guide for the planning, execution, and reporting of flight testing rotary wing aircraft for stability, control, and flying qualities. The FTM is intended for use as a primary instructional tool at the U.S. Naval Test Pilot School (USNTPS) and as a reference document for those conducting related helicopter flight testing at the Naval Air Warfare Center Aircraft Division (NAWCAD) or similar organizations interested in rotary wing flight testing. It is not intended to be a substitute for helicopter stability and control textbooks. Rather, the FTM summarizes applicable theory to the extent necessary to permit an understanding of the concepts, techniques, and procedures involved in successful flight testing. The FTM is directed at test pilots and flight test engineers (FTE); it deals with the more practical and prominent features of stability and control issues, sometimes sacrificing exactness or completeness in the interest of clarity and brevity.

The FTM does not replace the Naval Air Warfare Center Aircraft Division *Report Writing Handbook*. The FTM contains examples of stability, control, and flying qualities parameters discussed in narrative and graphic format. It contains discussions of the effect on mission performance and suitability of the various parameters, and a discussion of specification compliance where applicable. The examples in this manual show trends extracted from current helicopters and are in the format used at USNTPS.

This FTM, since it is a text for USNTPS, contains information relative to operations at USNTPS and NAWCAD; however, it does not contain information relative to the scope of a particular USNTPS syllabus exercise or to the reporting requirements for a particular exercise. Detailed information for each flight exercise will vary from time to time as resources and personnel change and will be briefed separately to each individual class.

1.2 ORGANIZATION

1.2.1 Manual Organization

The FTM is organized to simplify access to desired information. Although there is some cross referencing, in general, each chapter stands as a discrete unit. Discussions of flight test techniques are presented together with pertinent background analytic presentations. Most of the information is generic in nature although specific examples are given where appropriate. The contents are organized in a classical grouping and follow the stability and control syllabus at USNTPS.

The FTM puts in perspective flying qualities testing relative to open loop statics and dynamics documentation typical of Military Specification (MilSpec) compliance testing.

Chapter 1, Introduction, is an overview of the FTM.

Chapter 2, Pilot Flying Qualities Evaluations, details the pilot flying qualities evaluation process (closed loop flight testing) and establishes piloted flying qualities as the focus of supportive stability and control open loop flight testing. Evaluating mission relevant tasks using Handling Qualities Ratings (HANDLING QUALITIES RATING) is discussed.

Chapter 3, Open Loop Testing, outlines the various elements in open loop testing: the specialized test inputs for statics and dynamics testing, a summary of data reduction techniques, and the context for presenting results of open loop testing such as MilSpec. In the open loop system, the pilot interfaces within closed loop testing, including the augmented aircraft characteristics and the control system characteristics. Open loop testing is aimed at determining the transfer characteristics of these elements.

Chapter 4, Rotor Characteristics, contains stability and control theory without any discussion of test techniques or procedures. The purpose of this chapter is to reacquaint the reader with the rotor characteristics which provide control of the helicopter. No attempt is made to develop rigorously the various equations which govern the stability and control of the rotor. Rather the terminology, reference systems, and equations are presented with a discussion of the factors which influence the results. Simplifying assumptions are made and discussed.

Chapter 5, Flight Control System Characteristics, covers helicopter control system mechanical characteristics with emphases on the irreversible control system. Understanding and documenting the flight control system characteristics is fundamental to any flying qualities or closed loop flight evaluation. Since the flight control system is part of the control loop, the test pilot must have a detailed understanding of the components of the flight control system and their function. The understanding begins with a thorough flight control system description and proceeds with documenting the flight control system characteristics. Test methods, data requirements, and data presentation are addressed.

Chapter 6, Forward Flight Longitudinal Stability, Control, and Flying Qualities, covers longitudinal characteristics in powered forward flight above 35 kn. The lateral-directional responses of the helicopter are constrained to minimize interference with the longitudinal characteristics. Coupling between these axes is ignored in the analytic discussions. Cross axis coupling is explicitly treated in Chapter 9. Chapter 6 details the various elements which comprise the longitudinal open loop characteristics, how to test for these elements, and how they yield particular flying qualities.

Chapter 7, Forward Flight Lateral-Directional Stability, Control, and Flying Qualities, is the lateral-directional equivalent of Chapter 6. Lateral-directional characteristics in powered forward flight above 35 kn are discussed. Coupling between the longitudinal and lateral-directional axis is constrained.

Chapter 8, Hover and Low Airspeed Stability, Control, and Flying Qualities, covers the longitudinal and lateral-directional characteristics in the low speed regime, from hover to about 35 kn. The focus is on in and out of ground effect (IGE, OGE) characteristics for hover and vertical maneuvering. The upper end of the relative wind spectrum encompasses translational velocities. Takeoff and landing characteristics are included.

To simplify discussion, the longitudinal and lateral-directional characteristics are treated as decoupled wherever practical. For example, the lateral-directional responses of the helicopter are constrained to minimize their influence on the longitudinal characteristics.

Chapter 9, Coupled Longitudinal and Lateral-Directional Stability, Control, and Flying Qualities, aggregates the important coupling effects between the pitch, roll, and yaw axes dependent on attitude, rate, acceleration, or control coupling. This chapter discusses stability, control, and flying qualities when longitudinal and lateral-directional coupling is present for hover, low airspeed, and forward flight. Because of the helicopter's asymmetric configuration, coupling is a significant factor in assessing mission related flying qualities in all phases of flight. Understanding and developing meaningful test procedures and documenting any undesirable coupling effects on flying qualities is essential in evaluating a helicopter's ability to perform its mission.

Chapter 10, Sudden Engine Failures, Autorotative Flight, and Autorotative Landings, deals with the theory of autorotative flight and the flight test techniques pertinent to this flight regime. This chapter discusses helicopter stability, control, and flying qualities during engine failure, autorotation entry, steady autorotative flight, and landing following a successful autorotation. Although the emphasis is on flying qualities, knowledge of helicopter performance requirements and power deficits is essential to understanding the total autorotation evaluation starting with engine failure and ending with a satisfactory landing. Flying qualities and performance characteristics are both required in analyzing and testing autorotation capability.

1.2.2 Chapter Organization

Each chapter has the same internal organization where possible. Following the chapter introduction, the second section gives the purpose of the test. The third section is a review of the applicable theory. The fourth section discusses the test methods and techniques, data requirements, and safety precautions applicable to those methods. The fifth section discusses data reduction. The sixth section pertains to the data analysis. The seventh section covers relevant mission suitability aspects of the stability, control, and flying qualities parameters. The eighth section discusses specification compliance. The ninth section is a glossary of terms introduced in the chapter. Finally, the tenth section lists applicable references.

1.3 EFFECTIVE TEST PLANNING

In order to plan a test program effectively, a sound understanding of the theoretical background of the tests to be performed is necessary. This knowledge helps the test team establish the optimum scope of tests, choose appropriate test techniques and data reduction methods, and present the test results effectively. Time and money are scarce resources;

therefore, test data should be obtained with a minimum expenditure of both. Proper application of theory ensures the tests will be performed at the proper conditions, with appropriate techniques, and using efficient data reduction methods.

1.4 RESPONSIBILITIES OF TEST PILOT AND FLIGHT TEST ENGINEER

Almost every flight test and evaluation team will be composed of one or more test pilots and one or more project engineers. Team members bring together the necessary expertise in qualitative testing (flying qualities evaluation) and quantitative evaluation (knowledge of theory, instrumentation, and specifications). To perform the necessary tests and evaluations, the test pilot must have knowledge of theory, test methods, data requirements, data analysis, instrumentation, and specifications. The FTE must possess a thorough knowledge of the pilot tasks required to perform a total mission in order to participate fully in the planning and execution of the test and evaluation program.

1.4.1 The Test Pilot

The competent test pilot must be highly proficient to obtain accurate data. He must have well developed observation and perception powers to recognize problems and adverse characteristics. He must have the ability to analyze test results so that he understands them and can explain the significance of his findings. To fulfill these expectations, he must possess a superior knowledge of:

1. The test helicopter and rotary wing aircraft in general.
2. The total mission of the aircraft and the individual pilot tasks required to accomplish the mission.
3. Theory and associated test techniques required for qualitative testing and quantitative evaluation.
4. Specifications relevant to the evaluation program.
5. Technical report writing.

The test pilot must understand the test aircraft in detail, particularly the flight control system, to do a creditable job of stability and control testing. He must consider the effects of external configuration changes on flying qualities. The successful test pilot should have flight experience in many different types of aircraft. By observing diverse characteristics exhibited by a variety of aircraft, the test pilot can make accurate and precise assessments of design concepts. Further, by flying many different types, he develops adaptability. When flight test time is limited by monetary and time considerations, the ability to adapt is invaluable.

The test pilot must understand clearly the total mission of the helicopter. The test pilot must know the specific operational requirements on which the design was based, the detail specification under which the design was developed, and other planning documents. Knowledge of the individual pilot tasks required for total mission accomplishment is derived from recent operational experience. Additionally, he can gain knowledge of the individual pilot tasks from talking with other pilots, studying operational and tactical manuals, and visiting replacement pilot training squadrons.

A qualified engineering test pilot executes a flight test task and evaluates the validity of the results to determine whether he needs to repeat the test. Often the test pilot is the best judge of an invalid test point and can save the test team wasted effort. The test pilot's knowledge of theory, test techniques, relevant specifications, and technical report writing may be gained through formal education or practical experience. The most effective and efficient method is through formal study with practical application at an established test pilot school such as the USNTPS. This education provides a common ground for the test pilot and FTE to converse in technical terms concerning stability, control, and flying qualities phenomena.

1.4.2 The Flight Test Engineer

The successful FTE must have general knowledge of the same items for which the test pilot is mainly responsible. Additionally, he must possess superior knowledge of:

1. Instrumentation requirements.
2. Planning and coordination aspects of the flight test program.
3. Data acquisition, reduction, and presentation.
4. Technical report writing.

These skills are necessary for the FTE to form an efficient team with the test pilot for the planning, executing, analyzing, and reporting process.

Normally, the FTE is responsible for determining the test instrumentation. This involves determining the ranges, sensitivities, frequency response required, and developing an instrumentation specification or planning document. In the context of flying qualities testing, the FTE assists the test pilots with establishing the appropriate evaluation tasks, the standard of performance (desired/adequate), and the pilot comment cards. For open loop stability testing, the FTE must determine the pertinent test inputs necessary to generate time histories which can be reduced to yield the desired stability parameters and control sensitivities.

The FTE is in the best position to coordinate all aspects of the program because he does not fly in the test aircraft often and is available in the project office. The coordination involves aiding in the preparation and revision of the test plan and coordinating the order in which flights will be conducted. Normally, the FTE prepares all test flight cards and participates in all flight briefings and debriefings.

A great deal of the engineer's time will be spent working with flight and ground test data. He must review preliminary data from wind tunnel studies and existing flight tests. From this data, critical areas may be determined prior to actual military flight testing. During the flight tests, the engineer may monitor and aid in the acquisition of data through telemetry facilities and radio, or by flying in the test aircraft. Following completion of flight tests, the engineer coordinates data reduction, data analysis, and data presentation.

The FTE uses his knowledge of technical report writing to participate in the preparation of the report. Usually, the FTE and the test pilot will proofread the entire manuscript.

1.5 STABILITY AND CONTROL SYLLABUS

1.5.1 Overview

The stability and control syllabus at USNTPS consists of academic instruction, flight briefings, demonstration flights, practice flights, exercise flights, reports, and evaluation flights. The stability and control phase of instruction concludes in an individual evaluation flight. The final exercise at USNTPS is a simulated Navy Developmental Test

IIA (DT IIA). This exercise incorporates all the performance, stability, control, flying qualities, and airborne systems instruction into the total evaluation of an airborne weapon system.

The stability and control syllabus includes exercises in flight control systems, mechanical characteristics, hover and low airspeed stability and control, forward flight stability and control, and autorotative flight. The syllabus is presented in a step-by-step, building block approach which allows concentration on specific objectives and fundamentals. Unfortunately, this approach focuses on individual characteristics at the expense of evaluating the total weapon system. Progress through the syllabus must be made with the end objective in mind, the evaluation of the helicopter as a weapon system in the mission environment. The details of the current syllabus is contained in *U.S. Naval Test Pilot School Notice 1542*.

1.5.2 Flight Briefings

To form the basis of the applicable theory, each exercise in the stability and control syllabus is preceded by academic instruction. Printed and oral flight briefings are presented by the principal instructor for the exercise. The flight briefing gives specific details of the exercise and the exercise requirements. The flight briefing covers the purpose, references, test conditions, method of test to be used, the scope of test, test planning, and report requirements. The briefing also covers the applicable safety requirements for the exercise as well as administrative and support requirements.

1.5.3 Demonstration Flights

The demonstration flight is preceded by thorough briefings including: theory, test techniques, analysis of test results in terms of mission accomplishment and specification requirements, and data presentation methods. The student must prepare for the demonstration flight by review of briefing notes, appropriate technical literature, and relevant specifications. Thorough preparation is essential for maximum benefit.

The demonstration flight crew consists of one or more students and an instructor, with the students sharing airborne instructional time. If the students are not qualified in the demonstration aircraft, the instructor pilot may handle all preflight, ground operations, takeoffs, and landings. The students are required to know the operational characteristics of the demonstration aircraft. In flight, the instructor will demonstrate both qualitative and quantitative test techniques, use of special instrumentation, and data recording procedures. After the student observes and understands each technique, he has the opportunity to practice until attaining a reasonable level of proficiency. Throughout the demonstration flight, the instructor discusses the significance of each test, implications of results, and variations in the test techniques appropriate for other type helicopters. Students are encouraged to ask questions during the flight; many points are explained or demonstrated easier in flight than on the ground. A thorough post flight discussion between instructor and students completes the demonstration flight. During the debrief, the data obtained in flight are plotted and analyzed.

1.5.4 Practice Flights

Each student is afforded the opportunity to practice the test methods and techniques in flight after the demonstration flights and prior to the exercise or data flight. The purpose of the practice flights is to gain proficiency in the test techniques, data acquisition, and crew coordination necessary for safe and efficient flight testing.

1.5.5 Exercise Flights

Each student will fly one or more flights as part of each exercise. The student is expected to plan the flight, have the plan approved, and fly the flight in accordance with the plan. The purpose of the flight is to gather both quantitative and qualitative data as part of an overall flying qualities evaluation. The primary inflight objective is safe and efficient flight testing. Under no conditions will flight safety be compromised.

1.5.6 Reports

A fundamental purpose of USNTPS is developing the test pilot/FTE’s ability to report test results in clear, concise, unambiguous technical terms. After completing the exercise flight, the student reduces the data. Data reduction transposes the format of data gathered during flight to data used for engineering analysis. The student analyzes the data for both mission suitability and specification compliance. The data are presented in the proper format and a report is prepared. The report process combines factual data gathered from ground and flight tests and analysis of its effect on mission suitability. The report conclusions answers the questions implicit in the purpose of the test.

Theory should be used as the foundation for data analysis and technical reporting of test results. However, theory is often misused to explain why something desirable or undesirable happened. The question, “What caused the occurrence?” often receives more attention than reporting the actual occurrence and its impact on mission suitability. The cause may be completely irrelevant. For example, a helicopter exhibiting stable longitudinal maneuvering stability with a forward center of gravity (CG) is unstable with an aft CG. Such a trend is not unusual and no theoretical discussion is in order. A statement as to the degree of instability uncovered and its significance to mission suitability or safety of flight is needed.

1.5.7 Progress Evaluation Flight

The progress evaluation flight is both an evaluation exercise and an instructional flight. It is a check flight on the phase of study just completed. A numerical grade will be assigned by the instructor.

One student and one instructor comprise the flight crew. The student develops a flight plan considering the real or simulated aircraft mission and appropriate specification requirements. The student conducts the flight briefing, including the mission, a brief flight control system description, discussion of test techniques, and specification requirements.

As the student demonstrates his knowledge of qualitative and quantitative test techniques in flight, he is expected to comment on the impact of the results on the real or simulated mission. The instructor will comment on validity of the results obtained, errors or omissions in test procedures, and may demonstrate variations in test techniques not introduced previously. The instructor may ask the student to perform a flying qualities evaluation of a mission task. The student is expected to define the task, the expected performance, evaluate the workload and pilot compensation, and assign a Handling Qualities Rating.

During the debrief the student presents, analyzes, and discusses the test results. The discussion includes the influence of the results on aircraft mission suitability.

1.6 FLIGHT SAFETY

1.6.1 Incremental Build-Up

The concept of incremental build-up is one of the most important practical and philosophical approaches to flight testing. Build-up is the process of proceeding from the known to the unknown in an incremental, methodical pattern. Flight tests should be structured with build-up in mind. Testing begins with the best documented, least hazardous data points and proceeds toward the desired end points, always conscious of the aircraft, pilot, and evaluation limits. There should be no surprises in flight test. In the event a data point yields an unexpected result or a series of data points creates an unexpected trend, evaluation stops until the results are analyzed and explained.

1.6.2 Unusual Attitude Recovery

For USNTPS purposes, the term Unusual Attitude refers to the in flight pitch, roll, and yaw attitudes and/or rates resulting from an intentional control input commanding the aircraft to deviate from its trimmed flight condition. Applicable recovery procedures depend upon the aircraft being flown and the severity of the maneuver being executed. At all times the pilot must remain in the loop actively flying the aircraft and analyzing the aircraft response.

Recommended pilot recovery techniques from unusual attitudes for aircraft currently operated at USNTPS are delineated below.

1.6.3 Teetering Rotor System Aircraft

1.6.3.1 RECOVERY FROM NOSE HIGH UNUSUAL ATTITUDE

As the pitch attitude reaches the nose up limit or the data point is complete, yaw and roll right. The amount of right lateral cyclic will depend on the roll induced by the yaw rate. The goal of this recovery is to roll out to the right of the initial heading, accelerating nose down toward the trim airspeed. If the yaw rate appears to be excessive, apply left pedal to slow the response. As the yaw rate diminishes, add left lateral cyclic to roll wings level and fly away smoothly. If the pitch rate appears to be excessive or if the absolute allowable pitch up attitude is going to be exceeded, a small amount of forward cyclic may be added at the beginning of the recovery to slow the pitch rate. Do not attempt to stop the pitch rate, a low normal acceleration condition may result. If a small amount of forward cyclic and right pedal are applied simultaneously, it is likely that the resulting yaw rate will be greater than the yaw rate had the forward cyclic not been added.

1.6.3.2 RECOVERY FROM NOSE LOW UNUSUAL ATTITUDE

As the pitch attitude reaches the nose down limit or the data point is complete, smoothly apply aft cyclic to wash out the pitch rate. As the aircraft attitude stabilizes, apply appropriate control inputs to accelerate the aircraft and fly back to the trim condition. Pay attention to normal acceleration, rotor speed, and torque so as to not exceed limits on pull out.

1.6.3.3 RECOVERY FROM AN UNUSUAL ROLL ATTITUDE

As the roll attitude reaches the angle of bank limit or the data point is complete, apply pedal in the direction of the roll to reduce the sideslip angle and roll rate. Smoothly apply lateral cyclic to restore wings level and longitudinal cyclic as required to return to the trim condition.

1.6.3.4 RECOVERY FROM LOW NORMAL ACCELERATION MANEUVER

Flight near or below zero normal acceleration is prohibited. If a low normal acceleration condition is encountered, the helicopter may exhibit a tendency to roll to the right. Such things as sideslip, weight and location of wing stores, and airspeed will affect the severity of the right roll. The right roll occurs throughout the normal operating airspeed range and becomes more violent at progressively lower load factors.

If a low normal acceleration condition is encountered, smoothly apply aft cyclic to restore normal loading. Lateral cyclic will not affect recovery from a roll due to low normal acceleration and it may cause severe main rotor flapping. As normal loading is restored, lateral cyclic to return to wings level is appropriate. Do not move collective or directional controls.

1.6.4 Articulated Rotor System Aircraft

1.6.4.1 RECOVERY FROM A NOSE HIGH UNUSUAL ATTITUDE

Once sufficient data has been obtained, or no later than reaching the nose up limit, apply a slight forward cyclic stick input to stop the pitch rate. Once the pitch rate has been arrested, continue with forward cyclic to return the pitch attitude to slightly below the horizon to regain airspeed. In the more extreme pitch up attitudes, the recovery can be expedited by rolling slightly and applying pedal in the direction of cyclic to yaw the aircraft back to or below the horizon. For the roll, follow the direction the aircraft is tending to roll, if any. If the pitch is straight up, consider rolling right for US type rotors and left for those rotating in the opposite direction. When recovering from a nose high unusual attitude in a hover, apply up collective to arrest any undesirable rate of descent only after the pitch attitude has been returned to the horizon.

Horizontal stabilizers on some helicopters are programmed as a function of airspeed. The nose high recovery in forward flight needs to be quick enough so as to not attain low airspeed and risk further nose up pitch as the aircraft potentially settles by the tail.

In any recovery, pay particular attention to parameters such as normal acceleration limits, torque changes, onset of blade stall, and rotor speed.

1.6.4.2 RECOVERY FROM A NOSE LOW UNUSUAL ATTITUDE

Recovery techniques are the same as those described in Section 1.6.3.2 for the teetering rotor system aircraft.

1.6.4.3 RECOVERY FROM AN UNUSUAL ROLL ATTITUDE

Recovery techniques are the same as those described in Section 1.6.3.3 for the teetering rotor system aircraft.

1.7 GLOSSARY

1.7.1 Notations

CG	Center of gravity
DT IIA	Developmental Test IIA
FTE	Flight Test Engineer
FTM	Flight Test Manual
HQR	Handling Qualities Rating
IGE	In ground effect
kn	Knots
MilSpec	Military Specification
NAWCAD	Naval Air Warfare Center Aircraft Division
OGE	Out of ground effect
USNTPS	U.S. Naval Test Pilot School

1.8 REFERENCES

1. NATC Instruction 5213.3F, *Report Writing Handbook*, NAWCAD, Patuxent River, MD, 16 August 1984.

2. *U.S. Naval Test Pilot School Notice 1542*, Subj: Academic, Flight and Report Curriculum for Class --, U.S. Naval Test Pilot School, Patuxent River, MD.

CHAPTER TWO

PILOT FLYING QUALITIES EVALUATIONS

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PILOT FLYING QUALITIES EVALUATIONS

2.1 INTRODUCTION

The objective of aircraft service suitability test and evaluation (whether flight, ground, or laboratory testing) is to determine the aircraft's capability to accomplish the assigned mission in a combat environment. Performance testing provides answers to many questions such as how high, fast, and for how long. Stability, control, and flying qualities testing determines the pilot's ability to employ the aircraft in the mission environment. Can the pilot place the aircraft in the proper location on the battlefield for sufficient duration to bring its mission systems to bear?

The answer to this question can be determined through flying actual missions in a combat environment and evaluating performance. While this method will provide the answers, it may not be cost effective and there may not be a handy combat environment. Flying qualities testing is a controlled, quantifiable method of evaluating and reporting mission suitability of an aircraft.

To understand flying qualities testing, it is helpful to compare and contrast flying qualities testing with stability and control testing.

2.1.1 Stability, Control, and Flying Qualities Testing

Flying qualities determine the ease and accuracy of accomplishing the tasks or maneuvers which constitute an aircraft's mission. Flying qualities are the result of a closed loop or pilot-in-the-loop control concept where the pilot actively makes control inputs to accomplish a desired task. The stability and control characteristics directly affect the aircraft's flying qualities. The stability and control characteristics link the cockpit flight controls, through the flight control system, to the aircraft aerodynamic response characteristics which the pilot desires to control. Flying qualities are pilot perceived results of how easily and accurately an aircraft is commanded to perform assigned mission tasks.

The contributing elements of stability and control are measurable characteristics of the helicopter system which influence pilot perceived flying qualities. What matters is the ease and accuracy with which the pilot can perform particular tasks with a helicopter. Stability and control characteristics are the objective design parameters implemented to achieve flying quality objectives.

Flying qualities evaluations are closed loop tests where the pilot actively manipulates the flight controls to accomplish a task; stability and control characteristics are evaluated through open loop test.

2.1.2 Closed Loop and Open Loop Testing

Stability, control, and flying qualities testing consists of both closed loop testing and open loop testing. Closed loop testing consists of performing mission relevant flight tasks and evaluating the resulting man-machine performance. Closed loop testing (Figure 2.1) consists of a maneuver requirement, the pilot input through the flight control system to achieve the maneuver requirement, the aircraft reaction in accordance with its aerodynamic characteristics, and the response. The pilot observes the response through a feedback path and determines the error between the desired and actual response. If an error exists, the pilot provides additional inputs to reduce or eliminate the error. Additionally, the environment provides inputs to the aircraft through external disturbances. This closed loop process continues until the error is zero or until another maneuver is required. As long as the environment provides external disturbances, an error exists and the pilot continues the inputs to null the error.

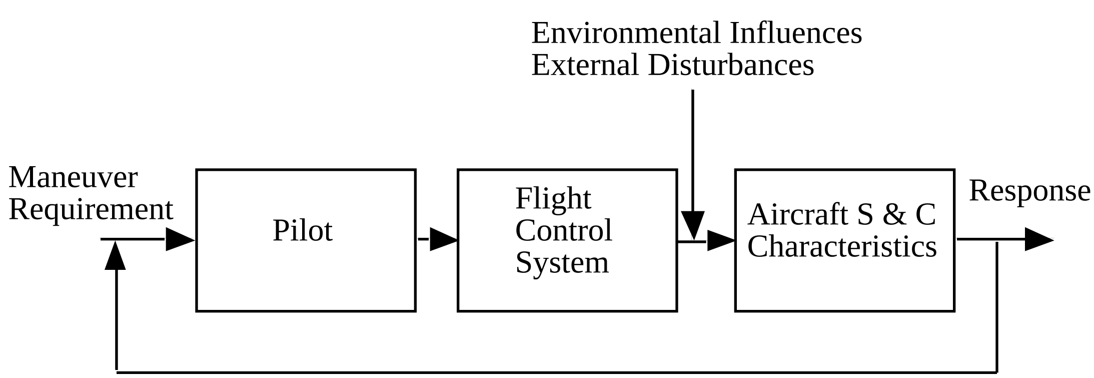


Figure 2.1
Closed Loop Control System

The process of performing closed loop flight evaluations, where the pilot actively manages the flight controls to accomplish a desired task in a real or simulated mission environment, is termed flying qualities testing.

Open loop testing is conducted without the pilot actively managing the flight controls. The objective of open loop testing is to determine and quantify the stability and control characteristics of the aircraft. The open loop tests quantify the stability and control characteristics of the aircraft. In open loop testing (Figure 2.2), the pilot makes one input into the aircraft through the flight controls and the response is observed. There is no preconceived command and the pilot does not evaluate the response against a desired result. No error is determined. The input could be generated by the environment, by the pilot, or through test equipment connected to the flight controls.

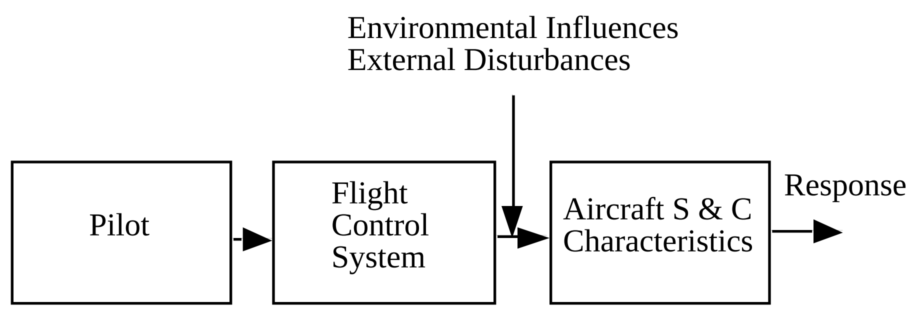
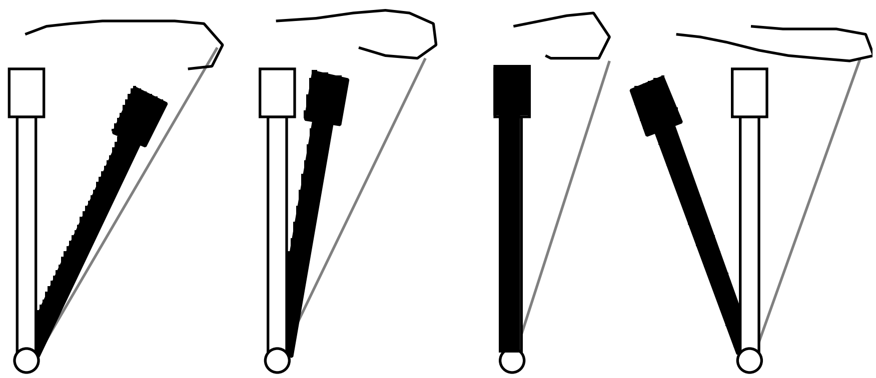


Figure 2.2
Open Loop Control System

2.1.2.1 THE CONCEPT OF CONTROLLABILITY

"Controllability may be defined as the capability of the airplane to perform, at the pilot's wish, any maneuvering required in total mission accomplishment. The characteristics of the airplane should be such that these maneuvers can be performed precisely and simply with a minimum of pilot effort." (Reference 2.3). This definition implies controllability is a closed loop or pilot-in-the-loop concept and is related to the concept of flying qualities.

The quote continues, “The pilot's opinion of controllability is shaped by several factors. The most apparent of these factors are the initial response of the airplane to a control input and the total attitude change which results. In addition, the cockpit control forces and deflections required to accomplish necessary pilot tasks are extremely important. These factors depend on the static and dynamic stability of the airplane and the characteristics of the flight control system. The complexity or degree of difficulty which the pilot encounters during maneuvering tasks is directly dependent on the stability characteristics of the airplane.” (Figure 2.3)



(a) (b) (c) (d)

Initial Position

Interim Position

Control Movement

Final Position

(a) Stable Aircraft
(b) Weakly Stable Aircraft
(c)Neutrally Stable Air craft
(d) Unstable Aircraft

Figure 2.3
Control Movement Required in Changing
From One Steady State Flight Condition to Another

The control movements required to maneuver a stable helicopter are simple. The simple control movements for the stable aircraft significantly reduces the pilot attention devoted to directly flying the aircraft. Thus, he can devote more of his attention to mission tasks, which may involve placing weapons precisely on target, navigating from point to point, or hovering (Figure 2.4).

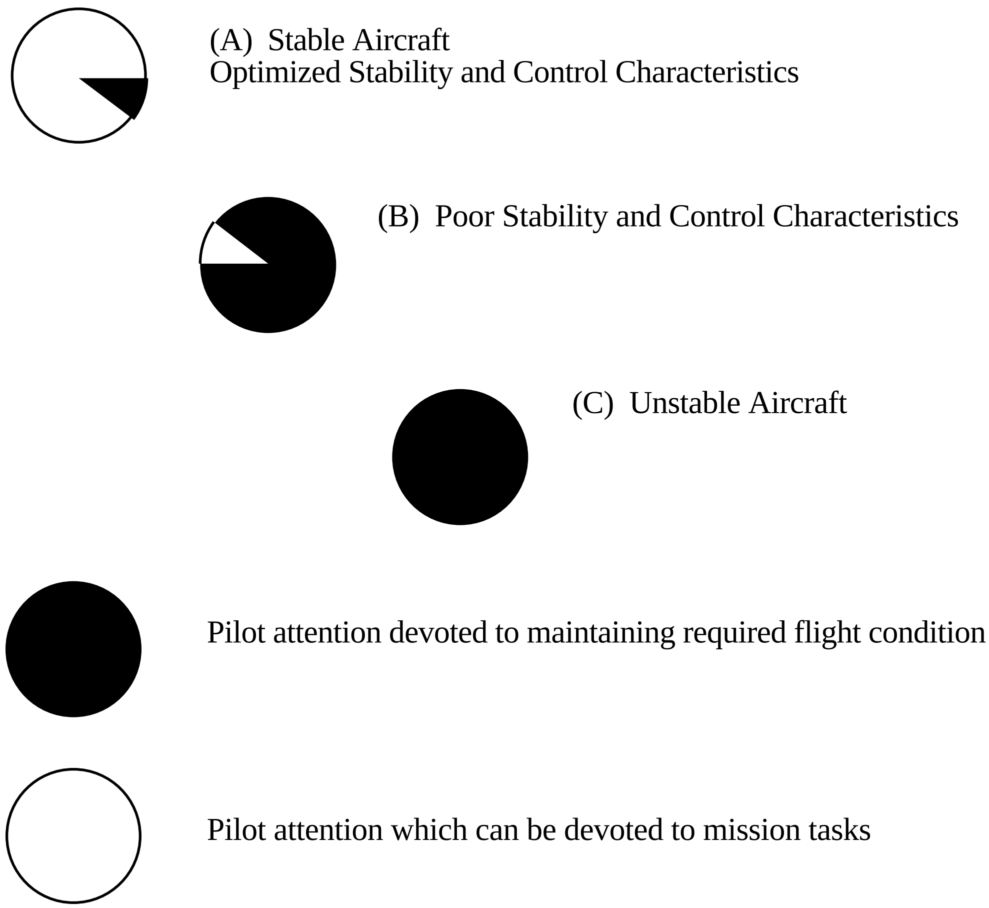


Figure 2.4
Typical Patterns of Pilot Attention Required as a Function of Aircraft Stability and Control Characteristics

2.2 PURPOSE OF TEST

The purpose of flying qualities testing is to evaluate the mission suitability of an aircraft’s piloted flying qualities in a real or simulated mission environment.

2.3 THEORY

Pilot flying qualities evaluation is based on the principle of selecting mission representative tasks, performing the tasks in a simulated mission environment, observing the pilot workload required to accomplish the task, and determining if the performance and workload are acceptable for the mission.

2.3.1 Flying Qualities

Flying qualities are defined as, “Those qualities or characteristics of an aircraft that govern the ease and precision with which a pilot is able to perform the tasks required in support of an aircraft role.” (Reference 2.3). Generally, the definition of flying qualities is the same as handling qualities. Either term is used throughout the Flight Test Manual.

Piloted flying qualities are a qualitative evaluation which is controlled and quantified as much as possible. However, the evaluation requires the subjective judgement of a trained pilot.

Flying qualities are a comprehensive pilot evaluation of the workload required to accomplish a task. Factors which influence a pilot's evaluation of flying qualities are the aircraft stability and control characteristics, flight control system, cockpit interface (displays and controls), the environment (weather conditions, visibility, turbulence), total mission requirements, and stress. The pilot’s evaluation may be affected by the displays which form part of the feedback loop. The aircraft stability and control characteristics may not change but the piloting tasks can be affected by a change in any element of the control loop (Figure 2.1). To understand the flying qualities evaluation process and communicate the results clearly, the following related concepts must be understood.

2.3.2 Mission and Role

The mission and role of an aircraft are defined in the operational requirements documents and the aircraft specification. The mission of the aircraft is defined as, “The composite of pilot-vehicle functions that must be performed to fulfill operational requirements. May be specified for a role, complete flight, flight phase, or a flight subphase.” (Reference 2.3). The important element in the definition of mission is operational requirements. The term mission refers to the operational environment in which the aircraft is intended to be used. The role of the aircraft is defined as, "The function or purpose that defines the primary use of an aircraft." (Reference 2.3). The role of the aircraft defines only the general use of the aircraft.

2.3.3 Task

The term task is defined as, "The actual work assigned to a pilot to be performed in completion of or as representative of a designated flight segment." (Reference 2.3). As an example, flight segment may be designated as ground taxiing. The tasks involved in the flight segment or phase, ground taxi, could be defined as maintaining taxi speed and ground track (Table 2.I).

Table 2.I
Relationship Between Mission, Role, Flight Phases, and Tasks

Mission	The composite of pilot-vehicle functions that must be performed to fulfill operational requirements							
Role	The function or purpose that defines the primary use of an aircraft							
Flight Phases	Ground	Takeoff	Climb	Cruise	Special phase determined by role	Descent	Approach	Landing
Sub Phases (Typical)	Taxiing, (ground or water), Catapult position, Confined areas	Accel - V_1, V_1 - V_R, V_R - V_{LO}, Aborted takeoff	1st segment, 2nd segment, Other segments	VOR tracking, Enroute climb, Altitude holding, Altitude change,	Air to Air, Dive bombing, Low level bombing, Refueling	Enroute descent	Terminal holding, Localizer capture, Glide path capture, ILS, tracking, Wave off	Flare to landing, Offset correct, Crosswind landing
Tasks	The actual work assigned a pilot which is to be performed in designated flight phases							

2.3.4 Performance

Flying qualities performance is defined as, “The precision of control with respect to aircraft movement that a pilot is able to achieve in performing a task.” (Reference 2.3). Performance is a quantifiable measure. For the example task of maintaining ground track, performance is quantified as maintaining taxi line and runway center line between the main landing gear.

The evaluation process involves judgement of pilot workload relative to degree of performance. The degree of precision is defined initially by the test team prior to the evaluation. Two degrees of precision are necessary for the process of assigning flying qualities ratings, desired and adequate performance.

2.3.4.1 DESIRED PERFORMANCE

Desired performance is the level of performance required to accomplish the task successfully in all mission environments. In the ground taxiing example, desired performance may be defined as maintaining nose gear on taxi line and runway center line within ± 5 ft. The degree of precision required for desired performance is determined from operational requirements. Our example aircraft may be required to operate in confined areas where taxi space is severely limited. In this environment, a high degree of precision in taxiing is required. Achieving desired performance indicates the task can be performed in the complete mission environment.

2.3.4.2 ADEQUATE PERFORMANCE

Adequate performance is a lower level of performance achievable only with the application of a higher level of pilot workload. Adequate performance is determined from operational requirements as well. Adequate performance indicates the task can be performed in a portion of the mission environment, but not in the complete environment. In our example, adequate taxi performance may be defined as maintaining nose gear on taxi line and runway center line ± 10 ft. Therefore, the aircraft could not be taxied in confined areas where the degree of precision required is ± 5 ft.

2.3.5 Compensation

Compensation is defined as, "The measure of additional pilot effort and attention required to maintain a given level of performance in the face of less favorable or deficient characteristics." (Reference 2.3). In our taxiing example, the compensation necessary to maintain desired performance could be described as the frequency and magnitude of directional control activity when attempting a straight ahead taxi.

2.3.6 Workload

The total pilot workload consists of workload due to compensation for aircraft deficiencies plus workload due to the task. In the taxi example, the total workload could be described as the frequency and magnitude of directional control activity required to initiate a taxi turn while maintaining desired performance. Generally, when describing a task, the evaluation pilot describes the total workload as follows:

> "Maintaining directional control during ground taxi in calm wind conditions over level prepared surfaces with the nose wheel ± 5 ft of the desired taxi line was easy, requiring ± 1/4 in directional control inputs every 8 to 10 s."

In this example, the pilot described the task, the environment, the desired performance, and the pilot workload.

2.3.7 Environment

The environment, both inside and outside the cockpit, influences the pilot's ability and workload required to complete the task. The internal environment is the sum of the tasks the pilot must perform simultaneously to accomplish the mission. It is one task to taxi the aircraft without performing other functions and another task to taxi the aircraft while tuning radios, making radio calls, and copying clearances. The taxi task can be significantly different when taxiing on a smooth, level, prepared surface in calm winds as compared to taxiing on unprepared surfaces or in gusty crosswinds.

Such conditions as day, night, Visual Meteorological Conditions (VMC), Instrument Meteorological Conditions (IMC), and turbulence level are environmental factors that affect visual cues, disturbance levels, and pilot workload. These factors affect pilot opinion of flying qualities and they should be specified in the test conditions.

Evaluating the task in a realistic operational environment minimizes the test pilot's extrapolation of his qualitative assessment. Generally, it is impractical to reproduce the environment in full fidelity. When some aspects of the environment are unavailable for the test program, it might be beneficial to introduce carefully designed, artificial aspects to approximate the operational environment and reduce extrapolation. For example, the test pilot might be required to perform a secondary task simultaneous with the evaluation task. Performing the secondary task simulates a realistic high stress environment.

2.3.8 Composite or Specific Ratings

Since the purpose of piloted flying qualities evaluations is to determine mission suitability of an aircraft, we may be tempted to assign an overall evaluation to the aircraft for the complete mission, but this composite approach is not practical. The complete mission is composed of many different flight phase, sub-phases, and tasks; each performed in different environments. Therefore, little useful information can be communicated about specific aircraft capabilities by assigning one rating. Similarly, if an entire flight phase is evaluated, there is little likelihood of the evaluation being performed in the same environment with the aircraft in the same state.

On the other hand, a separate, distinct task can be evaluated under one set of environmental conditions. Evaluating the aircraft for the complete mission using separate tasks, involves specifying and evaluating all the tasks which constitute the complete mission. Obviously, resources are not available to complete a flight test program of such magnitude. Therefore, task selection must consider the most important, critical, and representative tasks for the aircraft's mission.

2.3.9 Handling Qualities Rating Scale

Having defined the elements which are the foundation of the Handling Qualities Rating (HQR) scale, let us examine the scale and the pilot decisions required to use the scale (Table 2.II). The pilot, having defined the mission (selected the task, established the performance requirements, both desired and adequate, and determined the environment), performs the task. The pilot uses the maximum compensation necessary to obtain desired performance. If the desired performance is obtained easily, the pilot uses a lower level of compensation and total workload to accomplish the task; he does not interact frequently in the aircraft control loop. If the pilot does not achieve the desired performance, he must

Table 2.II
Handling Qualities Rating Scale

Pilot Rating

1

2

Aircraft Characteristics

3

YES

Excellent
Highly desirable

4

5

Is it satisfactory without improvement

NO

Good
Negligible deficiencies

6

Deficiencies warrant improvement

Fair Some mildly unpleasant deficiencies

YES

Adequacy for Selected Task or Required Operation*

Minor but annoying deficiencies

Demands on the Pilot in Selected Task or Required Operation*

7

8

NO

Pilot compensation not a factor for desired performance

Moderately objectionable deficiencies

9

Is adequate performance attainable with a tolerable pilot workload

Very objectionable but tolerable deficiencies

Pilot compensation not a factor for desired performance

Deficiencies require improvement

Major deficiencies

Minimal pilot compensation required for desired performance

10

YES

Is it controllable?

NO

Desired performance requires moderate pilot compensation

Major deficiencies

Major deficiencies

Adequate performance requires considerable pilot compensation

Improvement manditory

Adequate performance requires extensive pilot compensation

Major deficiencies

Pilot Decisions

Adequate performance not attainable with maximum tolerable pilot compensation. Controllability not in question

Considerable pilot compensation is required for control

Intense pilot compensation is required to increase his compensation trying to obtain desired performance. He may have to reduce the time spent on auxiliary or supporting tasks and direct his entire concentration to achieving desired performance for the evaluation task.

The pilot must maintain situational awareness at all times and operate the aircraft safely. If the pilot is unable to achieve desired performance with appropriate compensation (some auxiliary tasks cannot be ignored), he attempts to achieve adequate performance. Again the level of compensation used is that required to achieve adequate performance. This discussion assumes the pilot can control the aircraft and is able to perform the task. If controllability is in question, the pilot must direct all his attention to basic aircraft control and is unable to perform any mission tasks.

While the pilot performs the task, he describes task performance, compensation, and total workload. The pilot makes his comments using the simplest piloting terms possible. Do not analyze the results or aircraft response in complex stability and control terminology. Keep the comments simple and direct. The comments should be recorded concurrent with performing the task and shortly after task completion; ideally, recorded on a voice recorder and summarized on kneeboard cards. Comments should include a description of the environment at the time of task performance.

Immediately after task completion and after making all relevant comments, the pilot assigns the HQR. Starting at the lower left hand corner of the HQR scale (Table 2.II), the pilot enters the decision matrix. The pilot follows the decision matrix, moving upward and branching to the right until he assigns a HQR.

Is it "Controllable"?

To control is to exercise direction of, to command, or to regulate. The determination as to whether the airplane is controllable must be made within the framework of the defined mission or intended use. An example of the considerations of this decision would be the evaluation of an aircraft's handling qualities during which the evaluation pilot encounters a situation in which he can maintain control only with complete and undivided attention. The aircraft is "controllable" in this situation in the sense that the pilot can maintain control only by restricting the tasks and maneuvers he is called upon to perform and by giving the configuration undivided attention. However, to answer, "Yes, it is controllable in the flight phase or task," the pilot must be able to retain control in all mission oriented tasks and other required operations without sacrificing effort and attention to his overall duties. If the final overall answer to these questions is no, the pilot will assign an HQR-10.

A major division in the rating scale is made between HQR-9 and HQR-10. If control is or will be lost, the rating is 10. If control is or can be maintained, the rating is 9 or below. A rating of 9.5 is not permitted; therefore, a decision on control must be made here.

Is adequate performance attainable with a tolerable pilot workload?

There are really two questions asked here. First, is adequate performance attainable? If the pilot attains at least the adequate tolerance, the answer to the first part is yes. However, if adequate performance cannot be achieved, the answer is no. The second part of the question is, "Is the workload tolerable while attaining adequate performance?" Tolerable workload describes the capability to perform auxiliary tasks such as communication, navigation, and maintaining situational awareness. The pilot must be able to complete auxiliary tasks in addition to the evaluation task to maintain safe flight conditions. The pilot may consider that the designated flight phase or task can be accomplished but the effort, concentration, and workload required are of such magnitude that the pilot rejects the aircraft for this phase of its intended use. If the answer to either of the two questions is "No", then the aircraft is not suitable for the purpose, though it is controllable.

An HQR-7 states that adequate performance is not attainable with maximum tolerable pilot compensation, but controllability is not in question. The pilot workload is beyond what is tolerable although adequate performance may have been attained. The pilot's workload requires a major portion of his attention, but the issue is not one of controllability.

An HQR-8 is applied to a piloting requirement which can only be done with a minimum of cockpit duties. Considerable pilot compensation is then required for control of the task.

An HQR-9 may enter the dangerous control area being just controllable with complete attention to the task at hand. Intense pilot compensation is required to retain control. The rating appears similar to the HQR-10, but the pilot has already determined that it was controllable.

Is it satisfactory without improvement?

The next major question to be answered is “Is it satisfactory without improvement?” If the answer is yes, the pilot’s definition might be that it isn’t perfect, or even good, but that it is good enough to not require changes at least in this model. It meets a standard; it has sufficient goodness; it’s of a kind to meet all pilot demands for the intended use. If the answer to this question is no, the implication is that adequate or desired performance can be achieved only with a workload which impacts the pilot’s capability to perform auxiliary, or secondary, tasks simultaneously with the evaluation task. In this case, a deficiency exists which warrants improvement and the pilot selects the rating which best describes the situation.

HQR-4 means the desired level of performance was reached but with some work categorized as moderate pilot compensation. The aircraft characteristic is minor but annoying to the piloting task. If challenged enough, pilots can achieve desired levels of performance but the workload seems more than moderate. The words for an HQR-5 may better describe the workload although desired performance was attained. A division between desired and adequate occurs within this major section of the scale. At times it appears an HQR-5 is a better rating for a desired performance level. Discussions over many years have lead to two approaches. One allows an HQR-4.5 when a desired level of performance is attained but the workload is higher than described. The other allows an HQR-5 to be assigned to a desired level of performance at higher than described workload. Assigning an HQR-6 to a desired level of performance is too extreme and normally means the task and tolerances need to be better defined.

When desired performance is not attainable, an HQR-5 or 6 is the rating assigned within the major division. The pilot has determined that the characteristic is not satisfactory and the desired level of performance is not attainable. The rating now rests on the words for demands on the pilot and/or aircraft characteristics. Both should be considered when choosing the rating. For the demands on the pilot, adequate performance requires considerable (HQR-5) or extensive (HQR-6) pilot compensation. For the aircraft, the deficiencies are moderately objectionable (HQR-5) or very objectionable but tolerable (HQR-6).

Yes, it is satisfactory without improvement.

If task performance is satisfactory without improvement, desired performance was attained. The pilot selects the rating which best describes the situation. "Excellent, Highly desirable. Pilot compensation not a factor for desired performance," HQR-1. This rating is selected if the pilot makes one control input, the desired performance is achieved without error, and additional pilot inputs are not required. "Good Negligible deficiencies. Pilot compensation not a factor for desired performance," HQR-2. This rating implies that desired performance was not achieved on the first pilot input, but subsequent inputs were very minimal. "Fair, Some mildly unpleasant deficiencies. Minimal pilot compensation required for desired performance," HQR-3. This rating implies that a deficiency exists but its correction is not required to accomplish the mission.

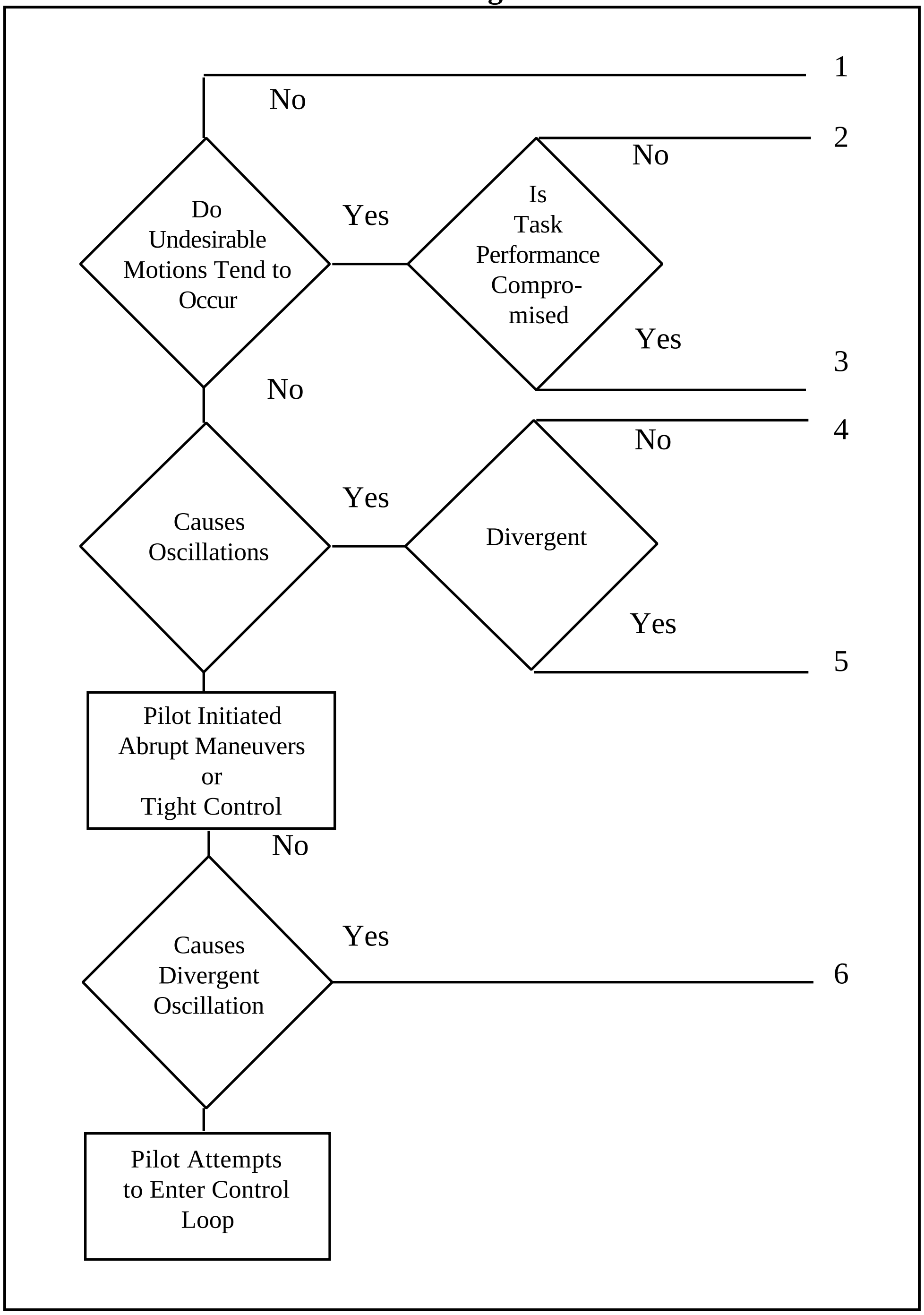
Half ratings are discouraged! The rating scale is broad enough to cover contingencies from highly desirable flying qualities to situations where control will be lost. Specifically, half ratings are not permitted between major decision points. HQRs assigned by trained test pilots to well defined tasks with defined performance in the same environment are repeatable. The HQRs are repeatable for the same test pilot over time or for different test pilots. Studies show the average standard deviation for well constructed flying qualities evaluations using HQRs is less than one.

2.3.10 Additional Rating Scales

An aid to assist the pilot describing performance, workload, and environment is the Pilot Induced Oscillation (PIO) rating scale (Table 2.III). This rating scale helps to classify the susceptibility of the task to PIO.

Table 2.III

PIO Rating Scale



The Turbulence Rating Scale (Table 2.IV) is a tool used to describe the environment. This scale, extracted from the *Flight Information Handbook* (Reference 2.1), describes conditions which exist during task performance.

Table 2.IV
Turbulence Rating Scale

INTENSITY	AIRCRAFT REACTION	REACTION INSIDE AIRCRAFT
LIGHT		
Turbulence Chop	Turbulence that momentarily causes slight, erratic changes in altitude and/or attitude. Turbulence that causes slight, rapid and somewhat rhythmic changes in altitude or attitude	Occupants may feel a slight strain against seat belts or shoulder straps. Unsecured objects may be displaced slightly. Food service may be conducted and little or no difficulty is encounter in walking.
MODERATE		
Turbulence Chop	Turbulence that causes changes in altitude and/or attitude, but with the aircraft remaining in positive control at all times. It usually causes variations in indicated airspeed. Turbulence that causes rapid bumps or jolts without appreciable changes in aircraft altitude or attitude.	Occupants feel definite strains against seat belts or shoulder straps. Unsecured objects are dislodged. Food service and walking are difficult.
SEVERE		
Turbulence	Turbulence that causes large, abrupt changes in altitude and/or attitude. It usually causes large variations in indicated airspeed. Aircraft may be momentarily out of control.	Occupants are forced violently against seat belts or shoulder straps. Unsecured objects are tossed about. Food service and walking are impossible.
EXTREME		
Turbulence	Turbulence in which the aircraft is violently tossed about and is practically impossible to control. It may cause structural damage.	
FREQUENCY	DEFINITION	
Occasional	Less than 1/3 of the time	
Intermittent	1/3 to 2/3 of the time	
Continuous	More than 2/3 of the time	

The Vibration Rating Scale (Table 2.V) is another rating scale the pilot uses to describe the environment. This scale classifies, quantifies, and communicates in-flight observations.

Table 2.V
Vibration Rating Scale

Degree of Vibration	Description	Pilot Rating
No Vibration		0
Slight	Not apparent to experienced aircrew fully occupied by their tasks, but noticeable if their attention is directed to it or if not otherwise occupied.	1 2 3
Moderate	Experienced aircrew are aware of the vibration but it does not affect their work, at least over a short period.	4 5 6
Severe	Vibration is immediately apparent to experienced aircrew even when fully occupied. Performance of primary task is affected or tasks can be done only with difficulty.	7 8 9
Intolerable	Sole preoccupation of aircrew is to reduce vibration.	10

Based on the Subjective Vibration Assessment Scale developed by the Aeroplane and Armament Experimental Establishment, Boscombe Down, England.

2.4 TEST METHODS AND TECHNIQUES

2.4.1 General

Since the objective of flying qualities evaluations is communicating the test pilot's qualitative assessment of the aircraft mission suitability, an orderly process should be observed. As is true for any communication, format should be standardized so data gathered by different pilots can be compared with the least chance of misinterpretation. The guidelines to accomplish the evaluation process are outlined in subsequent sections.

2.4.2 Test Planning

During the planning stage, the test team refers to specifications and requirements documents to determine the aircraft mission. The team determines the environment in which the aircraft will operate and which they will use for task evaluation. The team determines the mission representative tasks for evaluation and the priority for task evaluation. The team establishes the performance tolerances required for task completion, both desired and adequate. The team relies on fleet experienced pilots to guide task and performance selection.

Following mission definition, task selection, and performance tolerance determination, the team prepares the flight test data cards. The flight test data cards detail the specific task/maneuver sequences for the pilot evaluation. Usually an optimistic approach is taken in planning, including more tasks than can be accomplished during the evaluation. However, the tasks are prioritized and sequenced in a logical manner. The sequence is arranged for the most efficient use of the flight time.

The data card is used to prompt the pilot for all desired information. In general, cards that solicit narrative responses as opposed to one word answers are preferred.

2.4.3 Test Techniques

Following the flight test data card, the test pilot executes the evaluation tasks. The pilot performs the tasks with the same level of aggressiveness applicable to the mission environment. The test pilot uses the HQR Scale, PIO Rating Scale, Turbulence Rating Scale, and Vibration Rating Scale to quantify his observations. All are recorded on the data card.

The test pilot makes comments extemporaneously while performing the evaluation tasks. The data card lists items of interest. Generally, the pilot refers to the data card after finishing the task and completes his observations. The substance of evaluation data lies in the pilot comments.

Recording comments and rating data simultaneously is both difficult and time consuming while performing the evaluation tasks. Therefore, it is advantageous to carry some form of voice recorder on board. A voice activated unit is most efficient.

The following summarizes the evaluation process.

1. Fly the task/maneuver as planned.
2. Observe both the vehicle's responses and the nature of control inputs, relate the two in the closed control loop. What the aircraft did and how the pilot responded?
3. Make extemporaneous comments of pertinent observations during task performance.
4. Use plain piloting language to describe direct observations not technical or engineering language (e.g., frequency, damping, etc.). Do not interpret the results.
5. Be complete in comments, when the voice tape is transcribed leave no room for doubt about in-flight comments.
6. Refer to the data card at task completion and ensure all items are covered.
7. Assign a HQR using the rating scale each time. Assign the HQR as soon as the task is completed, while everything is in perspective. Complete the data card. Review the rating. If a change in rating is considered, justify it. Summarize the primary reasons for the rating.
8. Describe any pertinent environmental factors such as weather, turbulence, etc., which qualify the evaluation.

As soon as feasible, the test pilot conducts a debrief with the flight test team. He transcribes his comments from the voice tape and data cards and clarifies any confusion in the comments while the evaluation is fresh in his mind. The transcription is accomplished with one of the test engineers and put into a format for analysis.

2.4.4 Data Required

The data required for a piloted flying qualities evaluation are task definition, definition of desired and adequate performance, actual performance achieved, task environment, pilot compensation and total workload required for the task, HQR, any additional numerical ratings assigned to clarify the HQR and the transcribed pilot comments. These data are obtained from the flight test data cards and the voice recorder.

2.4.5 Safety Considerations/Risk Management

The principal consideration for flight test is flight safety. A test pilot or flight test crew can become so absorbed in performing, observing, and recording a task evaluation that basic aircraft control, situational awareness and flight discipline are lost. Under no circumstances is task evaluation more important than maintaining safe operating conditions. If flight safety is compromised, stop the evaluation. Regain basic flight discipline, situational awareness, and flight safety before continuing. If the task cannot be accomplished without further compromise, stop the evaluation and replan the flight on the ground.

2.4.6 Special Factors Affecting Evaluation in Helicopters

A number of factors peculiar to helicopters influence the evaluation process. The following list contains some factors which should be considered in planning and executing helicopter flight test.

1. Ambient vibration levels, primarily from the rotor system.
2. Cockpit field of view downward for ground reference maneuvers.
3. The accelerations perceived by the pilot due to seat location relative to the center of gravity may be particularly important in terms of the cues the pilot senses in closed loop tasks.
4. Tail cone or nose boom strike by the rotor.
5. Tip path plane dynamics due, for example, to a particular aerodynamic loading.
6. Effectiveness of RPM governor. For particular tests, the governor characteristics may need to be accommodated to satisfy test requirements.
7. Control and motion coupling. Pilots may be required to suppress such coupling.
8. Control nonlinearities. Open-loop testing should account for such possibilities.

2.5 DATA PRESENTATION

The data are presented in narrative format in the results and discussion section of the report. A sample narrative presentation is a follows:

> "Maintaining directional control during ground taxi in calm wind conditions over level prepared surfaces with the nose wheel ± 5 ft of the desired taxi line was easy, requiring ± 1/4 in directional control inputs every 8 to 10 s (HQR 3)."

Piloted flying qualities evaluation data for a large number of mission tasks may be summarized in tabular format and included in an appendix. Data for the same task performed under different environmental conditions, or by different pilots, may be summarized in tabular format. Comparative data are best presented in graphical format.

Preserve the perspective that flying qualities data is bottom line information and the results of open loop testing, including Military Specification compliance testing, supports this bottom line. Present the closed loop or handling qualities results first. Use the open loop data to support the pilot qualitative evaluation. Too often the quantitative testing results are presented. The flying qualities or qualitative data is presented in less detail or in a less definitive format. This form of presentation can misdirect the emphasis. Furthermore, it does not provide the proper focus for the open loop data. We are interested in open loop data primarily for their explanatory function with respect to the observed flying qualities.

2.6 GLOSSARY

2.6.1 Notations

ft	Foot
HQR	Handling Qualities Rating
IMC	Instrument Meteorological Conditions
PIO	Pilot Induced Oscillation
s	Second
VMC	Visual Meteorological Conditions

2.7 REFERENCES

1. *Flight Information Handbook*, DOD Flight Information Publication, Defense Mapping Agency Aerospace Center, St. Louis, MO, 23 August, 1990.

2. Naval Test Pilot School Flight Test Manual, *Fixed Wing Stability and Control, Theory and Flight Test Techniques*, USNTPS-FTM-No.103, U. S. Naval Test Pilot School, Patuxent River, MD, 1 January, 1975, Revised 1 November, 1981.

3. NASA Technical Note, NASA TN D-5153, “The Use of Piloted Rating In The Evaluation Of Aircraft Handling Qualities”, National Aeronautics and Space Administration, Washington, D.C., April, 1969.

CHAPTER THREE

OPEN LOOP TESTING

CHAPTER THREE

FIGURES

CHAPTER THREE

EQUATIONS

		PAGE
$m\ddot{x} + B\dot{x} + Kx = 0$	*eq 3.1*	3.7
$\lambda^2 + \frac{B}{m}\lambda + \frac{K}{m} = 0$	*eq 3.2*	3.7
$\lambda_{1,2} = -\frac{B}{2m} \pm \sqrt{\left(\frac{B}{2m}\right)^2 - \frac{K}{m}}$	*eq 3.3*	3.8
$x(t) = C_1 e^{\lambda_1 t} + C_2 e^{\lambda_2 t}$	*eq 3.4*	3.8
$x(t) = Ae^{-\zeta\omega_n t} \sin \omega_n \sqrt{1 - \zeta^2}\, t$	*eq 3.5*	3.9
$B_{CRIT} = 2m\left(\frac{K}{m}\right)^{\frac{1}{2}}$	*eq 3.6*	3.10
$\omega_n = \left(\frac{K}{m}\right)^{\frac{1}{2}}$	*eq 3.7*	3.12
$\zeta = \frac{B}{B_{CRIT}}$	*eq 3.8*	3.13
$\lambda^2 + 2\,\omega_n \zeta\lambda + \omega_n^2 = 0$	*eq 3.9*	3.13

$$\lambda_{1,2} = -\zeta\omega_n \pm i\,\omega_n \left(1 - \zeta^2\right)^{\frac{1}{2}}$$

eq 3.10 3.13

$$\zeta = \sin E_d$$

eq 3.11 3.14

$$\cos E_d = \sqrt{1 - \zeta^2}$$

eq 3.12 3.14

$$\omega_d = \omega_n \sqrt{1 - \zeta^2}$$

eq 3.13 3.14

$$\zeta\omega_n = \frac{\ln_e N}{T_{1/N}}$$

eq 3.14 3.14

$$\dot{w} - Z_w\, w = 0$$

eq 3.15 3.15

$$w = C_1\, e^{\lambda t}$$

eq 3.16 3.16

$$\tau = -\frac{1}{\lambda} = -\frac{1}{Z_w}$$

eq 3.17 3.17

$$C_{1/2} = \frac{T_{1/2}}{P}$$

eq 3.18 3.23

$$T_{1/N} = \frac{\ln_e N}{\zeta\omega_n}$$

eq 3.19 3.23

$$C_{1/N} = \frac{\ln_e N}{2\pi \dfrac{\zeta}{\sqrt{1 - \zeta^2}}}$$

eq 3.20 3.24

CHAPTER THREE

OPEN LOOP TESTING

3.1 INTRODUCTION

The major reason for conducting flying qualities evaluations is to determine if the pilot-aircraft combination can perform safely and precisely the tasks required for the mission. This determination can be made by the pure qualitative approach to stability and control testing via flying qualities evaluation or closed loop testing. However, the flying qualities evaluation is only part of the complete test program. Open loop testing is conducted to quantify the stability and control parameters and support the test pilot's qualitative evaluation. A balance between qualitative pilot opinion and quantitative testing must be achieved in any stability and control test and evaluation program.

The open loop system the pilot interfaces with in closed loop testing includes the augmented aircraft characteristics and the control system characteristics. Open loop testing is aimed at determining the transfer characteristics of these elements.

3.2 PURPOSE

The purpose of open loop testing is to quantify stability and control evaluations. Specific objectives of open loop testing are:

1. Substantiate the pilot's qualitative opinion.
2. Document characteristics of the aircraft-control system combination.
3. Provide data for comparing the aircraft characteristics with others and for formulating future designs.
4. Provide baseline data for expansion of flight and center of gravity envelope.
5. Determine compliance or noncompliance with specifications.
6. Provide data for predictive closed loop analysis.
7. Provide data for simulator applications.

3.3 THEORY

3.3.1 General

To exhibit satisfactory flying qualities, the aircraft must possess a balance of both stability and control. The optimum blend depends on the aircraft's mission. A degree of stability is necessary to reduce pilot workload. However, too much stability can degrade the pilot's ability to maneuver the aircraft. An optimum blend of stability and control is one goal of the aircraft designer. When the optimum mix of stability and control is attained, the aircraft flying qualities enhance the pilot's ability to perform the mission.

Stability and control can be viewed as opposite ends of a spectrum, with stability on one end and control on the other. While both characteristics are desirable, too much of one degrades the contributions of the other. A helicopter with a high degree of stability is easy to fly straight and level but is difficult to maneuver quickly and precisely. On the other hand, a helicopter which responds quickly to control inputs may lack sufficient stability for flight in instrument conditions. Static and dynamic stability partially determine the pilot's ability to control the aircraft. While static instability about any axis is generally undesirable, excessively strong positive static stability can degrade maneuverability to an unacceptable degree. For some tasks, neutral static stability is acceptable or even desirable. The optimum level of static stability depends on the specific mission oriented tasks of the aircraft.

3.3.2 Stability

The helicopter is a dynamic system. It is a body in motion under the influence of forces and moments producing or changing motion. In order to investigate the motion of the helicopter, it is necessary to establish a condition of equilibrium between opposing forces and moments. When the equilibrium condition is achieved, the stability characteristics can be determined.

3.3.2.1 STATIC STABILITY

Static stability is the sum of the forces and moments following a disturbance from an equilibrium flight condition. The helicopter is statically stable (positive static stability) if forces and moments are generated to restore the aircraft toward equilibrium. The helicopter is statically unstable (negative static stability) if the sum of forces and moments following a disturbance from an equilibrium flight condition cause a further divergence from equilibrium. The helicopter has neutral static stability if the forces and moments following a disturbance from an equilibrium flight condition cause the helicopter to continue in the direction of the disturbance without diverging from or converging toward equilibrium. Static stability is represented by the initial response of the aircraft when disturbed from an equilibrium flight condition. The aircraft has positive static stability if the initial response is toward the equilibrium flight condition. The aircraft has negative static stability if the initial response is away from the equilibrium condition. If the aircraft continues in the direction of the disturbance without a return toward or away from the equilibrium flight condition, the aircraft has neutral static stability.

Static stability characteristics must be investigated at equilibrium flight conditions in which all forces and moments are in balance. In wind tunnel tests, the forces and moments produced by a disturbance from an equilibrium flight condition can be measured directly. The direct in flight measurement of static stability parameters is not feasible in many instances. Therefore, the flight test team measures parameters which give indications of static stability. These indications are adequate to establish the mission effectiveness of the aircraft and are more meaningful to the pilot than the numerical value of actual static stability parameters. For example, static longitudinal stability is evaluated through the longitudinal control displacement required to counter or balance an off trim airspeed condition, not by measuring directly the moment produced by the off trim airspeed condition.

3.3.2.2 DYNAMIC STABILITY

Dynamic stability is the response of the aircraft over time after the initial reaction to a disturbance from an equilibrium flight condition. The time history of the aircraft response can be either oscillatory or aperiodic. The time history response can be either convergent, neutral, or divergent. The aircraft has positive dynamic stability if the response converges to the initial equilibrium flight condition. If the response diverges from the initial equilibrium flight condition, the aircraft has negative dynamic stability. The response can be aperiodic and convergent (positive dynamic stability), aperiodic and divergent (negative dynamic stability), oscillatory and convergent (positive dynamic stability), or oscillatory and divergent (negative dynamic stability). With neutral dynamic stability, the response neither converges or diverges. The true oscillatory neutral response is seldom found in aircraft dynamic responses. Static stability determines the initial response of the aircraft. If the aircraft has negative static stability, the initial response is away from trim and the dynamic response is aperiodic divergent.

As the pilot changes from one equilibrium flight condition to another, one or more of the aircraft's dynamic modes of motion are excited. Changes are initiated by pilot inputs which excite these dynamic modes. These modes of motion can be excited by external perturbations as well. The study of the characteristics of these modes of motion is the study of dynamics.

Time history parameters of the aircraft modes of motion define the dynamic stability characteristics. The most important parameters are the frequency and damping ratio of the motion. The frequency of the motion is the number of cycles per unit time and is a measure of the quickness of the motion. The term natural frequency, ω_n, used in analytical discussions, is the frequency the motion exhibits with no damping. However, the pilot observes the damped frequency, ω_d, usually referred to simply as frequency, ω.

3.3.3 Damping

Positive damping is the characteristic which causes progressive diminishing of the amplitudes or rates of the transient motion. Damping is a measure of the subsidence of the motion. Because it is observable by the pilot, the term damping ratio, ζ, is used rather than damping to describe the subsidence of an oscillatory motion. Both frequency and damping ratio are characteristics of a system response which affect flying qualities. For example, if the damping ratio is too low, aircraft motion is a nuisance when excited by pilot inputs or by atmospheric disturbances. If the damping ratio is too high, the aircraft response to control input appears sluggish. Desirable values of these characteristics depend on an aircraft’s mission. The corresponding characteristic of a first order response is the time constant, τ, defined as the time to reach 63.2 % of the steady state value.

3.3.4 Control Response

The pilot controls the helicopter through the dynamic modes of motion. The pilot first excites the dynamic mode, allows the helicopter to respond, and as the helicopter reaches the desired flight condition, the pilot suppresses the dynamic mode.

Control response testing is the evaluation of aircraft response following step displacement of a flight control. Control response testing is open loop testing of the dynamic response of the aircraft. Aircraft responses are often assumed to be either first or second order dynamic responses for data reduction purposes.

In discussions of control response, several terms are used. Control power is a measure of the moment produced per unit of control displacement. Typical units for control power are lbf in/in control displacement or Newton meter/% control displacement. Control sensitivity is the initial angular acceleration produced by a unit step control displacement. Typical units for control sensitivity are deg/s^2/in control displacement. Attitude control effectiveness (attitude control response) is the change in aircraft attitude achieved in one second following a unit step control displacement. Typical units for attitude control effectiveness are deg/in control displacement. Rate control effectiveness (rate control response) is the angular rate achieved one second following a unit step control displacement. Typical units for rate control effectiveness are deg/s/in control displacement.

3.3.5 Second Order System

A helicopter in flight can be represented by a mass restrained by springs, which are comparable to static stability, and dampers which are comparable to aerodynamic damping (Figure 3.1). For simplicity, consider a single degree of freedom to yield the equivalent spring-mass-damper system in Figure 3.1 (b).

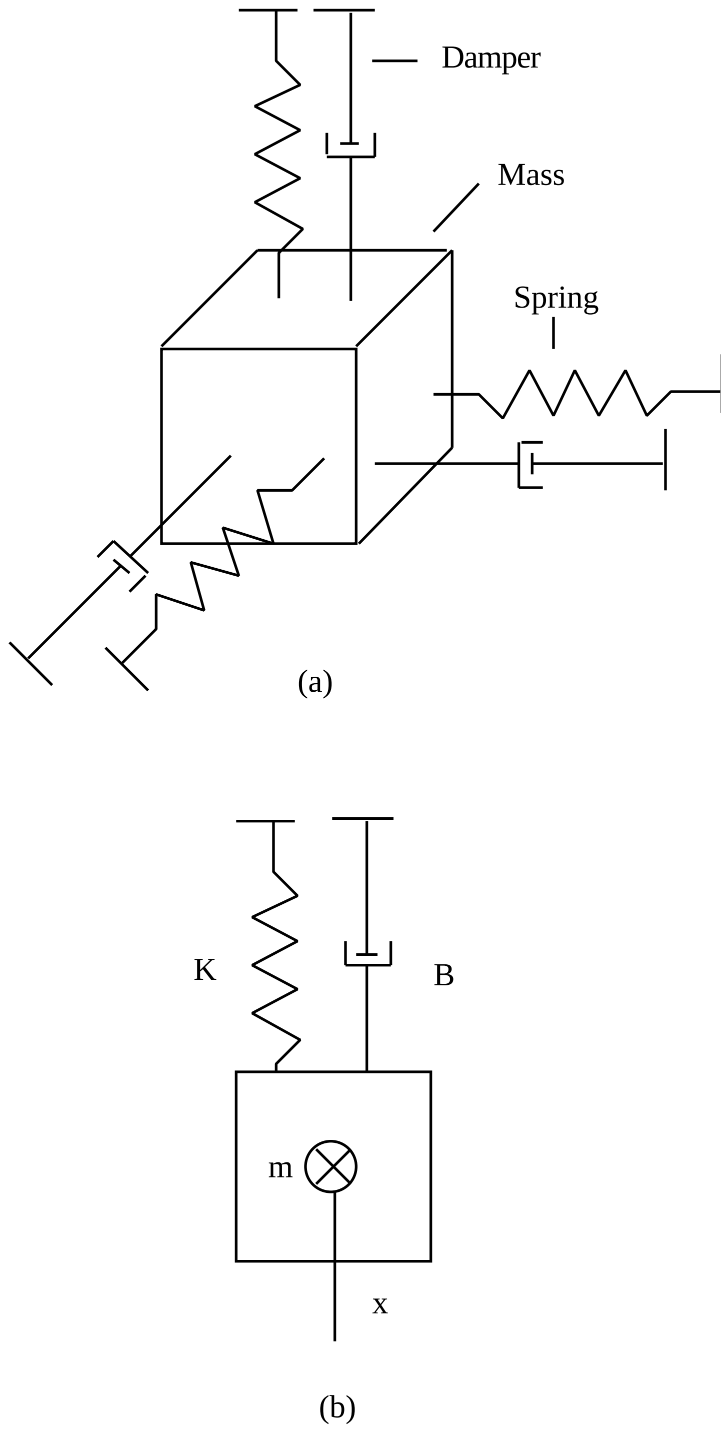


Figure 3.1
Spring Mass Damper System

The solution of the equation of motion for the system represented in Figure 3.1 (b) provides a useful insight into the solution of the more complex equations of motion of the helicopter. If the mass, m, is displaced from equilibrium an amount, x, and released, the subsequent free response of the system is described by the following second order linear homogeneous differential equation:

$$m\ddot{x} + B\dot{x} + Kx = 0 \qquad \textit{eq 3.1}$$

Where:

B	- Damping constant
K	- Spring constant
m	- Mass
x	- Displacement
$\ddot{x}$	- Acceleration
$\dot{x}$	- Velocity.

At the moment of release, the spring exerts a force, Kx, in the direction to restore the mass to its equilibrium position. After release, the viscous damper force opposing the motion is $B\dot{x}$. The inertia force is $m\ddot{x}$. The characteristic equation can be written as:

$$\lambda^2 + \frac{B}{m}\lambda + \frac{K}{m} = 0 \qquad \textit{eq 3.2}$$

Where:

B	- Damping constant
K	- Spring constant
λ	- Characteristic root
m	- Mass.

Solving the characteristic equation yields the following roots:

$$\lambda_{1,2} = -\frac{B}{2m} \pm \sqrt{\left(\frac{B}{2m}\right)^2 - \frac{K}{m}} \qquad \textit{eq 3.3}$$

Where:

B	- Damping constant
K	- Spring constant
$\lambda_{1,2}$	- Characteristic roots
m	- Mass.

The corresponding time responses are as follows:

1. If, $\left(\frac{B}{2m}\right)^2 > \frac{K}{m}$, λ_1 and λ_2 are real roots and the time response is a combination of first order (non-oscillatory) exponential responses of the form:

$$x(t) = C_1 e^{\lambda_1 t} + C_2 e^{\lambda_2 t} \qquad \textit{eq 3.4}$$

Where:

C_1 and C_2	- Constants affected by initial conditions of the motion
e	- Base of natural logarithm
$\lambda_{1,2}$	- Characteristic roots
t	- Time
x	- Displacement.

2. If, $\left(\frac{B}{2m}\right)^2 < \frac{K}{m}$, λ_1 and λ_2 are a complex conjugate pair and the time response is a sinusoidal response of the form:

$$x(t) = Ae^{-\zeta\omega_n t} \sin \omega_n \sqrt{1 - \zeta^2}\, t \qquad \textit{eq 3.5}$$

Where:

A	- Constant affected by initial conditions of the motion
e	- Base of natural log
t	- Time
ω_n	- Natural frequency
x	- Displacement
ζ	- Damping ratio.

The significance of the positions of the roots is shown in Figure 3.2.

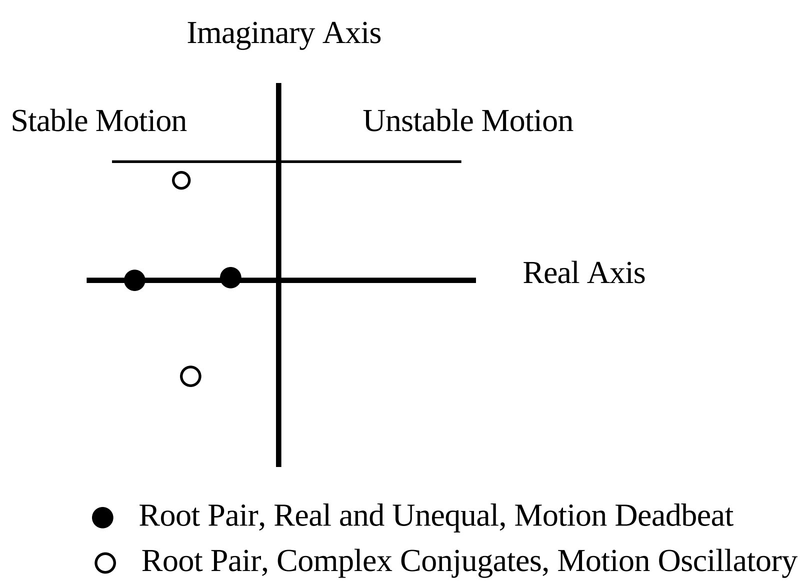


Figure 3.2
The Complex Plane

It is interesting to note the nature of these characteristic roots as the value of the spring constant, K, is increased from zero. The movement of the roots is shown in Figure 3.3. Point A, Figure 3.3, corresponds to K=0 or neutral static stability. The corresponding time response is shown in Figure 3.4 (a). For aperiodic roots lying on the real axis of the left half plane, Figure 3.4 (b) shows the subsidence time history. As long as the damping of the system is predominant, $\left(\frac{B}{2m}\right)^2 \geq \frac{K}{m}$, the roots lie along the real axis and the motion of the system is convergent without oscillation. This motion is described as aperiodic or deadbeat subsidence (the system is overdamped). If $\frac{K}{m} = \left(\frac{B}{2m}\right)^2$, the roots meet at point B, Figure 3.3, on the real axis. The motion of the system is still described as aperiodic or dead-beat subsidence. However, it is on the verge of becoming oscillatory and is critically damped. The value of the damping of the system at this point is called critical damping, B_{CRIT}.

$$B_{CRIT} = 2m \left(\frac{K}{m}\right)^{\frac{1}{2}} \qquad \textit{eq 3.6}$$

Where:

B_{CRIT}	- Critical damping
K	- Spring constant
m	- Mass.

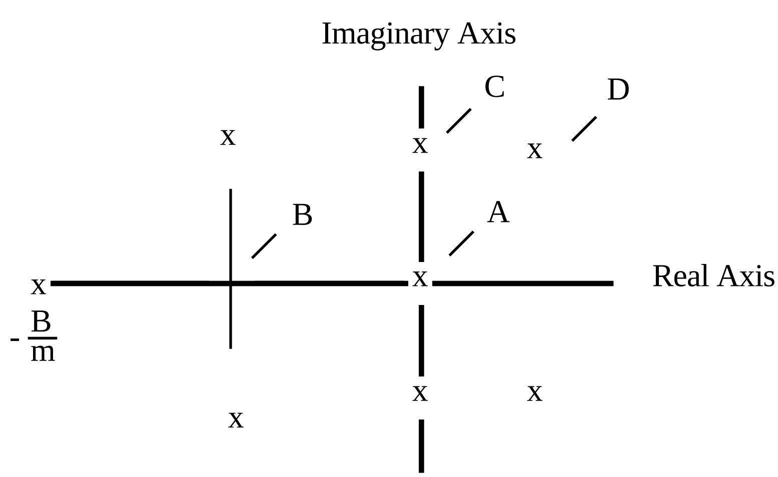


Figure 3.3
Effect of Increasing Spring Constant

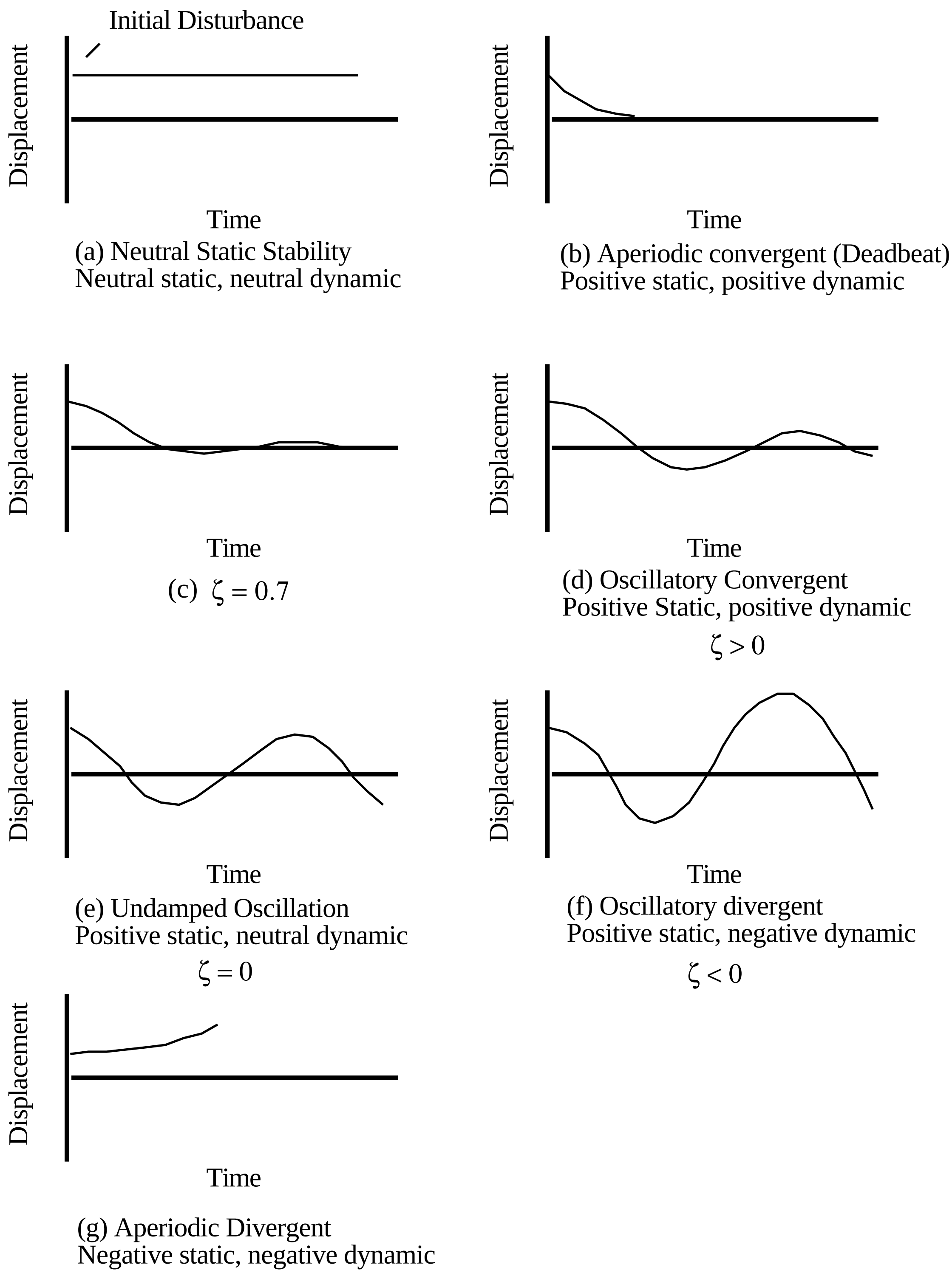


Figure 3.4
Time Histories of First and Second Order Systems

If the spring constant is increased further such that $\frac{K}{m} > \left(\frac{B}{2m}\right)^2$, the solutions to the equation of motion are composed of real and imaginary parts. The roots split at point B, Figure 3.3, the real part remains constant and as K increases, the imaginary part becomes larger. The motion of the system is now oscillatory and the frequency increases as K increases. However, for all values of K, the motion is damped after the disturbing force is removed.

The measure of the strength of the system to seek an equilibrium condition is called the system stiffness and is directly related to the square of the system frequency when damping is not present. This frequency is called the natural frequency, ω_n. It is usually a computed number since most systems have damping and the measured system frequency will be the damped natural frequency, ω_d. The natural frequency for the spring-mass-damper system is expressed as follows:

$$\omega_n = \left(\frac{K}{m}\right)^{\frac{1}{2}} \qquad \textit{eq 3.7}$$

Where:

K	- Spring constant
m	- Mass
ω_n	- Natural frequency.

The level of dynamic stability of a second order system is generally expressed in terms of the damping ratio, ζ. It is the ratio of the real system damping constant to the damping constant which makes the system critically damped and is related to the number of overshoots in free response:

$$\zeta = \frac{B}{B_{CRIT}}$$

eq 3.8

Where:

B	- Damping constant
B_{CRIT}	- Critical damping
ζ	- Damping ratio.

The characteristic equation for the spring-mass-damper system, written in terms of natural frequency and damping ratio, is as follows:

$$\lambda^2 + 2\,\omega_n\,\zeta\lambda + \omega_n^2 = 0$$

eq 3.9

The two roots of the equation are:

$$\lambda_{1,2} = -\,\zeta\omega_n \pm i\,\omega_n\left(1 - \zeta^2\right)^{\frac{1}{2}}$$

eq 3.10

Where:

i	- Imaginary index
$\lambda_{1,2}$	- Characteristic roots
ω_n	- Natural frequency
ζ	- Damping ratio.

These roots plotted on the complex plane are shown in Figure 3.5. Several important relationships are also presented:

$$\zeta = \sin E_d \tag{eq 3.11}$$

$$\cos E_d = \sqrt{1 - \zeta^2} \tag{eq 3.12}$$

$$\omega_d = \omega_n \sqrt{1 - \zeta^2} \tag{eq 3.13}$$

$$\zeta\omega_n = \frac{\ln_e N}{T_{1/N}} \tag{eq 3.14}$$

Where:

E_d	- Damping angle
$\ln_e$	- Natural logarithm
N	- Denominator of the fractional amplitude
$T_{1/N}$	- Time to decay to 1/N of maximum amplitude
ω_d	- Damped frequency
ω_n	- Natural frequency
ζ	- Damping ratio.

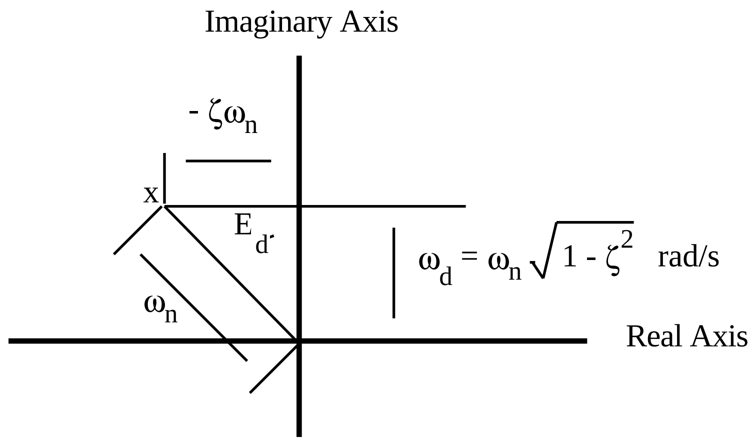


Figure 3.5
Relationship of Root Location on Complex Plane to Motion Characteristics

As the complex pair of roots separate vertically from the real axis, there is a point of interest where the damping angle, $E_d = 45^\circ$, and $\zeta \approx 0.7$. The corresponding time response is shown in Figure 3.4 (c). The oscillation has essentially one noticeable overshoot. This situation is considered to be the best compromise between quickness of initial response and minimum oscillation about steady state. Beyond this root location, $\zeta < 0.7$ and Figure 3.4 (d) applies.

If the characteristics of the system yield roots in location C on Figure 3.3, the time response is a constant amplitude oscillation as shown in Figure 3.4 (e). If the roots fall in position D on Figure 3.3, the response is a divergent oscillation as shown in Figure 3.4 (f).

3.3.6 First Order Systems

A first order system is described by a first order linear homogeneous differential equation. A mass whose motion is restrained by only viscous damping is a typical example. With no stiffness, the variable is velocity not displacement. The heave mode of a helicopter in a hover is a first order system with the following characteristic equation:

$$\dot{w} - Z_w w = 0 \qquad \textit{eq 3.15}$$

Where:

- w - Translational velocity component along z axis
- Z_w - Vertical force due to vertical velocity
- $\dot{w}$ - Linear acceleration along the z axis.

It has a real root $\lambda = Z_w$ where Z_w, a negative number, yields the root location on Figure 3.6.

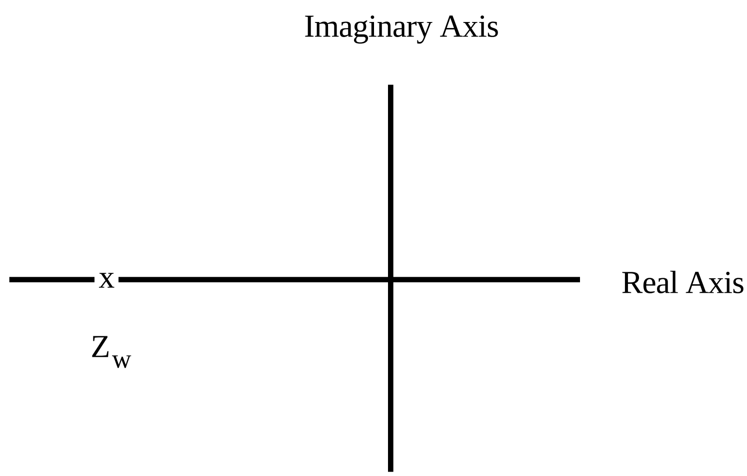


Figure 3.6
Z_w Root Location on Complex Plane

The characteristic equation yields the following time response, shown in Figure 3.7:

$$w = C_1 e^{\lambda t} \qquad \textit{eq 3.16}$$

Where:

C_1	- Constant affected by initial conditions of the motion
e	- Base of natural logarithm
λ	- Characteristic root
t	- Time
w	- Translational velocity component along z axis

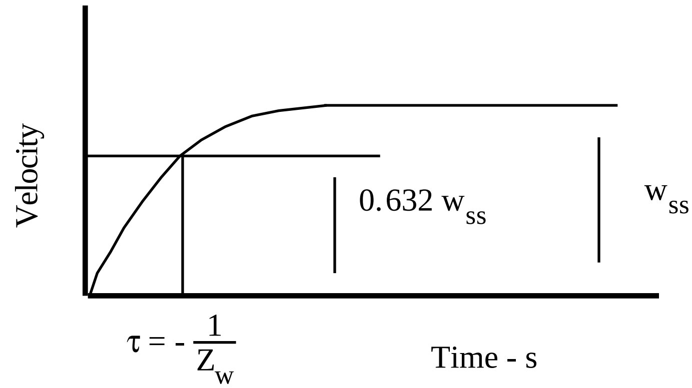


Figure 3.7
Typical Stable First Order Response

The time required to reach 63.2 % of a steady state first order response is called the time constant, τ:

$$\tau = -\frac{1}{\lambda} = -\frac{1}{Z_w}$$

eq 3.17

Where:

λ	- Characteristic root
τ	- Time constant
Z_w	- Vertical force due to vertical velocity.

The time constant is a function of heave damping; the greater the damping, the shorter the time constant. It takes about 5 τ 's for the response to approximate steady state. From a piloting viewpoint, τ relates to how quickly the response reaches steady state.

If the first order root, λ, is a positive number, it appears on the real axis in the right half of the complex plane. Systems with negative static stability yield an aperiodic divergent response as shown in Figure 3.4(g) when released from an initial off trim condition. The pertinent time history parameter is the time to double amplitude, T_d, expressed as: $T_d = 0.69/\lambda$.

3.4 TEST METHODS AND TECHNIQUES

3.4.1 General

Open loop dynamics data are obtained to clarify specific closed loop evaluation information, to document control system or aircraft response characteristics, or to check compliance with military specifications. The use determines the specific data to be acquired and the applicable flight conditions. The test pilot must be aware of the specific flight conditions and variable constraints to obtain useful open loop data.

3.4.2 Trim

The pilot must establish stabilized initial conditions to provide adequate reference for time histories. The initial conditions can be achieved either by trimming all control forces to zero (establishing a trim condition) or by holding the control fixed against the control force while maintaining equilibrium flight conditions. If the pilot intends to make a precise control input to create the disturbance from equilibrium conditions, the pilot trims all control forces to zero. If the pilot intends to use a control release from an off trim condition to create the disturbance, the pilot stabilizes at the equilibrium conditions by holding the off trim control forces.

3.4.3 Input Shape

Whether an input is a step, pulse, or doublet (Figure 3.8) depends on several factors including the nature of the response mode to be excited, whether it is to be excited selectively, the data reduction technique, and sometimes on specific data constraints. For the pulse and doublet, the controls must be returned to the initial trim position. For doublets, the test pilot makes a symmetrical input. The controls are returned to trim to eliminate any moment generated by the control displacement and to record the free response (without a forcing function) after the initial disturbance. Generally, step inputs are used for first order responses; while steps, pulses, and doublets are used for second order responses.

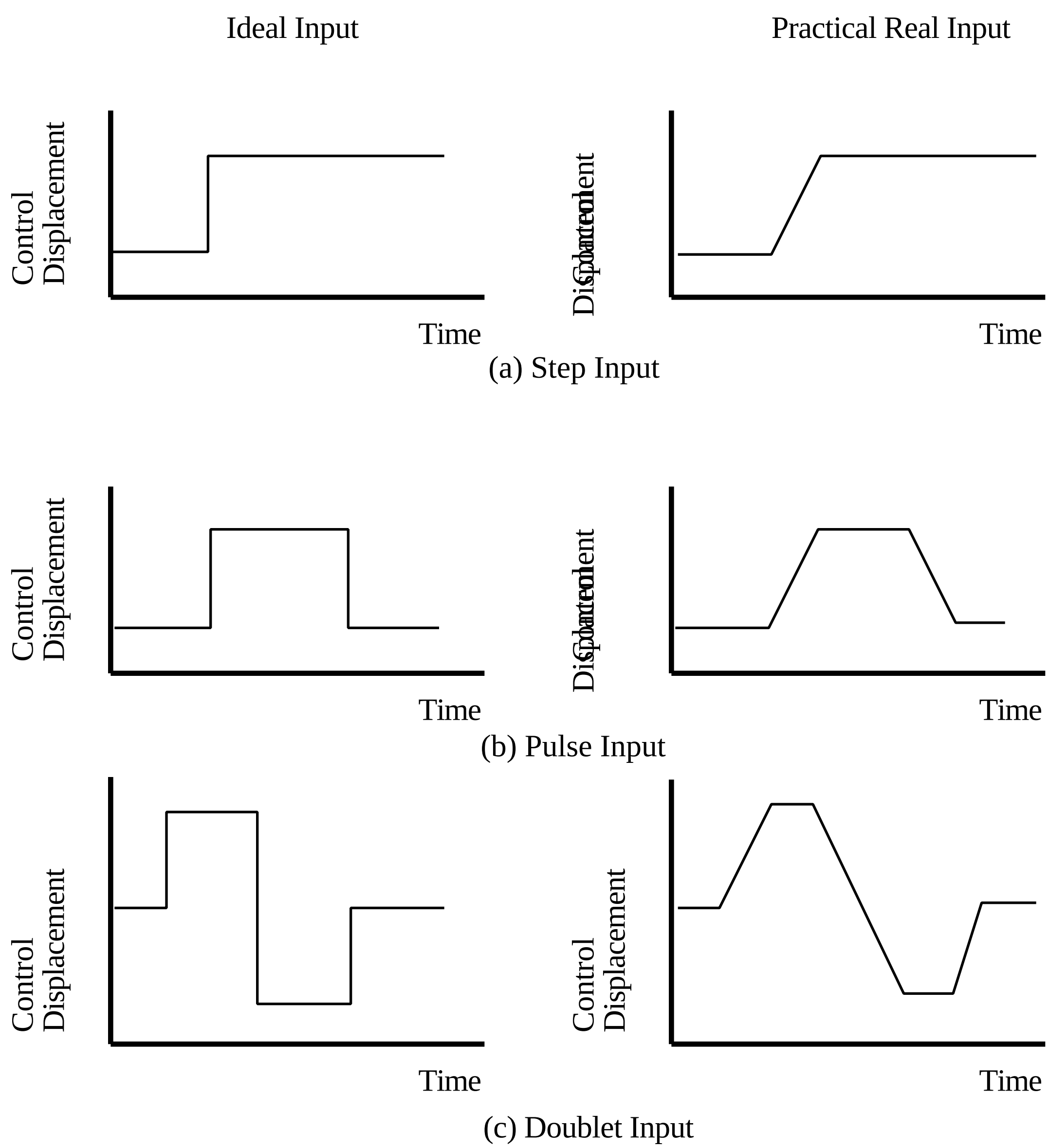


Figure 3.8
Examples of Test Inputs for Determination of Open Loop Dynamics

3.4.4 Input Amplitude

The appropriate input size is a function of the data desired. The input must be large enough to yield readable data with adequate signal to noise ratio. If the input is too large, nonlinearities of aerodynamics with airspeed might distort the data. Additionally, if the input size is large, the aircraft can reach a test limit without the pilot observing the complete response. On the other hand, it is desirable to correspond open loop data to realistic input size used to perform the closed loop tasks. The test team must explicitly consider input size. Because flight safety is always a constraint on input size, proper build up procedures must be followed.

3.4.5 Quickness of Input

The input frequency should maximize the disturbance at the response frequency. To approximate the natural frequency the pilot should make a series of sinusoidal, symmetrical control doublet inputs. Start at a low frequency and observe the response. Increase the frequency of the input and observe the response. Increase the frequency until the largest response is observed. Typically, bracketing the largest observed response with a low frequency and a high frequency will permit the pilot to find the correct input frequency. Once the input frequency is determined, use this frequency for the subsequent control inputs.

3.4.6 Excitation Methods

1. Natural excitation. No conscious excitation. Controls fixed without pilot input. Imperfect trim conditions or atmospheric disturbances may excite an aperiodic divergent mode or a lightly damped oscillatory mode. This is an unsatisfactory excitation technique if no response is obtained, but usually indicates a desirable aircraft characteristic.

2. Artificial excitation. Use the longitudinal control to accelerate or decelerate to an airspeed faster or slower than trim. Then smoothly return the control to the original trim position and record the aircraft motion. An off trim airspeed variation of 5-15 kn is normally used to excite the motion.

3. The method of excitation can determine the type of response documented. Natural disturbances resulting in a long term response are desirable but these responses are usually contaminated by another disturbance before the motion is completed. This contamination makes quantifying the mode of motion difficult.

4. Artificial methods are used to obtain time histories from engineering data. The excitation method chosen should result in an aircraft response similar to a response following a natural disturbance.

3.4.7 Test Technique

The longitudinal long term response in forward flight is used to illustrate the flight test techniques required for open loop dynamic evaluations. Subsequent chapters present the techniques applicable to a specific axis, mode of motion, and flight condition.

1. Stabilize at the desired trim airspeed and reduce all control forces to zero. Do not retrim control forces or move the collective during the test.

2. Record trim condition. Assure you can return and hold the controls at trim after exciting the response.

3. Determine if a long term response is excited by a natural disturbance. With the controls either fixed or free, note the open loop aircraft response. If no aircraft response is observed, an artificial excitation is used.

4. Excite the aircraft using a control input. Try several different methods in order to find the most representative aircraft response. Do not retrim any control forces and keep the collective constant during response.

5. If aircraft is flown hands off, obtain a control free response following excitation. Controls should be released at trim conditions so the subsequent control motions include the effect of attitude changes and gravity force acting on the controls during the response.

6. Record the resulting mode of motion using cockpit data and automatic recording system. Cockpit displayed boom airspeed is recorded at selected increments of time. Start the stop watch at the completion of the excitation or at a predetermined airspeed. The zero time reference point is arbitrary. Use time intervals small enough to define the shape of irregular responses.

7. The resulting mode of motion may not be a classical single axis response. Use lateral cyclic and directional pedals as necessary to maintain a single axis response for the initial excitation. If the lateral-directional inputs are significant, record a time history of the aircraft response with all controls fixed.

8. For hand held data, a plot of airspeed at regular intervals (5-10 s) adequately defines the response. An alternate method is to record peak (maximum and minimum) airspeeds and determine the period by timing the response as the helicopter passes through the trim airspeed.

3.4.8 Data Required

Time history of aircraft response to natural and/or artificial excitation with controls fixed and/or free. Parameters of interest are: V_o, boom and aircraft system, H_{P_o}, α, θ, q, control positions. If the excitation produces a coupled response which is not suppressed by the pilot, the attitudes and rates in the other axis are also of interest.

3.4.9 Test Criteria

1. Balanced (ball centered), wings level, unaccelerated trim condition.
2. All flight controls forces trimmed to zero.
3. Control returned exactly to trim following artificial excitation.
4. $H_{P_o} \pm 1000$ ft.

3.4.10 Data Requirements

1. Stabilize 5 s minimum prior to recording data.
2. Record sufficient data to include three complete cycles or until a test limit is reached.
3. $N_R \pm 1$ % at trim.

3.4.11 Safety Considerations/Risk Management

Aircraft and test attitude, rate, and acceleration limits must be established prior to test and strictly followed during the test. Artificial excitation and control displacement limits must be specified. Control displacement must follow proper build up techniques, progressing from small inputs to the test limit in small increments. Prior to the test, applicable recovery techniques must be reviewed and understood. The flight crew must practice the recovery techniques in a controlled build up manner until comfortable with the procedures.

3.5 DATA REDUCTION

3.5.1 General

The second order longitudinal long term oscillation responses is reported in the following terms:

1. The time required for the oscillation to subside to one-half amplitude, $T_{1/2}$, or the time to double amplitude, T_d.
2. Cycles required for the oscillation to subside to one-half amplitude, $C_{1/2}$, or the cycles to double amplitude, C_d.
3. Subsidence time, time required for the amplitude to decrease to a specific value after a specified disturbance. For example, the time required for the airspeed to return to within ± 1 kn of trim following a 10 kn disturbance.
4. Damping ratio, ζ.
5. Period of the oscillation.
6. Damped frequency, ω_d.

$T_{1/2}$, $C_{1/2}$, $T_{1/N}$, and $C_{1/N}$ can be determined using the graphical method shown in Figure 3.9 and/or the following relationships:

$$C_{1/2} = \frac{T_{1/2}}{P} \qquad \textit{eq 3.18}$$

$$T_{1/N} = \frac{\ln_e N}{\zeta\omega_n} \qquad \textit{eq 3.19}$$

$$C_{1/N} = \frac{\ln_e N}{2\pi \frac{\zeta}{\sqrt{1 - \zeta^2}}}$$

eq 3.20

Where:

$C_{1/2}$	- Cycles to one-half amplitude
$C_{1/N}$	- Cycles to 1/N amplitude
$\ln_e$	- Natural logarithm
N	- Denominator of the fractional amplitude
P	- Period
π	- Mathematical constant
$T_{1/2}$	- Time to one-half amplitude
$T_{1/N}$	- Time to decay to 1/N of maximum amplitude
ω_n	- Natural frequency
ζ	- Damping ratio.

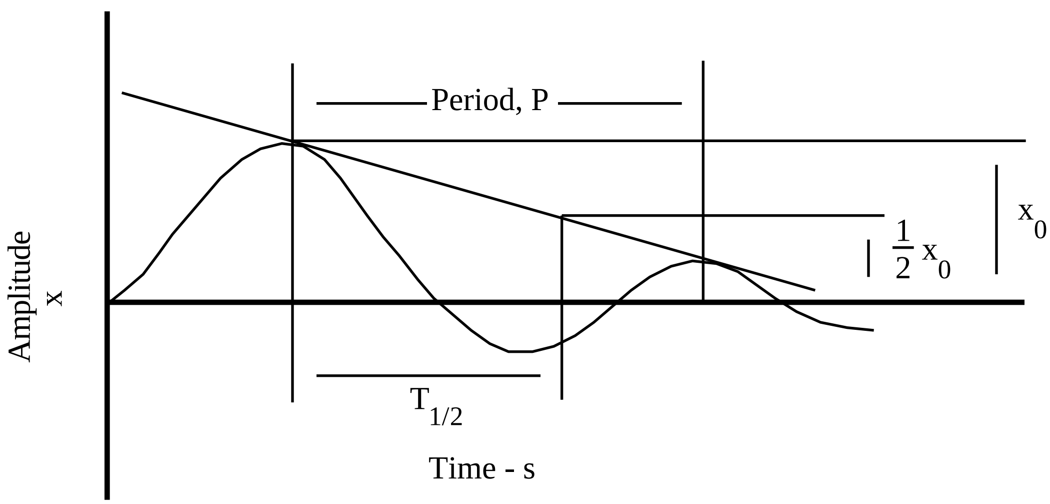
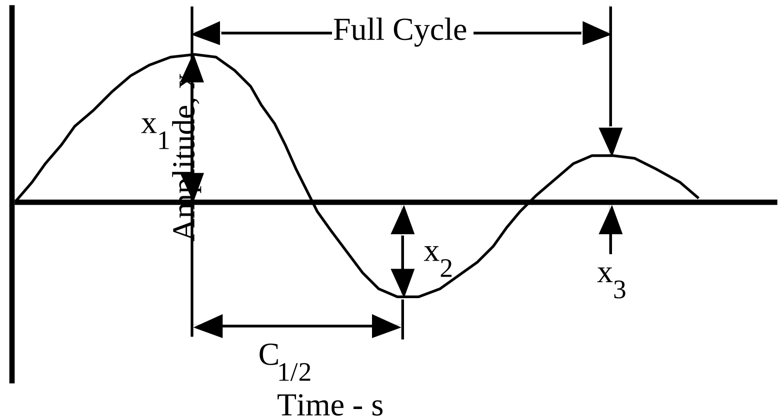


Figure 3.9
Graphic Method for Determining $T_{1/2}$ and $C_{1/2}$

Damping ratio can be determined using Figure 3.10. Entering this figure with the half cycle amplitude ratio provides a direct readout of ζ. For oscillatory divergent responses, change the horizontal scale to x_{n+1}/x_n and change the vertical scale to a negative sign.



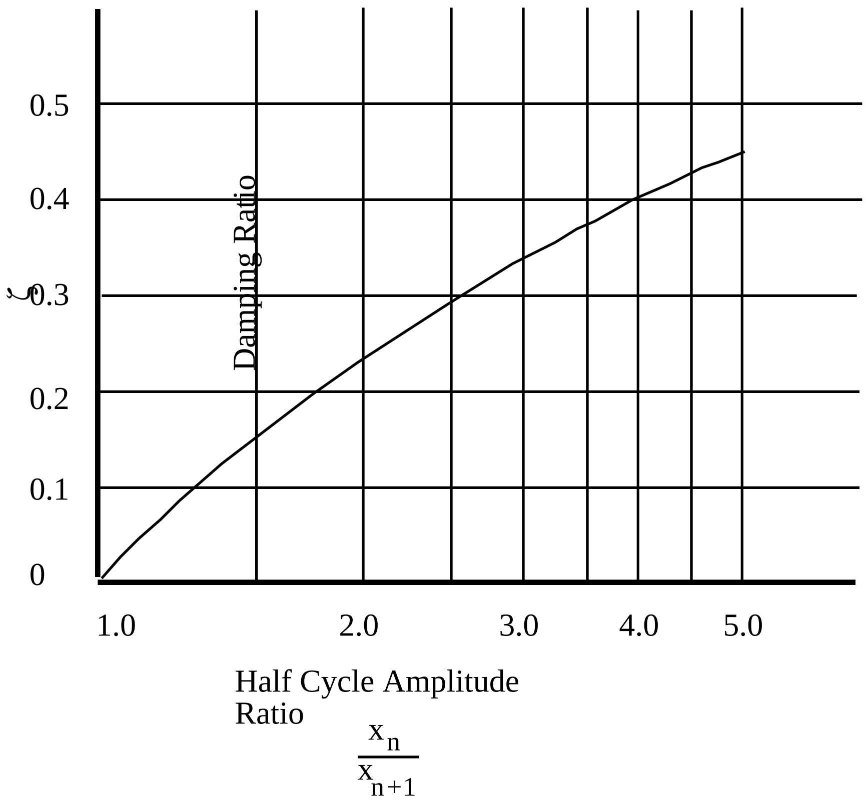


Figure 3.10
Graphic Solution for Damping Ratio

Plotting the roots of the characteristic equation on the complex plane as shown in Figure 3.5 allows for the graphic solution of natural frequency, ω_n; damped frequency, ω_d; and damping ratio, ζ, using Equations 3.11 and 3.13.

3.5.2 Data Presentation

1. Present an annotated time history of airspeed or altitude. Other parameters which can be plotted are pitch attitude or altitude deviations. Annotate the time history with the following engineering characteristics as appropriate: $C_{1/2}$, C_d, $T_{1/2}$, T_d, ω_d, period, and ζ.

2. Tables are used to summarize the engineering characteristics for several test conditions.

3.6 DATA ANALYSIS

In addition to the quantitative data reduction and presentation, the test pilot associates the open loop tests with the closed loop evaluation tasks discussed in Chapter 2. The following questions guide this association.

1. How easily excited is the long term response?
2. How does this motion influence the pilot's ability to perform mission tasks? The significance of the long term response is the manner in which it degrades or helps the pilot perform a specific task. Evaluate the pilot effort required to suppress or correct airspeed, attitude, and altitude variations about trim.
3. Can precise pitch attitude and airspeed changes be made?

3.7 MISSION SUITABILITY

Open loop testing supports the pilot qualitative evaluations made using closed loop test techniques. Mission suitability considerations are based on the closed loop flying qualities evaluations discussed in Chapter 2. The qualitative results of the open loop test support pilot opinion of mission suitability.

3.8 SPECIFICATION COMPLIANCE

Specification requirements are generally contained in military specification documents which put limits on open loop parameters as a function of flying qualities levels. Applicable documents are MIL-H-8501A for helicopters and the detailed or prime item specifications for each model helicopter. Subsequent chapters present the specification requirements relevant for that chapter.

3.9 GLOSSARY

3.9.1 Notations and Definitions

Attitude control effectiveness	Change in aircraft attitude achieved in one second following a unit step control displacement	deg/in
B	Damping constant	
B_{CRIT}	Critical damping	
$C_{1,2}$	Constants affected by initial conditions of the motion	
$C_{1/2}$	Cycles to one-half amplitude	
$C_{1/N}$	Cycles to 1/N amplitude	
C_d	Cycles to double amplitude	
Control power	Measure of the moment produced per unit of control displacement	lbf in/in
Control sensitivity	Initial angular acceleration produced by a unit step control displacement	deg/ s^2/in
deg	Degree	
e	Base of natural logarithm	
E_d	Damping angle	
H_{P_o}	Observed pressure altitude	
i	Imaginary index	
K	Spring constant	
kn	Knot	
ln_e	Natural logarithm	
m	Mass	
N	Denominator of the fractional amplitude	
N_R	Main rotor speed	
P	Period	
q	Angular velocity about y axis, Pitch rate	
Rate control effectiveness	Angular rate achieved one second following a unit step control displacement	deg/s/in
s	Second	
t	Time	
$T_{1/2}$	Time to one-half amplitude	
$T_{1/N}$	Time to decay to 1/N of maximum amplitude	
T_d	Time to double amplitude	

V_o	Observed airspeed
w	Translational velocity component along z axis
w_{ss}	Steady state translational velocity in the z direction
$\dot{w}$	Linear acceleration along the z axis
x	Displacement
x_0	Initial displacement
$\ddot{x}$	Acceleration
$\dot{x}$	Velocity
Z_w	Vertical force due to vertical velocity

3.9.2 Greek Symbols

α (alpha)	Angle of attack
λ (lambda)	Characteristic root
π (pi)	Mathematical constant
θ (theta)	Pitch attitude
τ (tau)	Time constant
ω (omega)	Frequency
ω_d	Damped frequency
ω_n	Natural frequency
ζ (zeta)	Damping ratio

3.10 REFERENCES

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CHAPTER FOUR

ROTOR CHARACTERISTICS

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EQUATIONS

Equation		PAGE
$\theta = A_0 - A_1 \cos\psi - B_1 \sin\psi - A_2 \cos 2\psi - B_2 \sin 2\psi \ldots - A_n \cos n\psi - B_n \sin n\psi$	*eq 4.1*	4.19
$\theta = \theta_C - A_1 \cos\psi - B_1 \sin\psi$	*eq 4.2*	4.20
$a_1 = a_{1s} + B_{1s}$	*eq 4.3*	4.20
$b_1 = b_{1s} - A_{1s}$	*eq 4.4*	4.20
$A_0 = A_{0s} = \theta_C$	*eq 4.5*	4.20
$\beta = a_0 - a_{1s} \cos\psi - b_{1s} \sin\psi - a_{2s} \cos 2\psi - b_{2s} \sin 2\psi \ldots - a_{ns} \cos n\psi - b_{ns} \sin n\psi$	*eq 4.6*	4.21
$\Delta\ CF = \Omega^2\ r\ m\ \Delta r$	*eq 4.7*	4.24
$\Delta M = \Delta CF\ r \sin\beta$	*eq 4.8*	4.25
$M = \int_0^R \Omega^2 r^2 m \beta\, dr = \Omega^2 \beta \int_0^R m r^2\, dr$	*eq 4.9*	4.25
$K = \frac{M}{\beta} = \Omega^2 \int_0^R m r^2\, dr$	*eq 4.10*	4.25
$I_f = \int_0^R m r^2\, dr$	*eq 4.11*	4.25

$$\omega_n = \sqrt{\frac{K}{I}} = \sqrt{\frac{\Omega^2 I_f}{I_f}} = \Omega$$ *eq 4.12* 4.25

$$\frac{\omega}{\omega_n} = \frac{\omega}{\Omega} = 1$$ *eq 4.13* 4.25

$$\Delta M = m\, \Delta r\, \Omega^2 (r' + e)\, r'\, \beta$$ *eq 4.14* 4.26

$$M = \Omega^2 \beta \int_0^{R-e} m\, (r' + e)\, r'\, dr'$$ *eq 4.15* 4.26

$$I_f = \int_0^{R-e} m\, r'^2\, dr'$$ *eq 4.16* 4.26

$$\frac{M_f}{g} = \int_0^{R-e} m\, r'\, dr'$$ *eq 4.17* 4.27

$$M = \Omega^2 \beta \left(I_f + e \frac{M_f}{g}\right)$$ *eq 4.18* 4.27

$$K = \frac{M}{\beta} = \Omega^2 \left(I_f + e \frac{M_f}{g}\right)$$ *eq 4.19* 4.27

$$I_f = m \int_0^{R-e} r'^2\, dr = \frac{mR^3}{3} \left(1 - \frac{e}{R}\right)^3$$ *eq 4.20* 4.27

$$\frac{M_f}{g} = m \int_0^{R-e} r'\, dr = \frac{mR^2}{2} \left(1 - \frac{e}{R}\right)^2$$ *eq 4.21* 4.27

$$\omega_n = \sqrt{\frac{K}{I_f}} = \Omega\sqrt{1 + \frac{eM_f}{g\,I_f}} = \Omega\sqrt{1 + \frac{\frac{3}{2}\frac{e}{R}}{1 - \frac{e}{R}}}$$ *eq 4.22* 4.27

$$\frac{\omega}{\omega_n} = \frac{\omega}{\Omega} = \sqrt{1 + \frac{\frac{3}{2}\frac{e}{R}}{1 - \frac{e}{R}}}$$ *eq 4.23* 4.27

$$I_f\ddot{\beta} + B\dot{\beta} + K\beta = 0$$ *eq 4.24* 4.28

$$\beta = -\frac{B}{2I_f} \pm \sqrt{\left(\frac{B}{2I_f}\right)^2 - \frac{K}{I_f}}$$ *eq 4.25* 4.29

$$B_{CRIT} = 2I_f\left(\frac{K}{I_f}\right)^{\frac{1}{2}}$$ *eq 4.26* 4.29

$$\zeta = \frac{B}{B_{CRIT}}$$ *eq 4.27* 4.29

$$M_A = \int_0^{R-e} r' \frac{\rho}{2}\, a\, c\, r'\, \dot{\beta}\, (r' + e)\, \Omega\, dr$$ *eq 4.28* 4.29

$$B = \frac{2M_A}{\partial\dot{\beta}} = \frac{c\rho a R^4}{8}\left(1 - \frac{e}{R}\right)^4 \Omega \left(\frac{1 + \frac{e}{3R}}{1 - \frac{e}{R}}\right)$$ *eq 4.29* 4.29

$$\zeta = \frac{\gamma}{16}\left(1 - \frac{e}{R}\right)^4 \left(\frac{1 + \frac{e}{3R}}{1 - \frac{e}{R}}\right) \frac{1}{\omega_n / \Omega}$$ *eq 4.30* 4.30

$$\gamma = \frac{c\rho a R^4}{I_f}$$

eq 4.31 4.30

$$\psi = \tan^{-1}\left[\frac{2\zeta\left(\frac{\omega}{\omega_n}\right)}{1 - \left(\frac{\omega}{\omega_n}\right)^2}\right]$$

eq 4.32 4.31

$$\mu = \frac{1}{\sqrt{\left[\left(1 - \left(\frac{\omega}{\omega_n}\right)^2\right)^2 + 4\,\zeta^2\left(\frac{\omega}{\omega_n}\right)^2\right]}}$$

eq 4.33 4.32

$$\frac{b_{1s}}{a_{1s}} = -\cot\psi$$

eq 4.34 4.34

$$\frac{b_{1s}}{a_{1s}} = -\frac{\frac{12}{\gamma}\,\frac{e}{R}}{\left[1 + \frac{e}{3R}\right]}$$

eq 4.35 4.34

$$a_{1s} = -\left(\frac{16}{\gamma}\right)\left(\frac{q}{\Omega}\right)$$

eq 4.36 4.36

$$b_{1s} = -\left(\frac{q}{\Omega}\right)$$

eq 4.37 4.36

$$a_{1s} = \frac{\dfrac{-\frac{16}{\gamma}\left(\frac{q}{\Omega}\right)}{\left(1 - \frac{e}{R}\right)^2} + \left(\frac{p}{\Omega}\right)}{1 - \frac{\mu^2}{2}} + \frac{\dfrac{\frac{12}{\gamma}\,\frac{e}{R}}{\left(1 - \frac{e}{R}\right)^3}\left[\dfrac{-\frac{16}{\gamma}\left(\frac{p}{\Omega}\right)}{\left(1 - \frac{e}{R}\right)^2} - \left(\frac{q}{\Omega}\right)\right]}{1 - \frac{\mu^4}{4}}$$

eq 4.38 4.36

$$b_{1s} = \frac{\dfrac{-\dfrac{16}{\gamma}\left(\dfrac{p}{\Omega}\right)}{\left(1 - \dfrac{e}{R}\right)^2} + \left(\dfrac{q}{\Omega}\right)}{1 - \dfrac{\mu^2}{2}} + \frac{\dfrac{\dfrac{12}{\gamma}\dfrac{e}{R}}{\left(1 - \dfrac{e}{R}\right)^3}\left[\dfrac{-\dfrac{16}{\gamma}\left(\dfrac{q}{\Omega}\right)}{\left(1 - \dfrac{e}{R}\right)^2} - \left(\dfrac{p}{\Omega}\right)\right]}{1 - \dfrac{\mu^4}{4}}$$ *eq 4.39* 4.36

$$\mu = \frac{V}{\Omega R}$$ *eq 4.40* 4.36

$$\beta = a_0 - a_{1s} \cos \psi - b_{1s} \sin \psi$$ *eq 4.41* 4.37

$$a_0 = \frac{\gamma}{8}\left[\theta_C\left(1 + \mu^2\right) + \frac{4}{3}\lambda\right] - \frac{M_W}{I\Omega^2}$$ *eq 4.42* 4.38

$$a_{1s} = -B_{1s}\,\frac{1 + \frac{3}{2}\mu^2}{1 - \dfrac{\mu^2}{2}} + \frac{\frac{8}{3}\theta_C\,\mu + 2\,\lambda\,\mu}{1 - \dfrac{\mu^2}{2}}$$ *eq 4.43* 4.38

$$b_{1s} = A_{1s} + \frac{\frac{4}{3}\mu\, a_0}{1 + \dfrac{\mu^2}{2}}$$ *eq 4.44* 4.38

$$\lambda = \frac{V \sin \alpha_s - v_i}{\Omega R}$$ *eq 4.45* 4.38

$$-F_{S_{\text{hub on blade}}} + \int_e^R dT \cos \beta = \int_e^R \ddot{\beta}\,(r - e)\, dm$$ *eq 4.46* 4.45

$F_{S_{\text{blade on hub}}} = T_b - \Omega^2 M_S \left(a_{1s} \cos \Omega t + b_{1s} \sin \Omega t\right)$	*eq 4.47*	4.45
$M_H(\psi) = -F_{S_{\text{blade on hub}}} (e \cos \psi)$	*eq 4.48*	4.46
$L_H(\psi) = -F_{S_{\text{blade on hub}}} (e \sin \psi)$	*eq 4.49*	4.46
$\overline{M_H} = \frac{b}{2\pi} \int_0^{2\pi} M_H(\psi)\, d\psi$	*eq 4.50*	4.46
$\overline{L_H} = \frac{b}{2\pi} \int_0^{2\pi} L_H(\psi)\, d\psi$	*eq 4.51*	4.46
$\overline{M_H} = \frac{1}{2} eb\, \Omega^2 M_S\, a_{1s}$	*eq 4.52*	4.46
$\overline{L_H} = \frac{1}{2} eb\, \Omega^2 M_S b_{1s}$	*eq 4.53*	4.46
$\sum M_{CG} = Th \sin\ a_{1s} + \overline{M_H} = Th + \frac{1}{2} eb\, \Omega^2 M_S\, a_{1s}$	*eq 4.54*	4.48
$H = T \sin a_{1s} \cos \varphi$	*eq 4.55*	4.50

CHAPTER FOUR

ROTOR CHARACTERISTICS

4.1 INTRODUCTION

Throughout the development of the helicopter, three fundamental problems faced the developers: reducing structural and engine weight sufficiently to provide a useful payload capability; counteracting rotor torque; and controlling the helicopter. The purpose of this chapter is to reacquaint the reader with the rotor characteristics which provide control of the helicopter. No attempt is made to develop rigorously the various equations which govern the stability and control of the rotor. Rather the terminology, reference systems, and equations are presented with a discussion of the factors which influence the results. Simplifying assumptions are made and discussed. References, containing complete development of the equations, are listed at the end of the chapter for those readers who wish further clarification.

The stability and control of the helicopter depends on the net effect of all the forces and moments applied to the helicopter from control inputs, helicopter motion, or external sources. The early development of rotary wing aircraft was slowed by the inherent stability and control problems of the single rigid rotor configuration. The basic difficulty was lift dissymmetry caused by a blade rotating with a constant pitch angle in forward flight, developing greater lift when advancing in the direction of motion than when retreating. Thus, the rigid rotor applies a rolling moment to the aircraft.

The problem of the lift dissymmetry in forward flight can be addressed in two ways. This problem was first solved by the introduction of blade flapping hinges which prevented the transmission of blade flapping moments to the aircraft. The unbalanced cyclic flapping moments resulted in a once per revolution blade flapping motion. The once per revolution blade flapping tended to make the aerodynamic lift and flapping moments on the blades approximately equal to the average values obtained with a rigid rotor.

The second method of overcoming lift dissymmetry in forward flight is through cyclic feathering of the blades; decreasing pitch on the advancing blades and increasing pitch on the retreating blade. In this case, the blades can be rigidly attached to the rotor shaft or incorporate flapping hinges. Hingeless blades flap relative to the rotor shaft and produce effects analogous to those obtained with flapping hinges.

The purpose of this chapter is understanding the flapping and feathering action of the rotor and how the pilot interacts to control the rotor. We begin the chapter with a review of various helicopter configurations.

4.1.1 Helicopter Configurations

There are five main types of helicopter configurations. Each type is presented along with its unique characteristics, advantages, and disadvantages.

The single rotor helicopter is the configuration most widely used. The major advantage is relative simplicity of one main rotor and one tail rotor. The single rotor helicopter can be powered by one or more engines combined through one transmission. The tail rotor typically uses about 8 to 10% of the total power available in a hover and 3 to 4% in forward flight. The simplicity and savings in weight make this method of countering rotor torque attractive. The few disadvantages of the single rotor configuration include the close proximity of the tail rotor to the ground, the relatively restrictive center of gravity (CG) range, and smaller payload capacity. These disadvantages have been overcome with advances in propulsion and materials, allowing design of larger multi-blade, multi-engine single rotor helicopters.

The tip driven rotor provides a simple solution to the problem of countering rotor torque. The rotor is powered at the tips rather than by the shaft, eliminating torque applied to the fuselage. The advantages are the elimination of the tail rotor and main transmission. The disadvantages are caused by the requirements of tip propulsion.

Coaxial rotors such as the Advancing Blade Concept (ABC) incorporate counterrotating main rotors to eliminate rotor torque to the fuselage. The advantage is the elimination of the tail rotor. The disadvantage is the weight and complexity of the rotor and hub. The interaction of the rotor flow causes vibration problems as well.

Side by side rotors offer the advantage of elimination of the tail rotor and a reduction in power required in forward flight since each rotor operates in relatively undisturbed air. The disadvantage is high fuselage parasite drag, greater weight, and complexity of rotor control systems.

Tandem rotors offer the advantage of large payload capability, larger CG range, and relatively clean fuselage design. The disadvantage is some increase in power required in forward flight due to the influence of the front rotor on the rear rotor.

Multi-rotors offer the advantage of increased lift capability and simplicity of control through varying thrust of each rotor. The disadvantages are large size and high weight.

A special type of helicopter configuration is the compound aircraft such as the V-22. This design offers the advantages of high speed flight while retaining vertical/short takeoff and landing capability. The disadvantages lie in overall complexity of design.

For the purpose of understanding rotor characteristics in this chapter, the discussion concentrates on the single rotor configuration with some discussion as applicable to the tandem rotor configuration.

4.1.2 Control Methods

To control the helicopter in flight, the forces and moments about all three axes of the helicopter must be controlled. This involves controlling three forces and three moments. Figure 4.1 presents a body fixed reference axis system and shows the usual convention for positive values. This orthogonal axis system is depicted initially aligned with the relative wind.

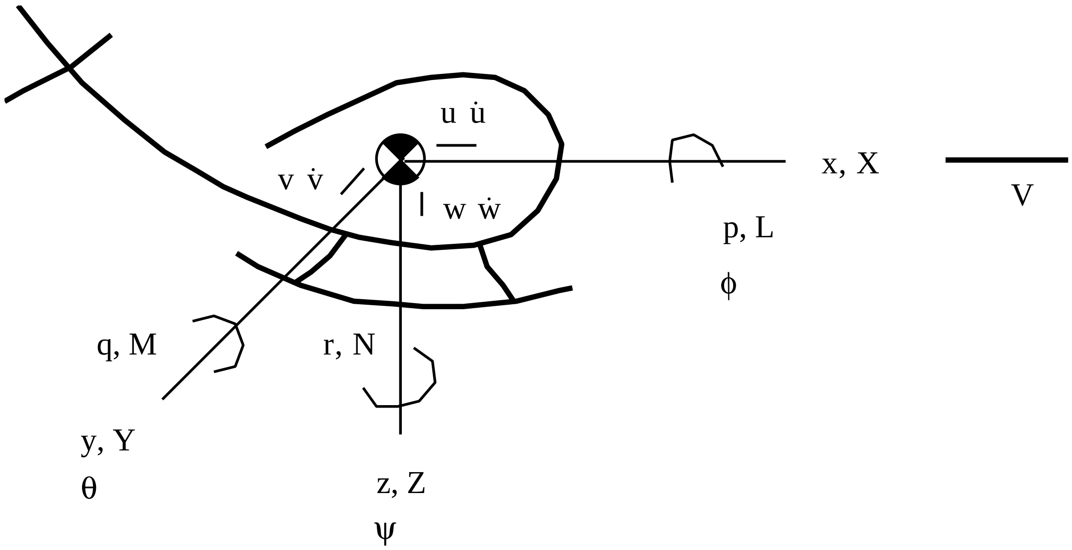


Figure 4.1
Body Fixed Orthogonal Stability Axis System

ϕ	- Roll angle
L	- Net moment about x axis, Roll moment
M	- Net moment about y axis, Pitch moment
N	- Net moment about z axis, Yaw moment
p	- Angular velocity about x axis, Roll rate
$\dot{p}$	- Angular acceleration about x axis
q	- Angular velocity about y axis, Pitch rate
θ	- Pitch angle
$\dot{q}$	- Angular acceleration about y axis
r	- Angular velocity about z axis, Yaw rate
$\dot{r}$	- Angular acceleration about z axis
u	- Translational velocity component along x axis
$\dot{u}$	- Linear acceleration along x axis
V	- Relative wind
v	- Translational velocity component along y axis
$\dot{v}$	- Linear acceleration along y axis
w	- Translational velocity component along z axis
$\dot{w}$	- Linear acceleration along z axis
X	- Resultant force in x direction

x	- Orthogonal direction along longitudinal axis of the aircraft
Y	- Resultant force in y direction
y	- Orthogonal direction along lateral axis of the aircraft
ψ	- Yaw angle
Z	- Resultant force in z direction
z	- Orthogonal direction along vertical axis of the aircraft.

Controlling the forces and moments in six independent degrees of freedom require six independent controls. The six required controls would be extremely cumbersome. Coupling the controls reduces the number of independent controls and provides a manageable control system. The sacrifice is complete freedom of controlling position and attitude. The degrees of freedom necessary to control the helicopter are reduced to four and controls are provided for each.

Vertical control is required to fix the helicopter vertically. Vertical control is provided through the collective control to increase/decrease the pitch of the rotor and increase/decrease thrust.

Directional control is used to fix the attitude of the helicopter about the z axis. Directional control is used to point the helicopter or if required by configuration, counteract main rotor torque. The directional control mechanism varies with configuration.

Lateral control is provided through control of both forces and moments. The lateral control produces a rolling moment about the CG which tilts the helicopter. However, the tilt of the thrust vector also produces a lateral or side force, depending on the configuration.

Longitudinal control consists of controlling both longitudinal forces and moments. In a tandem configuration, the longitudinal control is accomplished by controlling the thrust of the rotors independently or by tilting the rotors. Rotor tilt is reserved usually for trimming.

Control coupling or the application of control in one axis producing a force or moment in another axis is undesirable, unless the control coupling is designed to counteract aerodynamic coupling. The sources of aerodynamic coupling are introduced in subsequent sections and the requirement for control coupling or pilot action is discussed. Chapter 8 concentrates on coupling and the impact on stability and control.

4.1.3 Control Mechanics

Longitudinal and lateral control of the main rotor is exercised through tilting the thrust vector or by producing moments about the rotor hub. The pilot controls the rotor through cyclic pitch change. The cyclic pitch change is accomplished by the linkage from the cockpit flight control to the swashplate. The outer rotating race of the swashplate is connected through pitch change linkage to the pitch change horn. The pitch change horn controls the blade pitch or feathering through the feathering bearing. For rotors without flapping hinge offset or effective flapping offset, the tilt of the swashplate produces blade flapping as well as tip path plane (TPP) and thrust vector tilt. For rotors with flapping hinge offset or effective flapping offset, blade flapping produces a moment about the rotor hub as well as a tilt of the TPP and thrust vector.

4.1.4 Mechanical Controls

The two primary main rotor controls in a conventional helicopter are the collective and cyclic pitch controls. The pilot uses the collective pitch control to increase or decrease the average pitch angle of all the blades, controlling the helicopter vertically. The cyclic pitch control is used to change the pitch of the individual blades in relation to the feathering axis sinusoidally, once each revolution. One per revolution cyclic pitch change causes blade flapping, resulting in a tilt of the TPP and the thrust vector. The resulting tilt of the thrust vector produces pitching and rolling moments as well as longitudinal and lateral forces. A typical mechanism for collective and cyclic pitch control is illustrated in Figure 4.2. Cyclic pitch mechanisms customarily incorporate a swashplate and pitch change mechanisms as shown in Figure 4.3. Cyclic pitch change mechanisms usually are designed for a one to one relationship between the swashplate angle and the amplitude of cyclic feathering introduced to the rotor.

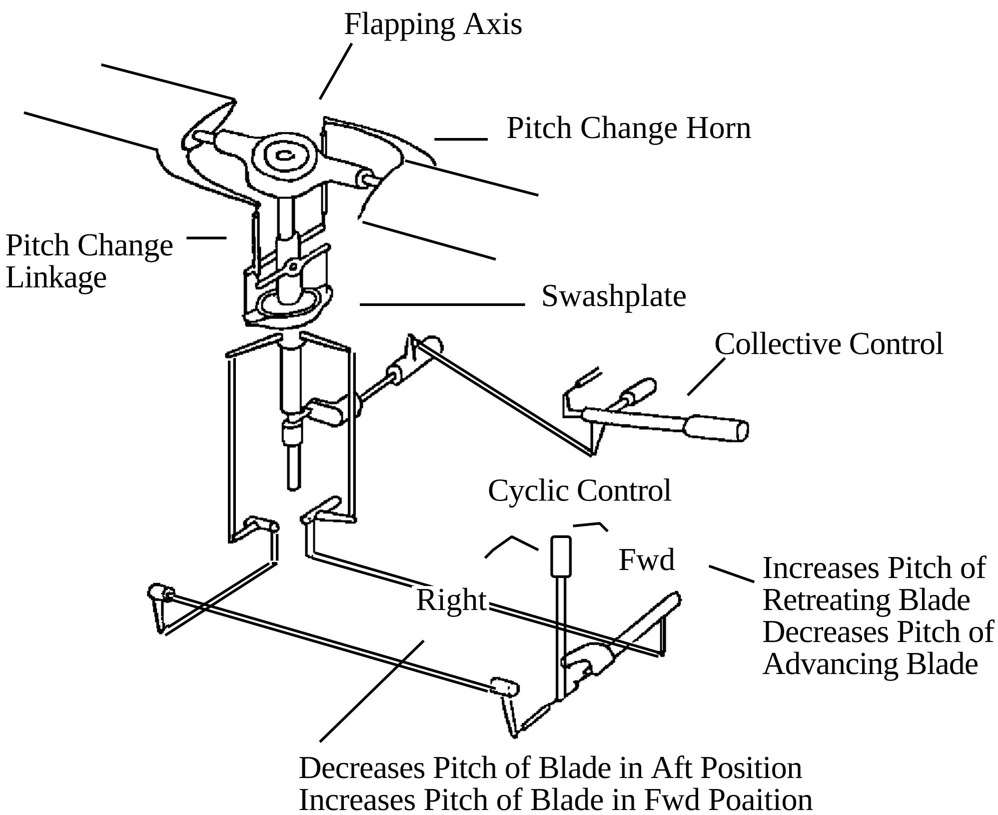


Figure 4.2
Typical Mechanism for Collective and Cyclic Pitch Control

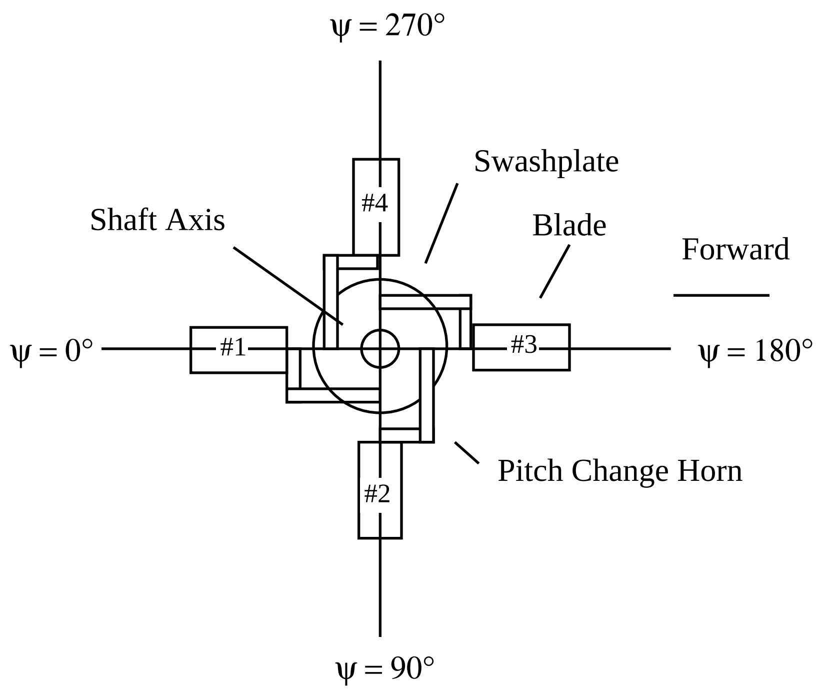


Figure 4.3
Four Bladed Rotor

4.1.5 Degrees of Freedom

Rotor systems were developed to accommodate rotation about the four degrees of freedom. Each of the of rotor systems discussed either provides for a hinge or bearing to permit rotation; or attempts to restrict the degree of freedom. Flexible, composite structures are used to provide effective hinges. The four degrees of freedom are:

1. Rotation about the shaft axis, denoted by the angle, ψ, with ψ equal to zero over the tail.
2. Rotation about an axis along the span of the blade (pitch or feathering axis) denoted by the feathering angle, θ. The feathering angle is also termed the blade pitch angle.
3. Rotation about an axis normal to the plane containing the shaft and feathering axes (flapping axis) denoted by the flapping angle, β.

4. Rotation about an axis parallel to the shaft (lead-lag axis) denoted by the lead-lag angle, ζ. Lead-lag motion is important for the study of vibration, rotor loads, and ground resonance but has little effect on stability and control. For this reason, lead-lag motion is not discussed in this chapter.

4.1.6 Rotor Types

Conventional helicopter rotor systems have blades hinged at or near the center of rotation to permit the blades freedom to flap up and down as they rotate. The two systems of articulation widely used are the fully articulated rotor and the semirigid or teetering rotor.

The fully articulated system incorporates three or more blades, each blade individually hinged near the hub as shown in Figure 4.4. The offset of the flapping hinge axis from the axis of rotation (shaft axis) effects the moments the rotor applies to the aircraft. The flapping frequency of the rotor blades, as well as the speed and phasing of their response to control inputs, depend on the flapping hinge offset. In addition to flapping motion, the articulated system incorporates lead-lag hinges which permit in-plane motion of the blade. The lag degree of freedom is ordinarily of secondary importance from the stability and control standpoint. Control of the rotor is accomplished by rotating the blade about the blade feathering axis.

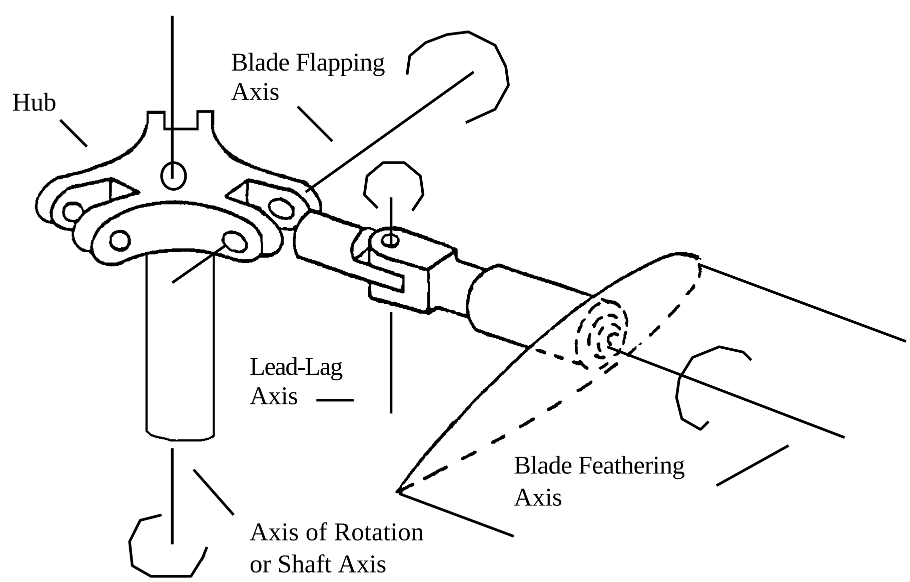


Figure 4.4
Fully Articulated Rotor

The two bladed semirigid or teetering rotor has the flapping hinge at the shaft centerline. The semirigid hub is free to teeter causing equal and opposite flapping motions on the two sides (Figure 4.5). Steady lift forces acting on articulated rotor blades tend to rotate the blades upward about their flapping hinges, while centrifugal forces tend to hold them flat. Under normal operating conditions, a balance is achieved with a small upward coning angle, a_0, on the order of 6 to 8°. Two bladed teetering rotors often incorporate a built in coning angle of this magnitude to relieve bending moments produced by these aerodynamic forces. The control moments supplied by the two bladed rotor are dependent on tilting the thrust vector since there is no flapping hinge offset from the shaft axis of rotation. The control moments are small when the thrust is reduced. Low g conditions reduce the thrust and may lead to ineffective cyclic control.

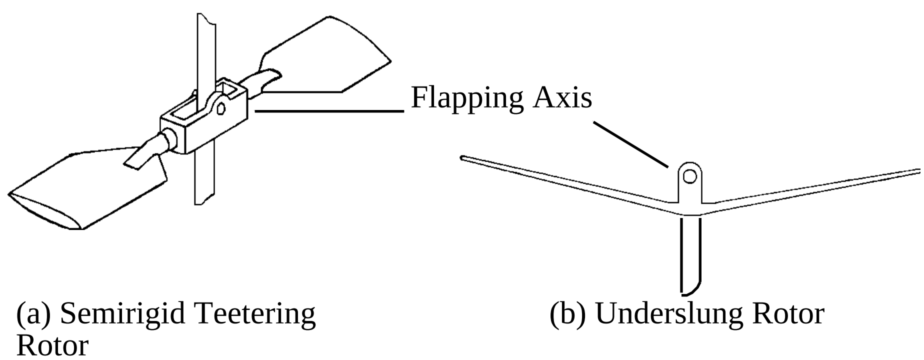


Figure 4.5
Semirigid Teetering Rotor

Many modern helicopter designs use hingeless rotors (Figure 4.6). No flapping or lead-lag hinges are incorporated. In these systems, lead-lag and flapping motions occur through hub and rotor blade flex. Bearings or special structural members are used which permit rotation about the feathering axis.

For stability and control purposes, hingeless rotors are modeled analytically as equivalent articulated rotors with large flapping hinge offset and/or auxiliary springs to restrain flapping motion. Both of these modifications to an articulated rotor model, increase blade flapping frequencies and the moments transmitted by the rotor blades to the rotor shaft; thus approximating hingeless rotor characteristics.

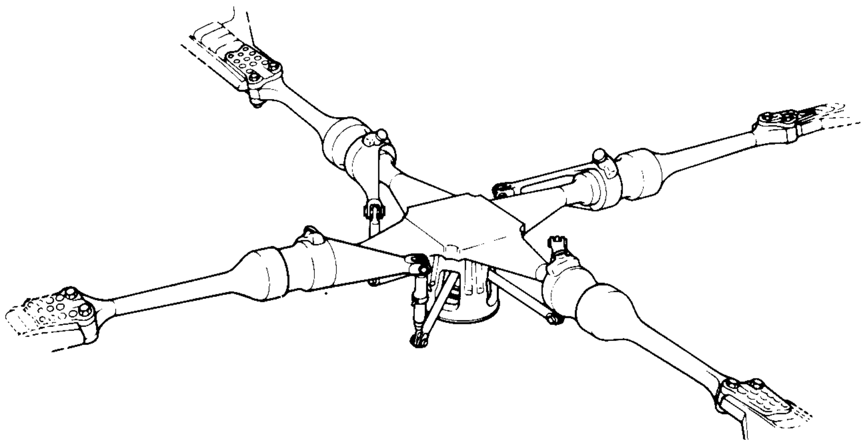


Figure 4.6
Hingless Rotor

Changing blade pitch about the feathering axis is the usual method of rotor control. Blade feathering hinges are usually located near the blade one-quarter chord to reduce blade pitching moments about the feathering axis. Another method of cyclic pitch change is structural twisting of the blade using a small, movable flap near the blade tip to control the twisting motion (Kaman rotor).

Various combinations of rotors and fixed airfoil surfaces are used in helicopters to improve performance (compound helicopter, X wing, retractable rotor, coaxial rotor, and tilt rotor). Auxiliary forward thrust devices are necessary with many of these configurations and each one has its own stability and control characteristics and problems. In general, the role of the rotors and lifting surfaces changes with the flight regime. Typical helicopter rotor control techniques are applicable in low airspeed flight. However, auxiliary airplane type controls are required in high speed flight because the rotors are unloaded (ABC) or transitioned (V-22) to delay compressibility effects and retreating blade stall.

4.2 REFERENCE SYSTEMS

4.2.1 Velocity Distribution on a Rotor

If the blades are not allowed to flap or feather, the velocity distribution on the rotor in flight causes uneven lift distribution across the rotor. The importance of the velocity distribution on the rotor in flight is the rolling moments imparted to the helicopter by the uneven lift distribution. The uneven lift distribution also causes high alternating blade loads. Flapping and feathering hinges relieve the uneven lift distribution. Figure 4.7 presents the velocity distribution on a rotor in flight.

The transverse velocity, u_T, on the advancing blade is the sum of the rotational velocity, Ωr , and the relative velocity, V. The transverse velocity on the retreating blade is the rotational velocity minus the relative velocity. The transverse velocity on the forward and aft blades is the rotational velocity. In general the transverse velocity is a function of blade azimuth position, $u_T = \Omega r + V \sin \psi$.

4.2.2 Blade Azimuth Angle

Blade azimuth angle is measured in a plane normal to the shaft of the rotor and denoted by the Greek letter ψ. Blade azimuth angle is zero when the blade is over the tail and is measured positive in the direction of rotation. For a counterrotating rotor, the advancing blade at the 90˚ position is over the right side; the forward blade is at the 180˚ position; and the retreating blade at the 270˚ position is over the left side (Figure 4.7).

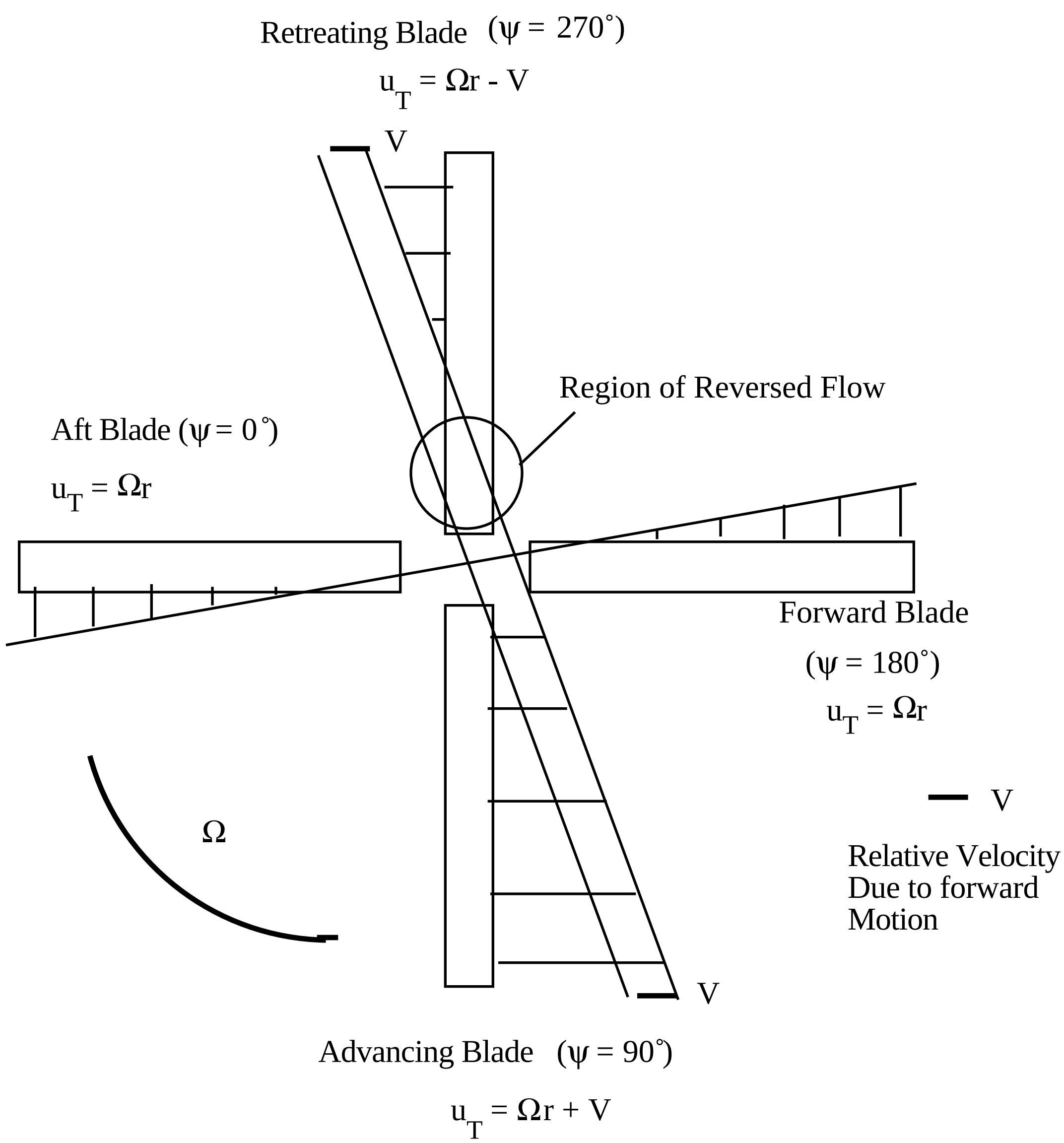


Figure 4.7
Velocity Distribution on a Rotor

4.2.3 Coning Angle

The coning angle, a_0, is the average blade flapping angle during one revolution of the blades. The coning angle can be thought of as the angle between the blades and the TPP (Figure 4.8). The coning angle remains the same whether the TPP is perpendicular to the shaft or not.

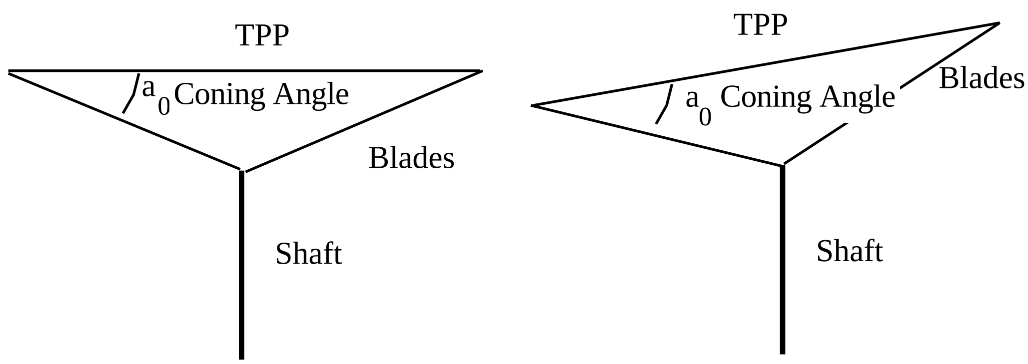


Figure 4.8
Coning Angle

4.2.4 Flapping Angle

The blade flapping angle, β, is the angle between a line drawn along the span of the blade and a plane normal to the shaft axis. Blade flapping angle is positive when the blade tip is higher than the blade cuff (Figure 4.9).

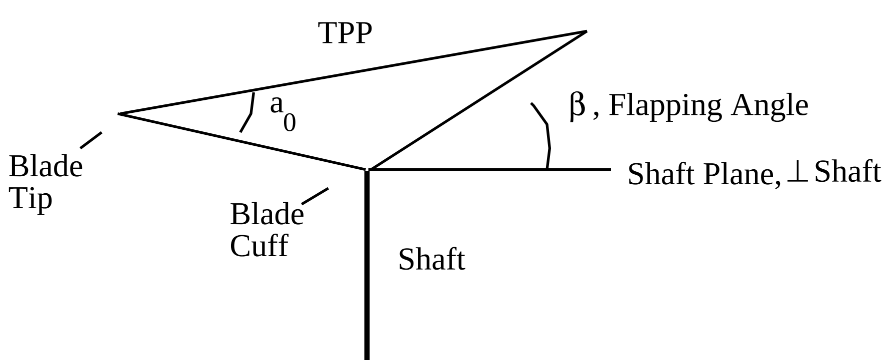


Figure 4.9
Flapping Angle

4.2.5 Blade Pitch Angle

The blade pitch angle, θ, is the angle between the blade zero lift line and a plane normal to the shaft axis. Blade pitch is positive when the leading edge is up (Figure 4.10).

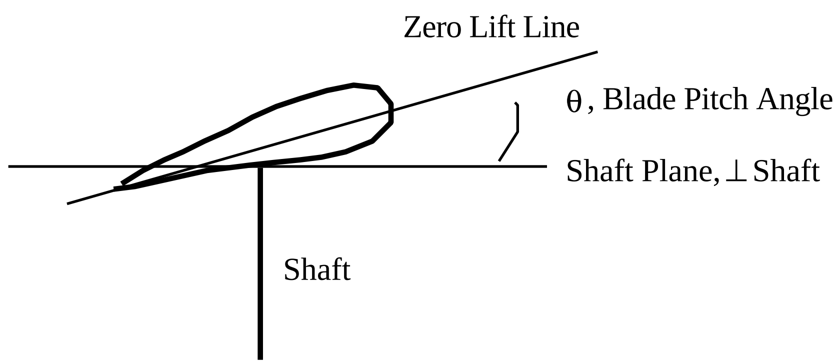


Figure 4.10
Blade Pitch Angle

4.2.6 Reference Axes System

Several axes systems are used in the discussion of rotor characteristics. The TPP axis is the axis normal to the TPP. The TPP axis is the axis of no flapping, the blades change pitch or feather with respect to the TPP axis but do not flap. The shaft axis is the axis along the shaft of the rotor system. Angles referenced to the shaft axis are subscripted by $_s$. The control axis is the axis of no feathering. The control axis is the physical axis of a flapping rotor, the blades are fixed in pitch or feather but are free to flap. The reference axis system does not effect the analysis of blade motion in fight. The axis system in use is denoted by subscripting if appropriate. The axes system is presented in Figure 4.11.

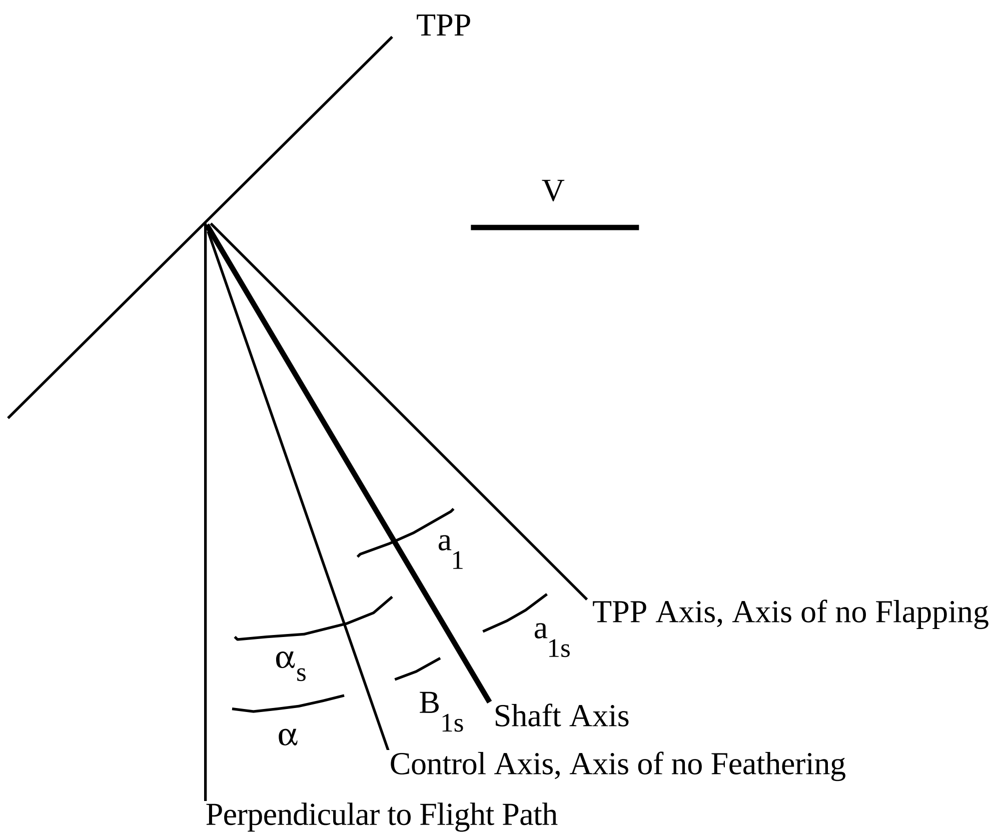


Figure 4.11
Reference Axes System

4.2.7 Rotor Angles

Just as with the reference axes system, the rotor angles are defined to aid the discussion of rotor characteristics. Figure 4.12 presents the rotor longitudinal angular relationships. Figure 4.13 presents the rotor lateral angular relationships. The angles referenced to the shaft plane (plane ⊥ to the shaft axis) are subscripted with an $_s$.

The longitudinal tilt of the TPP in relation to the shaft is the longitudinal flapping angle, a_{1s}. The longitudinal flapping angle is generated by longitudinal cyclic or external aerodynamic inputs. The longitudinal flapping angle is positive when the blade flaps up at $\psi = 180°$.

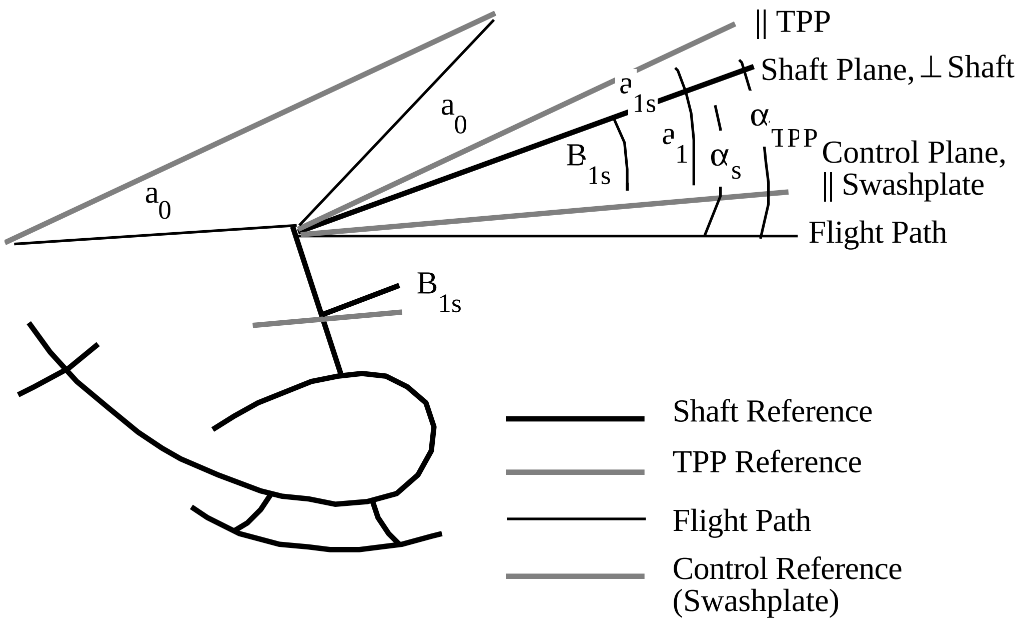


Figure 4.12
Rotor Longitudinal Angular Relationships

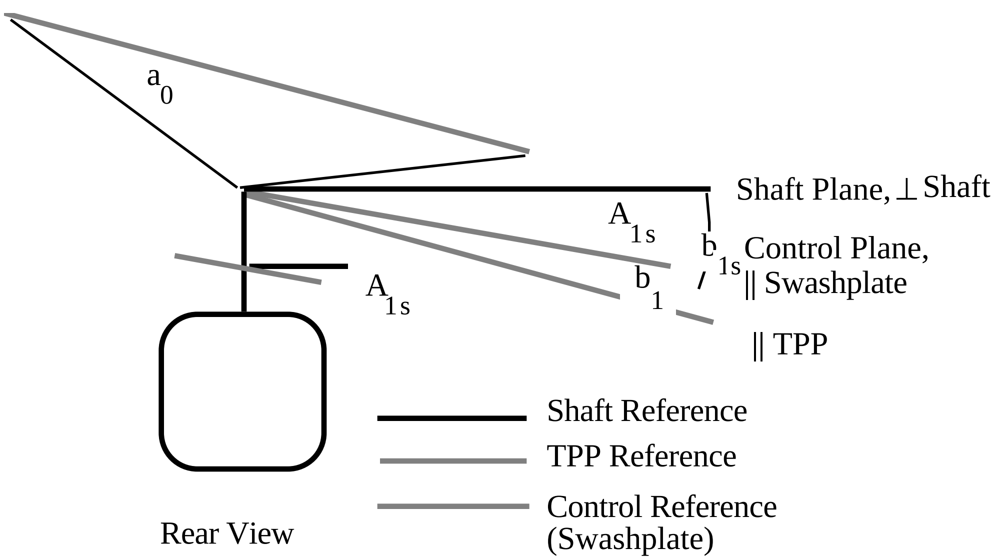


Figure 4.13
Rotor Lateral Angular Relationships

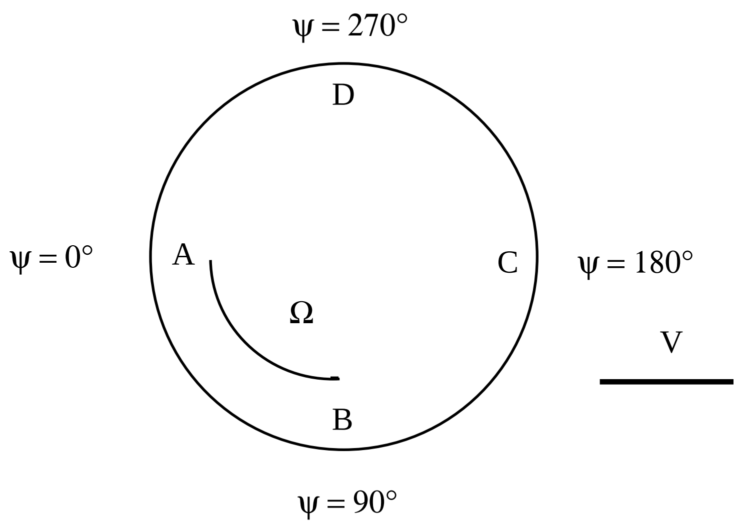
The longitudinal cyclic pitch angle, B_{1s}, is the angle between the shaft plane and the control plane or swashplate.

The lateral tilt of the TPP in relation to the shaft is the lateral flapping angle, b_{1s}. The lateral flapping angle is generated by lateral cyclic or external aerodynamic inputs. The lateral flapping angle is positive when the blade flaps down at $\psi = 90°$.

The lateral cyclic pitch angle, A_{1s}, is the angle between the shaft plane and the control plane or swashplate.

4.2.8 Relation of Flapping and Feathering

An observer riding on the control axis (axis of no feathering) and rotating with the blades observes the blade flap up and down each revolution, but they are fixed in pitch. Likewise, an observer riding on the TPP axis (axis of no flapping) and rotating with the blades observes the blades change pitch (feather) each revolution, but do not flap at all. The relation of flapping and feathering is shown in Figure 4.14.



Position	Blade Position With Respect to Control Axis	Blade Position With Respect to TPP
A	Flapping Full Down	Feathering Zero
B	Flapping Zero	Feathering Maximum Up
C	Flapping Full Up	Feathering Zero
D	Flapping Zero	Feathering Maximum Down
	(Assume 90˚ phase angle)	

Figure 4.14
Blade Position with Respect to Control and TPP Axes

4.3 ROTOR DYNAMICS

4.3.1 Blade Feathering Motion

Blade feathering motion in flight can be expressed as the variation of blade pitch angle (feathering angle) with respect to the TPP. The expression for blade pitch angle, θ, can be expressed in terms of the sinusoidal variation of θ with respect to azimuth angle, ψ. The reference axis is the TPP axis (axis of no flapping). The feathering motion is expressed as a Fourier series sum of simple harmonic motion:

$$\theta = A_0 - A_1 \cos\psi - B_1 \sin\psi - A_2 \cos 2\psi - B_2 \sin 2\psi \ldots - A_n \cos n\psi - B_n \sin n\psi \qquad \textit{eq 4.1}$$

Neglecting the higher harmonic motions, and replacing A_0, the constant pitch angle which does not vary with ψ, with collective pitch angle, θ_C, the expression is reduced to:

$$\theta = \theta_C - A_1 \cos\psi - B_1 \sin\psi \qquad \textit{eq 4.2}$$

This expression for the blade pitch angle indicates that the total blade pitch is a function of the fixed pitch imparted by the collective, θ_C, and the cyclic pitch variation caused by longitudinal and lateral cyclic pitch. The relationships between cyclic pitch angles and flapping angles are:

$$a_1 = a_{1s} + B_{1s} \qquad \textit{eq 4.3}$$

$$b_1 = b_{1s} - A_{1s} \qquad \textit{eq 4.4}$$

$$A_0 = A_{0s} = \theta_C \qquad \textit{eq 4.5}$$

Where:

A_0	- Collective pitch angle (not a function of ψ)
A_{0s}	- Collective pitch angle, shaft referenced (not a function of ψ)
A_1	- Lateral cyclic pitch angle
a_1	- Longitudinal flapping angle
A_{1s}	- Lateral cyclic pitch angle, shaft referenced
a_{1s}	- Longitudinal flapping angle, shaft referenced
B_1	- Longitudinal cyclic pitch angle
b_1	- Lateral flapping angle
B_{1s}	- Longitudinal cyclic pitch angle, shaft referenced
b_{1s}	- Lateral flapping angle, shaft referenced
θ	- Feathering angle, blade pitch angle
θ_C	- Collective pitch angle
ψ	- Blade azimuth angle.

The concept of phase angle is presented later. For now, assuming a 90˚ phase angle, the above expressions define the control of the blade flapping and hence the tilt of the TPP through the control of the cyclic pitch angle (feathering angle). The relationship between the feathering angle and flapping angle is presented again after presenting the flapping equation.

4.3.2 Blade Flapping Motion

To understand blade flapping motion in forward flight, the variation of flapping angle, β, with blade azimuth angle, ψ, is expressed in terms of a simple sinusoidal motion about the longitudinal and lateral axes. The reference axis is selected from those presented above. For this presentation the shaft axis is used. The angular relation to the control axis is presented in Figure 4.11. The flapping motion is expressed as a Fourier series sum of simple harmonic motion:

$$\beta = a_0 - a_{1s} \cos\psi - b_{1s} \sin\psi - a_{2s} \cos2\psi - b_{2s} \sin2\psi \ldots - a_{ns} \cos n\psi - b_{ns} \sin n\psi \qquad \textit{eq 4.6}$$

Where:

a_0 - Coning Angle
a_{1s} - Longitudinal flapping angle, shaft referenced
β - Flapping angle
b_{1s} - Lateral flapping angle, shaft referenced
ψ - Blade azimuth angle.

The coefficient a_0 (coning angle) represents the part of the flapping angle independent of blade azimuth. In a hover, without tilt of the TPP relative to the shaft, the flapping angle, β, is equal to the coning angle, a_0.

The coefficient a_{1s} represents the amplitude of a pure cosine motion. The motion represented by $\beta = -a_{1s} \cos \psi$, plotted against ψ, is presented in Figure 4.15. Figure 4.15 shows that β is maximum at $\psi = 180°$ and a minimum at $\psi = 0°$. Therefore, a_{1s} is the longitudinal flapping angle. The longitudinal flapping angle is positive in the downward direction at $\psi = 0°$ (aft tilt of the TPP). The rotor longitudinal angular relationships are presented in Figure 4.12.

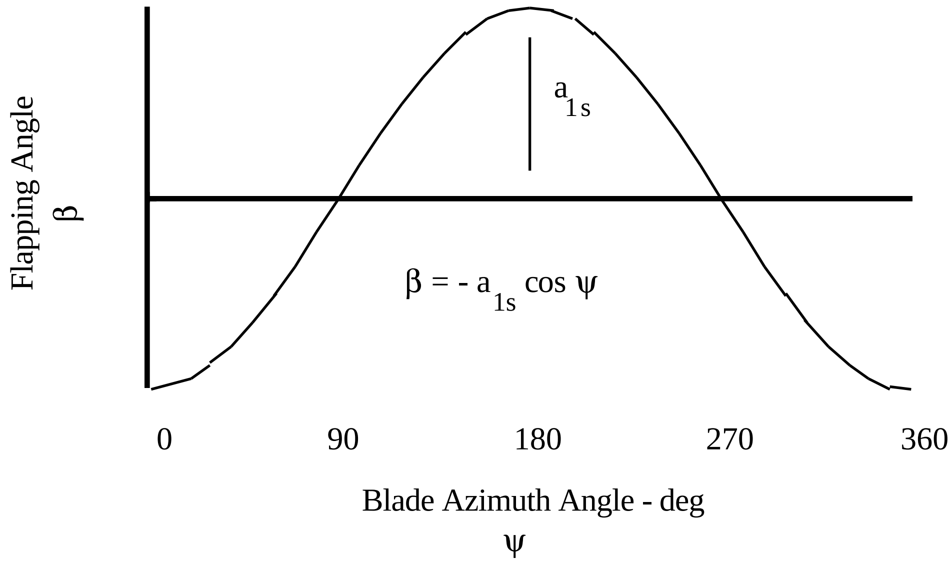


Figure 4.15
First Harmonic Cosine Flapping Motion, $\beta = a_{1s} \cos \psi$

The coefficient b_{1s} represents the amplitude of a pure sine motion. The motion represented by $\beta = -b_{1s} \sin \psi$, plotted against ψ, is presented in Figure 4.16. Figure 4.16 shows that β is maximum at $\psi = 270°$ and a minimum at $\psi = 90°$. Therefore b_{1s} is the lateral flapping angle. The lateral flapping angle is positive in the downward direction at $\psi = 90°$ (right tilt or the TPP). The rotor lateral angular relationships are presented in Figure 4.13.

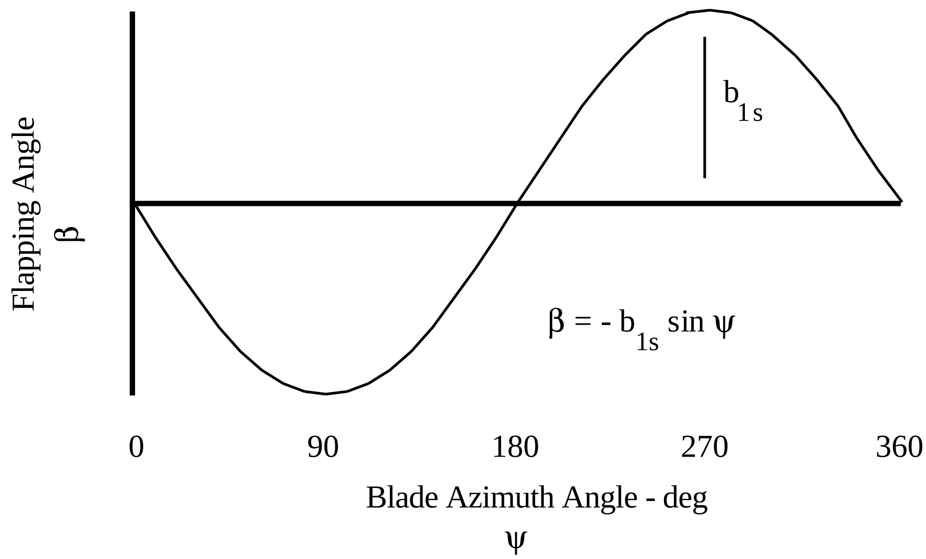


Figure 4.16
First Harmonic Sine Flapping Motion, $\beta = b_{1s} \sin \psi$

In equation 4.6, the coefficients a_{2s}, b_{2s}, … , a_{ns}, b_{ns} represent the amplitudes of the higher harmonic motion. Although work is being done on higher harmonic control systems, the higher harmonic motions are relatively small and have little effect on rotor torque and thrust. They are neglected for this discussion.

4.3.3 Frequency Ratio

The phase angle relationship between the feathering angle (cyclic pitch angle) and the flapping response is determined by the frequency ratio and the damping ratio. To determine the frequency ratio, blade flapping is represented by a spring-mass-damper system as discussed in Chapter 3. The blade has a mass and a restoring force provided by the centrifugal forces. The flapping motion is excited by the aerodynamic forces supplied by operation in the atmosphere and by cyclic change in blade pitch. The damping forces are supplied by aerodynamic forces proportional to flapping velocity.

For the hinged blade, the natural frequency is represented by ω_n and the forcing frequency is represented by ω. The flapping rotor, in equilibrium with and without flapping hinge offset from the axis of rotation, is presented in Figure 4.17.

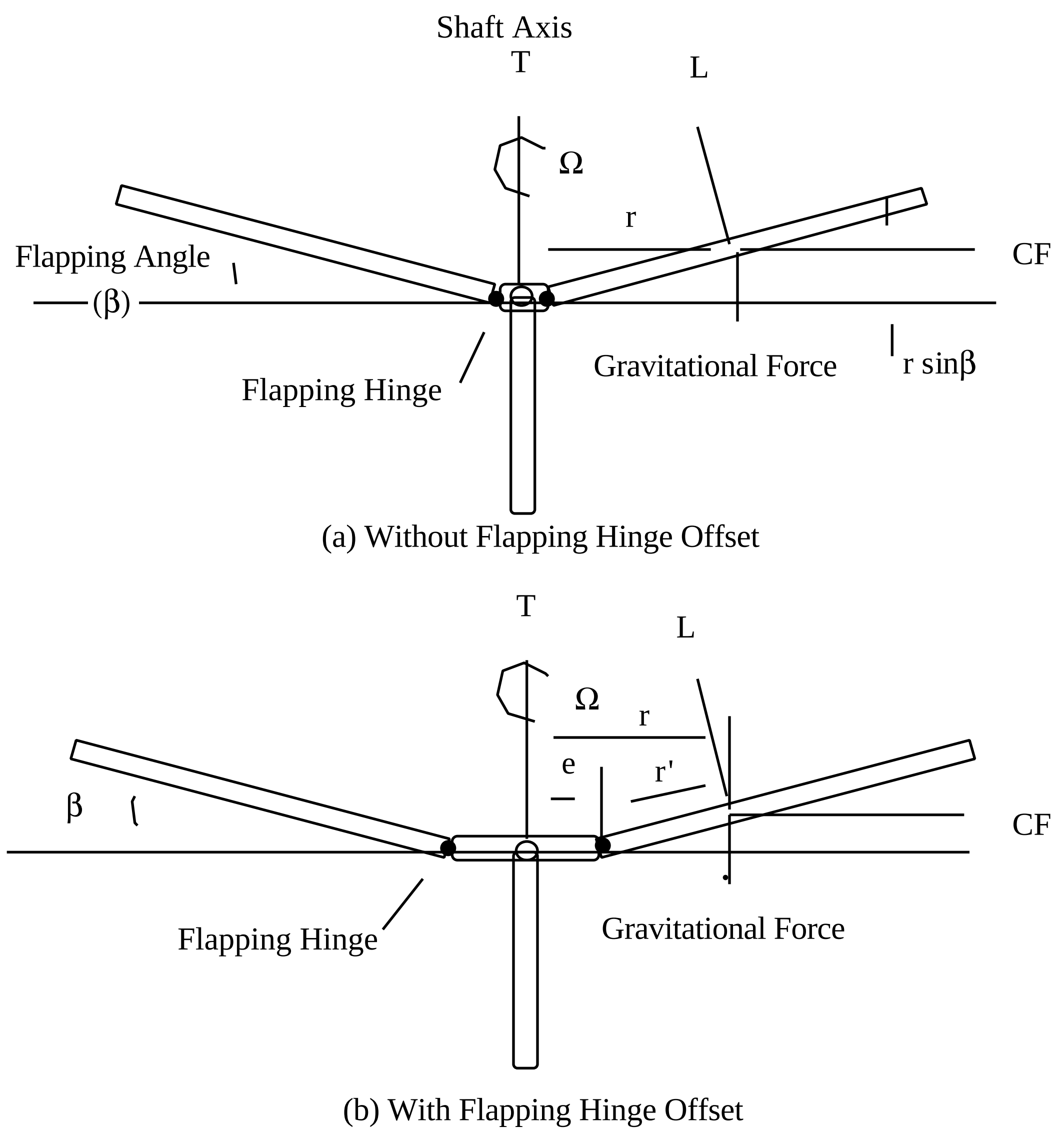


Figure 4.17
Flapping Rotor in Equilibrium

For the rotor without flapping hinge offset, the centrifugal force (CF) acting on a blade element a distance, r, from the center of rotation (shaft axis) is given by the expression:

$$\Delta\ CF = \Omega^2\ r\ m\ \Delta r \qquad \textit{eq 4.7}$$

The restoring moment due to this centrifugal force is given by:

$$\Delta M = \Delta CF\ r\ \sin\beta \qquad \textit{eq 4.8}$$

Using a small angle assumption, $\sin\beta = \beta$, the total moment is given by integrating with respect to r from 0 to R, the rotor radius, is:

$$M = \int_0^R \Omega^2\ r^2\ m\ \beta\ dr = \Omega^2\ \beta \int_0^R m\ r^2\ dr \qquad \textit{eq 4.9}$$

The spring constant for the equivalent spring-mass-damper system is:

$$K = \frac{M}{\beta} = \Omega^2 \int_0^R m\ r^2\ dr \qquad \textit{eq 4.10}$$

The moment of inertia about the flapping hinge is defined as:

$$I_f = \int_0^R m\ r^2\ dr \qquad \textit{eq 4.11}$$

Substituting into the equation for natural frequency of a spring-mass-damper, ω_n, the natural frequency of a rotor without spring offset is given by:

$$\omega_n = \sqrt{\frac{K}{I}} = \sqrt{\frac{\Omega^2\ I_f}{I_f}} = \Omega \qquad \textit{eq 4.12}$$

The frequency ratio of a rotor without hinge offset is given by:

$$\frac{\omega}{\omega_n} = \frac{\omega}{\Omega} = 1 \qquad \textit{eq 4.13}$$

Where:

β	- Flapping angle
CF	- Centrifugal force
I	- Moment of inertia
I_f	- Moment of inertia about flapping hinge
K	- Spring constant
M	- Moment
m	- Mass
R	- Rotor radius
r	- Radius along blade
Ω	- Rotor angular velocity
ω	- Frequency
ω_n	- Natural frequency.

The expression for the frequency ratio of a rotor without hinge offset says for a rotor without flapping hinge offset, the frequency ratio is 1; and the natural frequency is the same as the rotor angular velocity, Ω.

For the rotor with flapping hinge offset, the restoring moment due to the centrifugal force acting on a blade mass element at a distance r' from the flapping hinge such that r = e + r', is given by:

$$\Delta M = m \, \Delta r \, \Omega^2 \, (r' + e) \, r' \, \beta \qquad \textit{eq 4.14}$$

Using a small angle assumption, $\sin \beta = \beta$, the total moment is given by integrating with respect to r from 0 to R - e:

$$M = \Omega^2 \beta \int_0^{R-e} m \, (r' + e) \, r' \, dr' \qquad \textit{eq 4.15}$$

However:

$$I_f = \int_0^{R-e} m \, r'^2 \, dr' \qquad \textit{eq 4.16}$$

And:

$$\frac{M_f}{g} = \int_0^{R-e} m\ r'\ dr' \qquad \textit{eq 4.17}$$

Thus:

$$M = \Omega^2\ \beta\ (I_f + e\frac{M_f}{g}) \qquad \textit{eq 4.18}$$

The spring constant for the equivalent spring-mass-damper system is:

$$K = \frac{M}{\beta} = \Omega^2 \left(I_f + e\frac{M_f}{g}\right) \qquad \textit{eq 4.19}$$

The moment of inertia about the flapping hinge is defined as:

$$I_f = m \int_0^{R-e} r'^2\ dr = \frac{mR^3}{3}\left(1 - \frac{e}{R}\right)^3 \qquad \textit{eq 4.20}$$

And:

$$\frac{M_f}{g} = m \int_0^{R-e} r'\ dr = \frac{mR^2}{2}\left(1 - \frac{e}{R}\right)^2 \qquad \textit{eq 4.21}$$

Substituting into the equation for natural frequency of a spring-mass-damper, ω_n, the natural frequency of a rotor without spring offset is given by:

$$\omega_n = \sqrt{\frac{K}{I_f}} = \Omega\sqrt{1 + \frac{eM_f}{g\ I_f}} = \Omega\sqrt{1 + \frac{\frac{3}{2}\frac{e}{R}}{1 - \frac{e}{R}}} \qquad \textit{eq 4.22}$$

The frequency ratio of a rotor with hinge offset is given by:

$$\frac{\omega}{\omega_n} = \frac{\omega}{\Omega} = \sqrt{1 + \frac{\frac{3}{2}\frac{e}{R}}{1 - \frac{e}{R}}} \qquad \textit{eq 4.23}$$

Where:

β	- Flapping angle
CF	- Centrifugal force
e	- Flapping hinge offset
g	- Gravity
I	- Moment of inertia
I_f	- Moment of inertia about flapping hinge
K	- Spring constant
M	- Moment
m	- Mass
M_f	- Moment about flapping hinge
R	- Rotor radius
r	- Radius along blade
r'	- Radius along blade outboard of flapping hinge
Ω	- Rotor angular velocity
ω	- Frequency
ω_n	- Natural frequency.

The expression for frequency ratio for a rotor with flapping hinge offset says the frequency ratio is not 1; but, is a function of the flapping hinge offset. The natural frequency of the flapping motion is greater than the rotational frequency. For a rotor with a 5% hinge offset, the frequency ratio is 1.04. The influence of frequency ratio on phase angle is discussed later.

4.3.4 Damping Ratio

The damping ratio in combination with the frequency ratio determine the phase angle. The damping ratio of the rotor in motion is found from the equivalent spring-mass-damper system. The linear differential equation for the flapping motion is expressed as:

$$I_f\ddot{\beta} + B\dot{\beta} + K\beta = 0 \qquad \textit{eq 4.24}$$

The general solution for this equation is as follows:

$$\beta = -\frac{B}{2I_f} \pm \sqrt{\left(\frac{B}{2I_f}\right)^2 - \frac{K}{I_f}}$$

eq 4.25

If the term under the radical is negative, the motion is oscillatory. If it is positive, the motion is convergent without oscillation. Critical damping, B_{CRIT}, is the value of the damping constant, B, that makes the radical term zero:

$$B_{CRIT} = 2I_f \left(\frac{K}{I_f}\right)^{\frac{1}{2}}$$

eq 4.26

Further the damping ratio, ζ, is given by the following expression:

$$\zeta = \frac{B}{B_{CRIT}}$$

eq 4.27

Blade damping is evaluated from the aerodynamic hinge moments due to flapping velocity as follows:

$$M_A = \int_0^{R - e} r' \frac{\rho}{2} a\, c\, r'\, \dot{\beta}\, (r' + e)\, \Omega\, dr$$

eq 4.28

And:

$$B = \frac{2M_A}{\partial\dot{\beta}} = \frac{c\rho a R^4}{8} \left(1 - \frac{e}{R}\right)^4 \Omega \left(\frac{1 + \frac{e}{3R}}{1 - \frac{e}{R}}\right)$$

eq 4.29

Substituting Lock number, γ, and frequency ratio into the equation for damping, the damping ratio, ζ, is expressed as follows:

$$\zeta = \frac{\gamma}{16}\left(1 - \frac{e}{R}\right)^4 \left(\frac{1 + \frac{e}{3R}}{1 - \frac{e}{R}}\right) \frac{1}{\omega_n / \Omega}$$

eq 4.30

Where Lock number is defined as:

$$\gamma = \frac{c\rho a R^4}{I_f}$$

eq 4.31

Where:

a	- Lift curve slope
B	- Damping constant
β	- Flapping angle
B_{CRIT}	- Critical damping
$\ddot{\beta}$	- Flapping angle acceleration
$\dot{\beta}$	- Flapping angle rate
c	- Blade chord
e	- Flapping hinge offset
γ	- Lock number
I_f	- Moment of inertia about flapping hinge
K	- Spring constant
M_A	- Aerodynamic moment
R	- Rotor radius
ρ	- Density
r	- Radius along blade
r'	- Radius along blade outboard of flapping hinge
Ω	- Rotor angular velocity
ω_n	- Natural frequency
ζ	- Damping ratio.

Damping ratio is a function of flapping hinge offset, frequency ratio, and Lock number which, for a given design, is a function of air density. Damping ratio along with frequency ratio define the phase angle.

4.3.5 Phase Angle

The phase angle, ψ, is the angular lag between the input (forcing function) and the blade flapping response. The phase angle (from linear vibrating systems analysis) is expressed as:

$$\psi = \tan^{-1}\left[\frac{2\zeta\left(\frac{\omega}{\omega_n}\right)}{1-\left(\frac{\omega}{\omega_n}\right)^2}\right] \qquad \textit{eq 4.32}$$

Where:

ω	- Frequency
ω_n	- Natural frequency
ψ	- Phase angle
ζ	- Damping ratio.

Phase angle is a function of the damping ratio and the frequency ratio. Damping ratio is a function of flapping hinge offset, frequency ratio, and Lock number. Also, frequency ratio is a function of flapping hinge offset. Simply stated, the phase angle is a function of hinge offset and density. The effect of damping ratio and frequency ratio on phase angle is presented in Figure 4.18.

Figure 4.18 shows that for a rotor without flapping hinge offset $(\omega/\omega_n = \omega/\Omega = 1)$ the phase angle is 90˚, irrespective of the damping ratio. Therefore, the flapping response (tilt of the TPP) lags exactly 90˚ behind the input. The input can be the cyclic pitch input (feather) or aerodynamic input. For a rotor with flapping hinge offset, the phase angle is a function of both the frequency ratio and the damping ratio.

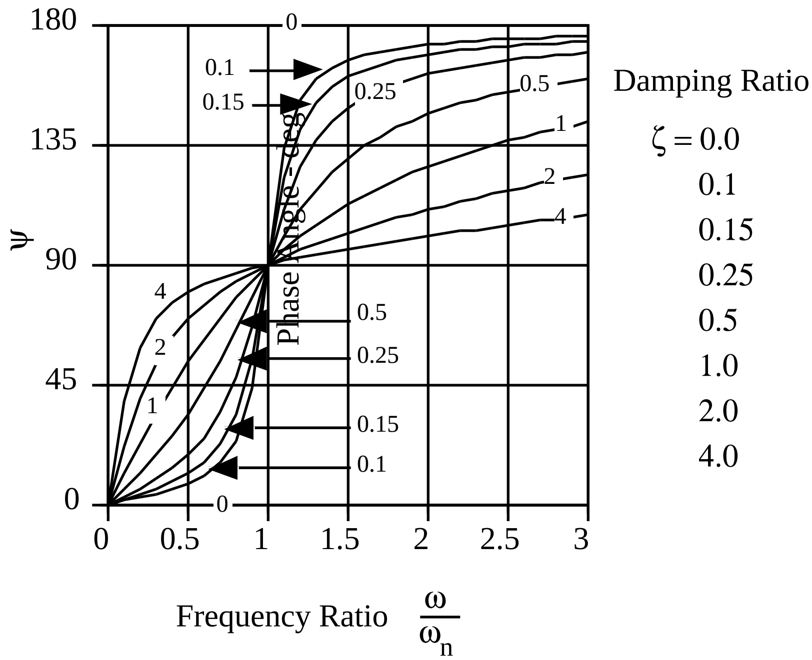


Figure 4.18
Phase Angle as a Function of Frequency and Damping Ratio

4.3.6 Amplitude Ratio

The amplitude ratio is the ratio of the response to the input. In the case of blade flapping, the ratio of flapping response to aerodynamic or control input is a function of the damping ratio and frequency ratio and is expressed as:

$$\mu = \frac{1}{\sqrt{\left[\left(1 - \left(\frac{\omega}{\omega_n}\right)^2\right)^2 + 4\,\zeta^2\left(\frac{\omega}{\omega_n}\right)^2\right]}} \qquad \textit{eq 4.33}$$

Where:

μ	- Amplitude ratio
ω	- Frequency
ω_n	- Natural frequency
ζ	- Damping ratio.

Amplitude ratio is a function of the damping ratio and the frequency ratio. And damping ratio is a function of the flapping hinge offset, frequency ratio, and Lock number. The frequency ratio is a function of flapping hinge offset. The effect of damping ratio and frequency ratio on the amplitude ratio is presented in Figure 4.19.

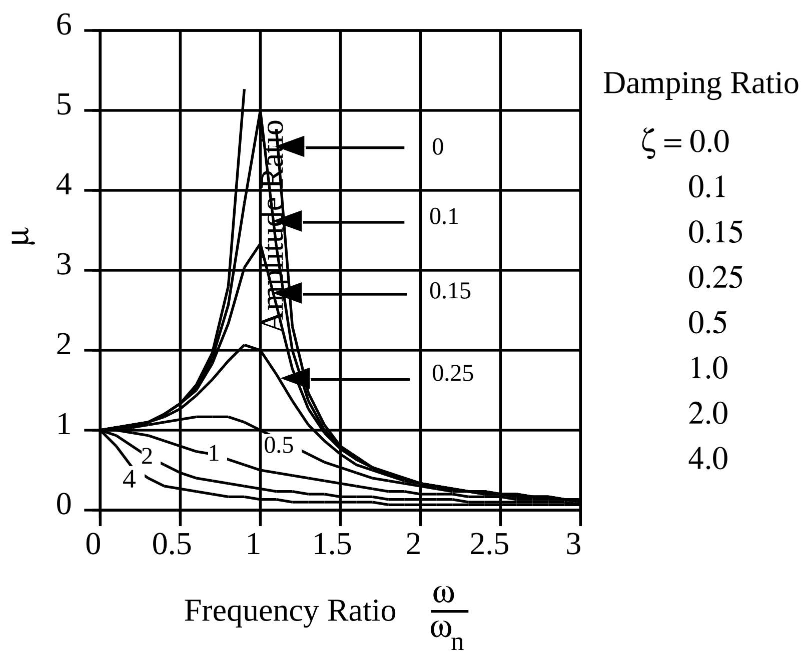


Figure 4.19
Amplitude Ratio as a Function of Frequency and Damping Ratio

Figure 4.19 shows that for a damping ratio greater than 1, the amplitude ratio decreases as the frequency ratio increases. For a damping ratio less than 1, the amplitude ratio is greatest at a frequency ratio of 1.

4.3.7 Cross Coupling

For the rotor with offset flapping hinges or effective offset, the frequency ratio and damping ratio define the phase lag between the input and the flapping response as presented above. The difference between the phase angle and 90˚ is one source of longitudinal/lateral flapping cross coupling. The magnitude of the coupling is given by the following expression:

$$\frac{b_{1s}}{a_{1s}} = -\cot\psi$$

eq 4.34

This expression says that as the phase angle becomes less than 90˚, the coupling increases. Substituting for phase angle, frequency ratio, and damping ratio gives the following expression for coupling:

$$\frac{b_{1s}}{a_{1s}} = -\frac{\frac{12}{\gamma}\frac{e}{R}}{\left[1 + \frac{e}{3R}\right]}$$

eq 4.35

Where:

a_{1s}	- Longitudinal flapping angle
b_{1s}	- Lateral flapping angle, shaft referenced
e	- Flapping hinge offset
γ	- Lock number
R	- Rotor radius
ψ	- Phase angle.

This expression is sometimes called acceleration cross coupling because it is associated with rotor moments which provide the initial acceleration during a maneuver. This cross coupling is a function of the hinge offset and Lock number. For a helicopter with a Lock number of 8.1 and a hinge ratio, e/R, of 0.05; a 1˚ change in longitudinal flapping is accompanied by -0.7˚ of lateral flapping. Thus, as the pilot uses aft longitudinal cyclic to tilt the TPP, he moves the cyclic slightly to the right to cancel the left roll otherwise generated.

4.3.8 Mechanical Correction for Coupling

The effect of phase angle on coupling due to control input is compensated for by changing the angular alignment between the control input and the swashplate. If the phase angle is constant, the angular alignment of the longitudinal and lateral inputs can correct for the angle. However, the phase angle is not constant but a function of Lock number, damping ratio, and frequency ratio. Thus, the designer has to make a compromise in establishing the angular alignment of the flight controls and the swashplate.

4.3.9 Flapping Due to Pitch and Roll Rates

The rotating rotor produces damping moments when the helicopter is subjected to pitch and roll rates. The damping produced by the tilt of the TPP lags behind the motion of the shaft by an amount proportional to the pitch or roll rate. The aerodynamic forces on the blade tend to stabilize the TPP with respect to the shaft. The TPP, attached to a shaft which is tilting, follows the shaft. To maintain a steady nose up pitch rate, the air load on the advancing blade is higher than on the retreating blade. This asymmetry is generated by downward flapping velocity with respect to the shaft at $\psi = 90°$ and an upward velocity at $\psi = 270°$. The maximum flapping amplitude with respect to the shaft is down at $\psi = 180°$ and up at $\psi = 0°$. The TPP follows the motion of the shaft at a lag angle that is proportional to the pitch rate and the rotor moment of inertia.

During a steady nose up pitch rate, some lateral flapping is generated because of the decreased angle of attack at $\psi = 180°$ and the increase at $\psi = 0°$. The change in angle of attack is caused by the pitch rate. This difference in angle of attack is compensated for by the lateral flapping which produces enough flapping velocity at these two blade locations to cancel out the effect of the pitch rate. The longitudinal and lateral flapping caused by a pure pitch rate is a source of cross coupling applicable to rotors with and without flapping hinge offset.

The expression for rate cross coupling for a hovering rotor without hinge offset experiencing only a pitch rate is given by the following simplified expression:

$$a_{1s} = -\left(\frac{16}{\gamma}\right)\left(\frac{q}{\Omega}\right)$$ *eq 4.36*

$$b_{1s} = -\left(\frac{q}{\Omega}\right)$$ *eq 4.37*

The complete equations for rate cross coupling are expressed as:

$$a_{1s} = \frac{\dfrac{-\dfrac{16}{\gamma}\left(\dfrac{q}{\Omega}\right)}{\left(1-\dfrac{e}{R}\right)^2} + \left(\dfrac{p}{\Omega}\right)}{1-\dfrac{\mu^2}{2}} + \frac{\dfrac{\dfrac{12}{\gamma}\dfrac{e}{R}}{\left(1-\dfrac{e}{R}\right)^3}\left[\dfrac{-\dfrac{16}{\gamma}\left(\dfrac{p}{\Omega}\right)}{\left(1-\dfrac{e}{R}\right)^2} - \left(\dfrac{q}{\Omega}\right)\right]}{1-\dfrac{\mu^4}{4}}$$ *eq 4.38*

$$b_{1s} = \frac{\dfrac{-\dfrac{16}{\gamma}\left(\dfrac{p}{\Omega}\right)}{\left(1-\dfrac{e}{R}\right)^2} + \left(\dfrac{q}{\Omega}\right)}{1-\dfrac{\mu^2}{2}} + \frac{\dfrac{\dfrac{12}{\gamma}\dfrac{e}{R}}{\left(1-\dfrac{e}{R}\right)^3}\left[\dfrac{-\dfrac{16}{\gamma}\left(\dfrac{q}{\Omega}\right)}{\left(1-\dfrac{e}{R}\right)^2} - \left(\dfrac{p}{\Omega}\right)\right]}{1-\dfrac{\mu^4}{4}}$$ *eq 4.39*

With the advance ratio, μ, defined as:

$$\mu = \frac{V}{\Omega R}$$ *eq 4.40*

Where:

a_{1s} - Longitudinal flapping angle, shaft referenced
b_{1s} - Lateral flapping angle, shaft referenced
e - Flapping hinge offset
γ - Lock number

μ	- Advance ratio
p	- Roll rate
q	- Pitch rate
R	- Rotor radius
V	- Relative velocity
Ω	- Rotor angular velocity.

This expression shows that for a positive nose up pitch rate the TPP lags behind longitudinally and tilts to the left laterally. For a conventional Lock number of 8, the lateral tilt is half the longitudinal lag angle. If the pitch rate is being produced by the pilot in a deliberate manner using cyclic pitch, both longitudinal and lateral flapping is essentially zero and the trim value of lateral cyclic is approximately half the longitudinal cyclic pitch. In this case, the longitudinal cyclic pitch angle, B_{1s}, is negative and the lateral cyclic pitch angle, A_{1s}, is positive to prevent the left roll.

In each of these equations (4.38, 4.39), the first term is the basic flapping due to pitch and roll rates. The second term represents the cross coupling due to hinge offset. Although offset is present in the first term, the impact of offset is primarily through the second term. In all cases, the coupling is a function of hinge offset, Lock number, and the advance ratio. The higher the advance ratio in forward flight the greater the coupling.

4.3.10 Blade Flapping in Forward Flight

The equations for blade feathering and flapping have been examined along with the impact of frequency ratio, damping ratio, and phase angle on flapping. We now turn our attention to solving for the blade flapping coefficients in forward flight and examining factors which influence flapping in forward flight. The expression for flapping was previously given as:

$$\beta = a_0 - a_{1s}\cos\psi - b_{1s}\sin\psi - a_{2s}\cos2\psi - b_{2s}\sin2\psi \ldots - a_{ns}\cos n\psi - b_{ns}\sin n\psi \qquad \textit{eq 4.6}$$

Ignoring the higher harmonic motion, the expression simplifies as:

$$\beta = a_0 - a_{1s}\cos\psi - b_{1s}\sin\psi \qquad \textit{eq 4.41}$$

The equation of motion for the blades was given as:

$$I_f\ddot{\beta} + B\dot{\beta} + K\beta = 0$$

eq 4.24

By differentiating equation 4.41 for $\dot{\beta}$ and $\ddot{\beta}$; assuming a steady state condition with $\dot{a}_{1s} = \dot{b}_{1s} = 0$; making the small angle assumption; ignoring reverse flow, blade twist, stall, and compressibility; and assuming blade motion consists only of coning and the first harmonic flapping motion; we solve for the flapping coefficients. The flapping coefficients are approximated as follows:

$$a_0 = \frac{\gamma}{8}\left[\theta_C\left(1 + \mu^2\right) + \frac{4}{3}\lambda\right] - \frac{M_W}{I\Omega^2}$$

eq 4.42

$$a_{1s} = -B_{1s}\,\frac{1 + \frac{3}{2}\mu^2}{1 - \frac{\mu^2}{2}} + \frac{\frac{8}{3}\theta_C\,\mu + 2\,\lambda\,\mu}{1 - \frac{\mu^2}{2}}$$

eq 4.43

$$b_{1s} = A_{1s} + \frac{\frac{4}{3}\mu\,a_0}{1 + \frac{\mu^2}{2}}$$

eq 4.44

With the inflow ratio defined as:

$$\lambda = \frac{V\sin\alpha_s - v_i}{\Omega R}$$

eq 4.45

Where:

a_0 - Coning angle
A_{1s} - Lateral cyclic pitch angle, shaft referenced
a_{1s} - Longitudinal flapping angle, shaft referenced
α_s - Angle of attack, shaft referenced

B	- Damping constant
β	- Flapping angle
$\ddot{\beta}$	- Flapping angle acceleration
$\dot{\beta}$	- Flapping angle rate
B_{1s}	- Longitudinal cyclic pitch angle, shaft referenced
b_{1s}	- Lateral flapping angle, shaft referenced
γ	- Lock number
I	- Moment of inertia
I_f	- Moment of inertia about flapping hinge
K	- Spring constant
λ	- Inflow ratio
μ	- Advance ratio
M_W	- Moment due to weight
θ_C	- Collective pitch angle
R	- Rotor radius
V	- Relative velocity
v_i	- Induced velocity
Ω	- Rotor angular velocity
ψ	- Blade azimuth angle.

Typically, the second term in the expression for a_0, $\frac{M_W}{I\Omega^2}$,is neglected because the weight of the blades is relatively small.

Several important relationships are contained in the above expressions. The coning angle, a_0, is a function of Lock number, advance ratio, and inflow ratio. The coning angle increases as forward flight speed increases and as inflow increases.

Longitudinal flapping angle (a_{1s}) is a function of the longitudinal cyclic pitch angle (B_{1s}). In accordance with the established sign convention, aft longitudinal cyclic pitch angle ($-B_{1s}$) produces aft longitudinal flapping ($+a_{1s}$) and aft tilt of the TPP.

Longitudinal flapping angle is a function of the advance ratio; as forward speed increases, a given ΔB_{1s} produces increased Δa_{1s}. In forward flight, the increase in a_{1s} due to μ, produces an increase in control power. Also, through the second term, as forward speed is increased for a fixed B_{1s}, the longitudinal flapping increases and the TPP tilts aft or blows back contributing to speed stability (M_u).

Longitudinal flapping angle is a function of collective pitch angle, θ_C. Increased θ_C produces increased a_{1s}. An increase in collective produces an increase in a_{1s} and the TPP tilts aft. The increase in a_{1s} due to collective is the reason forward cyclic is required as the collective is increased during termination of an approach.

Longitudinal flapping angle is a function of inflow ratio. Increased inflow such as caused by a vertical gust produces increased a_{1s}. Therefore, in a vertical gust the TPP tilts aft.

Lateral flapping angle (b_{1s}) is a function of lateral cyclic pitch angle (A_{1s}). In accordance with the established sign convention, right lateral cyclic pitch $(+A_{1s})$ produces right flapping $(+b_{1s})$ and right tilt of the TPP.

Lateral flapping angle is a function of the advance ratio. Increased forward speed produces increased lateral flapping and right tilt of the TPP. However, increased forward speed does not produce an increase in control power as is the case for longitudinal cyclic.

Lateral flapping is not a function of θ_C. However, increased collective produces increased coning, a_0, thereby increasing lateral flapping, resulting in a right tilt of the TPP.

Lateral flapping is not a function of inflow ratio. However coning is a function of inflow ratio. A vertical gust increases coning, a_0, thereby increasing lateral flapping, resulting in a right tilt of the TPP.

Figure 4.20 presents the TPP and blade flapping response to a longitudinal cyclic input in a hover. The representation is for a rotor system at a hover with a 90˚ phase angle.

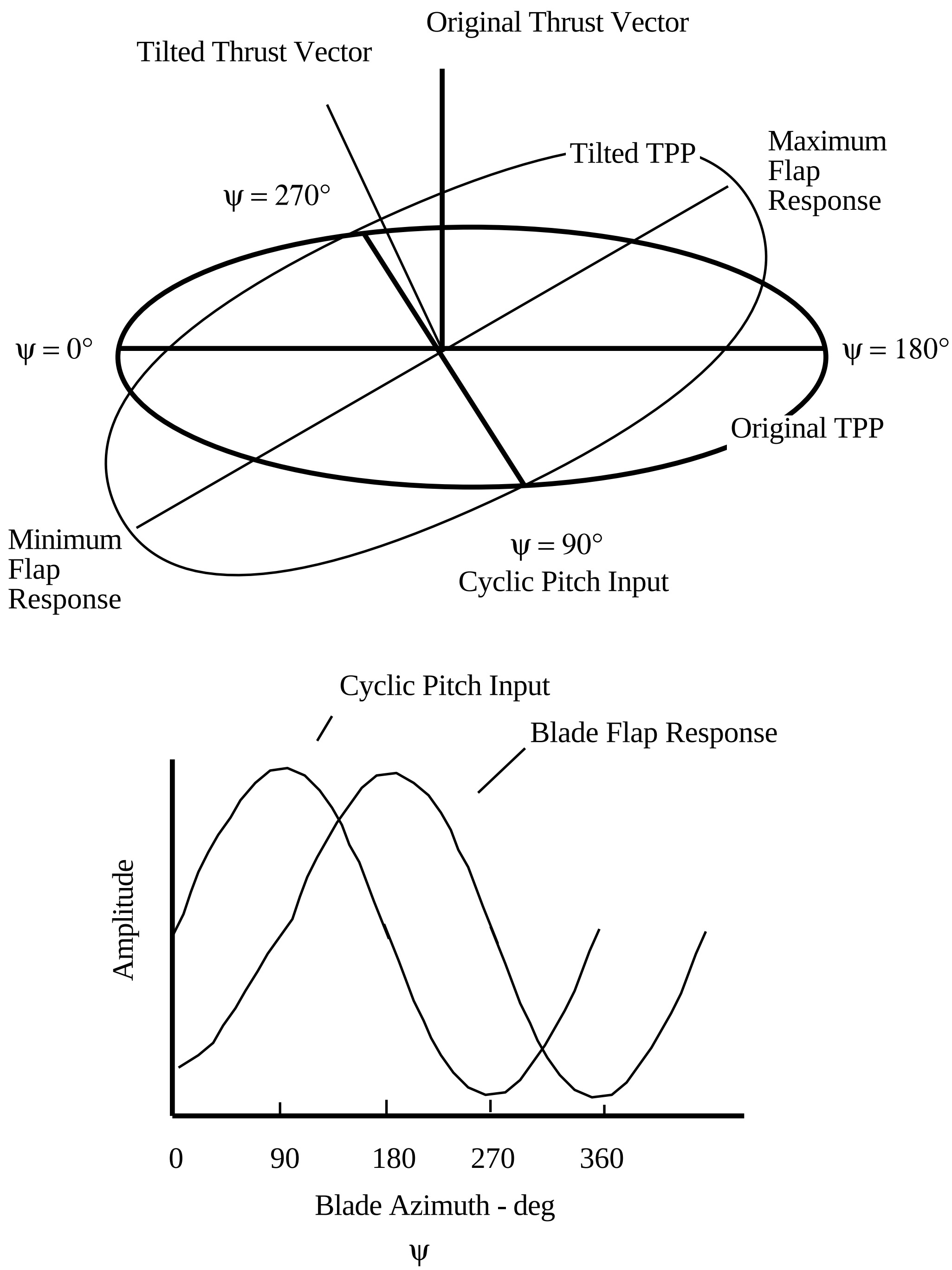


Figure 4.20
TPP Response to a Longitudinal Cyclic Input in Hover

As the TPP responds to a longitudinal cyclic input in hover, each blade of a multi-bladed rotor has the same flapping angle at a given azimuth position and lies in the new TPP. The new TPP is oriented with the maximum upward displacement 90˚ after the cyclic pitch input (phase angle = 90˚). Furthermore, the magnitude of the flapping (and TPP tilt) is equal to the magnitude of the cyclic control input. Since the thrust vector is perpendicular to the TPP, a moment is generated about the helicopter CG proportional to the product of mast height and TPP tilt.

4.3.11 TPP Response to Non-Uniform Downwash

The fore/aft variation of inflow is indicated schematically on Figure 4.21. A downwash variation from zero at the front of the disk, to maximum at the aft edge of the disk, is obtained by adding a uniform downwash to one varying linearly, with an up value at the front of the disk, and a down value at the aft edge of the disk. Figure 4.21 also shows the flow relative to a rotor blade with coning. A similar relative velocity situation exists. There is an up flow relative to a blade at the front of the disk and a down flow relative to the blade at the trailing edge of the disk. These velocities, normal to the TPP, change the effective blade angles of attack. The angle of attack is increased at the front of the rotor disk and decreased at the rear of the disk. The TPP tilts to the right if the pilot makes no compensating control inputs. This effect is most critical in the low airspeed transition region. In this region, the downwash effect is considerably larger than that due to coning.

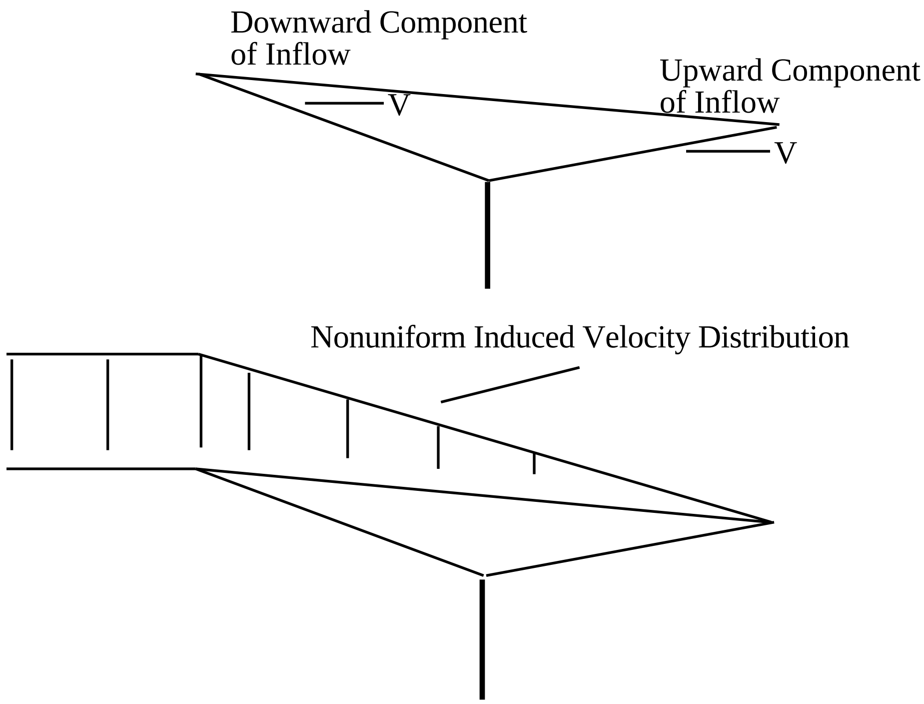


Figure 4.21
Two Fore and Aft Inflow Effects

4.3.12 TPP Response to Collective Step

When a step input in collective pitch is made for a helicopter in a hover, all rotor blades respond by flapping upward and stabilize at a greater coning angle. A graph of the coning angle and rotor thrust against time following a collective pitch step is shown on Figure 4.22. Figure 4.22 illustrates a rotor with Lock number, $\gamma = 8.82$, and rotor angular velocity, $\Omega = 20$ rad/s. At zero forward speed, application of collective pitch has little effect on the tilt of the TPP. Maximum lift is reached in about 0.16 s, or about half a revolution. The rotor effectively stabilizes at the new coning angle within 0.3 s, or about one revolution after application of the collective control.

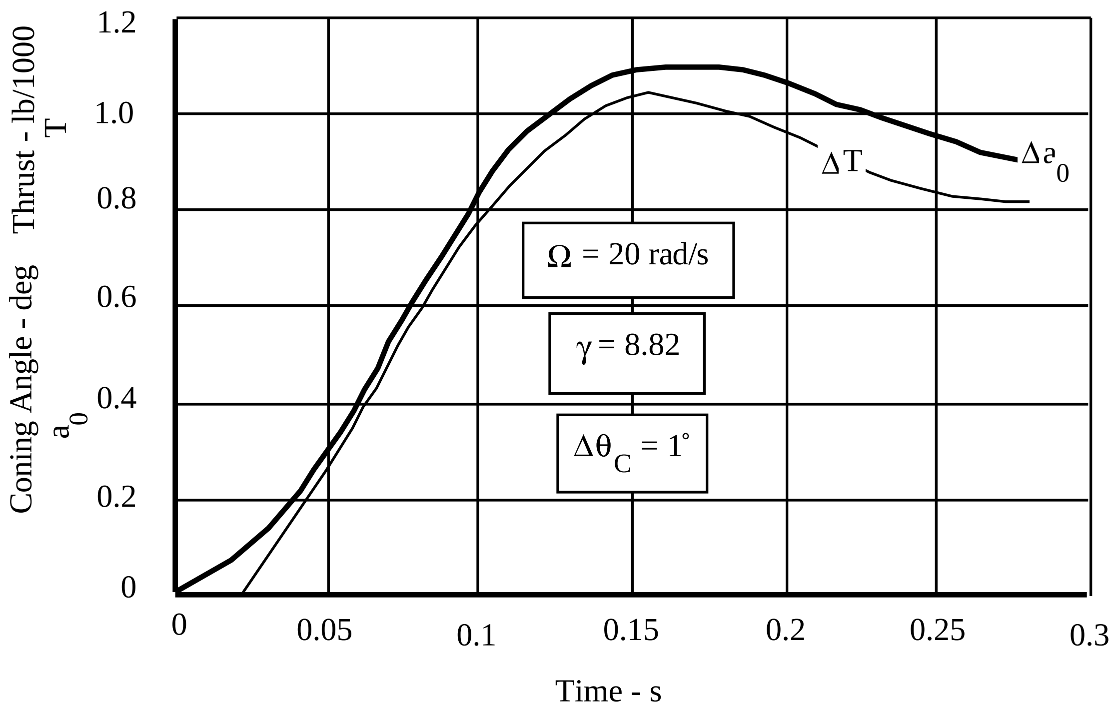


Figure 4.22
Rotor Response to Collective Pitch Step at Hover

4.3.13 TPP Response to Cyclic Step

When a step cyclic control input is made introducing a cyclic pitch angle change, aerodynamic flapping forces tilt the TPP. Because of the inertia of the blades, the TPP does not assume the new attitude instantaneously. Figure 4.23 shows the change in TPP angle with time and rotor revolutions following a step cyclic pitch input. The TPP moves about 85% of the way to its new position in about half a revolution, or 0.15 s and essentially stabilizes in the new position within about one revolution, or 0.3 s after the cyclic control application.

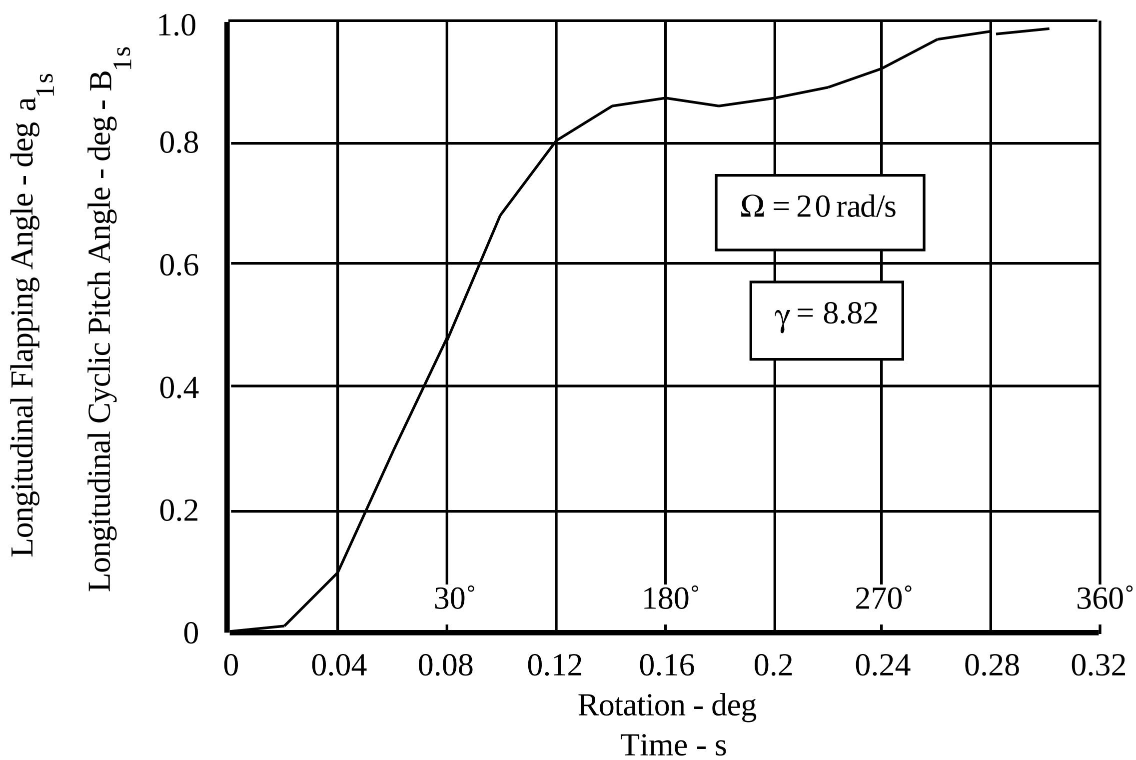


Figure 4.23
Rotor Response to Cyclic Pitch Step at Hover

4.3.14 Offset Hinge Moments

The rotor blades of a fully articulated rotor system normally do not have their flapping axis on the shaft axis. There is an offset between the blade flapping hinge and the shaft axis. Whenever the TPP forms an angle with the shaft plane, the offset allows centrifugal blade forces to impart a tilting moment to the shaft at the hub. Figure 4.24

shows the forces acting on the blade and the hub. The shear forces acting at the flapping hinge are denoted by F_S, for both the blade acting on the hub and the hub acting on the blade. The thrust force of the blade on the hub acting at the flapping hinge is denoted by F_T. The gravitational force and the centrifugal forces are shown.

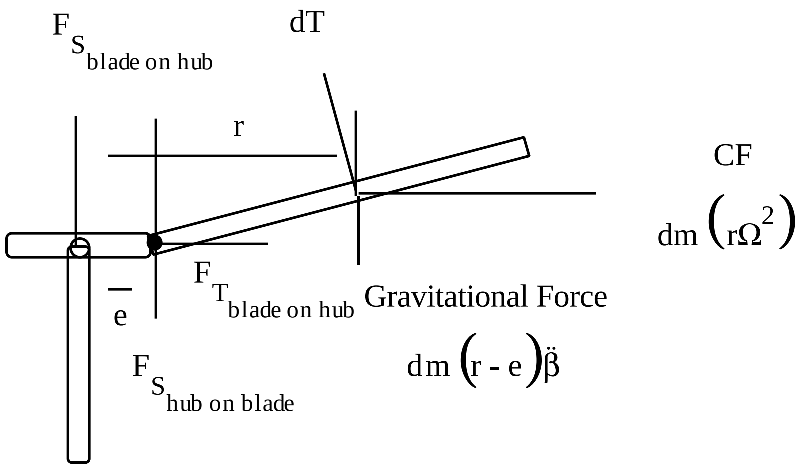


Figure 4.24
Moments Produced by Flapping

The sum of the forces acting on the blade is equal to the product of the mass of the blade and the linear acceleration given by the following expression:

$$-F_{S_{\text{hub on blade}}} + \int_e^R dT\cos\beta = \int_e^R \ddot{\beta}\,(r - e)\,dm \qquad \textit{eq 4.46}$$

Assuming small angles, $\cos\beta = 1$, substituting for $\ddot{\beta}$, the expression for F_S (blade on hub, the reciprocal of hub on blade) becomes:

$$F_{S_{\text{blade on hub}}} = T_b - \Omega^2 M_S\left(a_{1s}\cos\Omega t + b_{1s}\sin\Omega t\right) \qquad \textit{eq 4.47}$$

The pitching and rolling moment generated by the shear force about the hub varies as a function of azimuth as follows:

$$M_H(\psi) = -F_{S_{\text{blade on hub}}}(e\cos\psi) \quad \text{eq 4.48}$$

$$L_H(\psi) = -F_{S_{\text{blade on hub}}}(e\sin\psi) \quad \text{eq 4.49}$$

The average pitching and rolling moment for b blades is as follows:

$$\overline{M_H} = \frac{b}{2\pi}\int_0^{2\pi} M_H(\psi)\, d\psi \quad \text{eq 4.50}$$

$$\overline{L_H} = \frac{b}{2\pi}\int_0^{2\pi} L_H(\psi)\, d\psi \quad \text{eq 4.51}$$

Substituting and integrating, the average pitching and rolling moments become:

$$\overline{M_H} = \frac{1}{2}\, eb\, \Omega^2 M_S\, a_{1s} \quad \text{eq 4.52}$$

$$\overline{L_H} = \frac{1}{2}\, eb\, \Omega^2 M_S b_{1s} \quad \text{eq 4.53}$$

Where:

a_{1s}	- Longitudinal flapping angle
β	- Flapping angle
b	- Number of blades
b_{1s}	- Lateral flapping angle
$\ddot{\beta}$	- Flapping angle acceleration
e	- Flapping hinge offset
$F_{S\text{ blade on hub}}$	- Shear force of the blade on hub
$F_{S\text{ hub on blade}}$	- Shear force of the hub on blade
L_H	- Roll moment due to rotor hub forces
$\overline{L_H}$	- Average roll moment due to rotor hub forces for b blades

m	- Mass
M_H	- Pitch moment due to rotor hub force
$\overline{M_H}$	- Average pitch moment due to rotor hub force for b blades
M_S	- Blade mass moment
π	- Mathematical constant
R	- Rotor radius
r	- Radius along blade
T	- Thrust
t	- Time
T_b	- Thrust of b blades
Ω	- Rotor angular velocity
ψ	- Blade azimuth angle.

The expressions for $\overline{M_H}$ and $\overline{L_H}$ are positive when a_{1s} and b_{1s} are positive, that is aft and right tilt of the TPP. The hub moments are a function of the flapping hinge offset. The hub moments create an increase in control power. Figure 4.25 represents a simple example of a helicopter in a hover with the shaft axis through the CG.

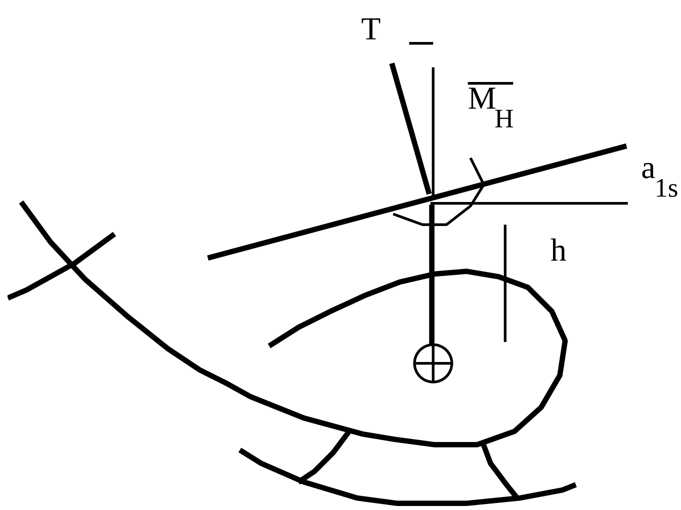


Figure 4.25
Longitudinal Moments About the CG

In this simple example, assuming small angles, the sum of the moments about the CG is as follows:

$$\sum M_{CG} = Th \sin\ a_{1s} + \overline{M_H} = Th + \frac{1}{2} eb\,\Omega^2 M_S\ a_{1s} \qquad \textit{eq 4.54}$$

Where:

a_{1s}	- Longitudinal flapping angle
b	- Number of blades
e	- Flapping hinge offset
h	- Height of hub above CG
M_{CG}	- Moment about CG
$\overline{M_H}$	- Average pitch moment due to rotor hub force for b blades
M_S	- Blade mass moment
T	- Thrust
Ω	- Rotor angular velocity.

The expression for the moment about the CG shows an increase in the M_{CG} and control power due to the hub moment produced by the flapping hinge offset.

4.3.15 H Force Due to Flapping

Rotor thrust is often assumed to be perpendicular to the TPP; however, the effect of inflow produces forces which act parallel to the shaft plane ($\perp$ to the shaft). Figure 4.26 represents a helicopter hovering with aft tilt of the TPP due to the CG being ahead of the shaft.

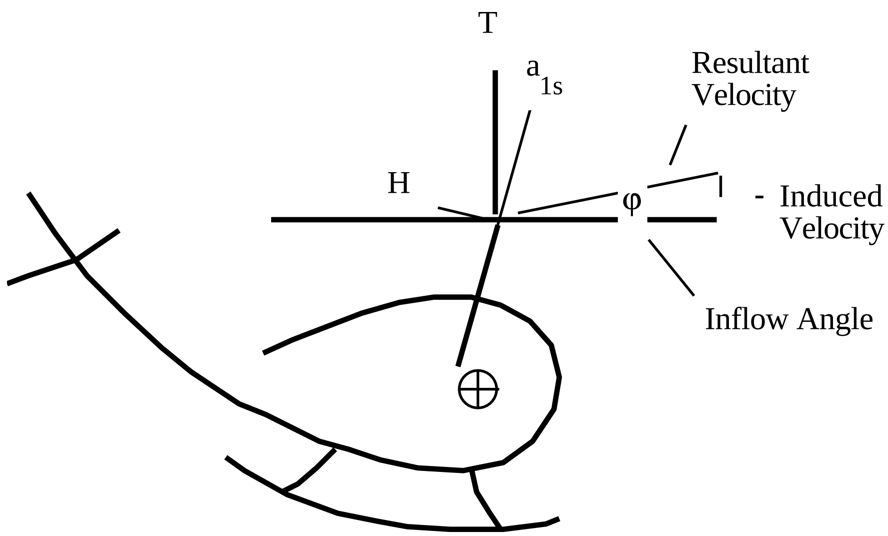


Figure 4.26
H Force Due to Inflow

The expression for the total H force is given by:

$$H = T \sin a_{1s} \cos \varphi \qquad \textit{eq 4.55}$$

Where:

- a_{1s} - Longitudinal flapping angle
- H - Rotor hub force, ⊥ to shaft
- φ - Inflow angle
- T - Thrust.

The total H force is quite small since the inflow angle is usually small with small longitudinal flapping angles.

4.3.16 Thrust Increment Due to Speed Change

A speed increment close to hovering produces a secondary change in the average dynamic pressure experienced by the rotor blades and a negligible change in thrust ($\Delta T \approx 0$). The average induced velocity remains nearly constant but a fore and aft downwash distribution develops in the low airspeed range which effects lateral flapping.

In forward flight, a thrust increment is produced by an increase in forward speed. The thrust increment is due to several factors whose relative magnitudes depends on the initial trim speed. An increase in average dynamic pressure in conjunction with collective pitch increases thrust. Decreased induced velocity with a forward speed increment increases the effective angle of attack of the blade sections and their lift. However, at higher forward speeds, the TPP is tilted forward so a speed increment produces a downward flow normal to the disk reducing the effective blade angle of attack and thrust. Typical thrust changes with speed increments are positive for low trim speeds, but become negative for high trim speeds (TPP tilted forward).

4.3.17 Effects of Compressibility and Retreating Blade

There are several ways compressibility effects handling qualities in high speed flight. Noise and vibration caused by compressibility are an annoyance to the pilot and often provide a high speed limitation on the helicopter. The possibility of encountering blade flapping or pitching flutter increases when blade aerodynamic characteristics are modified by compressibility. Although these instabilities may not be encountered, vibrations are amplified at flight conditions close to flutter.

The most significant effect of compressibility on handling qualities arises from large changes in the blade section pitching moment coefficient as a function of angle of attack. Since the local Mach number varies with blade azimuth angle, these changes cause large cyclic variations in pitching moments and result in corresponding twisting of the blades. A cyclic twisting of the blades has the same effect as a cyclic control input. Consequently, blade flapping occurs and undesirable pitching and/or rolling moments are applied to the helicopter. The pilot must apply cyclic control inputs to compensate for these undesirable moments. Blade pitching moments due to compressibility cause an additional problem for helicopters without irreversible controls where they result in forces being fed back through the control system.

Retreating blade stall is another phenomenon which tends to occur concurrently with advancing blade compressibility problems in high speed flight. While forward speed adds to the relative velocity of the advancing blade, it subtracts from the speed of the retreating blade. At high forward speeds a substantial portion of the rotor disc may be stalled, or operating in reverse flow. Figure 4.7 indicates the relative velocity of the advancing and retreating blades. The edge of the circular region on the retreating blade side of the rotor disc shows the outer limits of the reverse flow region. Since the boundary of this region corresponds to relative airspeed equal to zero, the region over which the blades are stalled is considerably higher.

The diameter of the reverse flow region is V/Ω at any forward speed; therefore, some portion of the rotor disc is operating in reverse flow. At low airspeeds, the effects are small because a substantial portion of the reverse flow region encompasses the inboard blade root region and because the contribution of the inboard blade sections to the total blade forces is small. At high speeds, this reverse flow region typically extends over 50% or more of the retreating blade radius, causing substantial loss of lift and undesirable pitching and rolling moments.

4.3.18 Characteristics of Tandem Rotor Configuration

Some aspects of stability and control of tandem rotor configurations are simpler than single rotor helicopters. The aircraft is supported by the thrust of two rotors. The thrust of each rotor is controlled through the conventional use of collective blade pitch. Aircraft pitching moment is produced through the use of differential collective between the fore and aft rotor. Yawing moment is produced by tilting the front rotor to the opposite side laterally from the rear rotor. Rolling moment is generated by tilting both rotors laterally in the same direction.

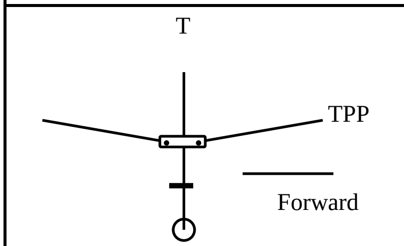
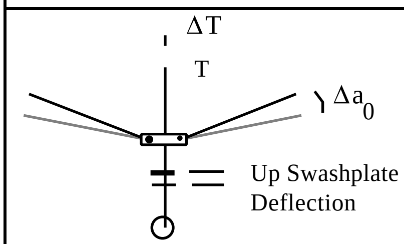
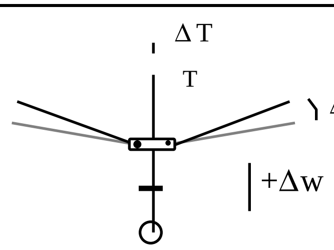
On the other hand, the analysis of the response of a tandem configuration to vehicle motions is complicated by the interference effects between the two rotors and between the rotors and the airframe. The significant effects of tandem rotors on the various stability derivatives of the vehicles, the corresponding aircraft dynamics, and the important differences from single rotor helicopters are covered at the end of the pertinent chapters.

4.3.19 Quasi-Static Approximation

Figures 4.22 and 4.23 indicate typical rotor responses to step collective and cyclic control inputs. In both cases, the rotor stabilizes in a new TPP within one revolution, or approximately 0.3 s. This suggests that dynamic blade responses have little effect on the flight dynamics of the helicopter because the time constants and periods of both the long term and short term flight dynamic response modes are much longer. Consequently, a quasi-static approximation for rotor response assumes that any time the TPP is tilted relative to the swashplate, the tilt is consistent with the instantaneous values of the airspeed, angle of attack, pitch rate, and roll rate. This approximation is valid for most unstabilized or lightly augmented helicopters. However, it should be applied with caution when high gain augmentation results in aircraft response modes with very short time constants and/or periods.

Tables 4.I and 4.II present summaries of quasi-static rotor response characteristics for reference in other chapters of this manual. The tables show the most significant responses of the rotor to specified input conditions and summarize the reasons for particular rotor contributions to stability derivatives.

Table 4.I
Quasi-Static Rotor Characteristics in Hovering Flight

	Input	Rotor Response	Rotor Force Applied to Aircraft
	(a) Nominal hover Nominal collective No cyclic	Steady state.	Thrust (T) acts along shaft axis through CG
	(b) Collective Increment, $+\Delta\theta_C$	Increased coning. $+\frac{\partial a_0}{\partial \theta_C}$	Thrust increase, $+\Delta T$
	(c) Downward rotor velocity or upward gust $+\Delta w$	Increased blade angle of attack. Increased coning. $+\frac{\partial a_0}{\partial w}$	Thrust increase, $+\Delta T$

Quasi-Static Rotor Characteristics in Hovering Flight (cont'd)

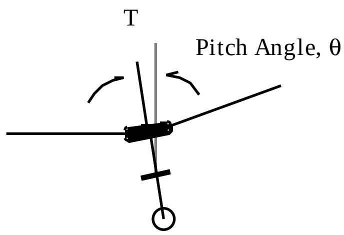
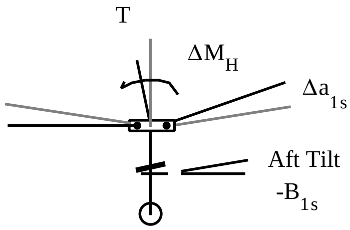
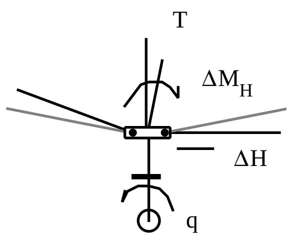
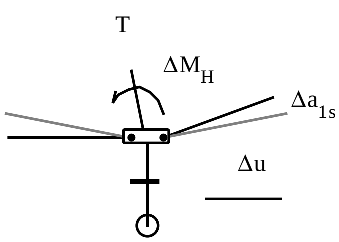
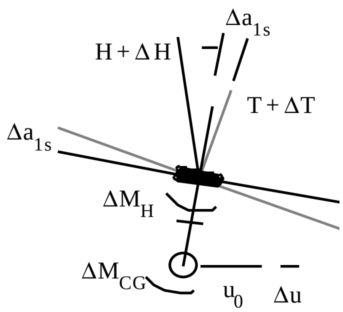
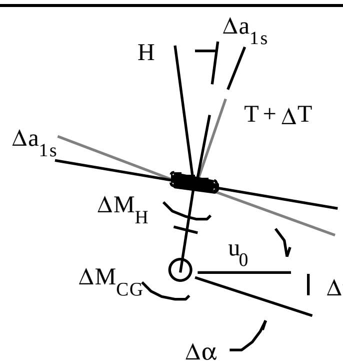
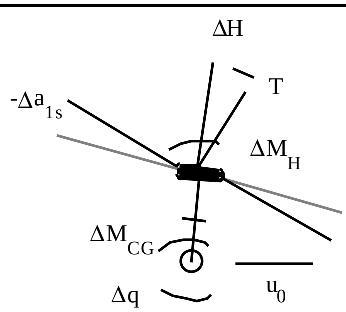
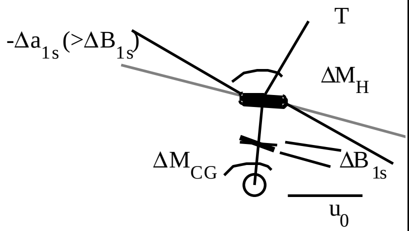
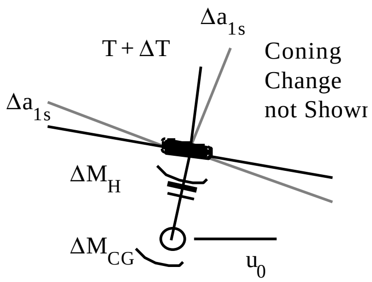
	(d) Shaft pitch angle, $+\Delta\theta$	TPP remains perpendicular to swashplate.	Thrust remains along shaft. No moment increase.
	(e) Cyclic pitch, $+\Delta B_{1s}$	TPP tilts due to cyclic pitch, nose up $-\frac{\partial a_{1s}}{\partial B_{1s}}$	Thrust vector tilts relative to shaft producing moment about CG. Offset hinge moment due to rotor tilt.
	(f) Pitch rate, $+\Delta q$	Rotor lags shaft by angle proportional to pitch rate $-\frac{\partial a_{1s}}{\partial q}$	Longitudinal thrust component damps pitch rate. H force reduces pitch damping. Offset hinge moment due to rotor tilt damps pitch rate.
	(g) Airspeed increase, $+\Delta u$	Rotor tilts away from relative velocity, nose up, stable (blowback) $+\frac{\partial a_{1s}}{\partial u}$	Thrust vector tilts back producing nose up pitching moment. Nose up offset hinge moment due to aft rotor tilt. Aft rotor H force.

Table 4.II
Quasi-Static Rotor Characteristics in Forward Flight

	Input	Rotor Response	Force and Moments Applied to Aircraft
	(a) Airspeed increase, +Δu	Rotor tilts away from relative velocity, nose up, stable, (blowback) $+\frac{\partial a_{1s}}{\partial u}$	ΔT = + (low u_0), - (high u_0). ΔH = + (aft). ΔM_H = + (nose up). $\Delta(M_{CG})$ = + (nose up, stable). $T(\Delta a_{1s})$ = + (aft, produce nose up moment).
	(b) Downward velocity, Δw $\Delta\alpha$ = $\Delta w/u_0$	Increased blade angle of attack. Increased coning. Increased lateral flapping due to increased coning. Increased longitudinal flapping. Nose up, unstable $+\frac{\partial a_0}{\partial w}$ $+\frac{\partial b_{1s}}{\partial a_0}$ $+\frac{\partial a_{1s}}{\partial w}$	ΔT =+ (all trim u_0) ΔH = 0, - (low α_{BE}) ΔM_H = + (nose up) $\Delta(M_{CG})$ = + (nose up, unstable) $T(\Delta a_{1s})$ = + (aft)
	(c) Pitch rate, +Δq, nose up	Rotor lags shaft by angle proportional to pitch rate, nose down, stable $-\frac{\partial a_{1s}}{\partial q}$	ΔT = 0 ΔH = + (aft, reduces pitch damping) ΔM_H = - (nose down damping moment) $\Delta(M_{CG})$ = - (nose down, stable, damping moment)

Quasi-Static Rotor Characteristics in Forward Flight (cont'd)

	(d) Cyclic pitch, $+\Delta B_{1s}$ nose down	TPP tilts due to cyclic pitch, nose down $-\frac{\partial a_{1s}}{\partial B_{1s}}$	ΔT = - (down force) ΔH = 0, - (low α_{BE}) ΔM_H = - (nose down control moment) $\Delta(M_{CG})$ = - (nose down pitch control moment)
	(e) Collective Increment, $+\Delta\theta_C$	Increased coning. Increased lateral flapping due to increased coning. Increased longitudinal flapping, nose up $+\frac{\partial a_0}{\partial \theta_C}$ $+\frac{\partial b_{1s}}{\partial a_0}$ $+\frac{\partial a_{1s}}{\partial \theta_C}$	ΔT = + ΔH = 0, + (low α_{BE}) ΔM_H = + (nose up) $\Delta(M_{CG})$ = + (nose up pitch moment)

Various aspects of helicopter aerodynamics are discussed academically by considering each component separately, but the ultimate goal is to evaluate the aircraft as a whole. The equations of motion for a helicopter are very intricate. In most cases, the longitudinal axis equations can be separated from the lateral/directional equations of motion. Analysis of these equations in linear form provides some insight to the relative influence of each parameter to the overall stability of the air vehicle. A listing of the important aerodynamic stability derivatives is summarized below. These derivatives are discussed in greater detail in subsequent chapters, but are included here as a convenient reference. The list provides the commonly accepted notation, a definition, the usual sign, and in most cases the associated common name.

Table 4.III
Aerodynamic Stability Derivatives

Notation	Definition	Usual sign	Common Name
X_u	Longitudinal force due to longitudinal velocity	-	Longitudinal velocity damping
X_w	Longitudinal force due to vertical velocity	Usually insignificant	N/A
X_q	Longitudinal force due to pitch rate	+	N/A
$X_{B_{1s}}$	Longitudinal force due to longitudinal cyclic pitch angle	+	N/A
X_{θ_C}	Longitudinal force due to collective pitch angle	Hover, 0 Forward, -	N/A
Y_v	Side force due to lateral velocity	- Auto, +	Side force
Y_p	Side force due to roll rate	- for a high tail	N/A
Y_r	Side force due to yaw rate	+	N/A
$Y_{A_{1s}}$	Side force due to lateral cyclic pitch angle	+	N/A
$Y_{\theta_{TR}}$	Side force due to tail rotor pitch angle	+	N/A
Z_u	Vertical force due to longitudinal velocity	Low speed, - High speed, +	N/A
Z_w	Vertical force due to vertical velocity	-	Vertical damping
$Z_{B_{1s}}$	Vertical force due to longitudinal cyclic pitch angle	Hover, 0 Forward, -	N/A
Z_{θ_C}	Vertical force due to collective pitch angle	-	Collective control sensitivity
L_v	Roll moment due to lateral velocity	Forward, - Auto, +	Dihedral effect or Lateral stability
L_r	Roll moment due to yaw rate	+	N/A
L_p	Roll moment due to roll rate	-	Roll rate damping
$L_{A_{1s}}$	Roll moment due to lateral cyclic pitch angle	+	Lateral control sensitivity

Aerodynamic Stability Derivatives (cont'd)

$L_{\theta_{TR}}$	Roll moment due to tail rotor pitch angle	+	N/A
M_u	Pitch moment due to longitudinal velocity	+	Speed stability
M_w	Pitch moment due to vertical velocity	Negligible	Angle of attack stability
M_q	Pitch moment due to pitch rate	-	Pitch rate damping
$M_{B_{1s}}$	Pitch moment due to longitudinal cyclic pitch angle	-	Longitudinal control sensitivity
M_{θ_C}	Pitch moment due to collective pitch angle	+	N/A
N_v	Yaw moment due to lateral velocity	+	Directional stability
N_p	Yaw moment due to roll rate	+	N/A
N_r	Yaw moment due to yaw rate	-	Yaw rate damping
$N_{A_{1s}}$	Yaw moment due to lateral cyclic pitch angle	-/+	Adverse/proverse yaw, usually adverse
$N_{\theta_{TR}}$	Yaw moment due to tail rotor pitch angle	-	Directional control sensitivity

4.4 GLOSSARY

4.4.1 Notations

a	Lift curve slope
A_0	Collective pitch angle (not a function of ψ)
a_0	Coning angle
A_{0s}	Collective pitch angle, shaft referenced (not a function of ψ)
A_1	Lateral cyclic pitch angle
a_1	Longitudinal flapping angle
A_{1s}	Lateral cyclic pitch angle, shaft referenced
a_{1s}	Longitudinal flapping angle, shaft referenced
ABC	Advancing Blade Concept
B	Damping constant
b	Number of blades
B_1	Longitudinal cyclic pitch angle
b_1	Lateral flapping angle
B_{1s}	Longitudinal cyclic pitch angle, shaft referenced
b_{1s}	Lateral flapping angle, shaft referenced
B_{CRIT}	Critical damping
c	Blade chord
CF	Centrifugal force
CG	Center of gravity
e	Flapping hinge offset
$F_{S\text{ blade on hub}}$	Shear force of the blade on hub
$F_{T\text{ blade on hub}}$	Tangential force of the blade on hub
g	Gravity
H	Rotor hub force, $\perp$ to shaft
h	Height of hub above CG
I	Moment of inertia
I_f	Moment of inertia about flapping hinge
K	Spring constant
L	Net moment about x axis, Roll moment, Lift

$L_{A_{1s}}$	Roll moment due to lateral cyclic pitch angle
L_H	Roll moment due to rotor hub forces
$\overline{L_H}$	Average roll moment due to rotor hub forces for b blades
L_p	Roll moment due to roll rate
$L_{\theta_{TR}}$	Roll moment due to tail rotor pitch angle
L_r	Roll moment due to yaw rate
L_v	Roll moment due to lateral velocity
M	Net moment about y axis, Pitch moment
m	Mass
M_A	Aerodynamic moment
$M_{B_{1s}}$	Pitch moment due to longitudinal cyclic pitch angle
M_{CG}	Moment about CG
M_H	Pitch moment due to rotor hub force
$\overline{M_H}$	Average pitch moment due to rotor hub force for b blades
M_q	Pitch moment due to pitch rate
M_{θ_C}	Pitch moment due to collective pitch angle
M_S	Blade mass moment
M_u	Pitch moment due to longitudinal velocity
M_W	Moment due to weight
M_w	Pitch moment due to vertical velocity
N	Net moment about z axis, Yaw moment
$N_{A_{1s}}$	Yaw moment due to lateral cyclic pitch angle
N_p	Yaw moment due to roll rate
$N_{\theta_{TR}}$	Yaw moment due to tail rotor pitch angle
N_r	Yaw moment due to yaw rate
N_v	Yaw moment due to lateral velocity
p	Angular velocity about x axis, Roll rate
$\dot{p}$	Angular acceleration about x axis
q	Angular velocity about y axis, Pitch rate
$\dot{q}$	Angular acceleration about y axis
R	Rotor radius

r	Angular velocity about z axis, Radius along blade, Yaw rate
r'	Radius along blade outboard of flapping hinge
$\dot{r}$	Angular acceleration about z axis
s	Second
T	Thrust
t	Time
T_b	Thrust of b blades
TPP	Tip path plane
u	Translational velocity component along x axis
u_0	Initial velocity
u_T	Transverse velocity
$\dot{u}$	Linear acceleration along x axis
V	Velocity, Free stream velocity, Relative velocity
v	Translational velocity component along y axis
v_i	Induced velocity
$\dot{v}$	Linear acceleration along y axis
w	Translational velocity component along z axis
$\dot{w}$	Linear acceleration along z axis
X	Resultant force in x direction
x	Orthogonal direction along longitudinal axis of the aircraft
$X_{B_{1s}}$	Longitudinal force due to longitudinal cyclic pitch angle
X_q	Longitudinal force due to pitch rate
X_{θ_C}	Longitudinal force due to collective pitch angle
X_u	Longitudinal force due to longitudinal velocity
X_w	Longitudinal force due to vertical velocity
Y	Resultant force in y direction
y	Orthogonal direction along lateral axis of the aircraft
$Y_{A_{1s}}$	Side force due to lateral cyclic pitch angle
Y_p	Side force due to roll rate
$Y_{\theta_{TR}}$	Side force due to tail rotor pitch angle
Y_r	Side force due to yaw rate
Y_v	Side force due to lateral velocity
Z	Resultant force in z direction

z	Orthogonal direction along vertical axis of the aircraft
$Z_{B_{1s}}$	Vertical force due to longitudinal cyclic pitch angle
Z_{θ_C}	Vertical force due to collective pitch angle
Z_u	Vertical force due to longitudinal velocity
Z_w	Vertical force due to vertical velocity

4.4.2 Greek Symbols

α (alpha)	Angle of attack
α_{BE}	Blade element angle of attack
α_s	Angle of attack, shaft referenced
β (beta)	Flapping angle
$\ddot{\beta}$	Flapping angle acceleration
$\dot{\beta}$	Flapping angle rate
ϕ (phi)	Roll angle
γ (gamma)	Lock number
φ (psi)	Inflow angle
λ (lambda)	Inflow ratio
μ (mu)	Advance ratio, Amplitude ratio
π (pi)	Mathematical constant
θ (theta)	Blade pitch angle, Pitch angle, Feathering angle
θ_C	Collective pitch angle
ρ (rho)	Density
Ω (Omega)	Rotor angular velocity
ω (omega)	Frequency
ω_n	Natural frequency
ψ (psi)	Blade azimuth angle, Phase angle, Yaw angle
ζ (zeta)	Damping ratio, lead-lag angle

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CHAPTER FIVE

FLIGHT CONTROL SYSTEM CHARACTERISTICS

CHAPTER FIVE

FIGURES

CHAPTER FIVE

TABLES

CHAPTER FIVE

FLIGHT CONTROL SYSTEM CHARACTERISTICS

5.1 INTRODUCTION

Since flight control systems are one element in the aircraft control loop, flight control system characteristics are very important to the pilot. The pilot interacts with the flight control system in maneuvering the aircraft to accomplish mission tasks. Pilot activity in the control loop depends on the flight control system characteristics, the aircraft stability and control characteristics, mission tasks, and atmospheric disturbances. Pilot opinion of aircraft flying qualities is influenced by the flight control system characteristics. A pilot working very hard to fly the aircraft is critical of the flight control system characteristics.

Understanding and documenting the flight control system characteristics is fundamental to any flying qualities or closed loop flight evaluation. Since the flight control system is part of the control loop, the test pilot must have a detailed understanding of the components of the flight control system and their function. The understanding begins with a thorough flight control system description and proceeds with documenting the flight control system characteristics.

Helicopter flight control systems are grouped into two general categories, irreversible and reversible. An irreversible control system has integral hydraulic or electro-mechanical power servos. These servos are an irreversible link between the cockpit flight controls and the aerodynamic control surfaces. Reversible control systems incorporate mechanical advantage or dynamic balancing to reduce the cockpit flight control forces without isolating the cockpit flight controls from the aerodynamic control surface loads.

The flight control system characteristics of an irreversible system can be tailored to provide a wide range of force, displacement, and pilot assist characteristics; while the reversible system is not as flexible. The ease of flying a helicopter with a reversible flight control system decreases with increase size and aerodynamic loads on the flight control surfaces. Reversible control systems have not proven satisfactory for helicopters with a gross weight (GW) exceeding about 8,500 lb.

5.2 PURPOSE OF TEST

The purpose of the flight control system characteristics evaluation is to document the control system characteristics in support of stability, control, and flying qualities evaluations, as well as specification compliance.

5.3 THEORY

5.3.1 Control Rigging

Every flight control system has a series of mechanical and electrical control interconnects between the cockpit flight controls and the flight control surfaces. Design specifications set the tolerances to which these controls are rigged. Procedures to adjust the control rigging are provided by the manufacturer and must be used to validate the flight control rigging prior to the flight control system evaluation.

5.3.2 Control Envelope

Each cockpit flight control has a range of motion influenced by the range of pilot motion and cockpit geometry. The control envelope is the range of displacement for each cockpit flight control. The control envelope is measured at the center of the control grip or point of application of control force by the pilot. The control envelope is measured without the trim system or Automatic Fight Control System (AFCS) engaged. The displacement can be limited by cockpit structure or by displacement of another control. The control envelope is a design consideration in combination with control gearing and control power which influence pilot opinion of flying qualities. Figure 5.1 presents a representative cyclic control envelope. In some cases, it may be important to further annotate location of the envelope within the cockpit by reference to a cockpit feature like a point on the instrument panel.

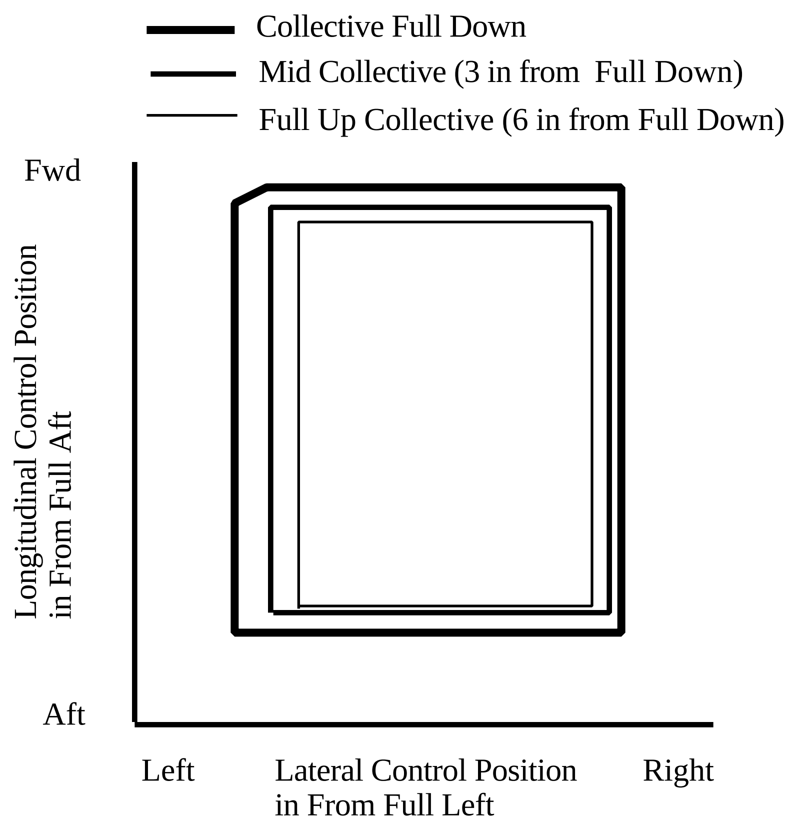


Figure 5.1
Cyclic Control Envelope

5.3.3 Control Mixing

Control mixing is the movement of an off axis control in response to movement of a cockpit flight control. Control mixing can be accomplished through mechanical mixing or through the AFCS (electrical mixing). Some aircraft incorporate both mechanical and electrical mixing. Control mixing is incorporated to correct for or mask aerodynamic mixing and produce an uncoupled control response. Control mixing can result in the movement of off axes cockpit flight controls or flight control surfaces. Control mixing can produce restrictions on the cockpit flight control envelope. Figure 5.1 presents a representative collective to cyclic control mixing at three collective positions and the restriction on the cyclic control envelope.

5.4 IRREVERSIBLE CONTROL SYSTEMS

Irreversible systems isolate the aerodynamic loads on the flight control surfaces from the cockpit flight controls. In such systems, there is no aerodynamically supplied force when the cockpit flight control is moved from one position to another. The absence of control force produces two problems. The first problem is the inability to release the control and perform other cockpit duties without the control moving due to the force of gravity or vibratory loads. The second problem is the absence of a cockpit flight control force cue which the pilot uses to sense displacement of the control from trim. A requirement exists to provide artificial cockpit flight control retention and force cues.

5.4.1 Trim Systems

Trim systems are incorporated into the flight control system to overcome the two problems of an irreversible control system. Trim systems provide for cockpit flight control centering, trim retention, and provide forces cues to control displacement from trim.

5.4.1.1 FRICTION SYSTEMS

Adjustable control friction is a simple flight control system trim device. Such devices normally have an adjustment through which the coefficient of sliding friction is varied, thereby adjusting the friction force in the flight control system. An example is the friction collar on the UH-60A collective control. Friction can be adjusted so that friction force is equal to or greater than any force in the control system tending to move the control from trim. For example, with all friction off, the pilot exerts a 4 lb lift force on the collective control to balance the gravity force acting on the control. If the collective control is released, the collective moves downward due to the gravity force. If the pilot increases the collective friction to 4 lb, the collective does not move when it is released. To raise the collective the pilot must exert 8 lb of lift force. To lower the collective, a few ounces of downward force is required. Precise control inputs are difficult with high friction forces. For this reason, there is a practical limit to the application of a variable friction trim device.

5.4.1.2 ELECTRO-MECHANICAL SYSTEMS

The spring and clutch combination is representative of mechanical trim systems which provide an artificial force gradient, control centering, and trim capability. A simplified system is presented in Figure 5.2. Zero force is exerted on the control in Figure 5.2 (a) at the initial trim condition. The control is displaced from trim in Figure 5.2 (b) and the spring is extended, exerting a force opposing the control displacement. In Figure 5.2 (c), the pilot activated a momentary trim release, disengaging the clutch and producing a sudden reduction of the opposing force. The pilot can fly with zero force (momentary trim release activated) or fly against a force, trimming as necessary.

Most trim systems incorporate switches to activate/deactivate and momentarily disengage the system. This capability is provided in case of malfunction or situations when the pilot does not desire to fly against a force. Current irreversible control helicopter trim systems use electro-mechanical or hydraulic servos to provide the force gradient and centering capabilities. An example is the SH-60 aircraft which uses electro-mechanical servos in the yaw and collective axis, and hydraulic servos in the pitch and roll axis. In all axes, the trim system is controlled through the AFCS system. The pilot must know what type of system is producing the force and trim characteristics to design and apply effective test procedures and document deficiencies.

5.4.1.3 FORCE VERSUS DISPLACEMENT

The trim system provides control centering and force cues for displacement from the trim position. The force versus displacement of the cockpit flight control is measured at the center of the cockpit flight control grip or the point of application of control force. Figure 5.3 presents a representative force versus control displacement curve. The helicopter force versus displacement characteristics should be approximately linear, without discontinuities or reversals, and symmetrical for displacement in both directions. The forces should be accommodated to the control axis and physiological capabilities of the pilot population. For a conventional center cyclic control, the forces in the longitudinal axis should be harmonious with those in the lateral cyclic axis. Additionally, for a multi-axis controller, the forces should be harmonious in all control axes and directions.

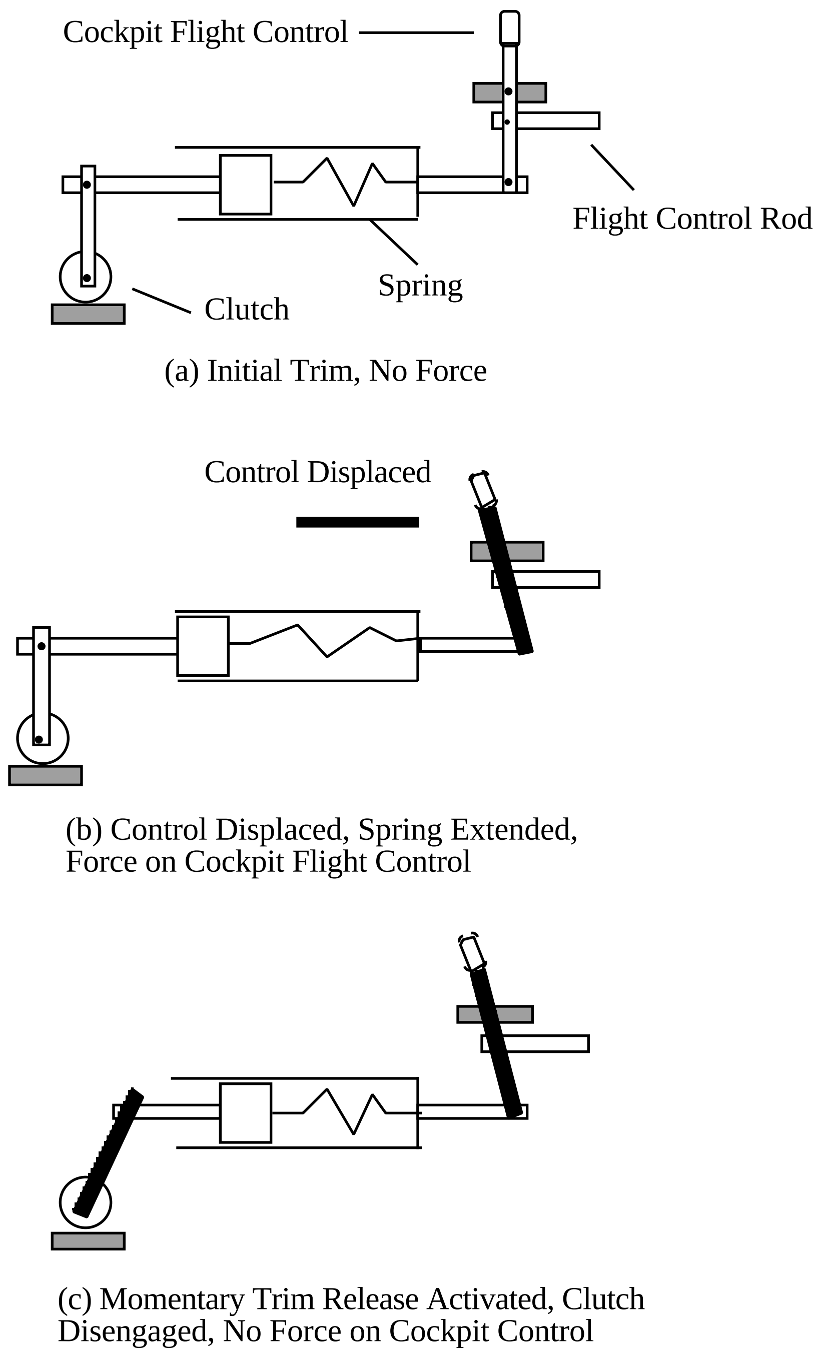


Figure 5.2
Representative Mechanical Trim System

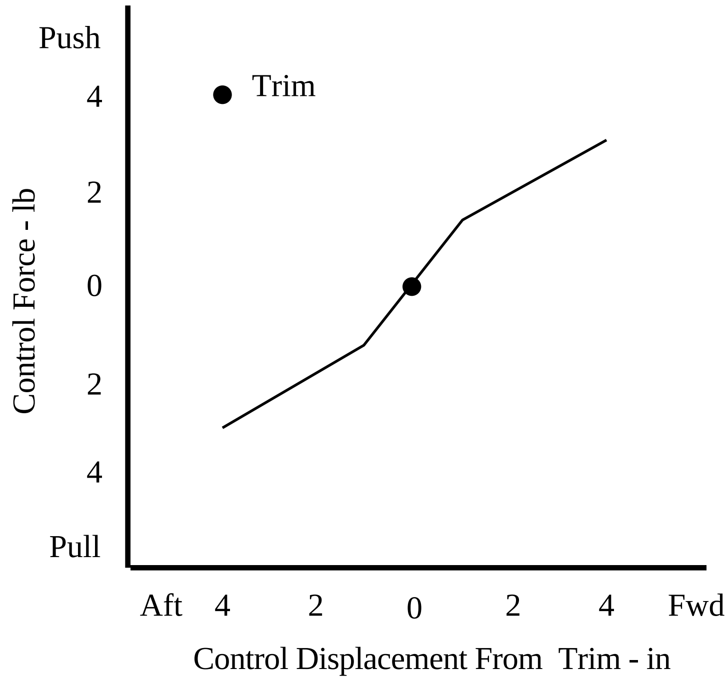


Figure 5.3
Control Characteristics with Trim System

5.4.1.4 FORCE GRADIENT

The force gradient is the average force per unit control displacement. The force gradient is calculated from the force versus displacement curve. Typically, the gradient for the first inch of travel from trim should be greater than the friction force and the gradient for the remaining control displacement. A representative linear force gradient is presented in Figure 5.3.

5.4.1.5 LIMIT CONTROL FORCE

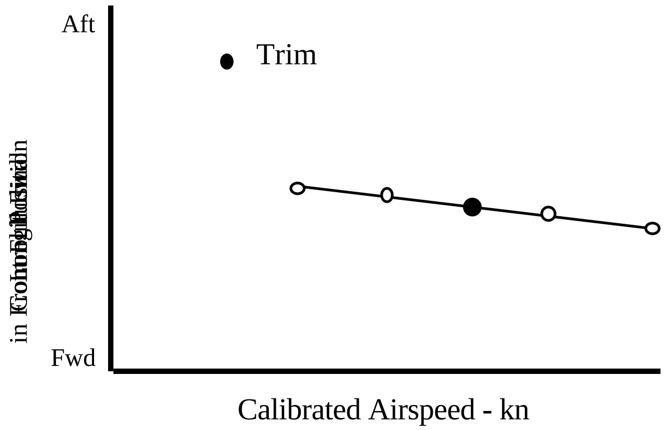
The maximum force required to hold the cockpit flight control in position displaced off trim is the limit control force. The limit force can be determined for any control range, not just the extremes of control travel. Correctly done, limit force would be determined after the aircraft was flown in most all, if not part, of the flight envelope to determine the approximate control throw. Then, measure the force required to move the control over the range expected and call it limit force. Obviously, full throw would be the extreme. For irreversible controls then, limit force measured stop to stop might be the most appropriate measure, especially when flight opportunities are limited. The graphical depiction most often presented for the Mil-H-8501A shows limit force as measured from the center of the control throw, although the description in paragraph 3.2.6 of the spec does not require measurement from center. When discussing limit force, it would be very important to cover the range measured. For spec compliance, the general agreement would be to use the forces measured from the center of control range out to each control throw extreme. A discussion of limit force for a control trimmed to one extreme of travel and force measured to the other extreme should be included, but not related, to specification requirements.

5.4.1.6 BEEPER TRIM SYSTEM

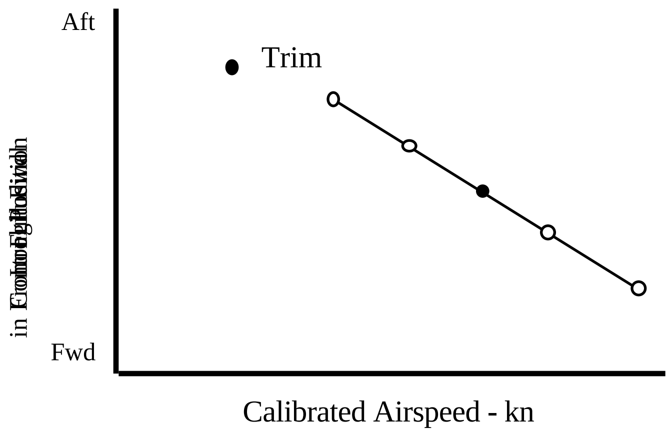
A beeper trim capability can be added to a trim system to provide vernier force adjustments. These systems are operated in two ways. One, to trim out control forces opposed by the pilot; or two, to move the cockpit flight control. The control switch for a beeper trim system should be located on the flight control being trimmed and oriented in the direction of control displacement. A beeper trim for pedals would be most conveniently mounted on the collective. The beeper trim system may have the capability of selecting single or multiple speeds. The beeper trim system should provide precise selection of control position as well as rapidly adjusting the control position and force required for maneuvering flight.

5.4.1.7 BEEPER TRIM RATES

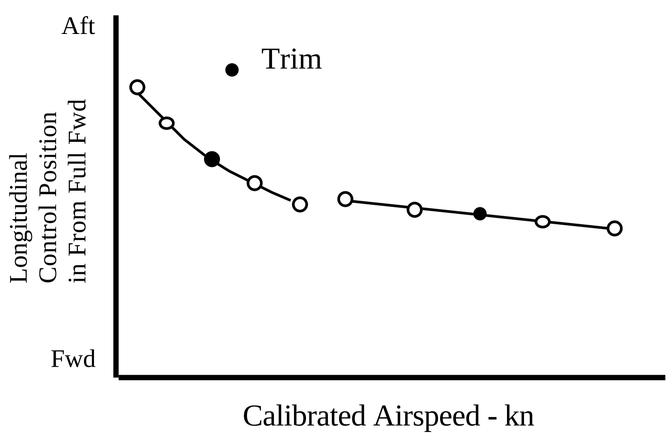
The beeper trim rate is the cockpit flight control displacement per unit engagement time of the beeper trim switch. The trim rate is measured at the cockpit flight control. Satisfactory beeper trim rates for the longitudinal, lateral, and directional controls depend in part on the slope of the control position gradients. Figure 5.4 (a) illustrates an aircraft with nearly neutral static stability as indicated by the shallow control position versus airspeed gradient. The beeper trim system in an aircraft with a shallow control position gradient must be capable of small, precise control displacements. A slight error in trim control position results in a large error in trim speed and the pilot must continuously retrim. Figure 5.4 (b) depicts a steep control position gradient. This type of gradient requires the beeper trim to operate at a fast rate to achieve modest airspeed changes in a satisfactory time period. Figure 5.4 (c) illustrates both near neutral and steep control position gradients in one aircraft. Slow speed maneuvering in this aircraft uses a momentary trim release system or one with a high beeper trim rate. Trimming at high speed is accomplished with a slow trim rate. Selection of the proper trim system in such an aircraft is not easy. A slow trim rate used in conjunction with a shallow control position gradient may give the illusion of



(a) Shallow Control Position Gradient



(b) Steep Control Position Gradient



(c) Steep and Shallow Control Position Gradients

Figure 5.4
Static Longitudinal Stability

increased positive static stability. This occurs because the pilot senses the time he is activating the beeper trim and the change in the control force or control position. Force or position is integrated with respect to time to produce a control cue similar to the cue provided by a steeper control position gradient.

5.4.1.8 TRIM LAG

Trim lag is the time delay between activation of the beeper trim switch and movement of the cockpit flight control. Trim lag is problematic when it is large enough to produce unpredictable results in response to beeper trim activation. Lags on the order of 0.3 to 0.5 s are generally unacceptable. If a pilot momentarily activates the beeper trim switch attempting to make a fine trim adjustment and there is no response in the trim system, the next input will be too long and will overshoot the desired displacement. Trim lag is particularly bothersome during Instrument Meteorological Condition (IMC) operations and is generally undesirable for all precise trim operations.

5.4.1.9 TRIM ENVELOPE

The trim envelope is the maximum control displacement wherein the cockpit flight control forces can be reduced to zero using the trim system. The control envelope determined using the momentary trim release is the trim release envelope. The control envelope determined using the beeper trim is the beeper trim envelope. The two trim envelopes may not be the same. Further, the trim envelopes may not be the same as the control envelope without the trim system engaged. Although most trim systems are capable of trimming forces to zero from one flight control stop to the other, there are some systems which have limits and do not provide a stop to stop trimming capability. In some cases, the trim limit is due to a design deficiency; however, in other situations the trim limit is purposely designed into the control system. A restricted trim envelope can be used to warn the pilot he is approaching the control displacement limit. Some aircraft use this technique to prevent the collective control from being trimmed full down. This low limit reminds the pilot not to operate on the ground with full down collective which helps to prevent droop stop pounding. Figure 5.5 presents a representative cyclic control trim envelope.

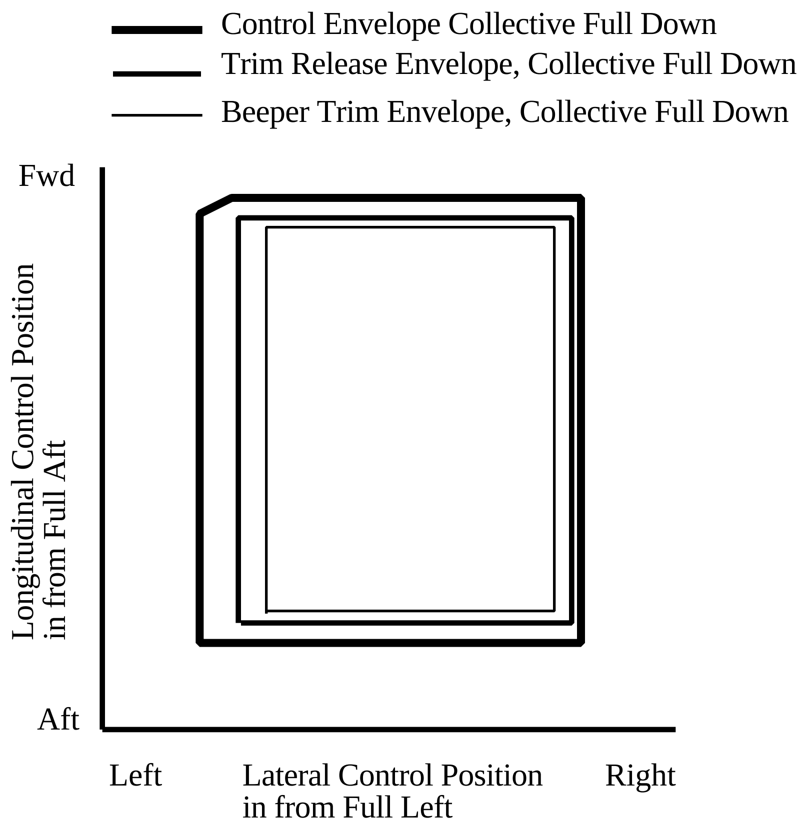


Figure 5.5
Cyclic Control Trim Envelope

5.4.1.10 CONTROL JUMP

Control jump is the undesired movement of the cockpit flight control that occurs when the momentary trim release is activated while holding a force against the trim system. Control jump is a function of displacement, force gradient, friction band, viscous dampers, and trim release characteristics. If upon activating the trim release, the trim force drops to zero faster than the pilot can relax his applied force, a force imbalance exists for a fraction of a second. The force imbalance results in undesired motion of the cockpit flight control. Although light forces do not produce significant problems, pre-trim forces of 3 lb or more result in small, rapid movement of the cockpit flight control. The end result is the control is displaced from the desired trim position, requiring retrimming the undesired motion.

Control jump can be avoided by activating the momentary trim release before making a control displacement and releasing the momentary trim when the new trim position is established. Control jump can also be avoided by depressing the momentary trim release continuously or deactivating the trim system. Either of these techniques to avoid control jump eliminates control centering and force displacement cues.

5.4.1.11 TRIM SYSTEM FREEPLAY

Trim system freeplay is the movement of the cockpit flight control without corresponding control force. Said another way, trim system freeplay is the lost motion between the cockpit flight control and the trim system. Trim system freeplay is independent of total system freeplay discussed in paragraph 5.4.10. Trim system freeplay is also known as force freeplay or force hysteresis and is illustrated in Figure 5.6.

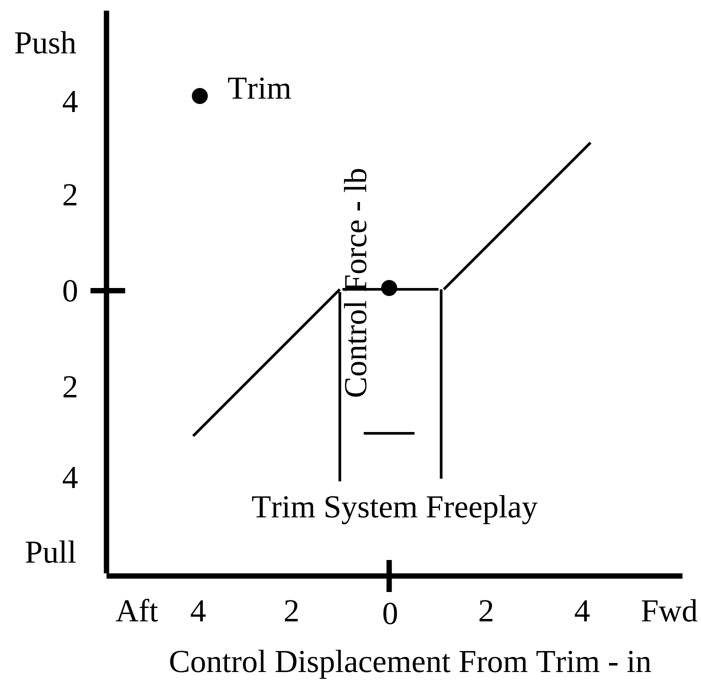


Figure 5.6
Control Characteristics with Trim System Freeplay

5.4.2 Friction

Friction as discussed here, is the built-in, net pilot adjustable friction force opposing movement of the cockpit flight control. Friction is composed of static and sliding friction. Static friction is the force opposing initial movement, and sliding friction is the force opposing continued movement. Most flight control systems possess some degree of friction; although it may be small. Friction comes from a number of sources, and in all cases resists control motion. The result of friction in the flight control system is illustrated in Figure 5.7. As a push force is applied to the flight control there is no motion until the 1 lb friction force is overcome. A 4 lb force is required to move the flight control 4 in forward. This force is equal to the sum of the 3 lb force provided by the feel spring and the 1 lb friction force opposing forward movement. If at 4 in forward the push force is gradually released, the control remains at 4 in until the force drops below 2 lb. At the point where the push force equals 2 lb, the feel spring is pulling on the control with a 3 lb force, the pilot is exerting a 2 lb force, and friction is supplying the remaining 1 lb force for equilibrium. A continued reduction of push force results in the control returning toward the initial trim condition. At a point 1.4 in forward of the initial trim position (Figure 5.7, Point A), friction equals the force provided by the trim system and the pilot no longer furnishes any force to achieve static equilibrium. A slight pull force is required to return the control to trim.

5.4.3 Trim Control Displacement Band

The trim control displacement band (TCDB) is the control displacement, about trim, within which the feel spring force is equal to or less than the friction force. Within the TCDB, the control has neutral control centering characteristics. A pilot has no force gradient cue to the location of trim within this band. The TCDB is caused by friction as illustrated in Figure 5.7. The seriousness of a TCDB is influenced by the static stability of the aircraft as indicated by the control position versus airspeed curve. In cases where small out of trim conditions produce large airspeed variations (shallow control position gradient), the problem is serious.

Aircraft which operate in IMC generally require flight control characteristics without a TCDB. Visual Meteorological Condition (VMC) missions are not as stringent and a small TCDB may be accepted. Interestingly, a TCDB may be desirable to a certain extent during VMC hovering missions when the pilot makes numerous small corrections about trim. In a precision hover with light atmospheric turbulence, a TCDB negates the requirement to trim out forces each time the control are moved. This occurs because the force falls to zero as soon as the displacement within the TCDB is complete, even if the new momentary trim is not identical to the last.

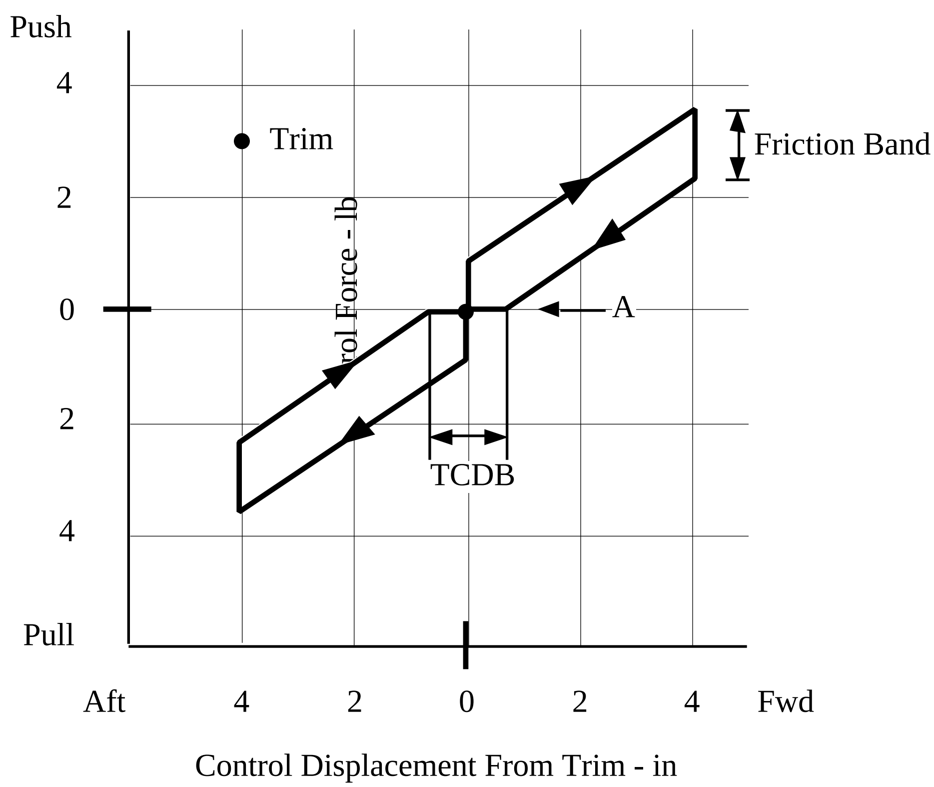


Figure 5.7
Control Characteristics with Friction

5.4.4 Breakout

The TCDB can be reduced by decreasing the friction force gradient or by introducing a breakout force equal to or greater than the friction force. Breakout force is the preload spring force in the force feel system. Figure 5.8 presents an example of control force characteristics with a 1.5 lb breakout. The control system is shown with and without friction force for clarity. The control force does not return to zero without the control returning to the initial trim condition.

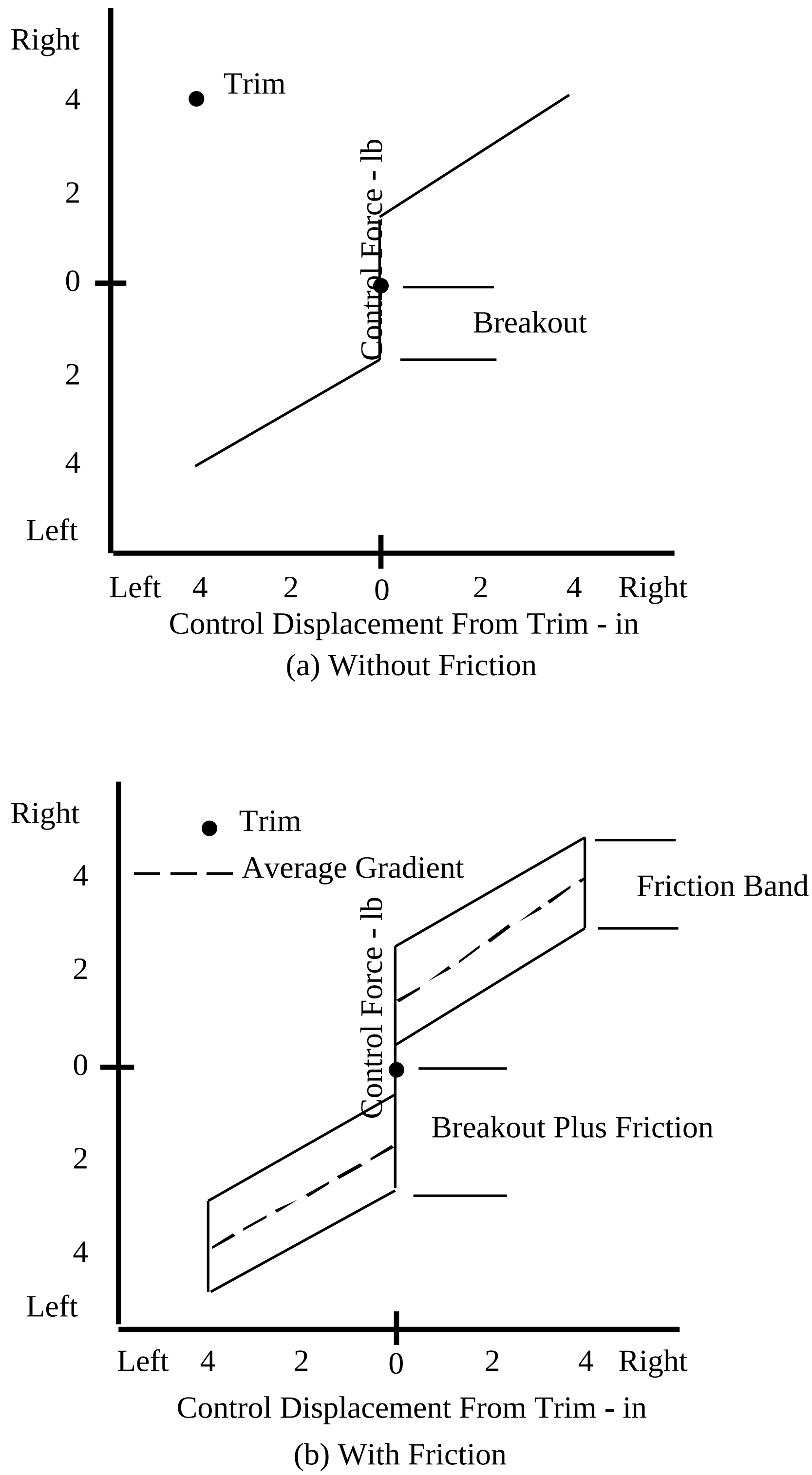


Figure 5.8
Control Characteristics with Breakout

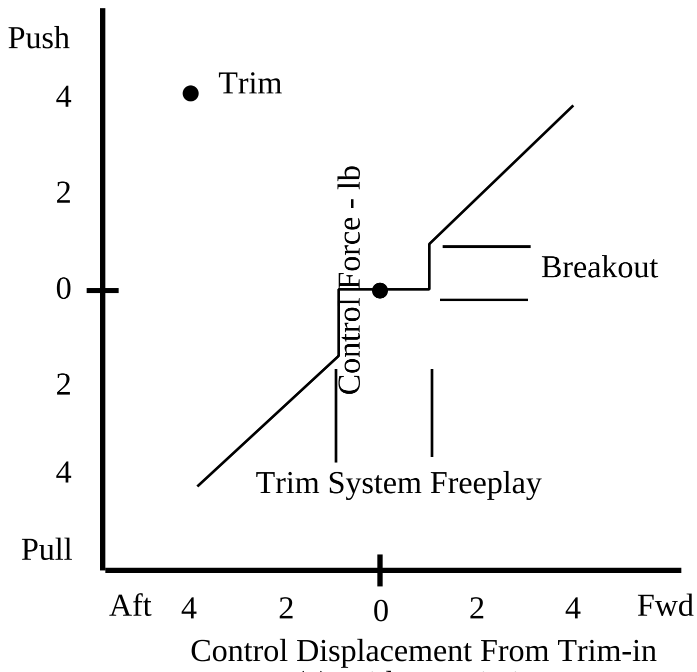
Large breakout forces introduce problems during maneuvering flight such as a roll reversal. This problem is illustrated by considering the system presented in Figure 5.8 (a). Assume a 1 in input is used to initiate a roll to the right, then the control is reversed to 1 in left to stop the right roll and commence a left roll. Assume as the control is reversed, the pilot tries to move the control at a constant rate from right to left. As the control moves through breakout, force quickly drops to zero, then rapidly increases in the other direction. A notch would be felt and the arm muscle does not readily accommodate to this force discontinuity in cockpit flight control motion. The discontinuity is much the same as produced by control jump.

5.4.5 Breakout Plus Friction

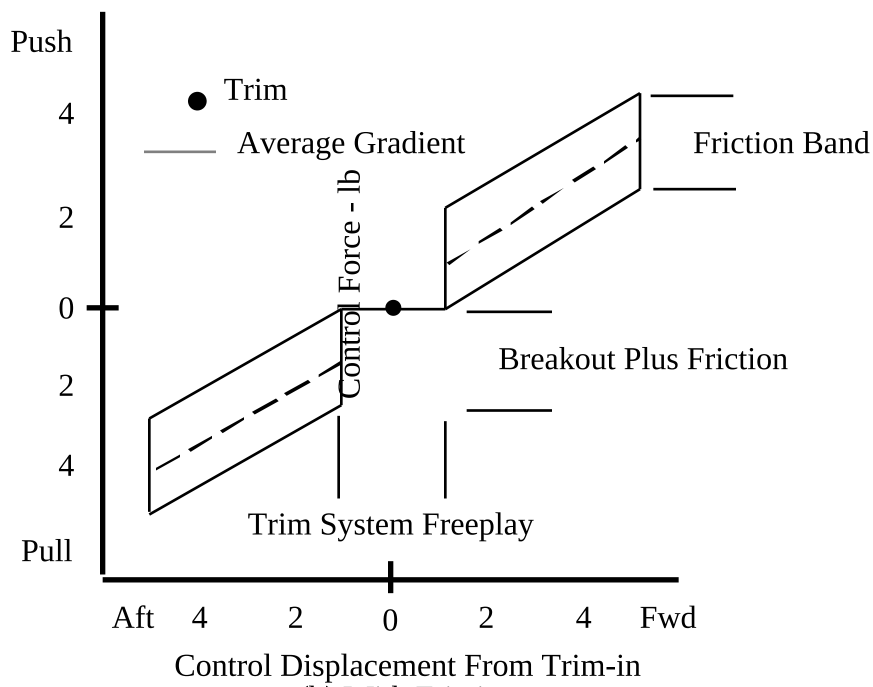
Both trim system freeplay and breakout can exist in the flight control system. Figure 5.9 (a) presents representative flight control system characteristics with trim system freeplay and breakout, but without friction. Figure 5.9 (b) presents representative flight control system characteristics with trim system freeplay and breakout plus friction. In this example, friction equals breakout.

The breakout plus friction force is the sum of the forces required to initiate control movement (breakout and friction). Figure 5.8 (b) illustrates a flight control system with breakout plus friction. The breakout plus friction force is 2.5 lb for both push and pull. For the smallest control displacement about trim, the pilot is required to operate across a five lb force differential. This is true even though the force gradient is a light 0.8 lb/in and alone requires less than a 0.4 lb differential force for a ± 1/4 in control displacement about trim.

Random cockpit flight control motions are required around trim for precise hovering in light gusty winds. This motion characteristic is present during control of lightly damped aircraft. The force differential characteristic associated with breakout plus friction becomes important when it is difficult to achieve smooth, low frequency cockpit flight control motions.



(a) Without Friction



(b) With Friction

Figure 5.9
Control Characteristics with Trim System Freeplay and Breakout

5.4.6 Control Mass Imbalance

Control mass imbalance can produce discontinuities in the breakout plus friction force. This problem normally is related to light breakout plus friction, light force gradient, heavy control stick, forward cyclic displacement, and nose down aircraft attitude. In some flight situations, the longitudinal flight control can be inclined up to 20˚ with respect to the earth. The inclined control mass, acted upon by gravity, produces a moment about the control pivot point. The moment produces an equivalent force at the pilot's grip. Equivalent forces of more than 1 lb are not uncommon. The force due to control stick mass tends to overcome the push breakout plus friction force, while it adds to the initial pull breakout plus friction force. There are cases where the longitudinal flight control moved forward after being trimmed and released for hands off flight. This control motion produced a nose down longitudinal divergence when the aircraft actually had a moderately damped oscillatory long term characteristic.

5.4.7 Control Force Coupling

Control force coupling is the force introduced in one cockpit flight control or axis when a second cockpit flight control is moved. Control force coupling can be intentional or unintentional. Force coupling is intentionally designed into an aircraft to offset a severe trim change or some other unusual characteristic. In one irreversible system, an unintentional control force characteristic was introduced in the longitudinal control system when the collective was raised or lowered, as a result of a control force per g augmentation system, providing the pilot with false force cues.

5.4.8 Control Centering

Many flight control system characteristics prevent the cockpit flight control from returning to a trim condition. The motion of the cockpit flight control following a displacement from trim is referred to as control centering. If the control moves unassisted towards its initial trim position, the control has positive control centering. If the control returns precisely to trim, the control has absolute control centering. If the control moves away from trim, the control has negative control centering. If the control does not move following the displacement from trim, the control has neutral control centering.

5.4.9 Control Dynamics

The motion of the cockpit flight control when dynamically disturbed from trim is referred to as control dynamics. The disturbance can be a displacement and sudden release, or a control rap. Control dynamics are important if the pilot inadvertently releases or raps the control. If the control dynamics are not well dampened, the resultant motion of the cockpit flight control could impart undesired aircraft motion.

5.4.10 Total System Freeplay

Total system freeplay is the movement of the cockpit flight control without movement of the corresponding flight control surface. Total system freeplay is the lost motion between the cockpit flight control and the control surface. Total system freeplay is also referred to as position freeplay or position hysteresis. Freeplay comes from several sources including: hydraulic system friction, sloppy control linkages, and control washouts due to AFCS inputs. If control freeplay is ± 1/4 in or more, the pilot can experience controllability problems. Total system freeplay is independent of trim system freeplay.

5.4.11 Viscous Dampers

Cockpit flight control rate damping is used to restrict the rate of cockpit flight control movement by incorporating viscous dampers in the control system. Viscous dampers are incorporated to prevent pilot induced oscillation (PIO), or otherwise prevent unpleasant characteristics related to rate of cockpit control movement. For example, a collective viscous damper was used in an aircraft because of a serious collective to pitch attitude dynamic coupling problem. In another situation, a damper was installed in the lateral cyclic control system of a tandem rotor helicopter to prevent pilot induced excitation of a lateral bending mode. A third common application is found in the directional control system of helicopters equipped with an antitorque tail rotor. The damper in this installation is intended to prevent unintentional overstressing of the tail boom.

The flight control system characteristics produced by viscous dampers can enhance or detract from flying qualities depending on the combined effects of the damper characteristics and the force gradient system. Rate damping gives the pilot a feel for how fast he is moving the control, and tends to take some of the jerkiness out of the control motion. When added to a system which has light breakout plus friction force and a light force gradient, a damper can provide an acceptable control system. Adding a rate damper to already high forces can be unacceptable. Rate damping of a control can reduce the adverse effects of control jump. Figure 5.10 illustrates two of many possible viscous damper installations. One parallel damper is connected directly to the flight control while the other damper is connected in series with the clutch and spring.

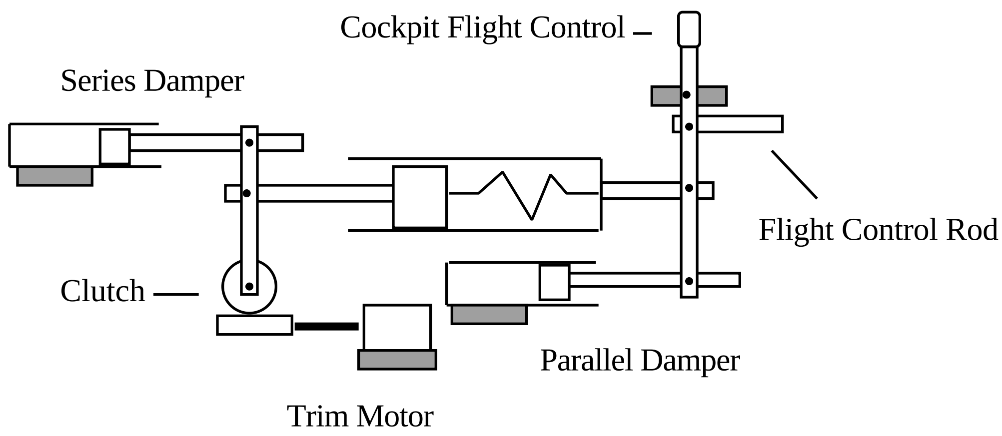


Figure 5.10
Parallel and Series Dampers

5.4.11.1 PARALLEL DAMPER

In a system with a parallel damper, the damper provides a control force when the control is moved. Depending upon damper design, the force produced by the damper can be so small it cannot be detected by the pilot unless high control rates are used. Normally the damper provides a force which is proportional to the rate of motion, but a non-linear system can be incorporated.

5.4.11.2 SERIES DAMPER

The damper shown in series with the force gradient spring is not effective with the clutch engaged. With the clutch engaged, motion of the control is resisted by the parallel damper. If the clutch is released with the spring under tension (or compression), the force does not immediately drop to zero, but decays as a function of the series damper design and control displacement. The time required for the force to drop to zero affords the pilot time to relax the force he is applying to the control. Ideally, the force being exerted by the pilot and the trim system drop to zero simultaneously, eliminating control jump.

5.4.12 Transient Control Forces

Transient control forces are the forces developed in a control system as the result of control velocity or some auxiliary control system force input. Transient control forces associated with rate of control motion are, in general, the result of viscous damper action, friction, or a servo system which cannot produce the rate or motion demanded by the pilot. The servo problem is not uncommon and can be identified by the sharpness of the transient force when it occurs. The hydraulic servo is limited by design and flow rate to support a maximum flight load and rate of control motion. A servo stalls or squeals when its capacity is exceeded due to maximum load or rate of motion, severely restricting aircraft control.

5.4.13 AFCS Inputs

Airspeed, sideslip, altitude, and normal acceleration among other flight parameters can be measured in flight and used to provide flight control system cues to the pilot. One application provides the pilot with increased maneuvering stability (control displacement and force per g), or in the case of airspeed, allow control free constant airspeed flight. The signals from the respective sensors are fed to a trim motor through the AFCS. For example, if an aircraft is trimmed at some airspeed then displaced by a gust into a nose down attitude, an airspeed transducer senses an increase in speed as the aircraft accelerates. The speed signal is fed to a trim motor to move the longitudinal control aft producing a corresponding pitch up of the aircraft. If the pilot is holding the control and does not let it move, he exerts a push force which is a force cue to an off trim condition. Depending upon sensitivity, the sensors may transmit random forces into the flight control system in response to gust or dynamic and static pressure changes. Such random forces are annoying to the pilot. Longitudinal control force per g may be troublesome with respect to random forces produced as a result of the basic aircraft gust response. If the random or transient forces are bothersome over a long term, the pilot becomes insensitive to the control force characteristics. Other essential longitudinal force cues may be disregarded or downgraded in the process.

5.5 REVERSIBLE CONTROL SYSTEMS

5.5.1 General

Reversible control systems are used when the flight control loads are low, and mechanical gearing is sufficient to obtain the desired control force characteristics. The reversible system is attractive because of low maintainability and high reliability. Some aircraft with irreversible systems provide an emergency capability to fly with a degraded system.

The reversible control system provides for direct control of the aerodynamic flight control surfaces. This system may incorporate a simple spring trim system to assist in reducing control forces once the control is fixed at a trim position. A large friction force may be required to assist in holding the control against the flight loads. The combined force with the friction inherent in a mechanical control system, often makes flight maneuvers difficult to perform precisely. If the aircraft has weak positive or negative static stability, the pilot workload may become excessive in light turbulence. There is no reason to expect any of the force cues necessary for IMC flight operations. Therefore, do not expect a reversible control system to provide anything more than an emergency IMC capability. An exceptionally stable and responsive aircraft may possess some IMC transit or cruise capability under ideal conditions with good flight instruments.

The forces experienced in a reversible control system are difficult to measure repeatedly in flight because of the nature of their source. Most of these forces result from blade pitching moments transmitted into the flight control system. These forces vary with flight conditions. Measurement of the control characteristics, such as position versus force, are difficult to attain qualitatively. A ground evaluation is also inappropriate for characteristics influenced by aerodynamic loads since the flight condition is not relateable. In some cases, aerodynamics balances, mechanical control balances, or springs are used to reduce flight loads transmitted to the cockpit flight controls. These artifacts are designed to minimize the forces transmitted from the flight control surface to the cockpit flight control for a given flight condition. Such counterbalances are usually insufficient during operations on one side of the design flight condition; and overcorrect during operations on the other side of the design flight condition. Thus control forces exist during off design operation.

Vibratory feedback from the rotor system may cause the cockpit flight controls to move. Such motion may be reduced in flight by increasing control friction or changing the operating conditions. Neither of these solutions is satisfactory.

5.5.2 Degraded Flight Control System Operation

Helicopters normally employ double or triple redundancy in flight control system augmentation. Primary hydraulic servos should be double redundant; and hydraulic pressure and supply systems should be at least double redundant. Augmented flight control systems which incorporate AFCS inputs should provide double redundant sensors and processors as well as error checking routines. Additionally, many flight control systems separate the pilot assist and boost functions from the primary flight control servos to increase reliability.

However reliable the flight control system, failures can occur. Some helicopters with irreversible flight control systems cannot be flown with a total failure of the flight control hydraulic boost system. However, most helicopters can be flown with a partial failure. When an irreversible flight control system is operated with a partial failure, the pilot generally changes his mode of control manipulations. This is because he has lost the advantages normally afforded by the flight control system, and acquired one with standing control forces, high friction, control force coupling, control freeplay, and other undesirable characteristics.

The ability of a pilot to transition into or accommodate a degraded flight control system operation is extremely important. Transition time and success attained after transition is related to how much the control characteristics differ with the degraded flight control system. When the change is severe, the result will be correspondingly poor unless the pilot has gained proficiency in flying the aircraft in a degraded mode.

In many cases, the problem of controlling the aircraft after a servo system failure is compounded by the simultaneous loss of stability and control augmentation. This compound failure can be more serious than a boost only failure. When there is a serious degradation of aircraft stability due to a servo system failure, it becomes important to have acceptable unaugmented flight control system characteristics.

5.6 TEST METHODS AND TECHNIQUES

5.6.1 General

Prior to any flight control system characteristics, flying qualities, or stability and control evaluation, the flight control system rigging must be verified and documented in accordance with the published maintenance practices for the aircraft. In addition to the primary flight control rigging check, auxiliary or secondary flight controls such as mechanically or automatically movable horizontal tails must be checked. The checks may involve inducing artificial signals to various sensors such as airspeed, normal or lateral acceleration, and turn rate.

The evaluation of flight control system characteristics involve measuring forces, displacements, and time. Typical flight test instrumentation used for stability and control evaluations provide cockpit control positions. Some instrumentation packages may include cockpit control force data as well. The instrumentation package may provide cockpit as well as recorded data, and may be capable of recording static as well as dynamic data.

Cockpit flight control position can be recorded and displayed as units of displacement or percent of total displacement. In either case, the control position data must be calibrated to displacement at the center of the cockpit flight control grip or point of application of pilot control force. Similarly, the control force data must be calibrated to the point of application of pilot control force.

If the cockpit flight control position and force data are manually recorded, the position data are measured along the arc described by the control grip, and the force must be applied normal to the control motion. If the total control displacement is small, the displacement along the arc can be approximated by the linear displacement.

Although several automatic devices are available for measuring cockpit flight control force and displacement on the ground, the following discussion assumes the use of a hand held force gauge and instrumented control position data.

The irreversible flight control system is quantitatively tested on the ground, and qualitatively evaluated in flight. The impact of flight control system characteristics on flying qualities is evaluated in flight. If appropriate, additional quantitative data may be obtained to support any deficiencies observed in flight.

Reversible flight control systems are evaluated primarily in flight. Some characteristics which do not depend on the effect of aerodynamic loads can be evaluated on the ground. Knowledge of the flight control system determines whether on deck or in flight evaluations are appropriate.

If the flight control system can be powered by external hydraulic and electrical power, ground tests of irreversible flight control systems are conducted without the engines or rotors turning. Another option for powering the flight control system is an on board auxiliary power unit (APU). If the APU can provide full hydraulic and electrical power, it may be used for ground tests. The least desirable method of powering the flight control system for ground tests is through turning the engines and rotor system. Turning engines and rotors restrict the control displacement due to droop stop pounding, rotor-fuselage interference, aircraft loads, or aircraft roll over. In all cases, the ground evaluation is conducted with the rotors spread. No matter what method is used to power the flight control system, all applicable restrictions such as ground operating time or temperature must be observed.

5.6.2 Ground Testing

5.6.2.1 CONTROL ENVELOPE AND CONTROL MIXING

The control envelope of each cockpit flight control is determined without the trim system activated. The control envelope for one control is determined with the other controls centered or at minimum displacement. If control mixing is present, the control envelope is determined with the off axis control displaced at several locations throughout its available range. Control envelope and control mixing data are presented in Figure 5.1.

5.6.2.1.1 Data Required

Cockpit flight control positions.

5.6.2.2 FORCE VERSUS DISPLACEMENT

The force versus displacement characteristics for the flight control trim system are determined from a representative trim position. The cockpit flight control is positioned at a representative trim position, or approximate center of the available control envelope by use of the momentary trim release. The force gauge is applied perpendicular to the displacement of the control grip. The force is applied at the center of the grip or at the point of pilot force application. The force is increased slowly, recording the force and displacement to the edge of the control envelope. At the edge of the envelope the force is slowly relaxed, recording the force and displacement as the control returns toward the trim position. The process is repeated in the opposite direction.

Several important points must be made. As the force is first applied, the force at which the control first moves is the breakout plus friction force. The difference in the force between the force at the edge of the control envelope and the force at which the control begins to move toward trim is the friction band or twice the friction force. If the control does not return to trim with zero force applied, apply a force in the opposite direction to return the control to trim. Always apply force in one direction as the control is moved away from or toward trim. If you allow the control to move in the opposite direction, you may contaminate the data since you are operating through the friction band.

Limit control force is determined from centered control to each extreme for specification compliance. If any discontinuities are noted in the force gradient, the limit force is determined for the discontinuities. Limit control force data are presented in narrative and tabular format (Table 5.I). The limit control force is also determined by trimming the control to one extreme of travel and measuring the force after the control is displaced to the other extreme of travel.

Representative force versus displacement data are presented in Figure 5.11. Annotate the breakout plus friction, the friction band at the edge of the control envelope, the average gradient for the first inch of travel, the average gradient for the remainder of travel, the TCDB, specification limits, freeplay, and total control travel.

5.6.2.2.1 Data Required

Cockpit flight control position and force.

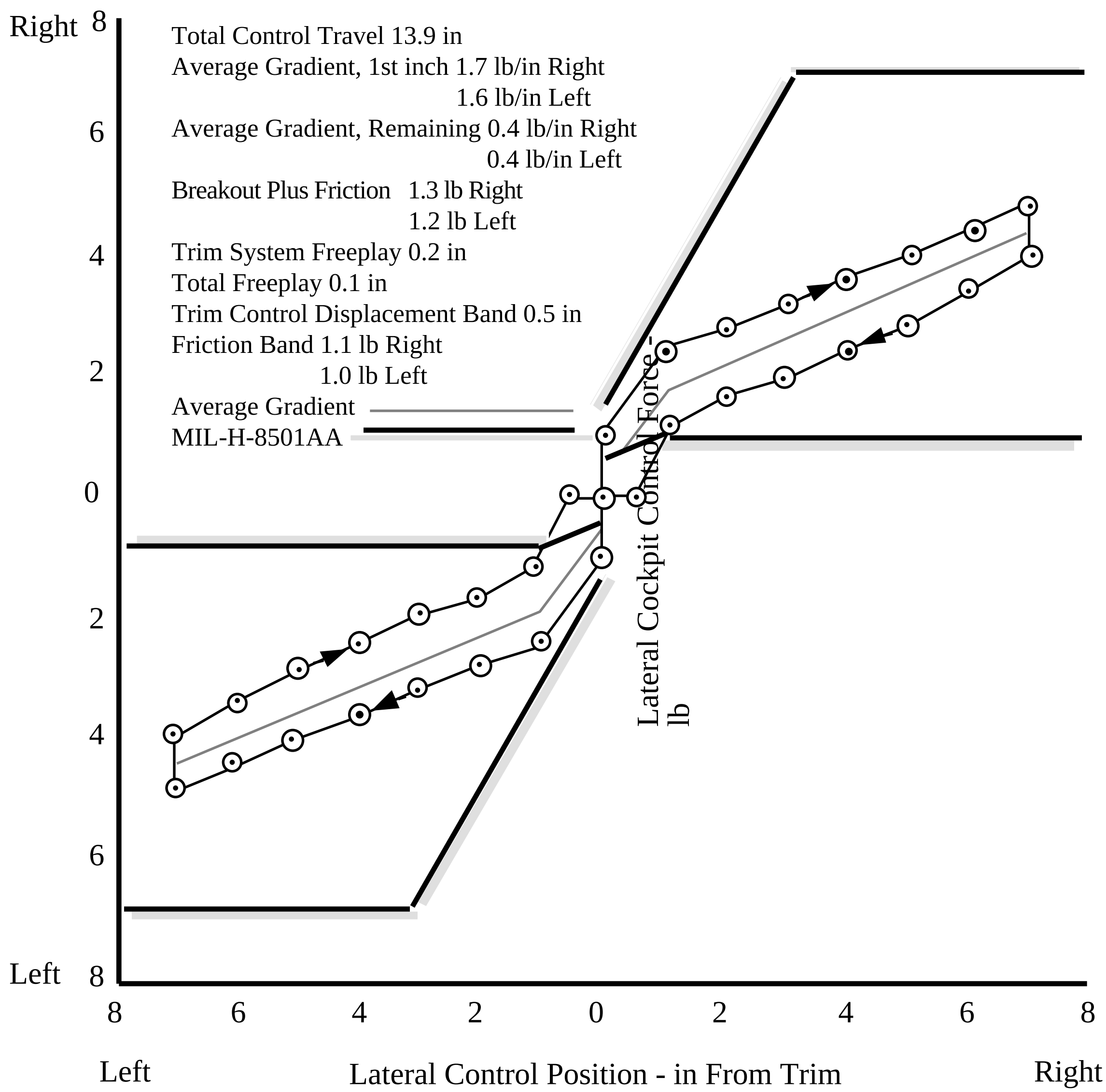


Figure 5.11
Lateral Cyclic Force Displacement Characteristics

5.6.2.3 TRIM SYSTEM

Evaluate the trim system freeplay by making small displacements around trim. If the control can be displaced with negligible force, record the freeplay. Trim system freeplay is annotated on the force versus displacement curve (Figure 5.11) and presented in narrative and tabular format (Table 5.I).

Evaluate the trim envelope for both the momentary trim release and the beeper trim system as appropriate. Representative trim envelope data are presented in Figure 5.5. Annotate the control envelope and trim envelope(s).

The trim lag is determined by timing the delay, if any, between activation of the beeper trim and cockpit control movement. Trim lag is determined for activation in both directions. Trim lag is presented in narrative or tabular format (Table 5.I), or annotated on the trim envelope. The trim rate is determined for movement in both directions. Time the control displacement through a known distance. Trim rate is determined at several locations throughout the trim envelope. The trim rate should be consistent throughout the envelope. Trim rate data are presented in narrative or tabular format (Table 5.I).

5.6.2.3.1 Data Required

Cockpit flight control position, force, and time.

5.6.2.4 CONTROL JUMP, CENTERING, AND DYNAMICS

Control jump is evaluated by trimming the cockpit flight control to a representative trim position through use of the momentary trim release or beeper trim. Using normal pilot-control interface, the control is displaced a known distance from trim and the momentary trim release is activated. The subsequent control motion is measured. The initial trim position and known distance should be representative of those used in flight. A nominal centered position and a 1 in displacement are used commonly. Evaluate control jump in both directions. If the force versus displacement data show any force discontinuities, evaluate control jump about the corresponding position. Control jump is presented in narrative and tabular format (Table 5.I). If automatic data recording is available, an annotated time history of control position can be presented as illustrated in Figure 5.12.

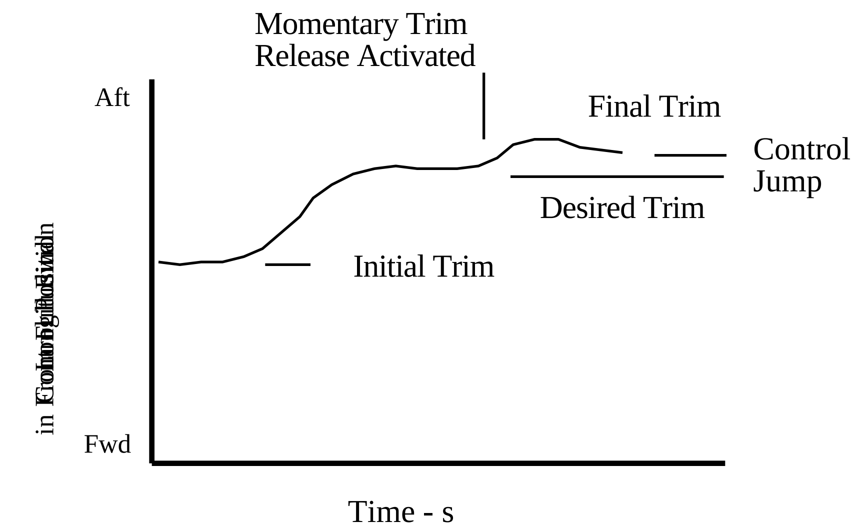


Figure 5.12
Control Jump

Control centering is evaluated around a representative trim position. Trim the cockpit flight control using either the momentary or beeper trim. Note the trim position. Displace the control from trim a representative distance. Slowly relax the control force to zero. Note the resulting control position. Quantify the control centering. Report the control centering in narrative and tabular format (Table 5.I).

Control dynamics are evaluated around a representative trim position established with either the momentary trim release or beeper trim. Displace the control from trim a known distance. Again, a 1 inch displacement is commonly used. Release the control and record the resulting motion. The dynamic motion can be analyzed using the techniques presented in Chapter 3; or if well dampened, described by the number of overshoots. Control dynamics can be evaluated by simply rapping the control. Present control dynamic data in narrative and tabular format (Table 5.I). If automatic data recording is available, an annotated time history of control dynamics can be presented as illustrated in Figure 5.13.

5.6.2.4.1 Data Required

Cockpit flight control position, force, and time. Time histories are useful.

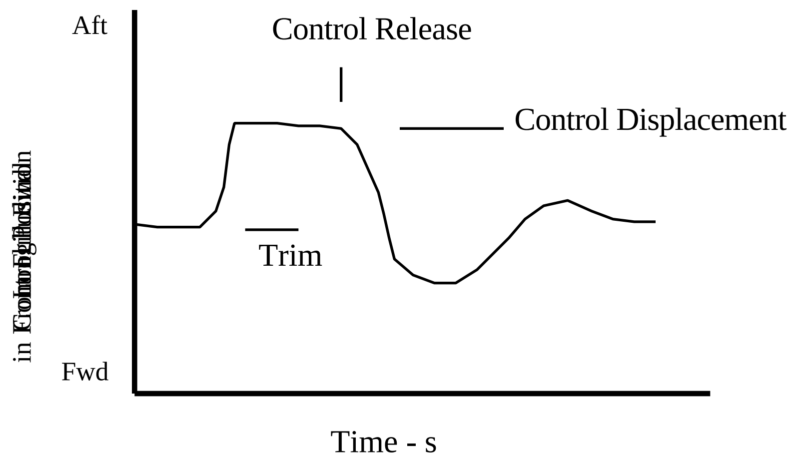


Figure 5.13
Control Dynamics

5.6.2.5 CONTROL FORCE COUPLING

Control force coupling is evaluated by applying a force in one control axis and measuring the force required to restrain motion in the off axes. Control force coupling is evaluated throughout the control envelope. Control force coupling data are presented as illustrated in Figure 5.14.

5.6.2.5.1 Data Required

Cockpit flight control position and force.

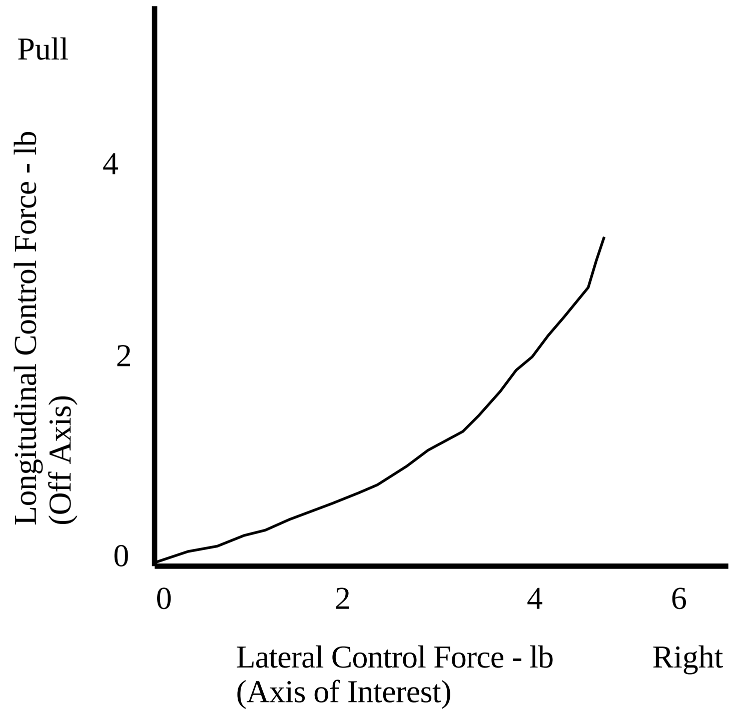


Figure 5.14
Control Force Coupling

5.6.2.6 TOTAL SYSTEM FREEPLAY

Total system freeplay is determined by displacing the cockpit flight control from a representative trim position and measuring the motion of the cockpit control prior to movement of the control surface. Use an observer to determine when the control surface moves. Evaluate total system freeplay for each control axis. Present total system freeplay data on the force versus displacement curve (Figure 5.11) and in narrative and tabular format (Table 5.I).

5.6.2.6.1 Data Required

Cockpit flight control position and flight control surface position.

5.6.2.7 VISCOUS DAMPER CHARACTERISTICS AND TRANSIENT FORCES

Viscous damper characteristics are evaluated by determining the control rate by timing control displacements with a stop watch, and recording the corresponding force exerted on the control. Apply a constant force to the control and observe the steady state rate of motion. To eliminate the influence of breakout, displace the control from a representative trim position prior to starting the test. Restrain the control in this position and apply a small force to the control in a direction to move the control away from trim. When the applied force is at the desired level, release the control and allow the control to acquire a velocity. Hold the applied force constant as the control moves and time the control movement through a known distance. Repeat the test for increasing force levels until the maximum rate of interest is tested. A time history can be used to determine the rate of motion. Present viscous damper data as a plot of rate versus force (Figure 5.15) or in narrative and tabular format (Table 5.I).

Rapid control displacements which produce transient control forces can be evaluated by making step control inputs and recording the resultant force and displacement time histories. The time constant of the force increase and decay is determined from the time history and used to quantify this characteristic. If time history data are not available, measure the time required for the force to increase or drop to an apparent steady state value (determined by the pilot's feel) after the control motion has started or stopped. Transient force decay characteristics are related to the magnitude of the control motion and the peak rate of motion, so these parameters are reported with the above data. Report transient force characteristics data in narrative or tabular format (Table 5.I).

5.6.2.7.1 Data Required

Cockpit flight control position, force, and time.

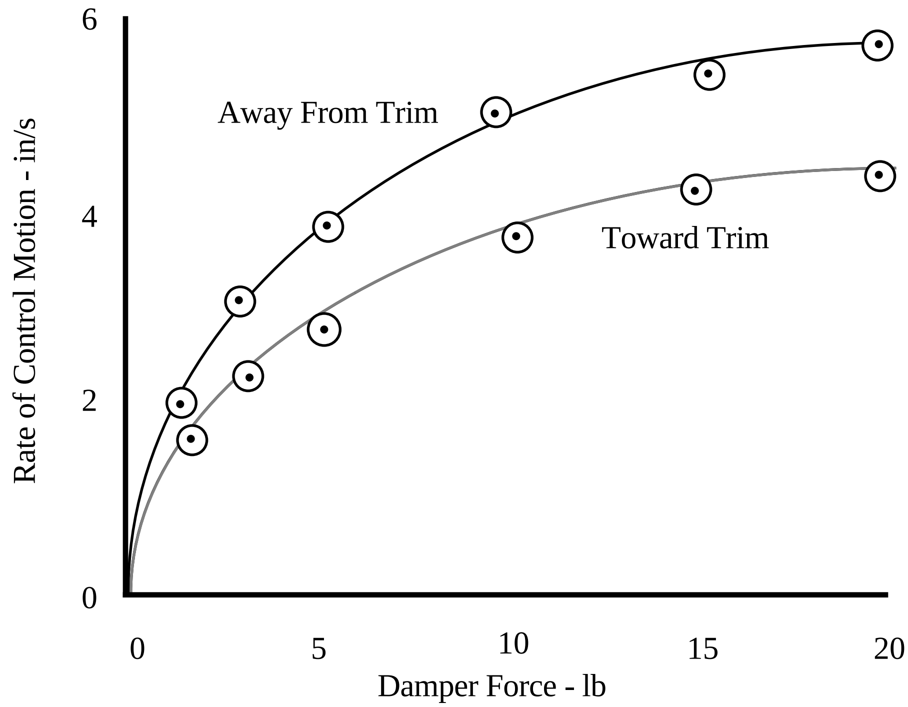


Figure 5.15
Viscous Damper Characteristics

5.6.2.8 AFCS INPUTS

The effects of AFCS inputs can be checked on deck. The reason for conducting such testing and the number of test which can be conducted are so vast, an attempt to discuss them all would be difficult. Instead, the example below illustrates a general approach.

In a hypothetical aircraft, the in flight test of longitudinal static stability produced control forces which were less than predicted for similar control displacements during ground test. The resultant force gradient was considered too weak and unacceptable. To check the problem, a ground test was conducted with the pitot static system pressurized to simulate the desired in flight trim speed. The control was trimmed at the control position corresponding to the in flight trim point used for the static stability test. The control was then displaced 1 in from trim against the force gradient. The airspeed system was then depressurized in 10 kn increments through the airspeed band of interest and the corresponding forces for the 1 in displacement were recorded. This test was repeated for 2 in aft and 1 and 2 in forward control displacement. The ground test data were compared to the in flight data and revealed an undesired AFCS input into the flight control system.

5.6.3 In Flight Testing

In flight evaluation of flight control system characteristics is performed to validate the results of the ground evaluation and to determine the effects of the control characteristics on flight parameters. Trim system characteristics such as freeplay, lag, and rate are evaluated for the effect on parameters such as airspeed, bank angle, or heading while performing representative mission tasks. Control jump and control dynamics are evaluated for the effect on aircraft dynamics. The limit control force is evaluated from one representative trim position to another.

In flight evaluation of the reversible flight control system is conducted in an environment with representative aerodynamic flight control loads. All control force versus displacement tests must be conducted for small displacements. Departure from the representative trim positions cause a change in the aerodynamic loads. Therefore, all displacements must be small changes from a trim condition. Control force data are difficult to record precisely in flight especially if there is a high vibration level in the cockpit. Regardless, all ground tests can be adapted for in flight use staying within the constraints of small displacements and the flight control envelope.

5.6.3.1 DATA REQUIRED

Cockpit flight control position, force, and time.

5.6.4 Safety Considerations/Risk Management

All ground tests must observe the operating limits of the ground power systems whether they are external or internal to the aircraft. Any ground test conducted with the rotors turning must use limited control displacements and rates to avoid rotor to airframe or rotor to ground contact or high aircraft structural load. Inflight tests are limited by the normal flight envelope. Displacements and rates may be limited. Normal build up procedures for force, displacement, or rate must be followed.

Table 5.I
Representative Flight Control System Characteristics

PARAMETER	CONTROL	RESULT	MIL-H-8501A		
			PARAGRAPH	REQUIREMENT	COMPLIANCE
Total System Freeplay (in)	Longitudinal Lateral Directional Collective	0.1 0.1 0.2 0	3.5.10	± 0.2	Met Met Met Met
Trim System Freeplay (in)	Longitudinal Lateral Directional Collective	0.2 0.2 0 0	N/A	N/A	N/A
Breakout plus Friction (lb)	Longitudinal Lateral Directional Collective	1.2 (Fwd) 1.5 (Aft) 1.0 4.0 1.5	3.2.7 3.3.13 3.3.13 3.4.2	0.5 - 1.5 0.5 - 1.5 3.0 - 7.0 1.0 - 3.0 	Met Met Met Met Met
Force Gradient first inch (lb/in)	Longitudinal Lateral	3.2 4.2	3.2.4 3.3.11	0.5 - 2.0 0.5 - 2.0	Failed, 1.2 lb/in (60%) Failed, 2.2 lb/in (110%)
Limit Control Force (lb) *(1)*	Longitudinal Lateral Directional Collective	9.0 11.0 25.0 1.5	3.5.9 3.5.9 3.3.12 3.4.2	8.0 7.0 15.0 7.0	Failed, 1.0 lb (12%) Failed, 4.0 lb (57%) Failed, 10.0 lb (66%) Met
Trim Lag (s)	Longitudinal Lateral Directional Collective	0.2 negligible negligible negligible	N/A	N/A	N/A
Trim Rate (in/s)	Longitudinal Lateral	1/4 1/4	N/A	N/A	N/A
Control Centering	Longitudinal Lateral Directional Collective	Absolute Positive Positive Positive	3.2.3 3.3.10 3.3.10 3.4.2	Positive Positive Positive No Creep	Met Met Met Met
Control Jump *(2)*	Longitudinal Lateral Directional Collective	0.2 0.3 0 0	3.2.3 3.3.10 3.3.10 3.4.2	No undesirable jump when trimming	Failed Failed Met Met
Control Dynamics *(3)*	Longitudinal Lateral Directional Collective	3 overshoots 2 overshoots deadbeat deadbeat	N/A	N/A	N/A

Notes: *(1) Measured from Cyclic and Pedal Centered, Collective from Full Down.*
(2) Displaced 1" from Trim and Trim Release Depressed.
(3) Control Released from 1" Displacement from Trim

5.7 DATA REDUCTION

The data reduction required for flight control system characteristic evaluations consists of calibrating the control position and control force data to the center of the cockpit flight control grip or the point of application of pilot control force. The data obtained from the flight test instrumentation, whether recorded or displayed in the cockpit, and the hand held data are corrected in accordance with this calibration. Inflight data are corrected as required for position error and instrument corrections. Data are presented in consistent units of measure and if control position data are presented as percent of total travel, a correlation between percent and units of displacement is presented.

5.8 DATA ANALYSIS

Flight control system characteristics data analysis consists principally of comparing the test results to the specification requirements and the impact on mission suitability. Quantitative data analysis techniques are not required.

5.9 MISSION SUITABILITY

While the majority of the flight control system evaluation is conducted statically on the ground, the impact of the characteristics must be evaluated inflight during representative mission maneuvers. Armed with a detailed knowledge of the flight control system and its characteristics, the test pilot can assess and report their impact during flying qualities evaluations.

5.10 SPECIFICATION COMPLIANCE

MIL-H-8501A, “Helicopter Flying and Ground Handling Qualities; General Requirements For”, contains general requirements for flight control system characteristics. Additionally, each aircraft specification or other procurement documents may contain additional or supplementary requirements. Research the specifications thoroughly to ensure contractual compliance and a complete evaluation. The following list identifies the paragraph number in MIL-H-8501A and a short description of the requirement.

3.2 Longitudinal characteristics
- 3.2.1 Control margin; helicopter and flight controls free from shake, vibration, roughness
- 3.2.3 Trimmability, control centering, control jump
- 3.2.4 Control force gradient
- 3.2.6 Limit control force
- 3.2.7 Breakout plus friction
- 3.2.8 Control force coupling coupling, transient control forces

3.3 Directional and Lateral characteristics
- 3.3.2 Control margins; helicopter and flight controls free from shake, vibration, roughness
- 3.3.10 Trimmability, control centering, control jump
- 3.3.11 Control force gradient
- 3.3.12 Limit control force
- 3.3.13 Breakout plus friction
- 3.3.14 Control force coupling, transient control forces

3.4 Vertical characteristics
- 3.4.2 Control characteristics, limit control force, breakout plus friction, trimmability
- 3.4.3 Control force coupling coupling

3.5 Autorotation, rotor characteristics, and miscellaneous requirements
- 3.5.6 Limit control force during transition to autorotation
- 3.5.8 Helicopters equipped with power-boosted or power-operated controls
- 3.5.9 Automatic stabilization and control or stability augmentation equipment
- 3.5.10 Total system freeplay
- 3.5.11 Control mixing
  - 3.5.11.1 Mechanical control mixing

3.7 Vibration characteristics
- 3.7.1 Flight control vibration, frequency and magnitude
- 3.7.2 Flight control vibration, forces

Requirements are included in the specification related to boosted controls, failure modes, automatic stabilization equipment, and vibrations. These paragraphs may apply to helicopters equipped with augmented systems or an AFCS. “A Graphical Summary of

Military Helicopter Flying and Ground Handling Qualities of MIL-H-8501A, Technical Report ASNF TN 68-3" provides guidance to the graphical presentation and interpretation of flight control system characteristics data.

5.11 GLOSSARY

5.11.1 Notations

AFCS	Flight Control System
APU	Auxiliary power unit
GW	Gross weight
IMC	Instrument Meteorological Condition
in	Inch
kn	Knot
lb	Pound
PIO	Pilot induced oscillation
s	Second
TCDB	Trim control displacement band
VMC	Visual Meteorological Conditions

5.12 REFERENCES

1. Griffin, John M. and Bellaire, Robert G., "A Graphical Summary of Military Helicopter Flying and Ground Handling Qualities of MIL-H-8501A", Technical Report ASNF TN 68-3, Deputy for Engineering, Aeronautical Systems Division, AFSC, Wright-Patterson Air Force Base, OH, 15 September, 1968.

2. Military Specification, "Helicopter Flying and Ground Handling Qualities; General Requirements For", MIL-H-8501A, with Amendment 1, 3 April, 1962.

3. Naval Test Pilot School Flight Test Manual, *Helicopter Stability and Control*, USNTPS-FTM-No. 101, U. S. Naval Test Pilot School, Patuxent River, MD, 10 June 1968, reprinted February 1978.

4. Naval Test Pilot School Flight Test Manual, *Helicopter Stability and Control, Theory and Flight Test Techniques*, USNTPS-FTM-No. 105, U. S. Naval Test Pilot School, Patuxent River, MD, November 1983, Preliminary Edition.

CHAPTER SIX

FORWARD FLIGHT LONGITUDINAL STABILITY, CONTROL, AND FLYING QUALITIES

FORWARD FLIGHT LONGITUDINAL STABILITY, CONTROL, AND FLYING QUALITIES

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FIGURES

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FORWARD FLIGHT LONGITUDINAL STABILITY, CONTROL, AND FLYING QUALITIES

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EQUATIONS

$$M_u = \frac{1}{I_{yy}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial a_{1s}}{\partial u} + \left(h' + ha_{1s}\right)\frac{\partial T}{\partial u} + h\,\frac{\partial H}{\partial u} + \frac{\partial M_{CG_{f+t}}}{\partial u}\right]$$

eq 6.6 6.11

$$M_q = \frac{1}{I_{yy}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial a_{1s}}{\partial q} + \left(h' + ha_{1s}\right)\frac{\partial T}{\partial q} + h\,\frac{\partial H}{\partial q} + \frac{\partial M_{CG_{f+t}}}{\partial q}\right]$$

eq 6.7 6.13

$$M_w = \frac{1}{I_{yy}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial a_{1s}}{\partial w} + \left(h' + ha_{1s}\right)\frac{\partial T}{\partial w} + h\,\frac{\partial H}{\partial w} + \frac{\partial M_{CG_{f+t}}}{\partial w}\right]$$

eq 6.8 6.15

$$\begin{bmatrix} X_u & X_w & -g & 0 \\ Z_u & Z_w & 0 & 0 \\ M_u & M_w & 0 & 0 \\ 0 & -\frac{1}{u_0} & 1 & -1 \end{bmatrix} \begin{bmatrix} \Delta u \\ \Delta w \\ \Delta\theta \\ \Delta\gamma \end{bmatrix} = - \begin{bmatrix} X_{B_{1s}} \\ Z_{B_{1s}} \\ M_{B_{1s}} \\ 0 \end{bmatrix} \Delta B_{1s} - \begin{bmatrix} X_{\theta_C} \\ Z_{\theta_C} \\ M_{\theta_C} \\ 0 \end{bmatrix} \Delta\theta_C$$

eq 6.9 6.26

$$\begin{bmatrix} X_{B_{1s}} & X_w & -g & 0 \\ Z_{B_{1s}} & Z_w & 0 & 0 \\ M_{B_{1s}} & M_w & 0 & 0 \\ 0 & -\frac{1}{u_0} & 1 & -1 \end{bmatrix} \begin{bmatrix} \Delta B_{1s} \\ \Delta w \\ \Delta\theta \\ \Delta\gamma \end{bmatrix} = - \begin{bmatrix} X_u \\ Z_u \\ M_u \\ 0 \end{bmatrix} \Delta u - \begin{bmatrix} X_{\theta_C} \\ Z_{\theta_C} \\ M_{\theta_C} \\ 0 \end{bmatrix} \Delta\theta_C$$

eq 6.10 6.26

$$\frac{\Delta B_{1s}}{\Delta u}\ \frac{\left[-g\left(Z_w M_u - M_w Z_u\right)\right]}{g\left(Z_w M_{B_{1s}} - M_w Z_{B_{1s}}\right)} = + \frac{M_u - M_w \dfrac{Z_u}{Z_w}}{-M_{B_{1s}} + M_w \dfrac{Z_{B_{1s}}}{Z_w}}$$ *eq 6.11* 6.28

$$\Delta M = \frac{\partial M}{\partial u}\Delta u + \frac{\partial M}{\partial w}\Delta w + \frac{\partial M}{\partial B_{1s}}\Delta B_{1s}$$ *eq 6.12* 6.28

$$\Delta M = I_{yy}\left[M_u \Delta u + M_w \Delta w + M_{B_{1s}}\ \Delta B_{1s}\right]$$ *eq 6.13* 6.29

$$\Delta w = -\frac{Z_u}{Z_w}\Delta u - \frac{Z_{B_{1s}}}{Z_w}\Delta B_{1s}$$ *eq 6.14* 6.29

$$\Delta M = 0 = I_{yy}\left(M_u - \frac{Z_u}{Z_w}M_w\right)\Delta u + I_{yy}\left(M_{B_{1s}} - \frac{Z_{B_{1s}}}{Z_w}M_w\right)\Delta B_{1s}$$ *eq 6.15* 6.29

$$\text{Static stability} = I_{yy}\left(M_u - \frac{Z_u}{Z_w}M_w\right)$$ *eq 6.16* 6.30

$$\left[X_u \Delta u + X_w \Delta w\right] + X_q \Delta q - g\Delta\theta + \left[X_{B_{1s}}\ \Delta B_{1s} + X_{\theta_C}\ \Delta\theta_C\right] = 0$$ *eq 6.17* 6.39

$$u_0 \Delta q + \left[Z_u \Delta u + Z_w \Delta w\right] + Z_q \Delta q + \left[Z_{B_{1s}}\ \Delta B_{1s} + Z_{\theta_C}\ \Delta\theta_C\right] = 0$$ *eq 6.18* 6.40

$$\left[M_u \Delta u + M_w \Delta w\right] + M_q \Delta q + \left[M_{B_{1s}}\ \Delta B_{1s} + M_{\theta_C}\ \Delta\theta_C\right] = 0$$ *eq 6.19* 6.40

$$-\frac{\Delta w}{u_0} + \Delta\theta - \Delta\gamma = 0$$

eq 6.20 6.40

$$g\,(\Delta n) + Z_w\,\Delta w + Z_{B_{1s}}\;\Delta B_{1s} = 0$$

eq 6.21 6.42

$$M_w \Delta w + M_{B_{1s}} \Delta B_{1s} + Mq \;\; g \; \frac{\Delta n}{u_0}$$

eq 6.22 6.42

$$q = g\,\frac{\Delta n}{u_0}$$

eq 6.23 6.42

$$\frac{\Delta B_{1s}}{\Delta q} = \frac{M_q\,Z_w - u_0\,M_w}{M_w\,Z_{B_{1s}} - Z_w\,M_{B_{1s}}}$$

eq 6.24 6.43

$$\frac{\Delta B_{1s}}{\Delta n} = \frac{g}{u_0}\,\frac{\left(M_q\,Z_w - u_0 M_w\right)}{\left(M_w\,Z_{B_{1s}} - Z_w\,M_{B_{1s}}\right)}$$

eq 6.25 6.43

$$q = \frac{g}{u_0}\left(n - \frac{1}{n}\right)$$

eq 6.26 6.44

$$g\Delta\left(n - \frac{1}{n}\right) + Z_w\,\Delta w + Z_{B_{1s}}\;\Delta B_{1s} = 0$$

eq 6.27 6.45

$$M_w\,\Delta w + M_{B_{1s}}\;\Delta_{B_{1s}} + M_q\,\frac{g}{u_0}\left(n - \frac{1}{n}\right) = 0$$

eq 6.28 6.45

$$\frac{dq}{dn} = \frac{g}{u_0}\left(1 + \frac{1}{n^2}\right) \qquad \text{for coordinated turns}$$

eq 6.29 6.45

$$\frac{dq}{dn} = \frac{g}{u_0} \quad \text{for a symmetric pull up}$$

eq 6.30 6.45

$$\frac{\Delta B_{1s}}{\Delta n} = \frac{g}{u_0} \frac{\left(M_q Z_w - u_0 M_w\right)}{\left(M_w Z_{B_{1s}} - Z_w M_{B_{1s}}\right)} \left(1 + \frac{1}{n^2}\right)$$

eq 6.31 6.45

$$\omega_n = \left(\frac{K}{m}\right)^{\frac{1}{2}} = \left[-g \frac{\left(M_u - \frac{Z_u}{Z_w} M_w\right)}{\left(M_q - \frac{u_0}{Z_w} M_w\right)} \right]^{\frac{1}{2}}$$

eq 6.32 6.48

$$\Delta n = \frac{1}{g} u_0 \dot{\gamma} = -\frac{1}{g}\left(\dot{w} - u_0 q\right) = -\frac{1}{g} u_0 \left(\dot{\alpha} - \dot{\theta}\right)$$

eq 6.33 6.51

$$\ddot{q} + \left(-Z_w - M_q\right)\dot{q} + \left(Z_w M_q - u_0 M_w\right)q = M_{B_{1s}} \dot{B}_{1s} + \left(M_w Z_{B_{1s}} - Z_w M_{B_{1s}}\right)B_{1s} = F_q$$

eq 6.34 6.53

$$\lambda_{1,2} = -\left(\frac{-Z_w - M_q}{2}\right) \pm \left[\left(\frac{-Z_w - M_q}{2}\right)^2 - \left(Z_w M_q - u_0 M_w\right)\right]^{\frac{1}{2}}$$

eq 6.35 6.53

CHAPTER SIX

FORWARD FLIGHT LONGITUDINAL STABILITY, CONTROL, AND FLYING QUALITIES

6.1 INTRODUCTION

This chapter discusses helicopter stability, control, and flying qualities in forward flight. The objective is understanding and documenting the helicopter's suitability for performing specific mission-oriented tasks. This chapter discusses open loop and closed loop response characteristics, and the test techniques used to assess and document the characteristics.

6.2 PURPOSE OF TEST

The purpose of these tests is to evaluate the forward flight longitudinal stability, control, and flying qualities of the helicopter. The engineering tests included in the evaluation are:

1. Trimmed flight control positions.
2. Static stability.
3. Maneuvering stability.
4. Long term dynamic stability.
5. Short term dynamic stability.
6. Control response.
7. Gust response.

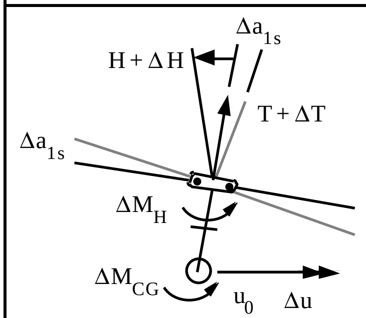
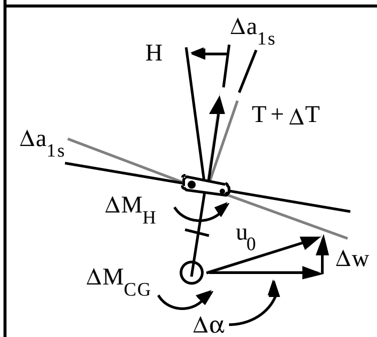
6.3 THEORY

6.3.1 Summary of Quasi-Static Rotor Characteristics

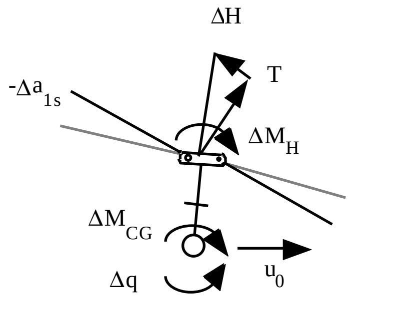
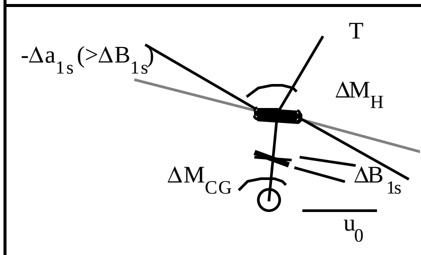
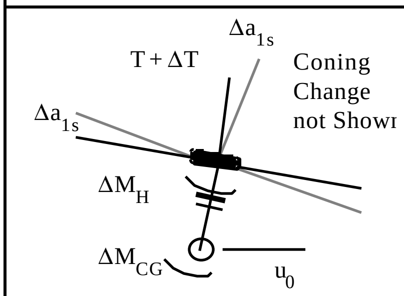
The basic rotor characteristics are discussed in Chapter 4 along with the dynamic responses to various inputs. The presentations indicated a quasi-static approximation for rotor responses is satisfactory for the discussion of helicopter stability, control, and flying qualities.

Table 6.I repeats the summaries of quasi-static rotor response characteristics in forward flight given in Chapter 4. It is included for reference in discussing the stability derivatives and the responses of the complete helicopter. The table indicates the most significant responses of the rotor to specified input conditions and summarizes the rotor contributions to stability derivatives.

Table 6.I
Quasi-Static Rotor Characteristics in Forward Flight

	Input	Rotor Response	Force and Moments Applied to Aircraft
	(a) Airspeed increase, $+\Delta u$	Rotor tilts away from relative velocity, nose up, stable, (blowback) $+\dfrac{\partial a_{1s}}{\partial u}$	$\Delta T = +$ (low u_0), - (high u_0). $\Delta H = +$ (aft). $\Delta M_H = +$ (nose up). $\Delta(M_{CG}) = +$ (nose up, stable). $T(\Delta a_{1s}) = +$ (aft, produce nose up moment).
	(b) Downward velocity, Δw $\Delta\alpha =$ $\Delta w/u_0$	Increased blade angle of attack. Increased coning. Increased lateral flapping due to increased coning. Increased longitudinal flapping. Nose up, unstable $+\dfrac{\partial a_0}{\partial w} \quad +\dfrac{\partial b_{1s}}{\partial a_0}$ $+\dfrac{\partial a_{1s}}{\partial w}$	$\Delta T =+$ (all trim u_0) $\Delta H = 0$, - (low α_{BE}) $\Delta M_H = +$ (nose up) $\Delta(M_{CG}) = +$ (nose up, unstable) $T(\Delta a_{1s}) = +$ (aft)

Quasi-Static Rotor Characteristics in Forward Flight (cont'd)

	(c) Pitch rate, $+\Delta q$, nose up	Rotor lags shaft by angle proportional to pitch rate, nose down, stable $-\frac{\partial a_{1s}}{\partial q}$	$\Delta T = 0$ $\Delta H = +$ (aft, reduces pitch damping) $\Delta M_H = -$ (nose down damping moment) $\Delta(M_{CG}) = -$ (nose down, stable, damping moment)
	(d) Cyclic pitch, $+\Delta B_{1s}$ nose down	TPP tilts due to cyclic pitch, nose down $-\frac{\partial a_{1s}}{\partial B_{1s}}$	$\Delta T = -$ (down force) $\Delta H = 0, -$ (low α_{BE}) $\Delta M_H = -$ (nose down control moment) $\Delta(M_{CG}) = -$ (nose down pitch control moment)
	(e) Collective Increment, $+\Delta\theta_C$	Increased coning. Increased lateral flapping due to increased coning. Increased longitudinal flapping, nose up $+\frac{\partial a_0}{\partial \theta_C} + \frac{\partial b_{1s}}{\partial a_0}$ $+\frac{\partial a_{1s}}{\partial \theta_C}$	$\Delta T = +$ $\Delta H = 0, +$ (low α_{BE}) $\Delta M_H = +$ (nose up) $\Delta(M_{CG}) = +$ (nose up pitch moment)

6.3.2 Longitudinal Equations of Motion

The three pertinent longitudinal equations of motion for small disturbances are presented below to set the context for discussions of the individual stability derivatives. These equations are referred to the stability axes and assume near level flight, $\cos \gamma_0 \approx 1$.

They do not include coupling with the lateral-directional motion of the aircraft. The two force equations, longitudinal and vertical, and the pitch moment equation are presented in terms of the rotor and cockpit control variables:.

$$m\left[\left(-\Delta\dot{u} - w_0\Delta q\right) + \left(X_u\Delta u + X_w\Delta w\right) + \left(X_q\Delta q\right) + \left(-g\cos\gamma_0\Delta\theta\right)\right]$$
$$= -m\left[X_{B_{1s}}\Delta B_{1s} + X_{\theta_C}\Delta\theta_C\right] = -m\left[X_{\delta_{Long}}\Delta\delta_{Long} + X_{\delta_C}\Delta\delta_C\right] \quad \textit{eq 6.1}$$

$$m\left[\left(-\Delta\dot{w} - u_0\Delta q\right) + \left(Z_u\Delta u + Z_w\Delta w\right) + \left(Z_q\Delta q\right) + \left(-g\sin\gamma_0\Delta\theta\right)\right]$$
$$= -m\left[Z_{B_{1s}}\Delta B_{1s} + Z_{\theta_C}\Delta\theta_C\right] = -m\left[Z_{\delta_{Long}}\Delta\delta_{Long} + Z_{\delta_C}\Delta\delta_C\right] \quad \textit{eq 6.2}$$

$$I_{yy}\left[\left(-\Delta q\right) + \left(M_u\Delta u + M_w\Delta w\right) + \left(M_q\Delta q + M_w\Delta w\right)\right]$$
$$= -I_{yy}\left[M_{B_{1s}}\Delta B_{1s} + M_{\theta_C}\Delta\theta_C\right] = -I_{yy}\left[M_{\delta_{Long}}\Delta\delta_{Long} + M_{\delta_C}\Delta\delta_C\right] \quad \textit{eq 6.3}$$

Conventional stability derivative notation is used in which the force derivatives are normalized with respect to mass (m), and the moment derivatives are normalized with respect to moment of inertia (I_{yy}):

$$X_{(\,)} = \frac{1}{m}\frac{\partial X}{\partial(\,)}$$

$$Z_{(\,)} = \frac{1}{m}\frac{\partial Z}{\partial(\,)}$$

$$M_{(\,)} = \frac{1}{I_{yy}}\frac{\partial M}{\partial(\,)}$$

Where:

B_{1s}	- Longitudinal cyclic pitch angle, shaft referenced
δ_C	- Collective control
δ_{LONG}	- Longitudinal control
g	- Gravity
γ_0	- Initial flight path angle
I_{yy}	- Moment of inertia about y axis, pitch moment of inertia
m	- Mass
$M_{B_{1s}}$	- Pitch moment due to longitudinal cyclic pitch angle
M_q	- Pitch moment due to pitch rate
M_{θ_C}	- Pitch moment due to collective pitch angle
M_u	- Pitch moment due to longitudinal velocity
M_w	- Pitch moment due to vertical velocity
$M_{\dot{w}}$	- Pitch moment due to vertical acceleration
θ	- Pitch angle
q	- Pitch rate
θ_C	- Collective pitch angle
u	- Translational velocity component along x axis
u_0	- Initial velocity
$\dot{u}$	- Linear acceleration along x axis
w	- Translational velocity component along z axis
w_0	- Initial velocity component along z axis
$\dot{w}$	- Linear acceleration along z axis
$X_{B_{1s}}$	- Longitudinal force due to longitudinal cyclic pitch angle
X_q	- Longitudinal force due to pitch rate
X_{θ_C}	- Longitudinal force due to collective pitch angle
X_u	- Longitudinal force due to longitudinal velocity
X_w	- Longitudinal force due to vertical velocity

$Z_{B_{1s}}$	- Vertical force due to longitudinal cyclic pitch angle
Z_q	- Vertical force due to pitch rate
Z_{θ_C}	- Vertical force due to collective pitch angle
Z_u	- Vertical force due to longitudinal velocity
Z_w	- Vertical force due to vertical velocity.

These three equations represent the equilibrium of force along the x and z axes of the aircraft, and the equilibrium of pitching moment about the center of gravity (CG). Control force and moment terms are transferred to the right hand sides of the equations. The first term in each of these expressions is an inertia term. The second terms are due to changes in velocity and angle of attack. The third terms are due to pitch rate and angle of attack. The forth terms are gravity force perturbations, and the expressions on the right sides are the control terms. The derivatives are functions of the initial trim variables, but the equations remain valid for typical small perturbations.

6.3.3 Longitudinal Stability Derivatives

The discussion in Chapter 4 indicated the rotor dynamic response is both rapid and well damped. Thus, the assumption of a quasi-static tip path plane (TPP) response is valid for many flight dynamic problems. The longitudinal stability derivatives were obtained using this approximation.

Figure 6.1 shows the forces and moments acting on the helicopter which contribute to the total moment about the CG. These include the fuselage/tail moment ($M_{CG_{f+t}}$), the pitch moment due to rotor hub force (M_H), the moments due to the rotor thrust (T) acting perpendicular to the TPP, and the rotor hub force (H), perpendicular to the shaft. The pitching moment due to thrust depends on a_{1s}, the aft tilt of the TPP relative to the shaft. The fuselage drag force (D_f), assumed to act through the CG, does not cause a pitching moment.

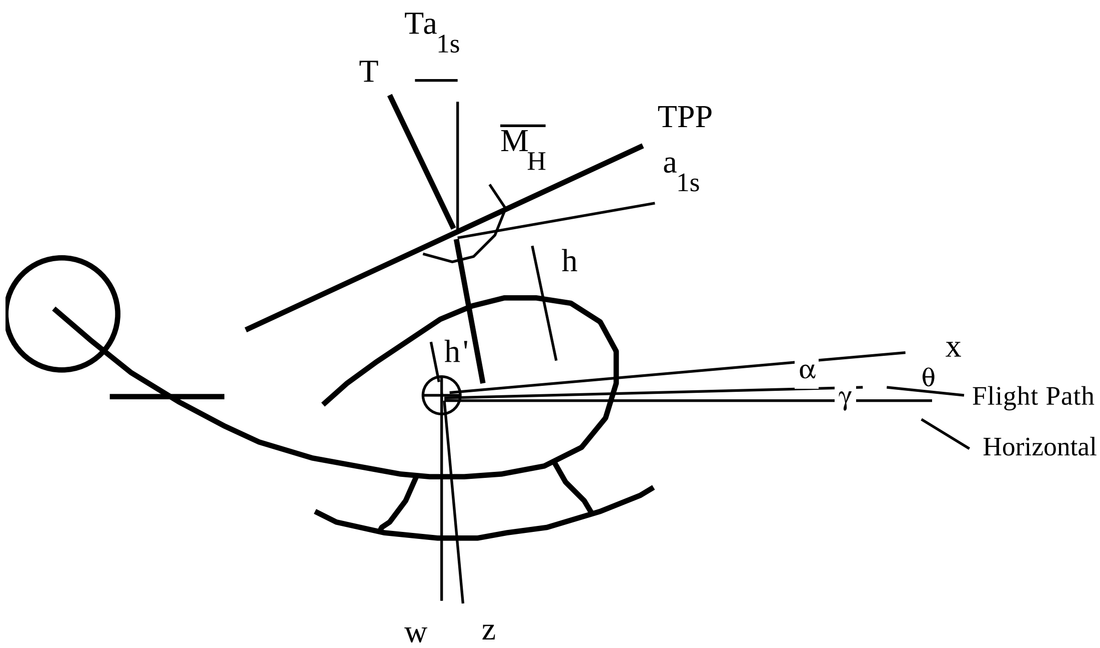


Figure 6.1
Forces and Moments Acting on the Helicopter

The moment stability derivatives give the change in the total aerodynamic pitch moment resulting from a change in one of the flight or control variables (u, w, q, b_{1s} or δ_{LONG}, θ_C or δ_C) while holding the remaining variables constant. These derivatives can be expressed as the sum of terms showing the effect of the changes in a_{1s}, T, H, and $M_{CG_{f+t}}$ with the variable in question. The incremental pitching moment, due to the incremental changes in Δa_{1s}, ΔT, ΔH, $\Delta M_{CG_{f+t}}$, is given by the expression:

$$\Delta M = \left[\underset{(1)}{\underset{\Uparrow}{\left(Th + \frac{ebM_S\Omega^2}{2} \right) \Delta a_{1s}}} + \underset{(2)}{\underset{\Uparrow}{\left(h' + ha_{1s} \right) \Delta T}} + \underset{(3)}{\underset{\Uparrow}{h\Delta H}} + \underset{(4)}{\underset{\Uparrow}{\Delta M_{CG_{f+t}}}} \right] \qquad \textit{eq 6.4}$$

Where:

a_{1s}	- Longitudinal flapping angle, shaft referenced
b	- Number of blades
e	- Flapping hinge offset
H	- Rotor hub force, $\perp$ to shaft
h	- Height of hub above CG
h'	- Longitudinal distance between the rotor shaft and the CG
M	- Net moment about y axis, Pitch moment
$M_{CG_{f+t}}$	- Pitch moment due to the aerodynamic forces on the fuselage/tail
M_S	- Blade mass moment
T	- Thrust
Ω	- Rotor angular velocity.

The physical explanation for the four terms is as follows:

(1) Term proportional to aft tilt of TPP (Δa_{1s}):

$Th\Delta a_{1s}$, Moment caused by aft thrust component due to TPP tilt. (Depends on height, h, of rotor above CG).

$\frac{ebM_S\Omega^2}{2}\Delta a_{1s}$, Offset hinge moment corresponding to aft TPP tilt.

(2) Term proportional to thrust (ΔT):

$(h' + ha_{1s})$, Moment arm from CG to line of action of thrust at trim condition. (Depends on height of rotor above CG, distance between rotor shaft and CG, and trim value of a_{1s}).

(3) Term proportional to rotor H force (ΔH). (Depends on height of rotor above CG).

(4) Term including pitching moment due to fuselage/tail.

Each of the stability derivatives discussed in the following section are treated as the sum of four terms corresponding to the four terms above.

6.3.3.1 CONTROL MOMENT DERIVATIVE

$$M_{B_{1s}} = \frac{1}{I_{yy}} \left[\left(Th + \frac{ebM_S\Omega^2}{2} \right) \frac{\partial a_{1s}}{\partial B_{1s}} + \left(h' + ha_{1s} \right) \frac{\partial T}{\partial B_{1s}} + h\,\frac{\partial H}{\partial B_{1s}} + \frac{\partial M_{CG_{f+t}}}{\partial B_{1s}} \right]$$

eq 6.5

$$\Uparrow \qquad\qquad \Uparrow \qquad\qquad \Uparrow \qquad\qquad \Uparrow$$
$$(1) \qquad\qquad (2) \qquad\qquad (3) \qquad\qquad (4)$$

Where:

a_{1s}	- Longitudinal flapping angle, shaft referenced
b	- Number of blades
B_{1s}	- Longitudinal cyclic pitch angle, shaft referenced
e	- Flapping hinge offset
H	- Rotor hub force, $\perp$ to shaft
h	- Height of hub above CG
h'	- Longitudinal distance between the rotor shaft and the CG
I_{yy}	- Moment of inertia about y axis, pitch moment of inertia
$M_{B_{1s}}$	- Pitch moment due to longitudinal cyclic pitch angle
$M_{CG_{f+t}}$	- Pitch moment due to the aerodynamic forces on the fuselage/tail
M_S	- Blade mass moment
T	- Thrust
Ω	- Rotor angular velocity.

In hovering flight, a longitudinal control input causes a corresponding tilt of the swashplate relative to the shaft and the TPP tends to assume a steady orientation parallel to the swashplate as discussed in Chapter 4. The longitudinal control input in hovering flight does not cause a rotor thrust or H force increment and the pitching moment applied to the helicopter is entirely due to the tilt of the thrust vector and the offset flapping hinge moment.

Two factors shown in Figure 6.2 result in differences between the control moment derivatives in hovering and forward flight. Figure 6.2 (a) shows an initial trim condition at airspeed u_0 with the TPP perpendicular to the shaft. The swashplate should be shown tilted forward for this condition to compensate for rotor blowback but is shown parallel to the TPP to clarify the effect of the longitudinal control input in Figure 6.2 (b). A longitudinal control input is introduced in Figure 6.2 (b) giving the swashplate an aft tilt. In hovering flight, the TPP remains parallel to the swashplate; however, in forward flight the aft tilt is increased by the angle Δa_{1s}.

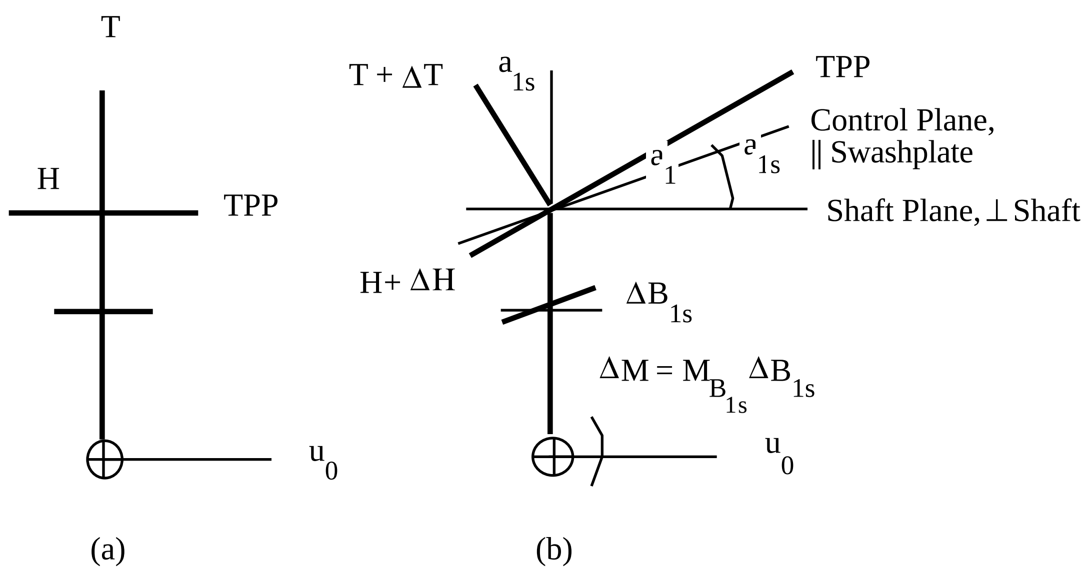


Figure 6.2
Control Force and Moment Derivatives

Term (1) in the expression for $M_{B_{1s}}$ is the primary source of the control moment due to a longitudinal control input. The moment due to the tilt of the thrust vector and the offset hinge moment is larger in forward flight than in hover.

Figure 6.2 assumes a case where the CG is on the shaft axis. If the CG is displaced from this axis, an additional control moment is obtained due to the thrust increment and this effect is included in term (2) for $M_{B_{1s}}$.

Term (3) in the expression for $M_{B_{1s}}$ is small because the control input causes little change in H force. Term (4) could give a contribution to the derivative if control inputs influenced the flow sufficiently to cause changes in the interference of the rotor with the fuselage or tail. However, this term is usually negligible.

6.3.3.2 SPEED STABILITY DERIVATIVE

$$M_u = \frac{1}{I_{yy}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial a_{1s}}{\partial u} + \left(h' + ha_{1s}\right)\frac{\partial T}{\partial u} + h\frac{\partial H}{\partial u} + \frac{\partial M_{CG_{f+t}}}{\partial u}\right] \qquad \textit{eq 6.6}$$

$$\underset{(1)}{\Uparrow} \qquad\qquad \underset{(2)}{\Uparrow} \qquad \underset{(3)}{\Uparrow} \qquad \underset{(4)}{\Uparrow}$$

Where:

a_{1s}	- Longitudinal flapping angle, shaft referenced
b	- Number of blades
e	- Flapping hinge offset
H	- Rotor hub force, ⊥ to shaft
h	- Height of hub above CG
h'	- Longitudinal distance between the rotor shaft and the CG
I_{yy}	- Moment of inertia about y axis, pitch moment of inertia
$M_{CG_{f+t}}$	- Pitch moment due to the aerodynamic forces on the fuselage/tail
M_S	- Blade mass moment
M_u	- Pitch moment due to longitudinal velocity
T	- Thrust
u	- Translational velocity component along x axis
Ω	- Rotor angular velocity.

The primary contribution of the rotor to the speed stability derivative (M_u) is given by term (1) which shows the effect of rotor blowback or aft tilt with increasing speed. This causes a positive or nose up pitching moment due to the combined effects of an offset hinge moment and the aft tilt of the thrust vector. The increased rotor profile drag with a speed increment also produces a positive nose up pitching moment (Term (3)). Terms (1) and (3) depend on the height of the rotor above the CG, but not on the fore and aft location of the CG. The change in rotor thrust with speed contributes a pitching moment depending on the fore and aft CG location and the product of the rotor tilt at trim and the rotor height. This term changes sign at high speed where the thrust variation with speed changes sign. In general, the change in thrust is small compared to the contribution due to rotor blowback.

The combined effect of the fuselage/tail on M_u, term (4), is strongly dependent on the tail incidence and the tail load at the trim condition. Negative incidence at trim produces a stabilizing down load. The down load becomes larger with a speed increase producing a nose up pitching moment. Conversely, an up load at trim makes a negative contribution to the speed stability derivative. However, a tail down load to obtain a positive increment in M_u results in some performance penalty.

Another problem with a horizontal tail is adverse interference with the main rotor or fuselage which may be encountered during some flight conditions. The induced velocities at the tail generated by the main rotor tend to be destabilizing. Downwash angle at the tail can be increased with increased speed due to main rotor wake impingement. The tail effective angle is increased, increasing tail lift and producing a negative or nose down moment. Other problems arise for flight conditions where the fuselage disturbs the flow at the tail. It may be impossible to find a tail location which avoids all interference difficulties. When a satisfactory tail location cannot be found for the desired M_u, it may be necessary to augment the aircraft to provide the desired airspeed variation with longitudinal control position.

6.3.3.3 PITCH DAMPING DERIVATIVE

$$M_q = \frac{1}{I_{yy}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial a_{1s}}{\partial q} + \left(h' + ha_{1s}\right)\frac{\partial T}{\partial q} + h\,\frac{\partial H}{\partial q} + \frac{\partial M_{CG_{f+t}}}{\partial q}\right]$$

eq 6.7

(1)	(2)	(3)	(4)
⇑	⇑	⇑	⇑

Where:

a_{1s}	- Longitudinal flapping angle, shaft referenced
b	- Number of blades
e	- Flapping hinge offset
H	- Rotor hub force, ⊥ to shaft
h	- Height of hub above CG
h'	- Longitudinal distance between the rotor shaft and the CG
I_{yy}	- Moment of inertia about y axis, pitch moment of inertia
$M_{CG_{f+t}}$	- Pitch moment due to the aerodynamic forces on the fuselage/tail
M_q	- Pitch moment due to pitch rate
M_S	- Blade mass moment
q	- Angular velocity about y axis, Pitch rate
T	- Thrust
Ω	- Rotor angular velocity.

The factors contributing to pitch damping in forward flight are indicated in Figure 6.3.

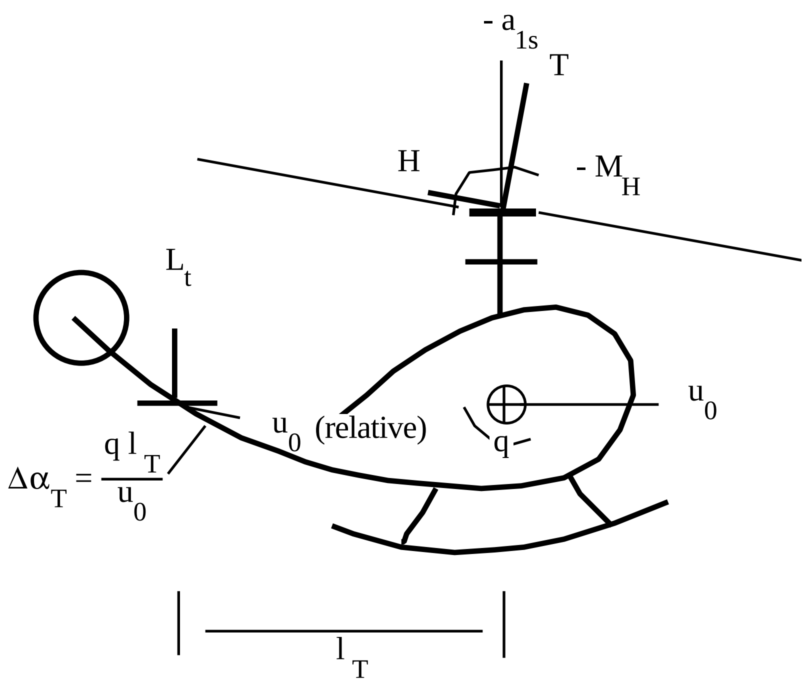


Figure 6.3
Forces and Moments Contributing to Pitch Damping

The helicopter rotates about the CG with a constant nose up pitch rate, q. The TPP and thrust vector are tilted forward (- a_{1s}) and lags behind the shaft in steady pitching motion. The thrust vector tilt and pitch moment due to rotor hub force (M_H) produce nose down pitching moments opposing the nose up pitch rate. Term (1) in the above equation for M_q includes these effects.

Pitching rate has little effect on the magnitude of the thrust vector. Consequently, term (2) in the expression for M_q is small and the CG position has little influence on the pitch damping derivative.

The increment in rotor H force due to pitch rate given by term (3) is opposite to the sign of term (1) and approximately one quarter of its magnitude. The nose up pitch rate about the CG induces a forward relative velocity at the rotor and a forward increment in rotor profile drag which acts in a positive damping sense.

Figure 6.3 indicates a pitch rate induces an upward relative velocity at the tail. The corresponding angle of attack increment causes a tail up load increment and a nose down pitch moment. This damping effect is included in term (4) of the expression for M_q.

6.3.3.4 ANGLE OF ATTACK DERIVATIVE

$$M_w = \frac{1}{I_{yy}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial a_{1s}}{\partial w} + \left(h' + ha_{1s}\right)\frac{\partial T}{\partial w} + h\frac{\partial H}{\partial w} + \frac{\partial M_{CG_{f+t}}}{\partial w}\right]$$

eq 6.8

$$\Uparrow \qquad\qquad \Uparrow \qquad\qquad \Uparrow \qquad\qquad \Uparrow$$
$$(1) \qquad\qquad (2) \qquad\qquad (3) \qquad\qquad (4)$$

Where:

a_{1s}	- Longitudinal flapping angle, shaft referenced
b	- Number of blades
e	- Flapping hinge offset
H	- Rotor hub force, $\perp$ to shaft
h	- Height of hub above CG
h'	- Longitudinal distance between the rotor shaft and the CG
I_{yy}	- Moment of inertia about y axis, pitch moment of inertia
$M_{CG_{f+t}}$	- Pitch moment due to the aerodynamic forces on the fuselage/tail
M_S	- Blade mass moment
M_w	- Pitch moment due to vertical velocity
T	- Thrust
w	- Translational velocity component along z axis
Ω	- Rotor angular velocity.

The first three terms in the above expression give the contributions of the rotor to the angle of attack derivative, M_w. There is little variation in the rotor H force with w, so the contribution of the third term is small. The relative contributions of the first and second terms depend on the CG location, flapping hinge offset, and the trim moments due to the fuselage/tail. Several simple situations are depicted in Figure 6.4 to illustrate the effect of these parameters.

Figure 6.4 (a) and (b) show the unstable tilt of the TPP resulting from a change in angle of attack for a configuration with the CG on the shaft axis (h' = 0), no flapping hinge offset (e = 0), and no fuselage or tail moments. The helicopter is flying to the right in Figure 6.4 (a) with velocity u_0 and zero angle of attack. The swashplate is tilted forward to compensate for rotor blowback and the TPP is perpendicular to the shaft. This initial condition is selected to simplify the diagram and disregards the need for a forward thrust component to balance drag forces. The longitudinal control is fixed and the swashplate is not shown on the other diagrams in Figure 6.4.

In Figure 6.4 (b), a positive angle of attack increment is introduced by changing the pitch attitude without changing the swashplate tilt relative to the shaft. Both α and θ are changing. This results in an even greater pitch angle of the TPP and an aft tilt of the TPP relative to the swashplate. The corresponding tilt of the thrust vector causes an aft force component giving a nose up pitching moment about the CG (unstable) which is proportional to the height of the rotor hub above the CG.

An additional nose up pitching moment due to a_{1s} is obtained if the rotor has offset hinges. Both of these moments are unstable because they tend to increase the pitch attitude of the aircraft, causing a further increase in angle of attack. This unstable contribution from TPP tilt is included in term (1) of the expression for M_w.

The angle of attack increase of the helicopter shown in Figure 6.4 (b) causes an upward relative velocity increment perpendicular to the TPP, increasing the blade average effective angles of attack. This increase, in conjunction with the average blade tangential velocity, results in a thrust increment (ΔT). However, the thrust increment has little effect on the pitching moment for this case because the thrust vector passes through the CG in the trim condition. The increment in relative velocity perpendicular to the rotor produces an increase in longitudinal flapping (a_{1s}), resulting in the aft tilt of the TPP shown in Figure 6.4 (b).

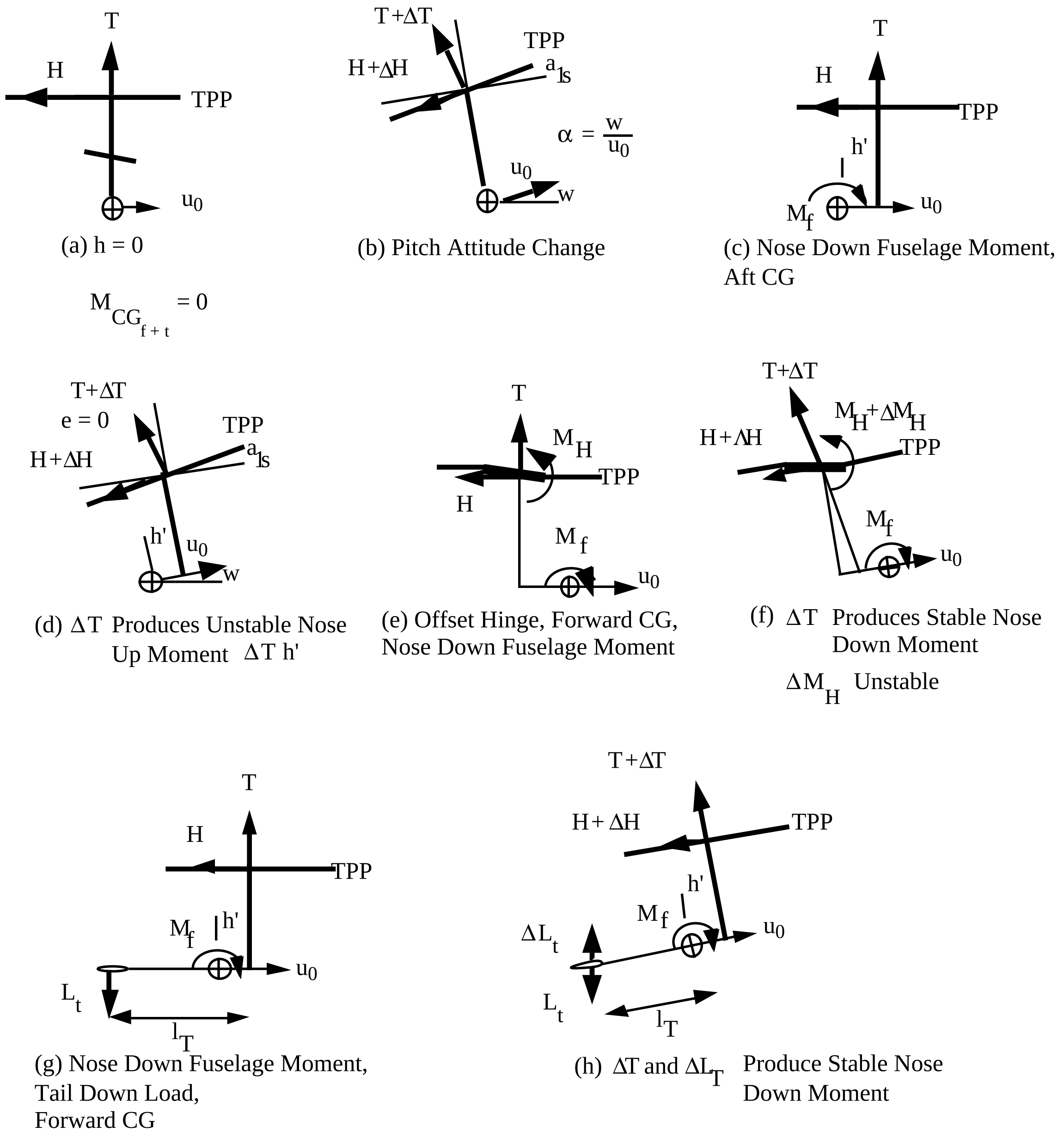


Figure 6.4
Factors Affecting Angle of Attack Derivative

Figure 6.4 (c) shows the trim condition for most helicopters with zero flapping hinge offset and a nose down fuselage moment. The fuselage moment is assumed independent of angle of attack. The moment is balanced by the thrust vector when the CG is a distance h' aft of the shaft. In Figure 6.4 (d), the angle of attack is increased by a change in aircraft pitch attitude. The destabilizing pitching moment due to TPP tilt (Figure 6.4 (b)) is also present in this case. The thrust increment (ΔT) due to the angle of attack change acting at a moment arm (h') gives an additional unstable nose up moment about the CG included by term (2) in the expression for M_w.

Figure 6.4 (e) shows the trim condition for a configuration with offset flapping hinges and the same nose down fuselage moment in Figure 6.4 (c). In this case, equilibrium is obtained with a forward CG location and the line of action of the thrust vector passing aft of the CG. Figure 6.4 (f) shows this configuration with an increased angle of attack. The thrust increment now produces a nose down or stabilizing pitching moment about the CG. This effect, included in term (2) of the equation for M_w, makes a stabilizing contribution to the angle of attack derivative.

Figure 6.4 (g) shows the initial trim condition with the fuselage moment balanced by a horizontal tail down load. Figure 6.4 (h) shows this configuration with an increased pitch attitude. The increase in the angle of attack of the tail gives a tail up load and a nose down stabilizing pitch moment. Term (4) in the expression for M_w is negative or stabilizing for this configuration.

6.3.4 Trim Characteristics

6.3.4.1 GENERAL

Longitudinal control position changes are required normally to achieve changes in rate of climb, descent, and forward airspeed. There are additional requirements for longitudinal trim changes during operations from in and out of ground effect. Investigation of trim changes is an important element of the helicopter's overall flying qualities evaluation. Trim changes refer to flight conditions where power is adjusted to obtain the desired flight path. The power adjustment is the element which makes trimmed control positions significantly different from the static longitudinal stability.

6.3.4.2 REQUIREMENTS FOR TRIM AND UNACCELERATED FLIGHT

Steady unaccelerated trim conditions are obtained for constant velocity in a vertical plane with zero pitching velocity. Such trim conditions are not restricted to level flight and can be obtained while climbing or descending at a constant flight path angle. Figure 6.5 indicates the principal parameters involved.

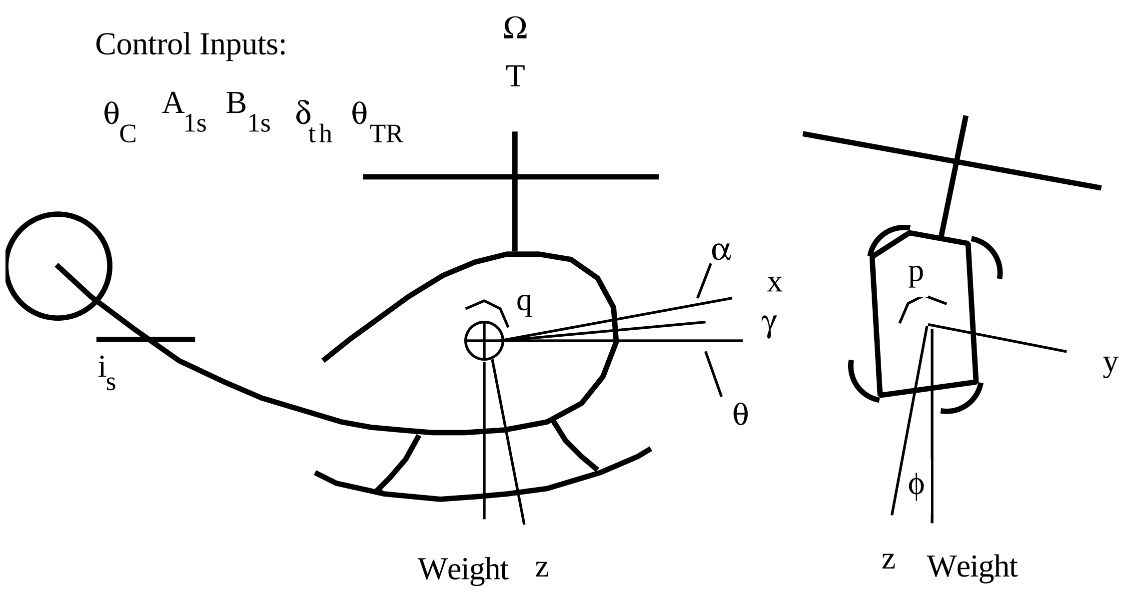


Figure 6.5
Parameters Involved in Trim

The requirement for establishing a steady trim condition is equilibrium of all aerodynamic forces, gravity forces, and moments acting on the helicopter. The total longitudinal, vertical, and side forces and the total pitching, rolling, and yawing moments are zero. In addition, the control inputs maintain constant rotor speed and blade flapping.

The five control inputs normally available on the single rotor helicopter are throttle control of engine torque ($Q = Q\delta_{th}$), collective pitch angle (θ_C), longitudinal cyclic pitch angle (B_{1s}), lateral cyclic pitch angle (A_{1s}), and tail rotor pitch angle (θ_{TR}). Some helicopters also permit a sixth control input to change horizontal stabilizer incidence angle

(i_s). These controls may not be independent. In many helicopters, linkage between the collective pitch and throttle controls are incorporated. In some helicopters, the stabilizer incidence is linked to the cyclic or collective pitch controls.

The aerodynamic forces and moments acting on the helicopter in steady flight conditions are determined by the six control input variables in conjunction with the rotor speed, the aircraft velocity components (u, v, w), and the aircraft angular velocity components (p, q, r). Although the forces and moments depend on the rotor downwash, longitudinal flapping, and lateral flapping; these variables are not considered in the present discussion of trim because they are determined by the control and flight path variables during steady flight conditions. Furthermore, the rotor response parameters are not ordinarily measured or monitored by the pilot. Pitch angle (θ) and roll angle (ϕ), shown on Figure 6.5 define the direction of the gravity force with respect to aircraft axes. The flight path angle (γ) is shown since it is an important variable in both maneuvering and non-maneuvering tasks. Thus, a total of 16 variables are involved in trimming the helicopter: Ω, V, α, β, p, q, r, θ, ϕ, γ, $Q(\delta_{th})$, θ_C, B_{1s}, A_{1s}, θ_{TR}, and i_s. Establishing a steady flight condition requires satisfying three conditions for equilibrium of force components, three conditions for equilibrium of moment components, and a condition for equilibrium of torque acting on the rotor. The kinematics of the situation provides an eighth condition relating the vertical velocity (or flight path angle), the aircraft attitude, and angle of attack. The 16 variables and eight general conditions involved, require the pilot to select eight independent variables to establish a steady flight condition. The need for specific values for some of the variables is obvious and generally they are selected intuitively. The following discussion provides examples of the complete selection of the eight independent variables to emphasize the constraints which exist in trimming the aircraft.

When steady rectilinear flight is required, the three angular velocities are zero. The five remaining independent variables could be control positions. However, a stabilized flight condition is not reached with arbitrarily selected control inputs unless the aircraft is statically and dynamically stable and not performance limited. Some of these five independent variables need not be control inputs and could be flight parameters. In this case, the pilot manipulates the controls as dependent variables to obtain the desired flight parameters.

Table 6.II shows sets of parameters which the pilot selects in setting up typical steady rectilinear flight conditions. The selected independent variables for each case are indicated with asterisks.

Table 6.II
Sets of Independent Variables for Trim in Steady Rectilinear Flight

Variable	Case					Remarks
	1	2	3	4	5	
Ω = Rotor angular velocity	*		*	*	*	
$V = \sqrt{u^2 + v^2 + w^2}$	*	*	*	*		
$\alpha = \frac{w}{u}$ = angle of attack						
$\beta = \frac{u}{V}$ = sideslip angle		*	*			
p = roll rate	*	*	*	*	*	p, q, r = 0
q = pitch rate	*	*	*	*	*	
r = yaw rate	*	*	*	*	*	
θ = pitch angle					*	
ϕ = roll angle	*			*	*	
γ = flight path angle	*					
Q (δ_{th}) = Applied torque		*				
θ_C = collective pitch angle		*	*	*	*	
A_{1s} = lateral cyclic pitch angle						
B_{1s} = longitudinal cyclic pitch angle						
θ_{TR} = tail rotor pitch angle						
i_s = stabilizer incidence	*	*	*	*	*	
* Independent Variable						

Total number of variables in table 6.II	16
Variables involved in:	
Force equilibrium	3
Moment equilibrium	3
Rotor torque equilibrium	1
Flight path kinematics	1
Variables remaining for rectilinear flight	3
Independent variables chosen by pilot	5

Table 6.II, Case 1 is the typical forward flight trim condition encountered in routine non-maneuvering tasks. Independent variables for this case include rotor speed, flight velocity, flight path angle, and roll angle. The fifth arbitrarily selected variable is stabilizer incidence which is assumed to be constant as with a fixed stabilizer.

All other control variables for Case 1 are dependent variables. The pilot cannot choose them arbitrarily but manipulates the controls until he finds the proper input values for $Q(\delta_{th})$, θ_C, A_{1s}, B_{1s}, and θ_{TR} to attain the desired velocity, climb angle, and roll angle. If the helicopter incorporates a rotor speed governor, the pilot is relieved of using the throttle to control rotor speed. The pilot accepts whatever values are obtained for the remaining dependent flight parameters. For example, when the stabilizer angle is fixed, the pilot cannot independently control the pitch angle. Also when the pilot chooses to fly with zero roll angle, he may be forced to fly with a nonzero sideslip angle. If instead he chose to fly with zero sideslip, he may have to fly with some bank angle.

Table 6.II, Case 2 corresponds to a test procedure used to obtain control position versus airspeed gradients to investigate static stability. Collective pitch and throttle are held constant at the initial trim positions. The pilot determines the longitudinal cyclic pitch angle required to stabilize at a new rectilinear flight condition differing from the original by a small speed increment. The speed increment is an arbitrarily selected independent variable for this test and longitudinal cyclic is a dependent variable. Pitch angle and flight path angle are dependent variables.

The complete definition of the longitudinal control position versus airspeed gradient test requires defining the lateral-directional control manipulations. In Case 2, sideslip angle is an independent variable, and lateral cyclic pitch angle and tail rotor pitch angle are dependent variables adjusted to obtain the desired sideslip.

Table 6.II, Case 3 corresponds to another test procedure which might be used in obtaining longitudinal control position versus airspeed gradients. In this case, rotor speed is an independent constant variable. This is accomplished by controlling the dependent variable, torque, via throttle adjustments. Comparatively small changes in rotor speed are obtained in tests corresponding to Case 2 and only small differences in results are expected using Case 2 or Case 3 techniques.

An alternative trim technique used in static stability tests is indicated by Case 4. This case differs from Case 3 because roll angle is treated as an independent variable rather than sideslip. Thus, Case 4 tests are flown with zero roll angle but with some sideslip.

Case 5 presents a technique commonly used for static longitudinal stability tests. Case 5 differs from Case 4 because pitch attitude is an independent variable which the pilot controls, and airspeed is a resulting dependent variable which the pilot accepts.

Although the collective pitch is constant for longitudinal control position gradients, both collective pitch and longitudinal cyclic are used to change trim conditions during non-maneuvering tasks. The Case 1 technique is applicable for changing trim speed in level flight while maintaining constant rotor speed. The primary control inputs are changes in longitudinal cyclic and collective pitch. However, small adjustments in throttle and lateral-directional controls might be required to hold the rotor speed and roll angle constant.

6.3.4.3 POWER EFFECTS

Power effects can produce three trim problems: reduction of control margin; nonlinear trim requirements; and excessive control displacement during normal maneuvering. All three of these problems are depicted in Figure 6.6.

Figure 6.6 indicates less than 3% control margin is available in a descent. There are two discontinuities in the high power portion of the curve. Almost 90% of the available longitudinal control envelope is required during the transition from autorotation to a full power climb.

6.3.4.4 AIRSPEED POSITION ERRORS

The example in Figure 6.6 represents a constant indicated airspeed situation. Indicated airspeed is used because the pilot flies the helicopter by reference to indicated airspeed. Some of the undesirable aspects of the curve may be the direct result of airspeed position error with power. This fact does not excuse the characteristic, the pilot cannot be concerned with position errors during operational flight conditions.

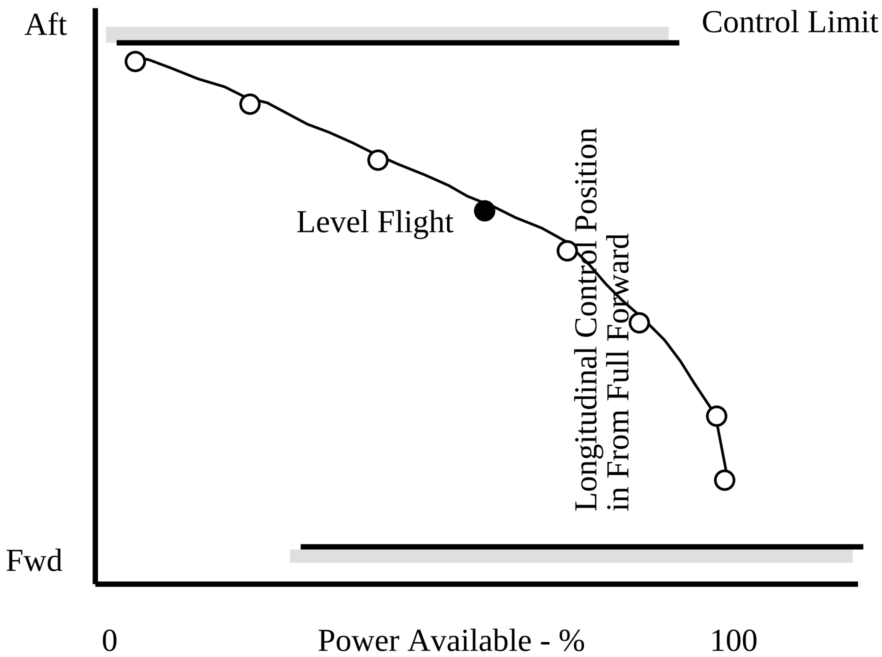


Figure 6.6
Longitudinal Trimmed Flight Control Positions Versus Power

6.3.4.5 EQUILIBRIUM FLIGHT CONDITION

Trimmed equilibrium flight is established by the pilot manipulating the flight controls to provide control forces and moments which balance the aerodynamic forces and moments. The pilot establishes the flight path for the equilibrium condition by adjusting power as required. If the flight condition is level flight, power for level flight is used. If a climb or descent is desired, corresponding power is used. Generally, the pilot maneuvers from one equilibrium flight condition to another or maneuvers about an equilibrium condition. The pilot generally trims all control forces to zero at the equilibrium flight condition. It is the control positions at the trimmed equilibrium flight conditions which are determined during the trimmed flight control positions tests.

6.3.5 Static Stability

The analysis of longitudinal control position versus airspeed and flight path is based on the changes in steady state control positions, control forces, and helicopter response variables required to change from a trimmed flight condition to a stabilized off trim condition. Static longitudinal stability is the sum of the moments generated by an off trim airspeed. The moments produced by an off trim airspeed can not be measured directly in flight. The helicopter can be maneuvered into an off trim equilibrium flight condition by providing an opposing moment generated by longitudinal control displacement. The moment produced by the longitudinal control displacement can not be measured directly in flight. The longitudinal control displacement from trim can be measured. Therefore, static stability is indicated by the gradient of longitudinal control position versus airspeed.

A large number of variables are required to define trim for a helicopter without explicitly including variables describing blade motions. The variables controlled by the pilot are independent and the remaining variables are dependent. The independent variables for each case of Table 6.II should be kept in mind when specifying flight test procedures to insure repeatable results. However, several simplifying assumptions are made to reduce the number of variables considered in this section.

Rotor speed (Ω), for helicopters in powered flight with rotor speed governors, is assumed constant. Consequently, it is unnecessary to include throttle position or applied rotor torque as variables. The tail incidence angle (i_s) is assumed fixed. Finally, any coupling of longitudinal with lateral-directional forces and moments and/or control inputs are not considered. After making the above assumptions, the remaining variables from table 6.II are: $V=\sqrt{u^2+w^2}$, $\alpha = w/u$, θ, θ_C, B_{1s}, and γ.

Tests for static stability are initiated from trimmed conditions in level, climbing or descending flight. All perturbation variables (Δu, Δw, and $\Delta\theta$) are zero at the original rectilinear trimmed flight condition. The pilot can disturb the aircraft from trim by longitudinal control input. Following the control input, a statically and dynamically stable aircraft undergoes a transient response and stabilizes at a new steady rectilinear flight condition with longitudinal speed increment, angle of attack increment, and pitch angle increment. If the aircraft is unstable, the pilot may establish a new momentary equilibrium condition to obtain flight test data by manipulating the controls until the time rate of changes of the perturbation variables are zero.

The following X force, Z force, and moment equations for the new trim condition are obtained by setting all time derivative terms equal to zero. Level flight with zero flight path angle ($\gamma_0 = 0$) is assumed for the trim condition. This assumption results in a simpler expression, emphasizing the physical aspects of the problem. A fourth kinematic equation relating the pitch angle, flight path angle, and angle of attack is added. Simplified matrix notation is used for convenience.

$$\begin{bmatrix} X_u & X_w & -g & 0 \\ Z_u & Z_w & 0 & 0 \\ M_u & M_w & 0 & 0 \\ 0 & -\frac{1}{u_0} & 1 & -1 \end{bmatrix} \begin{bmatrix} \Delta u \\ \Delta w \\ \Delta\theta \\ \Delta\gamma \end{bmatrix} = - \begin{bmatrix} X_{B_{1s}} \\ Z_{B_{1s}} \\ M_{B_{1s}} \\ 0 \end{bmatrix} \Delta B_{1s} - \begin{bmatrix} X_{\theta_C} \\ Z_{\theta_C} \\ M_{\theta_C} \\ 0 \end{bmatrix} \Delta\theta_C$$

eq 6.9

There are four equations and six variables (Δu, Δw, $\Delta\theta$, $\Delta\gamma$, ΔB_{1s} and $\Delta\theta_C$). A solution is obtained by selecting two of the variables as independent with the remaining four computed as dependent variables. As written above, the equations of motion are arranged with control variables or increments in longitudinal cyclic (ΔB_{1s}) and collective pitch ($\Delta\theta_C$) on the right hand side as independent variables.

An alternative arrangement of the equations for perturbed equilibrium conditions suitable for considering static longitudinal stability tests is:

$$\begin{bmatrix} X_{B_{1s}} & X_w & -g & 0 \\ Z_{B_{1s}} & Z_w & 0 & 0 \\ M_{B_{1s}} & M_w & 0 & 0 \\ 0 & -\frac{1}{u_0} & 1 & -1 \end{bmatrix} \begin{bmatrix} \Delta B_{1s} \\ \Delta w \\ \Delta\theta \\ \Delta\gamma \end{bmatrix} = - \begin{bmatrix} X_u \\ Z_u \\ M_u \\ 0 \end{bmatrix} \Delta u - \begin{bmatrix} X_{\theta_C} \\ Z_{\theta_C} \\ M_{\theta_C} \\ 0 \end{bmatrix} \Delta\theta_C$$

eq 6.10

Where:

B_{1s}	- Longitudinal cyclic pitch angle, shaft referenced
g	- Gravity
γ	- Flight path angle
$M_{B_{1s}}$	- Pitch moment due to longitudinal cyclic pitch angle
M_{θ_C}	- Pitch moment due to collective pitch angle
M_u	- Pitch moment due to longitudinal velocity
M_w	- Pitch moment due to vertical velocity
θ	- Pitch angle
θ_C	- Collective pitch angle
u	- Translational velocity component along x axis
u_0	- Initial velocity
w	- Translational velocity component along z axis
$X_{B_{1s}}$	- Longitudinal force due to longitudinal cyclic pitch angle
X_{θ_C}	- Longitudinal force due to collective pitch angle
X_u	- Longitudinal force due to longitudinal velocity
X_w	- Longitudinal force due to vertical velocity
$Z_{B_{1s}}$	- Vertical force due to longitudinal cyclic pitch angle
Z_{θ_C}	- Vertical force due to collective pitch angle
Z_u	- Vertical force due to longitudinal velocity
Z_w	- Vertical force due to vertical velocity.

In evaluating static stability through the longitudinal control position versus airspeed gradient, the speed increment (Δu) and collective pitch increment ($\Delta\theta_C$) are the independent variables. The test technique requires collective pitch fixed at the original trim value so there is no increment in collective ($\Delta\theta_C = 0$). The longitudinal cyclic input is a dependent variable with the value of ΔB_{1s} sufficient to stabilize at the desired speed increment (Δu). The longitudinal control position versus airspeed gradient is found from a plot of control position versus airspeed. Solution of the equilibrium equations gives the following expression for the control position versus airspeed gradient.

$$\frac{\Delta B_{1s}}{\Delta u}\;\frac{\left[-g\left(Z_w M_u - M_w Z_u\right)\right]}{g\left(Z_w M_{B_{1s}} - M_w Z_{B_{1s}}\right)} = +\,\frac{M_u - M_w \dfrac{Z_u}{Z_w}}{-M_{B_{1s}} + M_w \dfrac{Z_{B_{1s}}}{Z_w}} \qquad \textit{eq 6.11}$$

Where:

B_{1s}	- Longitudinal cyclic pitch angle, shaft referenced
g	- Gravity
$M_{B_{1s}}$	- Pitch moment due to longitudinal cyclic pitch angle
M_u	- Pitch moment due to longitudinal velocity
M_w	- Pitch moment due to vertical velocity
u	- Translational velocity component along x axis
$Z_{B_{1s}}$	- Vertical force due to longitudinal cyclic pitch angle
Z_u	- Vertical force due to longitudinal velocity
Z_w	- Vertical force due to vertical velocity.

Changes in angle of attack, pitch angle, and flight path angle are dependent variables whose values can be determined for each speed increment. Plots of pitch angle and flight path angle versus airspeed are used to determine pitch attitude and flight path angle stability. The gradients of longitudinal control position, pitch angle, and flight path angle versus airspeed are related and determined from data obtained in a single flight test.

Static longitudinal stability is discussed in terms of the change in pitching moment with speed in stabilized unaccelerated flight. The pitching moment increment for a perturbed steady rectilinear flight condition with constant collective pitch is:

$$\Delta M = \frac{\partial M}{\partial u}\Delta u + \frac{\partial M}{\partial w}\Delta w + \frac{\partial M}{\partial B_{1s}}\Delta B_{1s} \qquad \textit{eq 6.12}$$

$$\Delta M = I_{yy} \left[M_u \, \Delta u + M_w \, \Delta w + M_{B_{1s}} \, \Delta B_{1s} \right]$$ *eq 6.13*

Where:

- B_{1s} - Longitudinal cyclic pitch angle, shaft referenced
- I_{yy} - Moment of inertia about y axis, pitch moment of inertia
- M - Net moment about y axis, Pitch moment
- $M_{B_{1s}}$ - Pitch moment due to longitudinal cyclic pitch angle
- M_u - Pitch moment due to longitudinal velocity
- M_w - Pitch moment due to vertical velocity
- u - Translational velocity component along x axis
- w - Translational velocity component along z axis.

The first term gives the moment obtained with an airspeed change in a wind tunnel test where the model is supported by a balance system. However, in free flight, to maintain an unaccelerated 1 g condition, the angle of attack of the helicopter is changed to generate Z forces which compensate for increments in speed and cyclic pitch. The required angle of attack change from the Z force equilibrium equation is:

$$\Delta w = - \frac{Z_u}{Z_w} \Delta u - \frac{Z_{B_{1s}}}{Z_w} \Delta B_{1s}$$ *eq 6.14*

When this expression for Δw is inserted into the above equation for ΔM, it becomes:

$$\Delta M = 0 = I_{yy} \left(M_u - \frac{Z_u}{Z_w} M_w \right) \Delta u + I_{yy} \left(M_{B_{1s}} - \frac{Z_{B_{1s}}}{Z_w} M_w \right) \Delta B_{1s}$$ *eq 6.15*

As opposed to an indication of the static stability obtained from flight test data, the coefficient of the first term is used as a physically meaningful definition of static longitudinal stability:

$$\text{Static stability} = I_{yy}\left(M_u - \frac{Z_u}{Z_w} M_w\right)$$

eq 6.16

Where:

B_{1s}	- Longitudinal cyclic pitch angle, shaft referenced
I_{yy}	- Moment of inertia about y axis, pitch moment of inertia
M	- Net moment about y axis, Pitch moment
$M_{B_{1s}}$	- Pitch moment due to longitudinal cyclic pitch angle
M_u	- Pitch moment due to longitudinal velocity
M_w	- Pitch moment due to vertical velocity
u	- Translational velocity component along x axis
w	- Translational velocity component along z axis
$Z_{B_{1s}}$	- Vertical force due to longitudinal cyclic pitch angle
Z_u	- Vertical force due to longitudinal velocity
Z_w	- Vertical force due to vertical velocity.

The expression for the longitudinal control position versus airspeed gradient expressed in terms of TPP variables ($\Delta B_{1s}/\Delta u$) or cockpit control variables ($\Delta\delta_{LONG}/\Delta u$) obtained previously is proportional to the above expression. Furthermore, ($\Delta B_{1s}/\Delta u$) is inversely proportional to the coefficient of the second term in the expression for ΔM (Equation 6.15).

Usually the longitudinal control position versus airspeed gradient is defined in terms of the cockpit flight control (δ_{LONG}) rather than the change in longitudinal cyclic pitch angle (B_{1s}). Thus, flight test measurements include the gearing relating cyclic pitch angle to cockpit control. Helicopters are always designed so aft longitudinal cockpit flight control movement produces a positive pitching moment. Consequently, the sign of the longitudinal control position versus airspeed gradient is a reliable indication of the helicopter's static stability. However, the magnitude of the gradient does not show the degree of static stability because control effectiveness and gearing are included.

6.3.5.1 STATIC STABILITY AS INDICATED BY CONTROL POSITION VERSUS AIRSPEED

When a statically stable helicopter is given a speed increment above the trim speed, it has an initial tendency for nose up pitching motion which tilts the thrust vector aft and slows the helicopter. Static instability leads to an aperiodic divergent response.

Control position versus airspeed curves are referred to a particular trim airspeed. Therefore, the first step in performing a static stability evaluation is to establish a steady trimmed condition. Tests are conducted in wings level, steady heading, ball centered, collective fixed, unaccelerated flight. Any changes in power and rotor speed depend on the rotor speed governor. Ordinarily, these quantities are nearly constant in static stability tests.

The longitudinal control position data are recorded at the initial trim position and at incremental speeds above and below the trim speed. Data are taken at stabilized, equilibrium conditions. The pilot makes a small forward longitudinal cyclic control input to increase airspeed. The pilot observes the resulting decrease in pitch angle. If the helicopter has positive static stability, the helicopter will start to pitch up, returning toward the trim airspeed. The pilot makes additional longitudinal control inputs to maintain a fixed pitch attitude, allowing airspeed to stabilize at the increased value. If the helicopter has negative static stability, the pitch attitude will continue to decrease. The pilot uses aft longitudinal control to maintain the pitch attitude, allowing airspeed to stabilize. The pilot achieves the change in airspeed by flying a fixed pitch attitude, allowing airspeed to stabilize.

Vertical motion or angle of attack have little effect on small speed changes near zero airspeed or hovering flight since the coupling derivatives are small or zero. A more complicated situation exists when the initial trim is in forward flight. Then the Z forces due to speed change (Z_u) and control inputs ($Z_{B_{1s}}$) are no longer zero. They must be balanced by forces due to angle of attack changes (Z_w) to maintain one g flight. Pitching moments result from changes in angle of attack (M_w) and affect the control displacement versus airspeed variation of the helicopter.

Figure 6.7 depicts longitudinal control position and horizontal stabilizer load for several configurations having different static stability characteristics.

Forward longitudinal control displacement is required to stabilize at a higher airspeed for a helicopter with positive static stability (Figure 6.7 (a)). The nose down pitching moment due to forward longitudinal control displacement balances the speed stability of the rotor due to blowback.

Figure 6.7 (b) differs from Figure 6.7 (a) because the negative tail incidence results in a down load at the initial trim condition. The speed increment causes an increase in the down load and a corresponding nose up pitching moment. Thus, the longitudinal control is moved further forward for equilibrium.

Figure 6.7 (c) has positive tail incidence and tail up load. This up load becomes larger with increasing speed causing a nose down pitching moment which is exactly equal in magnitude to the nose up pitch moment due to the rotor. Consequently, Figure 6.7 (c) depicts a neutrally stable configuration where equilibrium is maintained with a 10 kn speed increment without changing longitudinal control position.

Figure 6.7 (d) depicts a configuration with greater positive tail incidence. The larger tail up load with increased speed produces a larger nose down moment than the stable nose up moment contributed by the rotor. The pilot uses aft longitudinal control displacement to achieve equilibrium at the higher speed, indicating negative static stability.

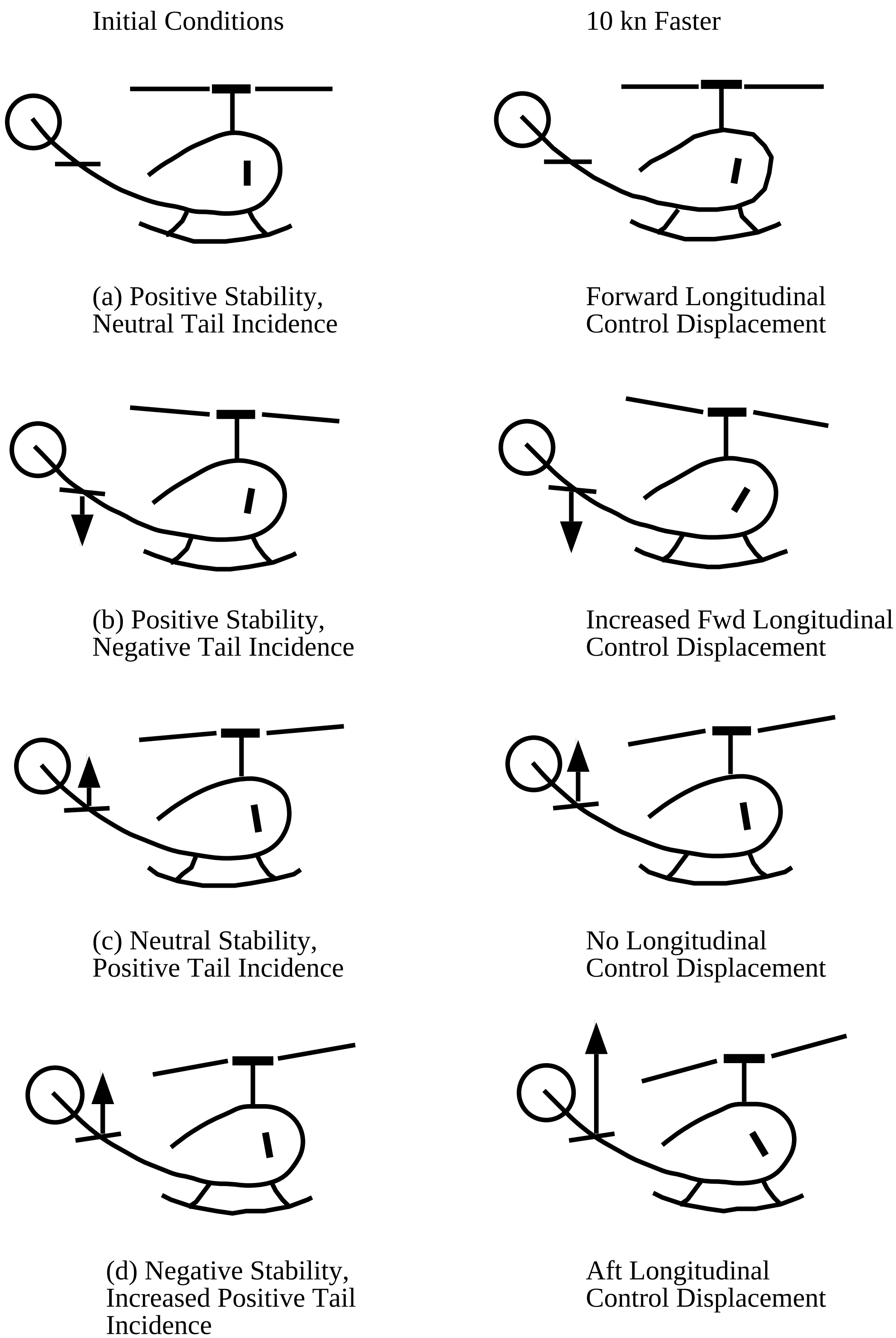


Figure 6.7
Static Stability Examples

Although the CG position of the helicopter affects the control positions required for trim, it has a comparatively small effect on the static stability. The pitching moment due to the tail is not sensitive to CG position because, in general, the tail moment arm is large compared to the CG travel. Pitching moment due to the thrust component parallel to the shaft remains almost constant in unaccelerated flight since this thrust component is approximately equal to the weight of the helicopter. The rotor force component perpendicular to the shaft changes for stabilized off trim conditions but the corresponding pitching moment depends on the vertical height of the rotor above the CG rather than the fore and aft CG location.

The effects of initial tail incidence and tail load on static stability are straight forward and are easily understood. However, there are many interference and downwash phenomena affecting static stability which are more difficult to anticipate and analyze. Figure 6.8 shows a case where the variation in downwash affects the static stability. In the initial condition, the rotor downwash results in a negative angle of attack of the stabilizer, a substantial tail down load, and a corresponding nose up pitching moment. The downwash angle at the tail is decreased with an increase in speed resulting in a smaller tail down load and a nose down pitching moment. Consequently, aft longitudinal control displacement is required to maintain pitch equilibrium with an increase in speed.

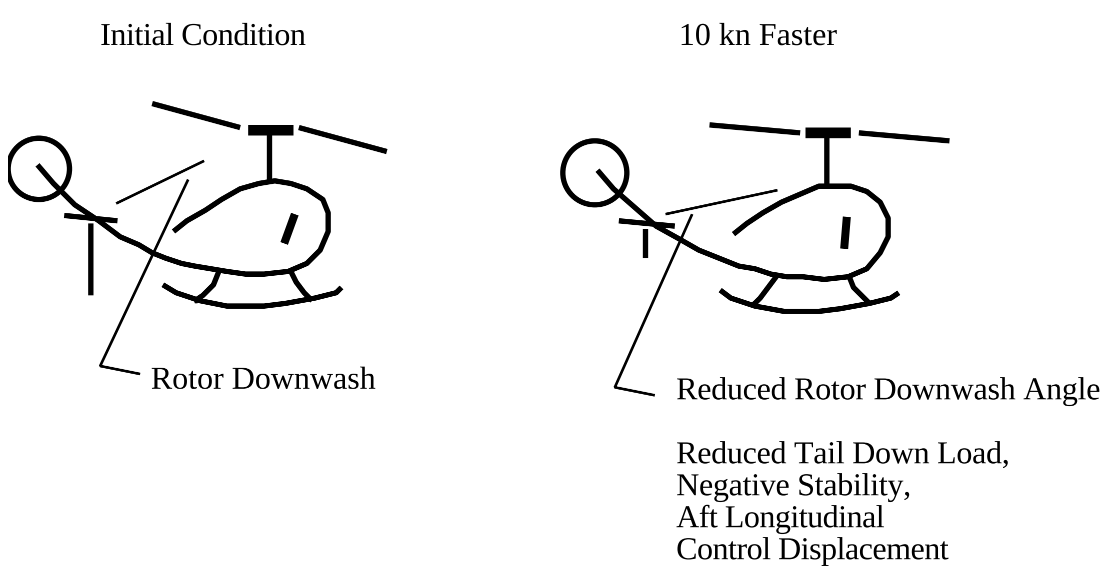


Figure 6.8
Destabilizing Effect of Reduced Rotor Downwash

Longitudinal control position versus airspeed data is presented in Figure 6.9. Positive static stability is indicated by forward longitudinal control displacement with an increase in airspeed. However, the control position versus airspeed curve includes the static stability of the helicopter, the longitudinal control effectiveness, and the longitudinal control system gearing. The same helicopter, with the flight control gearing modified to provide twice the pitching moment per unit of control displacement, can maintain the same increase in airspeed with about one-half the control displacement previously used. It is incorrect to say the first configuration possessed a higher level of longitudinal static stability than the second. It is correct to say the gradient of longitudinal control position versus airspeed is steeper for configuration (a) than (b). Control effectiveness plays a large part in the longitudinal static stability as indicated by the longitudinal control position versus airspeed gradient observed by the pilot. The sign of the slope of the longitudinal control position gradient indicates whether the helicopter is stable or not. However, the magnitude of the slope is not an indication of the level of stability.

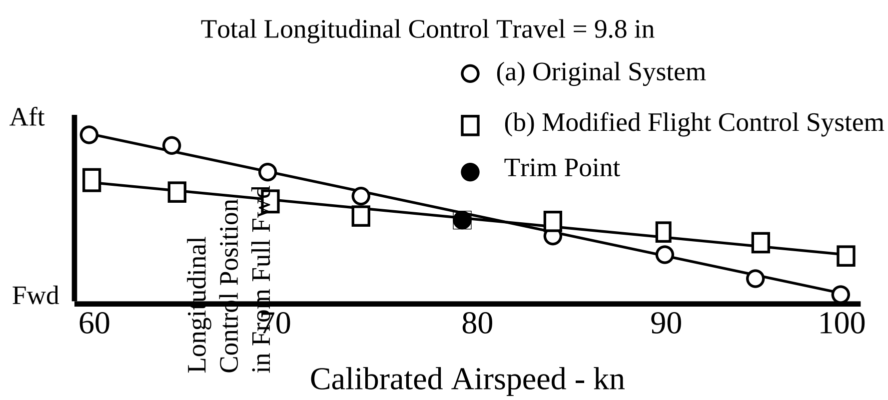


Figure 6.9
Static Stability as Indicated by Longitudinal Control Position Versus Airspeed

6.3.5.2 PITCH ATTITUDE VERSUS AIRSPEED

When the pilot holds the collective fixed and establishes a steady speed increment with longitudinal control displacement, the other dependent variables are recorded at equilibrium as well. The pitch attitude variation with airspeed around trim is used by the pilot as a cue to maintain airspeed. A logical cue to off trim is obtained when a nose down attitude is required to maintain increased speed and a nose up attitude is required for decreased speed. This cue is useful for instrument flight because the vertical gyro provides direct airspeed information. Pitch attitude stability is evaluated by plotting pitch attitude versus airspeed. Additionally, if the aircraft has positive flight path stability, pitch attitude provides a logical cue for controlling vertical velocity.

6.3.5.3 FLIGHT PATH ANGLE VERSUS AIRSPEED

Flight path stability is the variation in vertical velocity or flight path angle with airspeed at constant collective. Positive flight path stability is indicated by a reduction in flight path angle with a reduction in airspeed. Negative flight path stability is indicated by an increase in flight path angle with a reduction in airspeed. The data are obtained concurrent with static stability evaluations. Positive flight path stability is important for precision approaches. Positive flight path stability allows the pilot to associate longitudinal control inputs to flight path angle, attitude, and airspeed in a natural sense. This favorable relationship of flight path angle to attitude and airspeed is satisfied when the aircraft is operating on the front side of the power required curve (Figure 6.10, point A) where the slope of γ versus airspeed is negative. On the backside of the power required curve (Figure 6.10, point B), where the slope of γ versus airspeed is positive, this natural association of flight path angle with attitude and airspeed is reversed. Aft longitudinal control and nose up attitude change results in reduced airspeed (as on the front side) but more negative (steeper) glideslope angle. If the pilot uses front side control techniques to regulate flight path angle when operating on the back side, a closed loop instability results. The relationship on the back side is insidious because it takes some time to develop. The initial response appears correct; however, if no other control corrections are made and the power versus drag forces reach equilibrium, the backside long term flight path response is the reverse of the front side response.

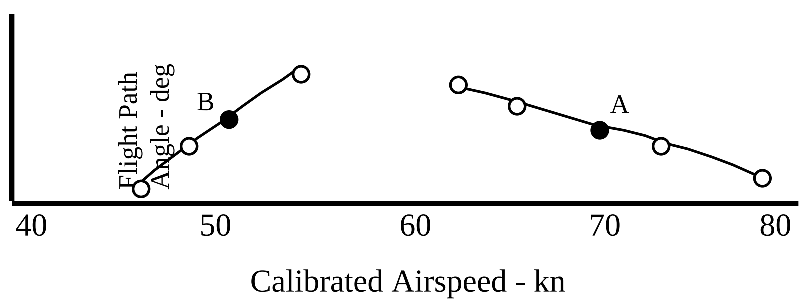


Figure 6.10
Flight Path Angle Versus Airspeed

6.3.5.4 STATIC STABILITY AS INDICATED BY CONTROL FORCE VERSUS AIRSPEED

The static stability of the helicopter is indicated by the longitudinal control position versus airspeed gradient. This variation of control position versus airspeed provides a cue to the pilot to off trim airspeed conditions. If the pilot desires to increase his airspeed, he moves the longitudinal control forward to accelerate. If the helicopter has positive static stability, he maintains a forward longitudinal control displacement at the increased airspeed. The control displacement is a cue to the pilot to the off trim airspeed condition. The force required to maintain the control displacement also provides a cue to the off trim condition.

In a helicopter with an irreversible flight control system incorporating an artificial trim system, the control force variation with airspeed is determined by the control position versus airspeed curve and the control force versus displacement characteristics of the control system. The control force versus airspeed curve is generated by cross plotting the control position versus airspeed curve with the control force versus displacement curve, indicated schematically on Figure 6.11 for control systems with various combinations of control force gradient, breakout force, and friction.

A linear control position versus airspeed curve is assumed for stabilized flight conditions close to the original trim position. When the cockpit force versus displacement curve is also linear, a linear force versus airspeed curve is obtained about the original trim. A negative slope of this force versus airspeed (like a negative slope of the control position versus airspeed curve) provides an indication of positive static stability to the pilot. Positive static stability is indicated by increasing push force for increasing airspeed.

The force feel system gradients must harmonize with the control position gradient in order to obtain good flying qualities. High force gradients are inappropriate for a helicopter which requires large control deflections to make small airspeed changes. The pilot generally senses forces on the cyclic control more acutely than displacement and thus cockpit control forces provide an indication of the control position and a cue to off trim conditions.

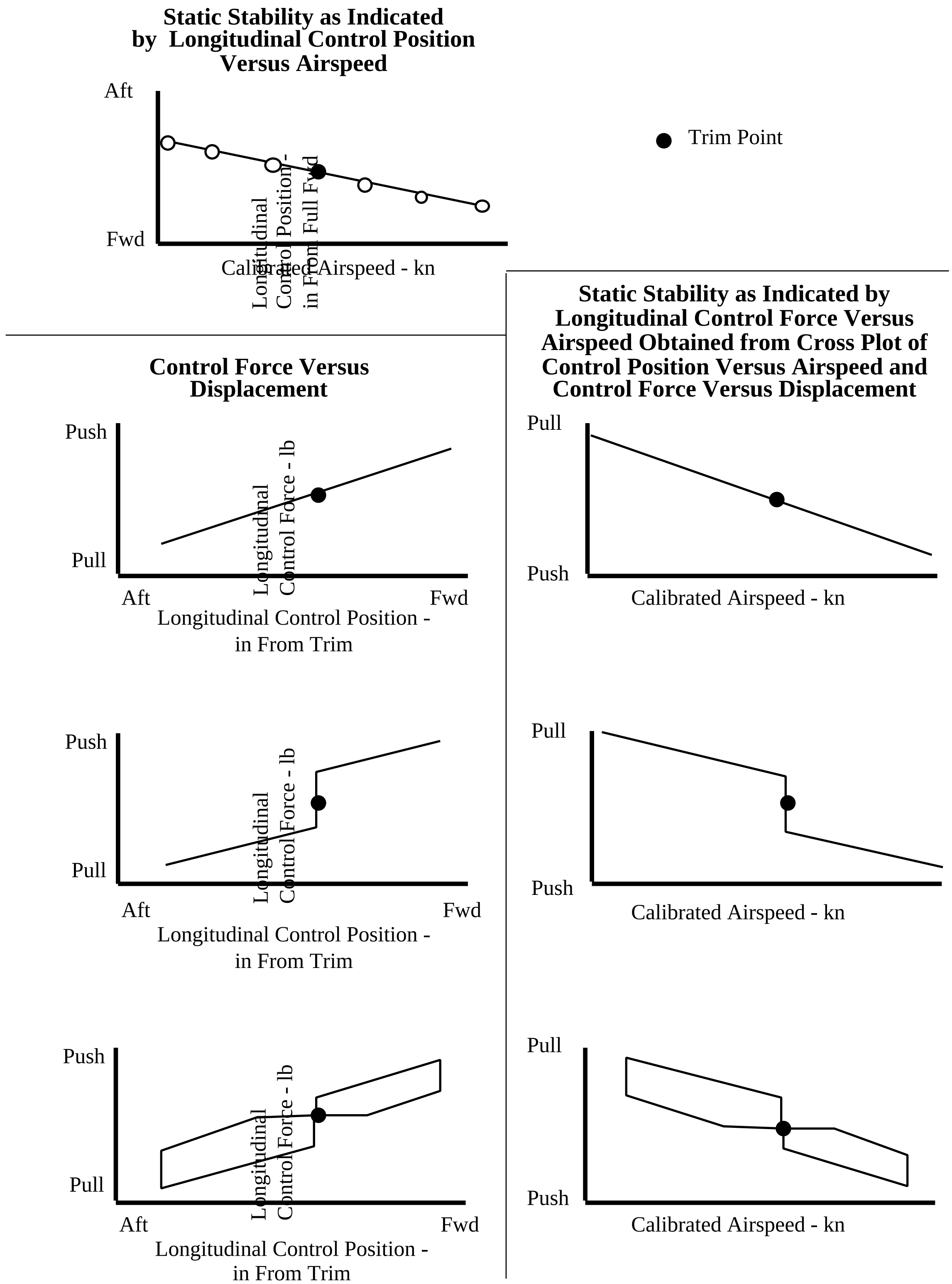


Figure 6.11
Static Stability as Indicated by Control Force Versus Airspeed

Control force versus airspeed gradients can be measured in flight as well as being obtained by a cross plot. In flight measurements are necessary to obtain accurate results for helicopters with reversible flight control systems. In this case, the flight dynamic and control system characteristics are coupled together. Wide variations in control force gradients are found in different flight conditions and may provide inconsistent cues to the pilot.

6.3.6 Maneuvering Stability

Maneuvering stability is the sum of the forces and moments acting on the helicopter due to a disturbance in normal acceleration. As in static stability, the forces and moments acting on the helicopter due to a disturbance can not be measured directly in flight. The forces and moments due to a disturbance in load factor are balanced by longitudinal control forces and moments. The longitudinal control displacement from trim is a measure of the maneuvering stability. A helicopter with positive maneuvering stability tends to return to one g flight following a disturbance. Positive maneuvering stability is indicated by aft longitudinal control displacement from trim for increasing g. A helicopter with negative maneuvering stability tends to depart in g when disturbed. Negative maneuvering stability is indicated by forward longitudinal control displacement from trim with increasing g.

6.3.6.1 SYMMETRICAL PULL UP/PUSH OVER

The linearized form of the longitudinal equations of motion used in discussing static stability are used to obtain a physical understanding of maneuvering stability as indicated by the longitudinal control position versus load factor gradient. In steady accelerated flight as in rectilinear flight, the acceleration time derivatives $\Delta\dot{u}$, $\Delta\dot{w}$, and $\Delta\dot{q}$ are zero, but the aircraft has a constant pitch rate (Δq). When terms in the equation which include Δq are retained, the linearized X force, Z force, and pitching moment equations are as listed below. The kinematic relationship between the variables Δw, $\Delta\theta$, and $\Delta\gamma$ remains unchanged.

$$\left[X_u \Delta u + X_w \Delta w \right] + X_q \Delta q - g\Delta\theta + \left[X_{B_{1s}} \Delta B_{1s} + X_{\theta_C} \Delta\theta_C \right] = 0$$

eq 6.17

$$u_0\,\Delta q + \left[Z_u\,\Delta u + Z_w\,\Delta w \right] + Z_q\,\Delta q + \left[Z_{B_{1s}}\,\Delta B_{1s} + Z_{\theta_C}\,\Delta\theta_C \right] = 0$$

eq 6.18

$$\left[M_u\,\Delta u + M_w\,\Delta w \right] + M_q\,\Delta q + \left[M_{B_{1s}}\,\Delta B_{1s} + M_{\theta_C}\,\Delta\theta_C \right] = 0$$

eq 6.19

$$-\frac{\Delta w}{u_0} + \Delta\theta - \Delta\gamma = 0$$

eq 6.20

Where:

B_{1s}	- Longitudinal cyclic pitch angle, shaft referenced
g	- Gravity
γ	- Flight path angle
$M_{B_{1s}}$	- Pitch moment due to longitudinal cyclic pitch angle
M_q	- Pitch moment due to pitch rate
M_{θ_C}	- Pitch moment due to collective pitch angle
M_u	- Pitch moment due to longitudinal velocity
M_w	- Pitch moment due to vertical velocity
θ	- Pitch angle
q	- Angular velocity about y axis, Pitch rate
θ_C	- Collective pitch angle
u	- Translational velocity component along x axis
u_0	- Initial velocity
w	- Translational velocity component along z axis
$X_{B_{1s}}$	- Longitudinal force due to longitudinal cyclic pitch angle
X_q	- Longitudinal force due to pitch rate
X_{θ_C}	- Longitudinal force due to collective pitch angle
X_u	- Longitudinal force due to longitudinal velocity
X_w	- Longitudinal force due to vertical velocity
$Z_{B_{1s}}$	- Vertical force due to longitudinal cyclic pitch angle
Z_q	- Vertical force due to pitch rate

Z_{θ_C} - Vertical force due to collective pitch angle

Z_u - Vertical force due to longitudinal velocity

Z_w - Vertical force due to vertical velocity.

The variables Δu and Δw are zero prior to a disturbance in g. The equations provide conditions which must be satisfied in steady accelerated longitudinal flight close to the original trim condition.

The $u_0(\Delta q)$ term in the above equation gives the upward acceleration in the -z direction resulting in flight path curvature. The $M_q(\Delta q)$ term is the pitch damping moment while the $X_q(\Delta q)$ and $Z_q(\Delta q)$ terms in the first two equations give the X and Z forces due to pitch rate.

The angle, θ, is the second of three Euler angles (ψ, θ, and ϕ) used to define the orientation of the aircraft relative to earth fixed axes. Hence, the Euler angle $\Delta\theta$ and angular velocity about the pitch body axis (Δq) are treated as separate variables. This is clearly the case in a steady turn where both body axis, q, and Euler θ can be constant. In a constant g, wings level, pullout maneuver, a particular $\Delta\theta$ (near the bottom of the arc) is required to satisfy the equations for the perturbed motion as shown in the following discussion. However, $\Delta\theta$ appears in the X force equation where it determines the component of the gravity force along the flight path which tends to give the aircraft a small forward or aft acceleration. Test results are not sensitive to small errors in $\Delta\theta$, provided they are obtained at the desired test velocity.

There are seven perturbation variables (ΔB_{1s}, Δw, $\Delta\theta$, $\Delta\gamma$, Δu, $\Delta\theta_C$, and Δq) in this system of four equations. Three variables must be treated as independent before the remaining four variables are determined.

Collective pitch is fixed at the trim value; therefore, $\Delta\theta_C = 0$. Maneuvering tasks are accomplished rapidly before appreciable velocity changes can occur. Consequently, Δu is assumed to be zero, requiring the pilot to obtain data at the specified test airspeed. The third independent variable is steady pitch rate or the corresponding incremental load factor.

Longitudinal cyclic pitch, ΔB_{1s}, is one of the dependent variables and the pilot must adjust the longitudinal control position to obtain a steady pitch rate or load factor at the test airspeed. The z axis velocity component (Δw), the pitch angle ($\Delta\theta$), and the flight path angle ($\Delta\gamma$) are dependent variables which are determined when the pilot adjusts the longitudinal control to obtain a stabilized load factor.

The above discussion emphasizes there are a number of independent and dependent variables involved in tests under steady accelerated flight conditions. However, an expression for the acceleration trim gradient can be obtained using only the Z force and pitching moment equations since they do not involve two of the variables, $\Delta\theta$ and $\Delta\gamma$. Neglecting the small contribution of Z_q and substituting for $u_0\Delta_q$, the case where Δu and $\Delta\theta_C$ are zero, the Z force and pitching moment equations are:

$$g\,(\Delta n) + Z_w\,\Delta w + Z_{B_{1s}}\,\Delta B_{1s} = 0 \qquad \textit{eq 6.21}$$

$$M_w\,\Delta w + M_{B_{1s}}\,\Delta B_{1s} + M_q\,g\frac{\Delta n}{u_0} = 0 \qquad \textit{eq 6.22}$$

Where:

$$q = g\,\frac{\Delta n}{u_0} \qquad \textit{eq 6.23}$$

The longitudinal control position versus pitch rate gradient and the longitudinal control position versus load factor gradient obtained by a simultaneous solution of these equations are as follows:

$$\frac{\Delta B_{1s}}{\Delta q} = \frac{M_q Z_w - u_0 M_w}{M_w Z_{B_{1s}} - Z_w M_{B_{1s}}}$$

eq 6.24

$$\frac{\Delta B_{1s}}{\Delta n} = \frac{g}{u_0} \frac{\left(M_q Z_w - u_0 M_w\right)}{\left(M_w Z_{B_{1s}} - Z_w M_{B_{1s}}\right)}$$

eq 6.25

Where:

B_{1s}	- Longitudinal cyclic pitch angle, shaft referenced
g	- Gravity
$M_{B_{1s}}$	- Pitch moment due to longitudinal cyclic pitch angle
M_q	- Pitch moment due to pitch rate
M_u	- Pitch moment due to longitudinal velocity
M_w	- Pitch moment due to vertical velocity
n	- Normal acceleration, Normal load factor
q	- Angular velocity about y axis, Pitch rate
u_0	- Initial velocity
w	- Translational velocity component along z axis
$Z_{B_{1s}}$	- Vertical force due to longitudinal cyclic pitch angle
Z_w	- Vertical force due to vertical velocity.

6.3.6.2 STEADY TURNS

The preceding discussion investigated maneuver stability as indicated by longitudinal control position versus normal acceleration obtained in cyclic pull ups with constant collective. Although steady pitch rates and angles of attack can be obtained using this procedure, the change in flight path angle and X gravity force component eventually causes a change in forward velocity. During the pull up, data must be taken during a short period of time when the longitudinal velocity is equal to the test airspeed. Therefore, steady turns are convenient for gathering maneuvering stability data.

An approximation of a level turn is given below to indicate some of the factors influencing the longitudinal control position versus normal acceleration obtained in a turn.

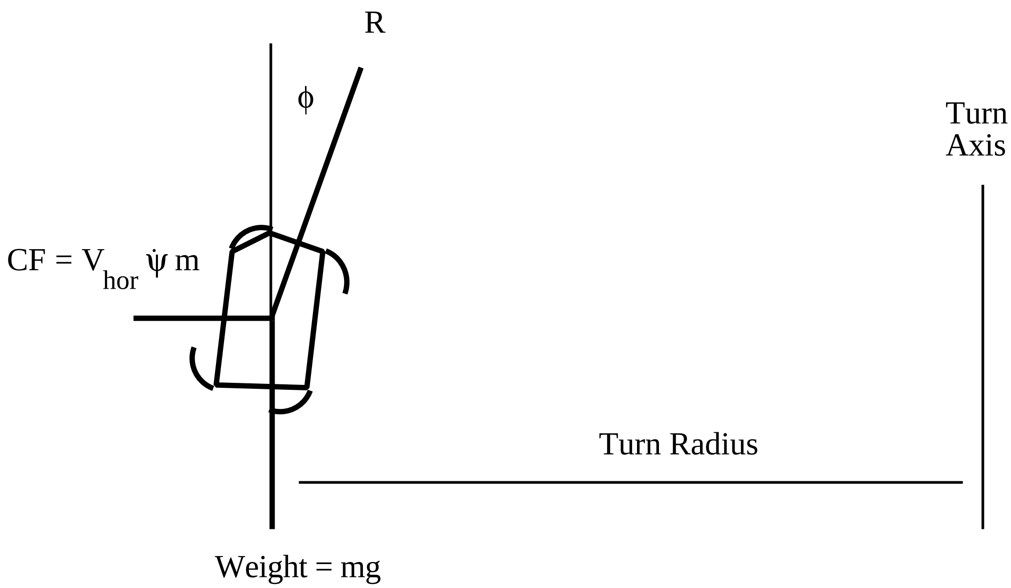


Figure 6.12
Relationships Between Variables in Steady Coordinated Turn

Figure 6.12 indicates the relationship between variables which define a steady coordinated turn. The centrifugal and gravity forces act in a plane passing through the turn axis and the CG of the helicopter. They are opposed by an aerodynamic force (R = n mg), inclined by an angle (ϕ) relative to the vertical axis. Pitch rate is related to load factor and bank angle:

$$q = \frac{g}{u_0}\left(n - \frac{1}{n}\right) \qquad \textit{eq 6.26}$$

Tests of longitudinal control position versus acceleration gradients in steady turning flight are conducted maintaining constant airspeed. The approximate vertical force and pitching moment equations in a steady turn for disturbances from a level flight trim condition similar to the one used in discussing pull up maneuvers are:

$$g\Delta\left(n - \frac{1}{n}\right) + Z_w \,\Delta w + Z_{B_{1s}} \,\Delta B_{1s} = 0$$

eq 6.27

$$M_w \,\Delta w + M_{B_{1s}} \,\Delta_{B_{1s}} + M_q \frac{g}{u_0}\left(n - \frac{1}{n}\right) = 0$$

eq 6.28

The relationship of pitch rate with load factor in a coordinated turn is:

$$\frac{dq}{dn} = \frac{g}{u_0}\left(1 + \frac{1}{n^2}\right) \qquad \text{for coordinated turns}$$

eq 6.29

This differs from the relationships given previously for a pull up:

$$\frac{dq}{dn} = \frac{g}{u_0} \qquad \text{for a symmetric pull up}$$

eq 6.30

The cyclic control versus load factor in a coordinated turn follows from the preceding results:

$$\frac{\Delta B_{1s}}{\Delta n} = \frac{g}{u_0} \frac{\left(M_q Z_w - u_0 M_w\right)}{\left(M_w Z_{B_{1s}} - Z_w M_{B_{1s}}\right)} \left(1 + \frac{1}{n^2}\right)$$

eq 6.31

Where:

B_{1s}	- Longitudinal cyclic pitch angle, shaft referenced
g	- Gravity
$M_{B_{1s}}$	- Pitch moment due to longitudinal cyclic pitch angle
M_q	- Pitch moment due to pitch rate
M_w	- Pitch moment due to vertical velocity
n	- Normal acceleration, Normal load factor
q	- Angular velocity about y axis, Pitch rate
u_0	- Initial velocity
$Z_{B_{1s}}$	- Vertical force due to longitudinal cyclic pitch angle
Z_w	- Vertical force due to vertical velocity.

The above expression for the cyclic pitch versus load factor gradient obtained in a coordinated turn with constant collective pitch differs from the one obtained for steady pull ups. As indicated, the gradient (dq/dn) is different in the two cases. The gradient of B_{1s}/n if multiplied by the factor $(1 + 1/n^2)$. At $n \approx 1$, the local gradient in a steady turn is approximately twice the gradient in a steady pull up. At high n, the local gradient in a steady turn approaches the local gradient in a pull up.

6.3.6.3 MANEUVERING STABILITY AS INDICATED BY CONTROL FORCE VERSUS LOAD FACTOR

Longitudinal control force versus load factor is a cue to the pilot to off trim load factor just as the longitudinal control force versus airspeed is a cue to off trim airspeed. When a longitudinal control position versus normal load factor curve is obtained for a helicopter with an irreversible control system, corresponding control force versus normal load factor curves are constructed by cross plotting cockpit flight control force versus displacement data. For a reversible flight control system, control force versus load factor data can be measured in flight. Positive maneuvering stability is indicated by increasing pull force for increasing load factor. Negative maneuvering stability is indicated by increasing push force for increasing load factor.

Trim systems which give a satisfactory control force versus airspeed gradient may provide an insufficient control force versus normal load factor gradient. Therefore, a bob weight or other flight control system artifact might be used to provide the desired control force versus load factor gradient. The bob weight does not change the control force versus airspeed gradient. However, a bungee spring may be required to counteract the bob weight under static conditions. Bob weights are not used widely in helicopters because of inherent drawbacks such as longitudinal control forces generated by vertical accelerations due to collective pitch inputs and inputs fed into the control system by turbulence.

Some helicopter trim systems introduce control forces proportional to pitch rate as a pilot cue in maneuvering flight and incorporate a minimum pitch rate limit so control forces are not affected by aircraft responses to small gust inputs.

6.3.7 Dynamic Stability

Dynamic stability is the helicopter motion over time following a disturbance. Both control response and gust response depend on the basic vehicle dynamics and are not independent considerations from the test pilot's viewpoint. One other distinction between the two responses might be mentioned. Control inputs generally do not contain high frequency components exciting blade dynamic response. Therefore, high frequency modes involving blade dynamics are ignored in considering control responses. On the other hand, the gust response may include high frequency dynamics.

6.3.7.1 LONG TERM DYNAMIC STABILITY

The significance of static stability on the nature of the long term response in forward flight is determined from an approximate second order equation for the longitudinal velocity (u). The longitudinal equations of motion are approximated on the basis the helicopter has a slow aperiodic or a low frequency oscillatory long term response, and the uncoupled u velocity response is much slower than the uncoupled w velocity or q = responses. This approximation is generally true for augmented helicopters and also for most unaugmented helicopters. However, the simplified formulas developed for the long term mode cannot be applied for cases having comparatively slow w and $\dot{\theta}$ responses. Thus w and $\dot{\theta}$ should be close to their quasi-static values, w_{QS} and $\dot{\theta}_{QS}$, obtained by solving the equations of motion assuming $\dot{w}$ and $\ddot{\theta}$ are zero as a first approximation. The time derivatives dw_{QS}/dt and $d\dot{\theta}_{QS}/dt$ are substituted into the equations as a second approximation for $\dot{w}$ and $\ddot{\theta}$.

Cyclic pitch inputs are assumed to be slow, $B_{1s} \approx 0$. Further simplifications are obtained by neglecting the $X_w w$ and $M_{\dot{w}} \dot{w}$ terms in the equations of motion. The equations are combined to obtain the 2nd order equation for u, which is analogous to the equation for a spring mass damper system. The analogous quantities are shown in Table 6.III.

If the spring constant term has a negative value, the characteristic equation for the system has a positive real root and a negative real root, corresponding to convergent and divergent aperiodic responses. Similarly, the effective stiffness for the helicopter u response is negative when the static stability parameter $g(M_w Z_u - M_u Z_w)$ is negative. Thus, the sign of this parameter determined from the longitudinal control position versus airspeed gradients indicates whether or not the helicopter has a divergent aperiodic long term response. The natural frequency of the oscillation in forward flight is:

$$\omega_n = \left(\frac{K}{m}\right)^{\frac{1}{2}} = \left[-g \frac{\left(M_u - \frac{Z_u}{Z_w} M_w\right)}{\left(M_q - \frac{u_0}{Z_w} M_w\right)} \right]^{\frac{1}{2}} \qquad \textit{eq 6.32}$$

Where:

g	- Gravity
K	- Spring constant
m	- Mass
M_q	- Pitch moment due to pitch rate
M_u	- Pitch moment due to longitudinal velocity
M_w	- Pitch moment due to vertical velocity
u_0	- Initial velocity
ω_n	- Natural frequency
Z_u	- Vertical force due to longitudinal velocity
Z_w	- Vertical force due to vertical velocity.

Table 6.III
Analogy of Long Term Response and Spring Mass Damper

Approximate Equations of Motion for Long Term Response	Assumptions
$\dot{u} - X_u + g\theta = X_{B_{1s}} B_{1s}$	$\dot{B}_{1s} = 0$ (Slow change in longitudinal cyclic pitch angle)
$-Z_u u \frac{d}{dt} w_{QS} - Z_w w - u_0 \dot{\theta} = Z_{B_{1s}} B_{1s}$	$w_{QS} = \frac{\left(-M_q Z_u + u_0 M_u\right) u}{\left(Z_w M_q - u_0 M_w\right)}$
$-M_u u - M_w w + \frac{d}{dt} \dot{\theta}_{QS} - M_q q = M_{B_{1s}} B_{1s}$	$\dot{\theta}_{QS} = \frac{\left(-Z_w M_u + M_w Z_u\right) u}{\left(Z_w M_q - u_0 M_w\right)}$
	$\dot{w} = \ddot{\theta} = B_{1s} = 0$ in 2nd and 3rd equations of motion

Second Order form of Equations of Motion for u:

$$\ddot{u} + \left[-X_u + g \frac{\left[-Z_w^2 M_u - M_w \left(u_0 M_u - M_q Z_u - Z_w Z_u \right) \right]}{\left(Z_w M_q - M_w u_0 \right)^2} \right] \dot{u}$$

$$+ \left[g \frac{\left(M_w Z_u - M_u Z_w \right)}{\left(Z_w M_q - M_w u_0 \right)} \right] u = \left[g \frac{\left(-Z_{B_{1s}} M_w + M_{B_{1s}} Z_w \right)}{\left(Z_w M_q - M_w u_0 \right)} \right] B_{1s}$$

Comparison of Spring Mass Damper and Helicopter Speed Response:

Spring Mass Damper	Helicopter Speed Response
x	u
$\frac{B}{m}$	$\left[-X_u + g \frac{\left[-Z_w^2 M_u - M_w \left(u_0 M_u - M_q Z_u - Z_w Z_u \right) \right]}{\left(Z_w M_q - M_w u_0 \right)^2} \right]$
$\frac{k}{m}$	$\left[g \frac{\left(M_w Z_u - M_u Z_w \right)}{\left(Z_w M_q - M_w u_0 \right)} \right]$
$\frac{F_\delta \delta}{m}$	$\left[g \frac{\left(-Z_{B_{1s}} M_w + M_{B_{1s}} Z_w \right)}{\left(Z_w M_q - M_w u_0 \right)} \right] B_{1s}$

This expression incorporates increments to the M_u and M_q derivatives which include the effect of changes in angle of attack with u to maintain Z force equilibrium.

Two other factors contribute to the damping of the long period mode in forward flight as shown by the expression for the damping term given in Table 6.III. The drag derivative (X_u) which is always negative, produces a positive damping increment. The third term in the damping expression results in an increase or decrease in the damping of the long term oscillatory mode depending on the sign of the angle of attack derivative, M_w. The rotor system makes an unstable or positive contribution to this derivative but most helicopters incorporate large enough horizontal tails to make the net M_w negative, producing positive damping.

The long term mode of the helicopter involves changes in altitude (Figure 6.13), but cannot be treated as a simple exchange of kinetic and potential energy because of the changes in angle of attack involved in the motion. The positive speed derivative (M_u) of the helicopter results in pitching moments and angle of attack changes when the speed varies.

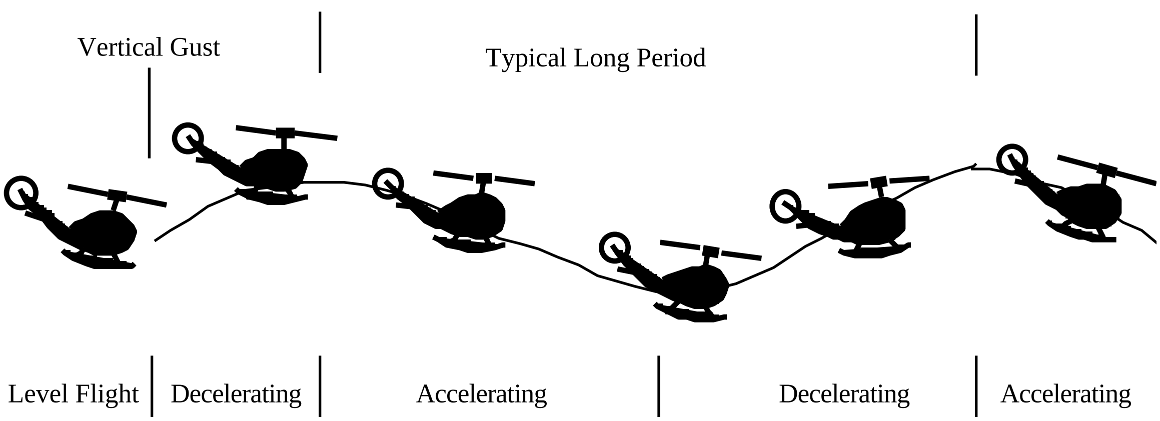


Figure 6.13
Helicopter Long Term Response

6.3.7.2 SHORT TERM DYNAMIC STABILITY

The short term response characteristics of a helicopter in forward flight influence the handling qualities in performing longitudinal maneuvering tasks. The pilot should be able to make changes in pitch attitude and flight path quickly and easily. Ordinarily, the speed remains essentially constant while the pilot makes short term pitch attitude changes. Consequently, changes in forward speed are neglected in discussing the short term response. This approximation is used with slower short term responses but in this case, neglecting speed changes is less valid.

Table 6.IV indicates the variables involved in the dynamics of the short term responses in forward flight. They include pitch angle (θ), pitch rate $\left(q = \dot{\theta}\right)$, z velocity component $\left(w = u_0\alpha\right)$, flight path angle (γ), incremental load factor (Δn) and the longitudinal cyclic pitch angle (B_{1s}).

The incremental normal load factor in dynamic maneuvers is proportional to the rate of change of the flight path angle:

$$\Delta n = \frac{1}{g} u_0 \dot{\gamma} = -\frac{1}{g}\left(\dot{w} - u_0 q\right) = -\frac{1}{g} u_0 \left(\dot{\alpha} - \dot{\theta}\right)$$

eq 6.33

Where:

$\dot{\alpha}$	- Time rate of change of angle of attack
g	- Gravity
$\dot{\gamma}$	- Time rate of change of flight path angle
n	- Normal acceleration, Normal load factor
q	- Angular velocity about y axis, Pitch rate
$\dot{\theta}$	- Rate of change of pitch angle
u_0	- Initial velocity
$\dot{w}$	- Linear acceleration along z axis.

The angle of attack is constant $(\dot{\alpha} = 0)$ in a steady pull up so the above expression is $\Delta n = 1/g\ u_0 q$ as used in discussing longitudinal control position versus g gradients.

Table 6.IV

Analogy of Short Term Response and Spring Mass Damper

First Order Coupled Equations of Motion:

$$\dot{w} - u_0\, q = \frac{\text{Z force}}{m} = Z_w\, w + Z_{B_{1s}}\, B_{1s}$$

$$\dot{q} = \frac{\text{Pitching Moment}}{I_{yy}} = M_w\, w + M_q\, q + M_{B_{1s}}\, B_{1s}$$

Substituting for w or q Yields Uncoupled Second Order Forms:

$$\ddot{w} + \left(-Z_w - M_q\right)\dot{w} + \left(Z_w M_q - u_0 M_w\right)w = \left[Z_{B_{1s}}\dot{B}_{1s} + \left(M_{B_{1s}} u_0 - Z_{B_{1s}} M_q\right)B_{1s} \right]$$

$$\ddot{q} + \left(-Z_w - M_q\right)\dot{q} + \left(Z_w M_q - u_0 M_w\right)q = \left[M_{B_{1s}}\dot{B}_{1s} + \left(M_w Z_{B_{1s}} - Z_w M_{B_{1s}}\right)B_{1s} \right]$$

Comparison of Spring Mass Damper and Helicopter Response:

Spring Mass Damper	Helicopter Response
x	w or q
$\frac{B}{m}$	$\left(-Z_w - M_q\right)$
$\frac{k}{m}$	$\left(Z_w M_q - u_0 M_w\right)$
F_x	$\left[Z_{B_{1s}}\dot{B}_{1s} + \left(M_{B_{1s}} u_0 - Z_{B_{1s}} M_q\right)B_{1s} \right]$ or $\left[M_{B_{1s}}\dot{B}_{1s} + \left(M_w Z_{B_{1s}} - Z_w M_{B_{1s}}\right)B_{1s} \right]$

The second order equations are obtained by eliminating w or q from the first order coupled equations. The pitching moment equation is differentiated and the above expression is used to eliminate w.

$$\ddot{q} + \left(-Z_w - M_q\right)\dot{q} + \left(Z_w M_q - u_0 M_w\right)q = M_{B_{1s}} \dot{B}_{1s} + \left(M_w Z_{B_{1s}} - Z_w M_{B_{1s}}\right)B_{1s} = F_q$$

eq 6.34

Where:

B_{1s}	- Longitudinal cyclic pitch angle, shaft referenced
F_q	- Pitch input
$M_{B_{1s}}$	- Pitch moment due to longitudinal cyclic pitch angle
M_q	- Pitch moment due to pitch rate
M_w	- Pitch moment due to vertical velocity
q	- Angular velocity about y axis, Pitch rate
$\ddot{q}$	- Second time derivative of pitch rate
$\dot{q}$	- Angular acceleration about y axis
u_0	- Initial velocity
$Z_{B_{1s}}$	- Vertical force due to longitudinal cyclic pitch angle
Z_w	- Vertical force due to vertical velocity.

The short term response of the helicopter, like the response of the spring mass damper system, is aperiodic if the characteristic roots are real and is oscillatory if the roots are complex numbers. The solution for the two roots of the short term mode are as follows:

$$\lambda_{1,2} = -\left(\frac{-Z_w - M_q}{2}\right) \pm \left[\left(\frac{-Z_w - M_q}{2}\right)^2 - \left(Z_w M_q - u_0 M_w\right)\right]^{\frac{1}{2}}$$

eq 6.35

Where:

λ	- Characteristic root
M_q	- Pitch moment due to pitch rate
M_w	- Pitch moment due to vertical velocity
u_0	- Initial velocity
Z_w	- Vertical force due to vertical velocity.

If $\left(\frac{-Z_w - M_q}{2}\right)^2 > \left(Z_w\ M_q - u_0\ M_w\right)$, λ_1 and λ_2 are real roots and the time response is a combination of first order (non-oscillatory) exponential responses.

If the angle of attack derivative is negative (stable) and large enough for $\left(\frac{-Z_w + M_q}{2}\right)^2 < \left(Z_w\ M_q - u_0\ M_w\right)$, λ_1 and λ_2 are a complex conjugate pair and the time response is a sinusoidal response. This condition is satisfied with a horizontal tail sufficiently large to compensate for the unstable contribution of the rotor to the angle of attack derivative. The vertical damping derivative (Z_w) and the pitch damping derivative (M_q) of the helicopter are inherently negative.

When the angle of attack derivative M_w is positive (unstable), the two roots of the characteristic equation are real numbers indicating aperiodic responses. One of these component responses is convergent or divergent depending on the sign of ($Z_w\ M_q$ - u_0 M_w).

The parameter, ($Z_w\ M_q$ - $u_0\ M_w$), might be termed the maneuvering stability level (MSL). When this parameter is positive, both stability roots are negative and a stable response is obtained. However, if the MSL parameter is negative, one of the real roots is positive and yields an exponential divergent short term response.

The factor Z_wM_q is always positive because both Z_w and M_q are negative quantities. Consequently, the MSL parameter is negative when the equivalent angle of attack stability derivative ($M_\alpha = u_0M_w$) is positive (destabilizing) and greater than Z_wM_q. The nature of the short term response is related to the MSL parameter which in turn depends on M_α = u_0M_w and, to a large extent, the size of the pitch damping derivative, M_q.

When the pilot introduces longitudinal cyclic inputs, the TPP tilts causing pitching moments and Z forces which excite the short term response of the helicopter. Figure 6.14 shows the pitch rate time histories due to a step change in cyclic pitch for configurations with different values of the MSL parameter. The responses are for step cyclic pitch inputs at 100 kn for a configuration without flapping hinge offset or stability augmentation.

Figure 6.14 (c) is based on derivatives for the AH-lG helicopter with Stability Control Augmentation System (SCAS) OFF. It has a stable angle of attack derivative, $M_w < 0$; the unstable contribution of the rotor is overbalanced by the effect of the normal tail. Figure 6.14 (b) assumes a smaller tail, $M_w = 0$. For Figure 6.14 (a), $M_w > 0$ and MSL = zero. The damping derivative is assumed constant for Figure 6.14 (a), (b), and (c) although it is recognized a change in tail size affects M_q.

Case	Z_w	M_q	M_w	$MSL = Z_w M_q - u_0 M_w$	
(a)	- 0.991	-0.385	0.00226	0	Neutral Stab
(b)	- 0.991	-0.385	0	0.381	Small Tail
(c)	- 0.991	-0.385	-0.0058	1.36	Normal Tail
(d)	- 0.991	-1.37	0	1.36	High Damping

All the pitch responses are normalized to the theoretical steady state pitch rate for case (b) of Figure 6.14; although this pitch rate is not reached within the 3.5 s time period in which forward speed is approximately constant. The steady state rates are inversely proportional to MSL parameters, and depend slightly on the change in effective control sensitivity with M_w.

When the pitching moment is independent of angle of attack, the pitch rate response in constant speed forward flight is independent of the w response and similar to first order hover response (Figure 6.14 (b)). The time constant for Figure 6.14 (b) (without stability augmentation) is 2.6 s, yielding a first order response with a long time constant. The condition for an oscillatory response $\left(\frac{-Z_w + M_q}{2}\right)^2 < \left(Z_w M_q - u_0 M_w\right)$ is satisfied with $M_w = -0.0058$. Therefore, a second order pitch rate response with some overshoot is obtained.

Figure 6.14 (d) presents a configuration with a small tail giving $M_w = 0$ as in Figure 6.14 (b). However, it is assumed the effective M_q is increased by augmentation so the same MSL parameter is obtained as with Figure 6.14 (c). The response for Figure 6.14 (d) with a time constant of 0.73 s is more rapid than for Figure 6.14 (b) and is comparable to that obtained with the normal tail.

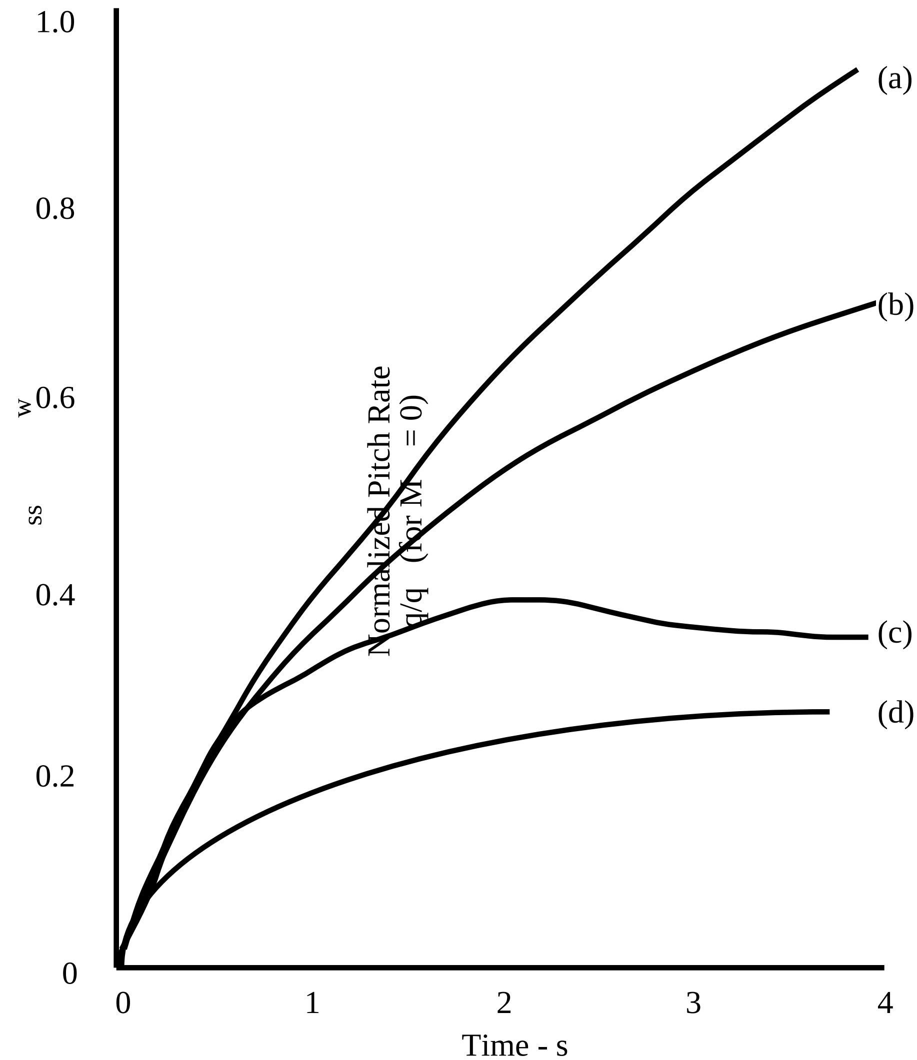


Figure 6.14
Pitch Rate to Step Cyclic Input

Thus, the effect of the M_w derivative is less pronounced when more pitch damping is present. One could augment the damping for good pitch response characteristics in hovering flight and this leads to improved pitch response in forward flight. The curve for zero MSL (Figure 6.14 (a)) shows the significance of this parameter in classifying short term response shapes. When the MSL parameter is zero, the effective stiffness term is zero in the equation for pitching velocity. Consequently, the pitch rate response to a step input approaches an acceleration command situation with a linear increase in q. Pitch rate responses with a positive MSL tend to approach steady pitch rates exponentially; while the responses with a negative MSL parameter diverge exponentially.

6.3.7.3 SHORT TERM NORMAL ACCELERATION RESPONSE

The incremental load factor, $\Delta n = (1/g)u_0\, q$, experienced by a pilot in a steady turn or pull up is proportional to u_0 and is small in low speed flight. The pitch rate, q, is a more important cue than load factor in performing low speed tasks as, for example, nap of the earth maneuvering.

However, at higher speeds, load factor developed in longitudinal maneuvers is a significant cue for the pilot, and the time history of the acceleration affects his judgment of handling qualities. Figure 6.15 presents normal acceleration time histories following a step cyclic pitch input. These correspond to the pitch rate responses shown on Figure 6.14.

Case	Z_W	M_q	M_W	$MSL = Z_W M_q - u_0 M_W$		Concave Down
(a)	- 0.991	-0.385	0.00226	0	Neutral Stab	
(b)	- 0.991	-0.385	0	0.381	Small Tail	t = 1.75 s
(c)	- 0.991	-0.385	-0.0058	1.36	Normal Tail	t = 1.2 s
(d)	- 0.991	-1.37	0	1.36	High Damping	t = 1.06 s

These responses are idealized by considering an instantaneous change in cyclic pitch due to an aft control displacement and do not include the effect of transient blade dynamics. The principal characteristics of interest are the initial load factor increment, the subsequent dip in load factor, and the portion of the response which becomes concave downward.

Aft control movement results in a positive pitch acceleration and a positive pitch rate indicated on Figure 6.14. This positive pitch rate tends to increase the helicopter's angle of attack and causes an increase in load factor. During the first one half second of the response, the decrease in load factor due to the upward w velocity is larger in magnitude than the increase caused by the pitch rate. This explains the dip in the load factor curves for this time interval (Figure 6.15).

This initial decrease in load factor response does not typically cause handling qualities problems and is generally masked when gradual control inputs are used. A more important question from a handling qualities standpoint is whether or not the load factor versus time plot is divergent. If the load factor time history becomes concave downward (for aft stick input), the response is not divergent. Extensive work by NASA led to the handling quality criterion which states the load factor response curve should become concave downward within two seconds after a control input.

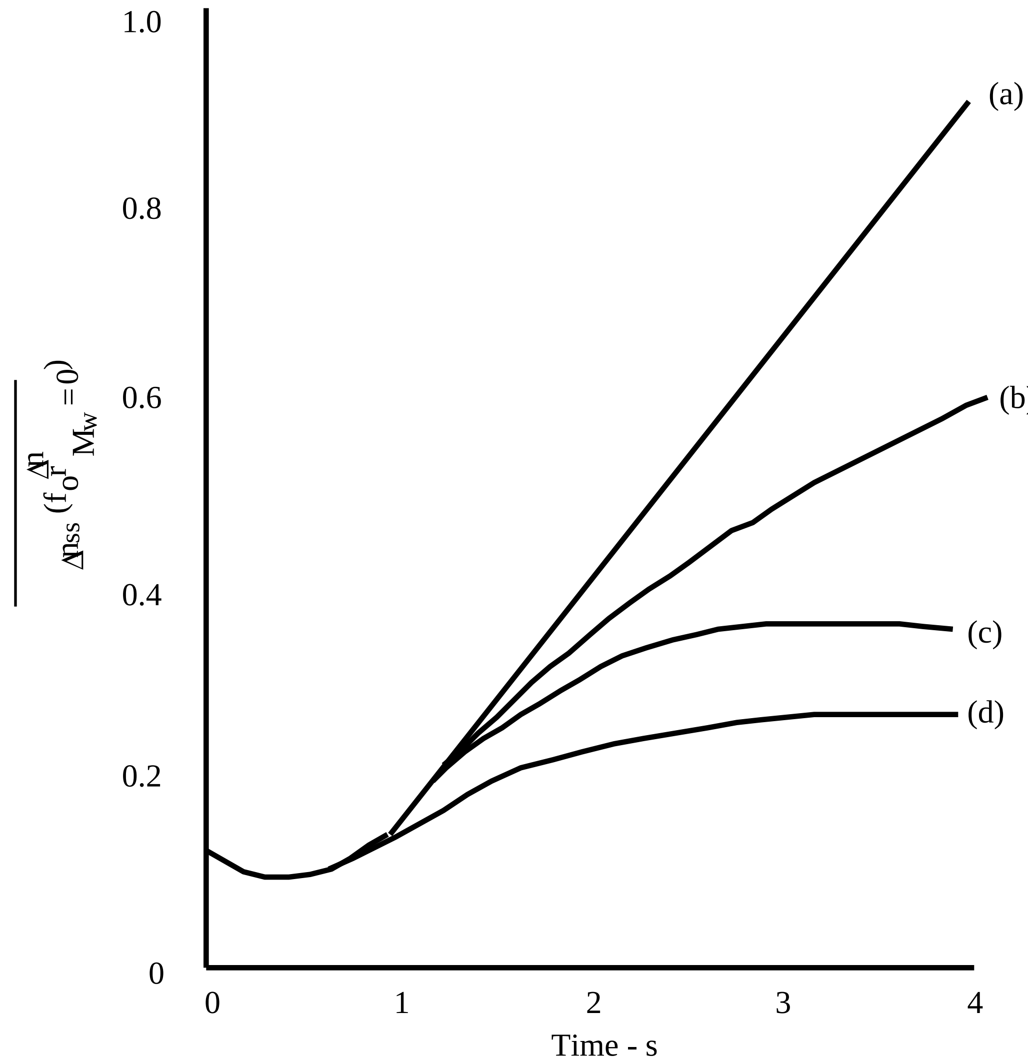


Figure 6.15
Normal Acceleration Response to Step Input in Cyclic Pitch

Figure 6.15 (b) becomes concave downward at t = 1.75 s and meets the NASA requirement although $M_W = 0$. Figure 6.15 (c) becomes concave downward even earlier at time t = 1.2 s. On the other hand, the curve for the zero MSL case (Figure 6.15 (a)) never becomes concave downward and the normal acceleration diverges with time. Figure 6.15

(d) is for the same MSL value as for curve (c) but with $M_w = 0$ and a more negative pitch damping derivative. The normal load factor response curve for this case becomes concave downward at t = 1.06 s.

The preceding discussion of the short term response started by assuming the u velocity component remains constant and no coupling exists with lateral and directional degrees of freedom. Analysis of the complete system requires consideration of the additional degrees of freedom with their corresponding equations of motion and results in additional characteristic frequencies and natural response modes. However, for the more complete case, the general motions for small disturbances from trim condition can be described by considering several elemental first and second order spring mass damper models. The time history of each of the flight variables resulting from a control input is the sum of several terms, each proportional to the response in one of the spring mass damper models used to represent the helicopter.

6.3.8 Control Response

Helicopter response to cockpit control inputs is termed control response. The short and long term responses of the helicopter are key ingredients of both control response and gust response. It suffices here to list those characteristics which are additional when considering pilot inputs. Helicopter designs include increasing use of augmentation and digital flight controls. The effects of these systems should be considered in the total control response.

Feed forward control loops may include the control boost system. The feedback loop may consist of pure gain on a response feedback signal such as pitch rate or both gain and dynamics from filters, etc. All elements directly in the control signal path such as the force feel system, any pertinent nonlinearities and stabilization loop feed forward elements affect the total control response directly. Elements in this path are sometimes called control augmentation. On the other hand, elements within the stabilization loop function as stability augmentation. An evaluation of the adequacy of control response in the context of operational tasks, therefore, implicitly includes a consideration of the effects of all pilot command path elements, in addition to the basic or stability augmented dynamics of the helicopter. The evaluation considers predictability of the total control response as well as adequate control sensitivity.

6.3.9 Pilot Induced Oscillation

Some helicopters exhibit pilot induced oscillation (PIO) tendencies when carrying out mission related pitch tasks requiring tight closed loop control. For example, in a conventional gun run task, the pilot uses the short term response to bring the aircraft into alignment with the target. This is accomplished by a series of short rapid inputs to correct for errors in attitude.

If the pilot makes a correction to bring the nose on target and the short term response and control system combination is such that the harder he tries to zero the tracking error, the poorer his closed loop performance and more oscillatory than the open loop response, he is in a PIO. If he persists, he is worse off than if he backed off in the closed loop. The existence of PIO tendencies is examined by the test pilot throughout the operating envelope.

6.3.10 Gust Response

A distinction can be made between the gust and short term responses of the helicopter. Pitch attitude and pitch rate changes are usually considered in characterizing the short term response. Although the pilot uses the short term response mode to make corrections for pitch attitude or rate changes resulting from encountering turbulence, this is not the complete gust response.

The evaluation of the gust response amounts to a subjective pilot judgment of the helicopter ride qualities when flying through turbulence. This judgment is influenced by the aircraft linear and angular accelerations due to turbulence and the location of the pilot relative to the CG. The time variations of these quantities influence the judgment of ride qualities.

The aerodynamic forces due to turbulence simultaneously excite responses of the rotor blade and fuselage structural modes as well as the short term and other rigid body modes. The pilot's judgment is based on the total perceived response.

Other factors involved in evaluating the gust response are how easily the aircraft is disturbed from trim, its tendency to return or depart from trim when disturbed, and the pilot compensation required to maintain a desired flight path while flying through the turbulence. These factors depend on the static stability, dynamic stability, and control sensitivity of the aircraft as well as any stabilization systems.

The real test for the helicopter ride qualities is obtained by flying through actual turbulence. However, tests are carried out without turbulence to evaluate both the aircraft response to simulated upsets as well as the ease of pilot recovery.

6.3.11 Tandem Rotor Characteristics

Longitudinal control of a tandem rotor helicopter is simpler than of the single rotor helicopter. Control pitching moment is produced through the use of differential collective blade pitch. The thrust is increased on one rotor and decreased on the other rotor to develop the desired moment. For example, the thrust of the forward rotor is reduced and the thrust of the aft rotor is increased to develop a nose down moment. Figure 6.16 illustrates longitudinal control of a tandem helicopter.

The individual rotors of tandem configurations respond to speed changes, pitch and roll rates, and angle of attack, the same as for the single rotor helicopter. However, there are important modifications in the responses because of aerodynamic interference.

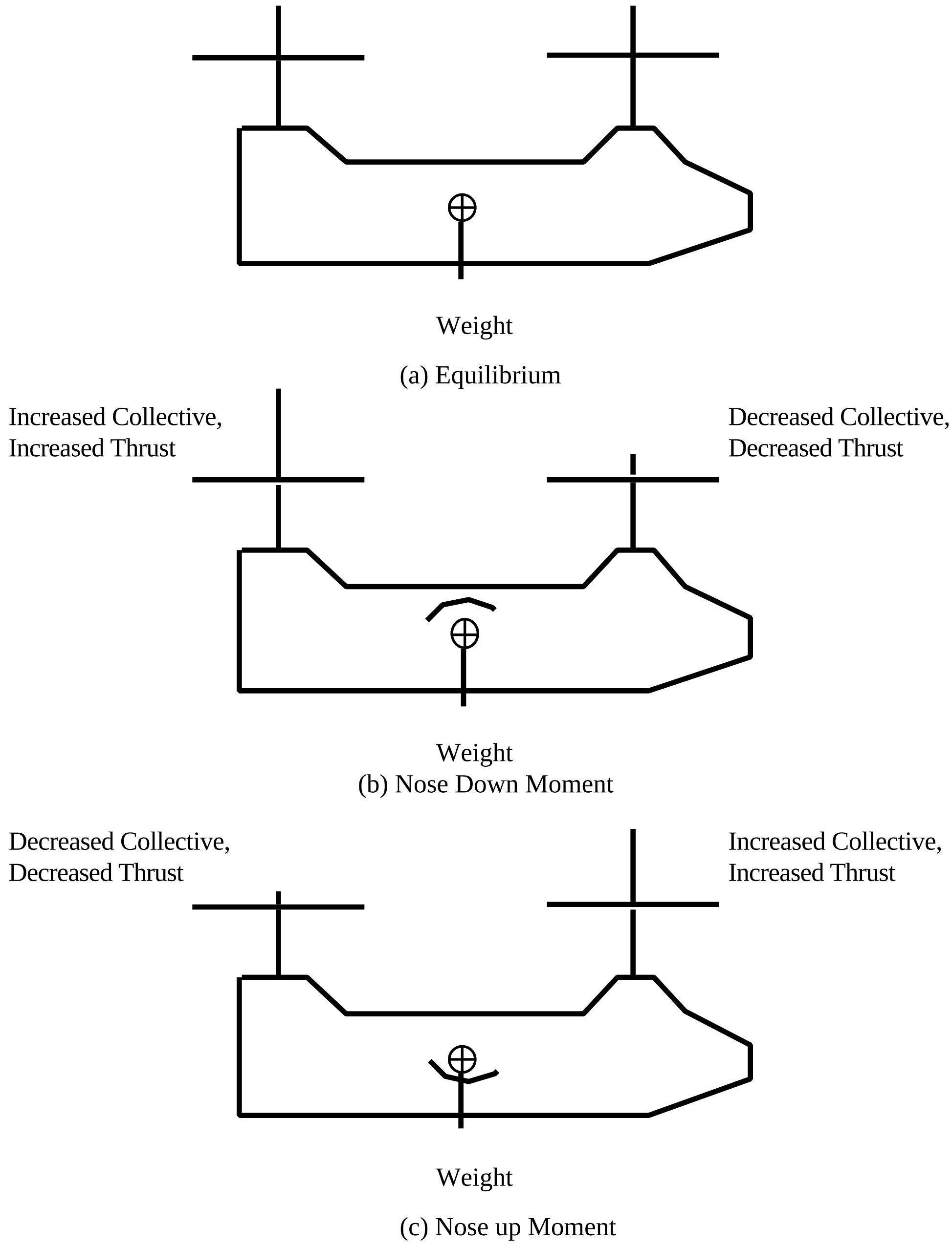


Figure 6.16
Longitudinal Control of a Tandem Rotor Helicopter

Figure 6.17 indicates some of the factors affecting the longitudinal stability of the tandem helicopter. The nature of the configuration is such that the pitching moment of inertia is much greater than a single rotor helicopter. However, this characteristic is offset by the higher control power and pitch damping of most tandem designs. The higher control power of the tandem configurations arises from the use of differential collective pitch on the front and rear rotors. Some control systems combine longitudinal cyclic pitch with differential collective pitch to obtain increased control power.

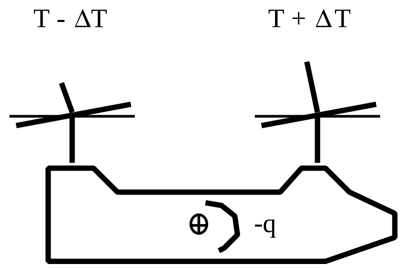
Figure 6.17 (a) indicates the major source of pitch damping of the tandem. For the case shown, the pitching velocity produces an upward increment of flow through the front rotor disc, and a downward increment of flow through the rear rotor disc. A higher average blade angle of attack of the front rotor increases its thrust; while the average blade angle of attack and thrust of the rear rotor are decreased. The damping moments about the CG produced by these changes in thrust are considerably larger than the moments due to the TPP inclination, which is the primary source of pitch damping of the single rotor helicopter.

A helicopter has positive static stability if an increase in speed results in a nose up pitching moment in opposition to the speed change. The aft tilting of the individual TPPs and thrust vectors due to a speed increment has a stabilizing effect. Unlike the single rotor helicopter, the speed stability of the tandem helicopter is affected appreciably by the CG location. The rate of change of thrust on the front and rear rotors with forward speed can be different because of differences in the trim values of collective pitch. A favorable speed stability condition is obtained with a forward CG location.

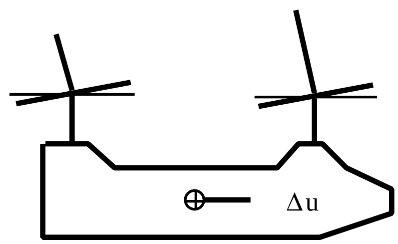
Tandem helicopters encounter some problems in handling qualities due to pitching moments which change with forward speed. In hovering, the downwash from both rotors moves straight down and there is negligible interaction. As forward speed is increased, the downwash on the forward rotor affects the flow over the rear rotor and consequently reduces its lift. This results in a nose up pitching moment as the speed is first increased. After moderate speed is gained, at an advance ratio of about 0.1, further increases in speed tend to decrease the downwash from the front rotor on the rear rotor. The rear rotor lift is increased by further speed increases. This latter condition sometimes results in an unstable change in pitching moment with speed. At high speeds the downwash becomes small enough to be unimportant from the speed stability standpoint.

Figure 6.17 (c) indicates a means to adjust the speed stability of the tandem helicopter by the use of swashplate dihedral. A dihedral of the type indicated could be obtained by appropriate rigging of the control system or by tilting of the shafts of the two rotors. For the configuration shown, a forward speed increment produces an upflow through the front rotor disc increasing the average blade angle of attack and a down flow through the rear rotor decreasing the average angle of attack of the rear rotor. The net result is an up thrust increment on the front rotor and a down thrust increment on the rear rotor giving an aft pitching moment. An outward tilting of the rotors has an opposite effect, decreasing the speed stability of the tandem helicopter. Since adjustments of the dihedral changes the speed stability, it also provides a convenient means of controlling the frequency of the long period oscillation.

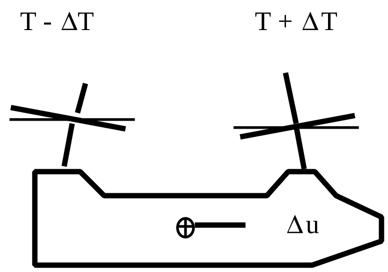
The angle of attack stability characteristics of the tandem helicopter are indicated on Figure 6.17 (d) and (e). Although a nose up increment in moment is a stable change in moment with forward speed, it represents an unstable change in moment with angle of attack. Aft tilting of the TPPs of the individual rotors with increasing pitch attitude results in unstable pitching moments similar to a single rotor configuration. These unstable moments can be compensated for by proper positioning of the CG because the angle of attack stability of the tandem helicopter, like the airplane, depends on the CG position relative to the aerodynamic center. When downwash effects are neglected, the aerodynamic center is midway between the rotor shafts. For this condition, an increase in fuselage angle of attack results in equal increments of thrust on the front and rear rotors and a resultant thrust increment acting at the aerodynamic center. This thrust increment produces a stable nose down pitching moment if the CG is forward of the aerodynamic center as indicated on Figure 6.17 (d). A second beneficial effect of a forward CG location is an unloading of the rear rotor, reducing the likelihood of rear rotor tip stalling which has a destabilizing influence.



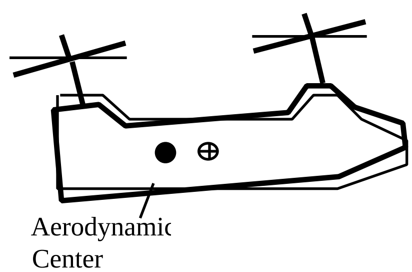
(a) Effect of Pitch Rate
Rotors tilt aft. Thrust of front rotor increased by up flow. Rear rotor thrust decreased by down flow.



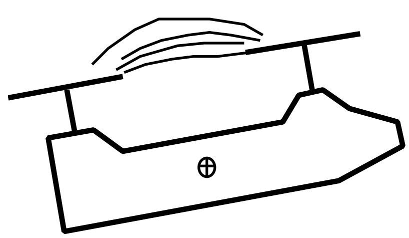
(b) Effect of Increase in Forward Velocity
(Downwash interaction neglected)
Rotors tilt aft due to positive speed stability of isolated rotor. Thrust of front rotor increased more than thrust of aft rotor.



(c) Effect of Increase in Forward Velocity with Swashplate dihedral.
Thrust of front rotor increased due to up flow. Thrust of rear rotor decreased due to down flow.



(d) Angle of Attack Stability
(Downwash interaction neglected)
Rotors tilt further aft than fuselage due to instability of isolated rotor. Total thrust increment acting through the aerodynamic center behind CG produces stable nose down moment.



(e) Effect of Downwash Interaction on Angle of Attack Stability
Smaller increase in rear rotor thrust with fuselage angle of attack due to downwash induced by front rotor resulting in a decrease in angle of attack stability.

Figure 6.17
Factors Affecting Longitudinal Stability of a Tandem Helicopter

The downwash induced by the front rotor at the rear rotor has an unfavorable effect on angle of attack stability as well as on speed stability and is largely responsible for the unsatisfactory maneuvering characteristics of some tandem configurations (Figure 6.17 (e)). An increase in front rotor angle gives a large front rotor thrust and a corresponding increase in downwash induced at the rear rotor. The average blade angle of attack of the rear rotor is changed less than the front rotor because of this downwash giving a smaller increment in rear rotor thrust. Thus, the lift curve slope of the rear rotor is less than the front rotor and this difference results in a decrease in angle of attack stability.

6.4 TEST METHODS AND TECHNIQUES

A thorough knowledge and understanding of the flight control system operation is important when designing the test conditions for flying qualities evaluations. Generally, the Stability Augmentation System (SAS) does not impact the results of the trimmed flight control positions and the static stability tests. However, if the Automatic Flight Control System (AFCS) system incorporates a device similar to the pitch bias actuator found in the SH-60 series aircraft, the AFCS can effect the trimmed flight control positions as well as the apparent static stability. In most helicopters, the AFCS does effect the dynamic and control response results. For this reason, the stability, control, and flying qualities test are flown with the AFCS fully operational in the mission representative manner. If time permits, the test can be repeated with the AFCS in alternate configurations, simulating possible pilot selected options or system degradation. Further, the tests can be performed with the AFCS secured, simulating total system failure and documenting the unaugmented helicopter responses, possibly for flight simulator development.

6.4.1 Trimmed Control Positions

The purpose of the test is an evaluation of the variation in control positions with changes in power and airspeed, the control margins, the ease of trimming the flight control forces to zero, and the qualitative evaluation of the importance of trim changes to the piloting task. The entire airspeed envelope is evaluated to V_{NE} and the complete power range is evaluated from autorotation to maximum power. For level flight, the helicopter is established in trimmed, wings level, unaccelerated flight, and the data are recorded. The airspeed is varied approximately 10 kn and the test is repeated.

For climbs and descents, one or more mission representative airspeeds are used. The helicopter is stabilized on airspeed and the power is varied from autorotation to maximum available. Airspeed remains constant while the power is varied. At least 5 power increments are used. The entire test is repeated at another airspeed. Usually, the airspeed for maximum rate of climb, or minimum rate of descent is evaluated. If large control displacements are observed when making a power change, the test can be conducted open loop with all controls fixed except for power. The resulting attitude and rate changes show the aircraft response which the pilot must counteract to maintain wings level, unaccelerated flight conditions. The results of the open loop test is used to relate the possible helicopter excursions when performing constant airspeed climbs and descents.

6.4.1.1 TEST TECHNIQUE

1. Establish wings level, unaccelerated, trimmed flight. Record the initial trim conditions.
2. Assign a Handling Qualities Rating (HQR) and Vibration Assessment Rating (VAR) to each data point.
3. Use the force trim and AFCS in a mission representative manner.

6.4.1.1.1 Level Flight

1. This data can be obtained concurrently with level flight performance testing.
2. Investigate entire airspeed profile (40 kn to V_H).

6.4.1.1.2 Diving Flight

1. If V_{NE} is greater than V_H, fix the collective at maximum power and dive the aircraft to obtain control position data at the higher airspeeds.
2. Control margins are particularly important.

6.4.1.1.3 Climbs and Descents (Open Loop)

1. Collective changes are made and the aircraft response observed without making any other control inputs.
2. Tests are conducted controls fixed and free.

6.4.1.1.4 Climbs and Descents (Closed loop)

1. Establish wings level, unaccelerated flight at the desired trim airspeed.
2. Vary power incrementally from autorotation to the maximum available.
3. Normally five increments define a satisfactory curve.
4. Altitude range, ±1000 ft about target.

6.4.1.2 DATA REQUIRED

Control positions, V_o, H_{P_o}, θ, ϕ, Q, N_R, fuel counts (FC), HQR, and VAR.

6.4.1.3 TEST CRITERIA

1. Stabilized, wings level, ball centered, unaccelerated flight.
2. No vertical velocity (level flight).
3. Constant vertical velocity (climb and descent).
4. All control forces trimmed to zero.

6.4.1.4 DATA REQUIREMENTS

1. Stabilize 15 s prior to recording data.
2. Record 10 s data.
3. H_{P_o} ± 1000 ft of the target test altitude.
4. V_v, level flight, ±10 fpm, climb and descent, stabilized ± 25 fpm.
5. N_R ± 1%.

6.4.1.5 SAFETY CONSIDERATIONS/RISK MANAGEMENT

Observe normal operating limitations. Use incremental build up procedures when approaching any envelope boundary. Conduct tests at a safe operating altitude over terrain suitable for forced landings. Maintain proper lookout procedures. Consider using a chase aircraft and sole use airspace.

6.4.2 Static Stability

The purpose of this test is to evaluate the static stability or the change in the total longitudinal moment generated by off trim airspeed changes. Static stability is the initial tendency of an aircraft to return or to depart from equilibrium if disturbed. Static stability is indicated by the variation of the longitudinal control position with airspeed. Positive static stability is indicated by increasing forward longitudinal control displacement with increasing airspeed. Since the fuselage/rotor system forces and moments can not be measured directly in flight, the purpose of this test is to determine the control positions and forces necessary to balance the pitching moments caused by airspeed variations from trim.

The test is accomplished by establishing a wings level, ball centered, unaccelerated trim condition. Without changing the collective position, trim settings, or rotor speed, the airspeed is varied and stabilized at incremental airspeed faster and slower then trim. The airspeed around trim is most important. Small 2 to 3 kn increments are used around trim. As airspeed is varied from trim, larger 5 kn increments are used. At each stabilized flight condition, lateral and directional controls are used to maintain wings level, ball centered flight. Collective is maintained constant. A stabilized climb or descent is accepted and altitude is maintained within ± 1000 ft of the test altitude. Alternating test points faster and slower than trim are used to remain within the test altitude band. If power must be changed to return to the test altitude, record all engine parameters and collective control position. Do not retrim the collective. When the test altitude is reestablished, return the collective control to trim.

The airspeed increments are arbitrary. It is not important to establish precisely a 3 or 5 kn increment. It is important to be in stabilized flight. A suggested procedure is to make maximum use of attitude flying by reference outside the cockpit. A good instrument scan is necessary. From the initial trim position, displace the longitudinal control forward, decreasing the pitch attitude slightly. As the helicopter accelerates, observe the pitch attitude. As it begins to change, apply forward control (Assuming positive static stability) to maintain a constant pitch attitude. When airspeed stabilizes, establish equilibrium flight conditions. Do not try to chase the airspeed indicator. Attempt to move the control in one direction to avoid operating through any force hysteresis band.

6.4.2.1 TEST TECHNIQUE

1. Stabilize at the initial trim airspeed and reduce all control forces to zero. Do not retrim control forces or adjust engine controls during the test.
2. Record the trim data.

3. Stabilize at airspeeds slower and faster than the trim airspeed. Do not retrim control forces or adjust the collective and engine controls. Generally airspeed variations of ±20 kn in 5 kn increments is acceptable. However, use ± 2 kn increments close to the trim airspeed.

4. Stabilize at each speed increment for 15 s and record data for approximately 10 s. Make qualitative comments on the force and displacement cues at the off trim conditions.

5. Hold attitude constant once a desired speed change is achieved. Pitch attitude change provides a cue to off trim conditions.

6. Repeat the procedure until the airspeed range is completed. Alternate airspeeds faster and slower than trim to remain in the altitude band.

7. If it is necessary to adjust altitude to remain within the test band, note the collective position, torque, and N_g, before moving the collective. Do not retrim any control forces. The test is continued after the collective control is returned to the original trim position.

6.4.2.2 DATA REQUIRED

Longitudinal control position, longitudinal control force, V_o, V_v, θ, Q, N_R, FC, H_{P_o}, T_o.

6.4.2.3 TEST CRITERIA

1. Wings level, ball centered, unaccelerated flight.
2. Collective fixed.
3. No retrimming.

6.4.2.4 DATA REQUIREMENTS

1. Stabilize 15 s prior to taking data.
2. Record 10 s stabilized data.
3. $V_o \pm 1$ kn.
4. Wings level, $\phi \pm 1^\circ$.

5. Ball centered.
6. H_{P_o} ± 1000 ft of the target test altitude.
7. N_R ± 0.5%.

6.4.2.5 SAFETY CONSIDERATIONS/RISK MANAGEMENT

Observe normal operating limitations. Use incremental build up procedures when approaching any envelope boundary. Conduct tests at a safe operating altitude over terrain suitable for forced landings. Maintain proper lookout procedures. Consider using a chase aircraft and using sole use airspace.

6.4.3 Maneuvering Stability

The maneuver stability test is a variation on the static longitudinal stability test. In both cases, longitudinal control inputs are used to stabilize the aircraft at off trim conditions. The static stability evaluation is conducted at 1.0 g and the airspeed is varied. In maneuvering stability tests the trim airspeed is maintained and the normal acceleration is varied. The maneuvering stability or pitching moment change with change in load factor is determined from the variation of the longitudinal control position and control force with normal acceleration obtained in stabilized conditions with constant airspeed and collective.

Three test techniques are used to cover the load factor envelope of the aircraft: steady turns, symmetrical pull-ups, and symmetrical pushovers. Symmetrical pull-ups and symmetrical pushovers encompass both the target cyclic and target “g” test methods.

6.4.3.1 TEST TECHNIQUE

6.4.3.1.1 Steady Turns

1. Stabilize in ball centered, coordinated level flight at the desired trim airspeed.
2. Reduce all control forces to zero and do not retrim the forces during the tests. Fix the collective position.
3. Record trim data.

4. Establish the desired bank angle, stabilizing on trim airspeed with a constant g. Do not retrim the control forces and maintain fixed collective. Accept any resultant climb or descent. Start the test at an altitude which permits a stabilized condition within the test altitude band.

5. Record data when the airspeed, bank angle, and normal acceleration are stabilized.

6. Repeat at incrementally increasing bank angles (15°, 30°, 45°, 60°) until the maximum load factor or a predetermined flight limit is reached. These are target bank angle, exact bank angles are not required. Evaluate turns in both directions.

7. Note collective position, torque and $N_{g,}$. Do not retrim the collective. Climb if required to stay in the altitude band and reset collective to the trim position before continuing the test.

6.4.3.1.2 Symmetrical Pull Up

The symmetrical pull up test technique requires considerable practice and proper timing to yield satisfactory results. Exercise caution with this method because an aircraft with neutral or unstable maneuvering stability characteristics continues to increase g when the longitudinal control is held against the control fixture.

1. Stabilize in ball centered, coordinated level flight at the desired trim airspeed.

2. Reduce all control forces to zero and fix the collective, noting its position.

3. If a control fixture is used, establish the fixture increment and place the fixture behind the cyclic. Start with a small increment which corresponds to a low value of normal acceleration.

4. Decelerate below the trim airspeed.

5. Lower the nose below the trim attitude to accelerate toward the trim airspeed.

6. Prior to reaching the trim airspeed, rapidly displace the longitudinal control and hold the control against the fixture. If a fixture is not used, displace the cyclic to obtain the desired g level.

7. The aircraft must pass through the trimmed attitude, wings level, stable g, and on trim airspeed.

8. Voice procedures for cockpit coordination are as follows: "Data on. Stand-by for a (displacement magnitude), (direction of input), (control) input on three. Thousand one, thousand two, thousand three." The input is executed on three. Example, "Data on, stand-by for a one inch aft longitudinal control input on three. Thousand one, thousand two, thousand three."

9. Turn on the automatic data recording prior to setting the fixture.

10. Use incremental increases in cyclic displacement (fixture increment) and repeat the procedure until the desired normal acceleration range is obtained or a predetermined limit is reached.

11. As the displacement and g level increase, it is necessary to decelerate more and use a lower nose attitude during the acceleration phase. This technique requires practice and crew coordination. Satisfactory results can be obtained through practice.

12. Announce the recovery to allow the copilot to remove the fixture.

6.4.3.1.3 Symmetrical Push Over

The symmetrical push over test technique requires considerable practice and proper timing to yield satisfactory results. Exercise caution with this method because an aircraft with neutral or unstable maneuvering stability characteristics continues to decrease g if the longitudinal control is held against a fixture.

1. Stabilize in wings level, ball centered, coordinated level flight at the desired trim airspeed.

2. Reduce all control forces to zero and fix the collective.

3. If a control fixture is used, establish the fixture increment and place the fixture in front of the cyclic. Start with a small increment which corresponds to a small change of normal acceleration.

4. Accelerate to an airspeed faster than trim.

5. Raise the nose above the horizon and allow the airspeed to decrease toward trim.

6. Prior to reaching the trim airspeed, rapidly displace and hold the cyclic against the fixture. If a fixture is not used, displace the cyclic to obtain the desired g level.

7. The aircraft must pass through horizontal with wings level, steady g, and on trim airspeed.

8. Increase the cyclic displacement (fixture increment) and repeat the procedure until the desired normal acceleration range is obtained or a predetermined limit is reached.

9. As the displacement and g level increase it becomes necessary to accelerate to a higher airspeed, and use a more nose high attitude during the deceleration phase.

10. Use cockpit voice procedures similar to pull ups for crew coordination.

6.4.3.2 DATA REQUIRED

Longitudinal control position, longitudinal control force, g, V_o, ϕ for steady turn method, V_v, θ, FC, H_{P_o}, T_o, Q, N_R.

6.4.3.3 TEST CRITERIA

1. Ball centered, balanced flight.
2. Collective fixed.
3. Constant bank angle.
4. Constant airspeed.
5. Data valid only as aircraft passes through the trimmed pitch attitude with constant g and control displacement for pull up and push over methods.

6.4.3.4 DATA REQUIREMENTS

1. Stabilize 15 s prior to taking data.
2. Record 10 s stabilized data.
3. Load factor, ± 0.05 g.
4. $V_o \pm 2$ kn.
5. $\phi \pm 2^\circ$.
6. Ball centered.
7. $H_{P_o} \pm 1000$ ft of the target test altitude.
8. $N_R \pm 1.0\%$.

6.4.3.5 SAFETY CONSIDERATIONS/RISK MANAGEMENT

Observe normal operating limitations. Use incremental build up procedures. Conduct test at a safe operating altitude over terrain suitable for forced landings. Maintain proper lookout procedures. Consider using a chase aircraft and sole use airspace. If a chase is used, ensure adequate separation is maintained throughout these maneuvers. Consider positioning the chase at the minimum altitude of the test band for low altitude recovery radio calls.

Maintain situational awareness. Wear parachutes when conducting maneuvering stability tests, particularly if aircraft limits are expected to be approached. The crew must be trained in the proper use and care of these devices.

Control fixtures can be as much of a hazard as they are a help. Brief, coordinate, and practice the use of control fixtures. There have been instances in which the control fixture was not properly removed at the appropriate time and/or the fixture became jammed in the control system, causing an exciting recovery from the maneuver. Preparation is the key.

6.4.4 Long Term Dynamic Stability

The purpose of this test is to evaluate the helicopter's long term airspeed and altitude variations as a function of time. The long term response is a nuisance mode of motion which must be suppressed by the pilot or Automatic Flight Control System (AFCS). Tests include a qualitative determination of the difficulty of suppressing the long term response, and conversely the ease of excitation. The test consists of recording the response of the aircraft to off trim conditions resulting from turbulence or disturbances introduced by the pilot. The test is conducted to determine the response in the longitudinal axis. Use the lateral and directional controls to obtain a single axis response. If the workload in the off axis is high or the aircraft is flown hands off, obtain a control fixed and free response.

6.4.4.1 EXCITATION METHODS

1. No conscious excitation. Controls fixed without pilot inputs. Imperfect trim conditions or atmospheric disturbance may excite a lightly damped mode. This is an unsatisfactory excitation technique if no response is obtained, but indicates a desirable aircraft characteristic.

2. Use the longitudinal control to accelerate or decelerate to an airspeed faster or slower than trim. Smoothly return the control to the original trim position and record the aircraft motion. An off trim airspeed variation of 5 to 15 kn normally is used to excite the motion.

3. The method of excitation can determine the type of response documented. A large input may cause the helicopter to reach a test limit, necessitating a recovery, without observing the entire response. Natural disturbances which result in a long term response are desirable, but these responses are usually contaminated by another disturbance before the motion is completed. This makes quantifying the mode of motion difficult.

4. Artificial methods are used to obtain time histories from which the engineering data are obtained. The excitation method chosen should result in an aircraft response similar to a response following a natural disturbance. For example, Figure 6.18 shows responses which result from airspeed deviations of different magnitudes. Figure 6.18 (a) shows an oscillatory divergent response obtained with a small excitation. Less than a complete cycle of oscillation is obtained with greater excitation in Figure 6.18 (b). The initial airspeed deviation used for Figure 6.18 (c) may be larger than obtained with a representative horizontal gust and the response for this case appears to be aperiodic.

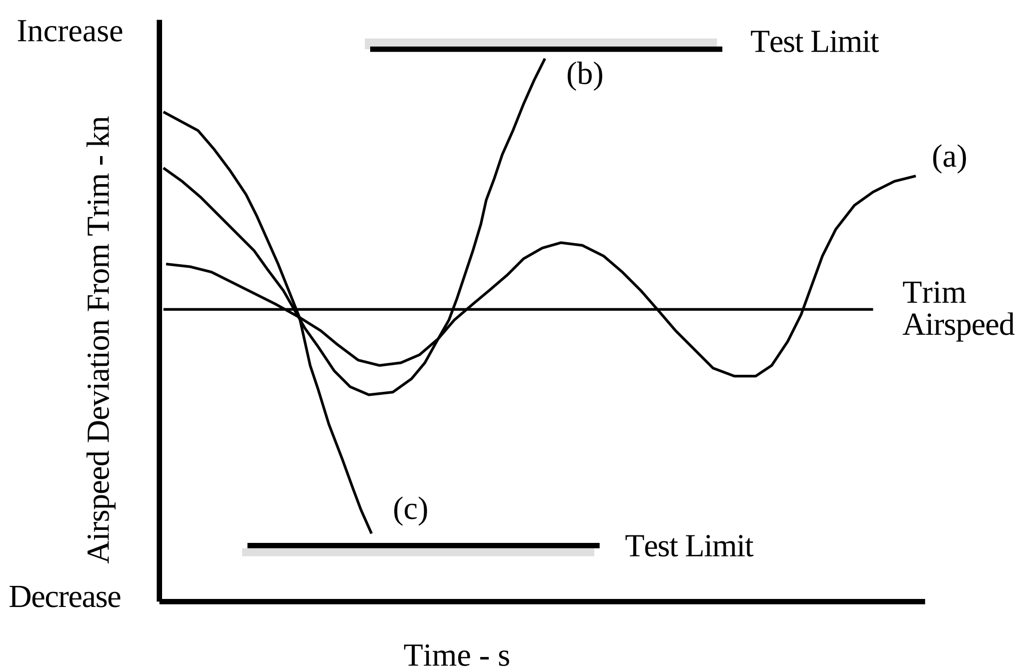


Figure 6.18
Time Histories Produced by Various Levels of Excitation

6.4.4.2 TEST TECHNIQUE

1. Stabilize at the desired trim airspeed (these tests can be conducted at trimmed conditions other than level flight), and reduce all control forces to zero. Do not retrim control forces or move the collective during the test.
2. Record trim conditions. Assure the ability to return the control to trim after exciting the response.
3. Determine if a long term response results from a natural disturbance. With the controls either fixed or free, note the open loop aircraft response. If no aircraft response is observed, an artificial excitation is used.
4. Excite the aircraft using an artificial input. It may be necessary to try several different inputs to find the most representative aircraft response Do not retrim any control forces and keep the collective constant during the response.
5. If the aircraft is flown hands off, obtain control free responses following the excitation. The controls are released at trim so the subsequent control motions indicate the effect of attitude changes and gravity force acting during the response.
6. Record the resulting mode of motion using cockpit data and automatic recording systems. Cockpit displayed airspeed or altitude is recorded at selected time increments. Start the stop watch at the completion of the excitation or at a predetermined airspeed. The zero time reference point is arbitrary. Use small enough time intervals to define the shape of irregular responses.
7. The resulting mode of motion may not be a classical single axis response. Use lateral cyclic and directional control as necessary to maintain a single axis response for the initial excitation. If you find the lateral-directional inputs are significant, record a time history of the aircraft response with all controls fixed.
8. For hand held data, a plot of airspeed at regular intervals (5 to 10 s) adequately defines the response. An alternate method is to record peak (maximum and minimum) airspeeds and determine the period by timing the response as the helicopter passes through the trim airspeed.

6.4.4.3 DATA REQUIRED

V_o, H_{P_o}, V_v, θ, FC, T_o, Q, N_R, cockpit control positions.

Following the excitation, record airspeed, altitude, or pitch attitude as a function of time. Use automatic recording systems to simplify this task.

6.4.4.4 TEST CRITERIA

1. Ball centered, balanced flight.
2. Longitudinal control fixed (at trim) and/or free.
3. Collective fixed.

6.4.4.5 DATA REQUIREMENTS

1. Stabilize 15 s prior to taking data.
2. Trim $V_o \pm 2$ kn.
3. $H_{P_o} \pm 1000$ ft of the target test altitude.
4. $\phi \pm 2^\circ$.
5. Ball centered.

6.4.4.6 SAFETY CONSIDERATIONS/RISK MANAGEMENT

Observe the normal operating limitations. A thorough knowledge of airspeed and attitude limitations is essential. Recovery must be initiated in sufficient time to prevent limit overshoot. Conduct tests at a safe operating altitude over terrain suitable for forced landings. Maintain proper lookout procedures. Consider using a chase aircraft and sole use airspace. Wear parachutes.

6.4.5 Short Term Dynamic Stability

The longitudinal short term mode is the mechanism used to make pitch attitude changes. Only small pitch attitude changes from unaccelerated flight conditions are classified under short term dynamic stability. Short term longitudinal responses resulting from gust inputs are evaluated and discussed under the classification of gust responses. The purpose of the test is to determine the characteristics of the short term response due to longitudinal control inputs and to evaluate the pilot's ability to make small attitude changes.

6.4.5.1 EXCITATION METHODS

The following excitation methods are used for longitudinal short term dynamic tests:

1. Pulse and doublet type cyclic inputs are used to evaluate the short term response.
2. Doublet inputs minimize inadvertent excitation of the long term response. A pulse input causes an attitude and airspeed change which tends to initiate a long term response and contaminate the short term response data.
3. A frequency sweep is accomplished first by applying sinusoidal inputs of constant amplitude at a series of frequencies and observing the pitch response. The short term frequency is found by starting with a low frequency input and increasing the input frequency while comparing the pitch attitude response. The short term frequency is indicated by a maximum amplitude of the pitch response to the inputs. Data are taken using doublets at the noted short term frequency with sufficient amplitude to disturb the pitch attitude.
4. Mission suitability tasks, such as changing the gun-target line, are evaluated using the required closed loop inputs.

6.4.5.2 TEST TECHNIQUE

1. Establish trim conditions and record trim data.
2. Conduct a frequency sweep at the trim airspeed to determine the short term frequency.
3. Make a doublet input at this frequency and record a time history of the pitch attitude response. Control fixtures and control position indicators are helpful for making good doublet inputs.
4. Investigate the ease or difficulty in making small, precise pitch attitude changes. The attitude indicator and/or grease pencil marks on the windscreen are helpful. Determine if pilot corrections are required for overshoots in pitch, which might compound problems of establishing a target line.
5. Many helicopters are well damped. Do not waste time trying to document a problem which does not exist.

6.4.5.3 DATA REQUIRED

V_o, H_{P_o}, V_v, θ, FC, T_o, Q, N_R, cockpit control positions. Time history of pitch attitude response to the excitation.

6.4.5.4 TEST CRITERIA

1. Wings level, ball centered, balanced flight.
2. Collective fixed.

6.4.5.5 DATA REQUIREMENTS

1. Stabilize 15 s prior to taking data.
2. Trim $V_o \pm 2$ kn.
3. $H_{P_o} \pm 1000$ ft of the target test altitude.

6.4.5.6 SAFETY CONSIDERATIONS/RISK MANAGEMENT

Observe normal operating limitations. Conduct tests at a safe operating altitude over terrain suitable for forced landings. Maintain proper lookout procedures. Maintain situational awareness. Consider using a chase aircraft and sole use airspace. Wear parachutes.

A thorough knowledge of airspeed and attitude limitations is essential. Recovery must be initiated in sufficient time to prevent exceeding a test limit. The use of control fixtures must be thoroughly briefed and practiced.

6.4.6 Control Response

Static and maneuvering stability tests are primarily concerned with evaluating the stability characteristics which keep the helicopter at a trim condition and the cues provided to the pilot of an off trim condition. Control response testing involves evaluating the aircraft response caused by rotor system moments generated from cyclic, pedal, and collective inputs. The purpose of these tests is to evaluate the aircraft response to flight control input. The tests are conducted by applying control step inputs of incrementally increasing size against a fixture and recording a time history of the aircraft response. Definitions of control response terms are shown on Figure 6.19.

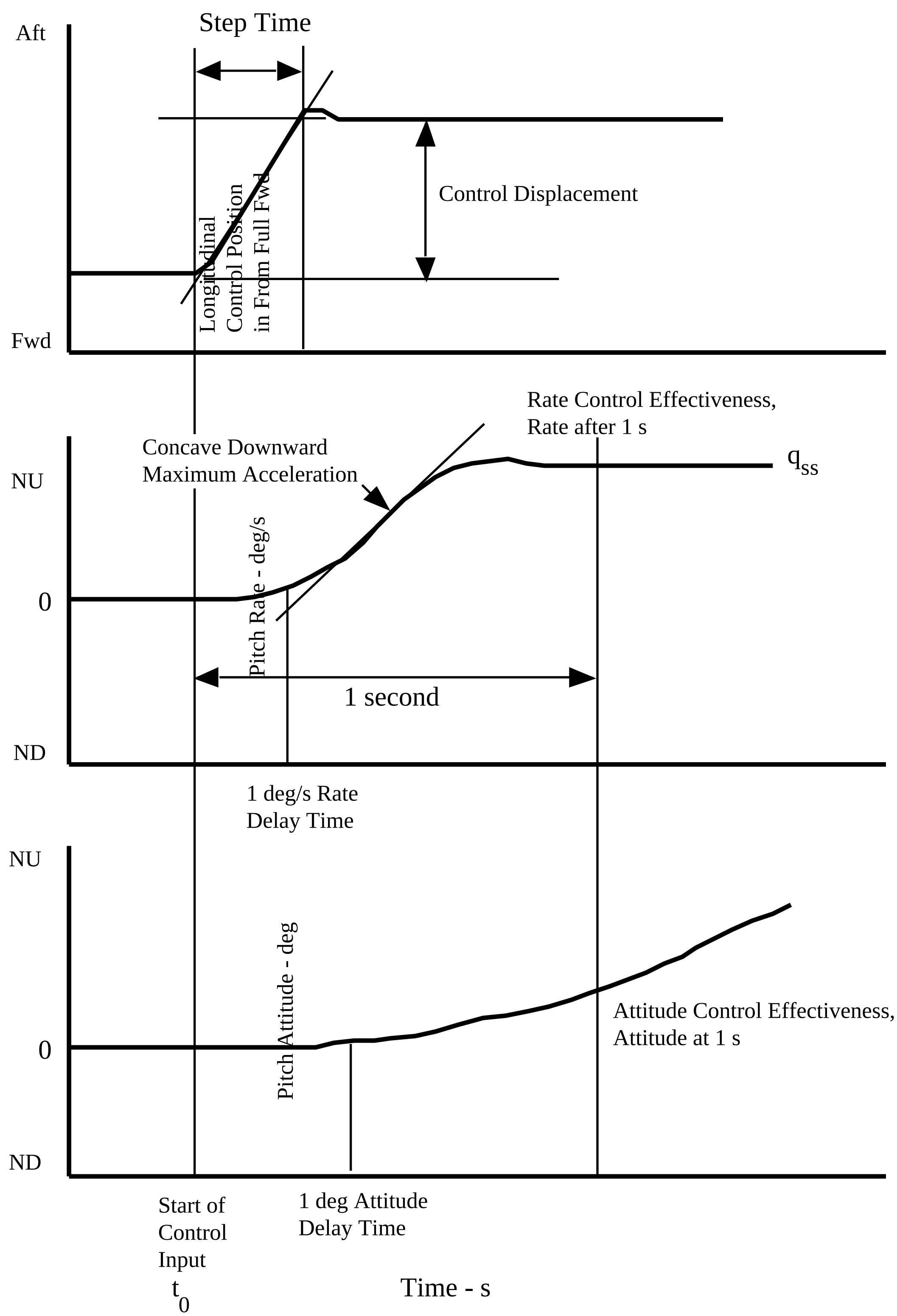


Figure 6.19
Measurement of Step Control Response Characteristics

6.4.6.1 TEST TECHNIQUE

1. Stabilize in wings level, ball centered, coordinated flight at the desired trim conditions. Maintain constant collective during the control response. Record trim data.
2. Set the control fixture for the desired displacement magnitude and direction. The control input direction and size is verbally and visually (if possible) verified between the pilot and copilot/engineer. Start with small displacements and always use an incremental buildup procedure.
3. A suggested voice procedure is: “Data ON, standby for a (displacement magnitude), (direction of input), (control) input on three. Thousand one, thousand two, thousand three.” For example: “Data ON, standby for a one inch forward longitudinal step input on three. Thousand one, thousand two, thousand three.”
4. Turn ON the data system prior to the countdown, allowing the initial conditions to be recorded.
5. After the countdown, a crisp step input is made against the fixture. Hold the input control rigidly against the fixture while maintaining all other controls fixed. When making cyclic inputs, use only wrist action and not the entire arm.
6. Do not attempt to decouple responses in other axes. The response lasts only seconds and attempts to decouple usually result in contaminating the desired response.
7. Hold the input control fixed and initiate recovery when a steady state rate is obtained or a predetermined flight limit is reached. When you have the required data or a limit is approached, recover and secure the automatic data recorder.
8. Restabilize at the trim conditions and repeat the procedure for incrementally larger inputs until the desired control input magnitude is obtained, or a predetermined flight limit is reached.
9. An incremental buildup in step input size is mandatory. Practice and experience are necessary for good results.

6.4.6.2 DATA REQUIRED

V_o, H_{P_o}, θ, FC, T_o, Q, N_R, cockpit control positions, control input size and direction.

Automatic recording systems are required for data collection. A time history of the longitudinal control position, pitch attitude, pitch rate, and pitch acceleration are the minimum essential data traces.

6.4.6.3 TEST CRITERIA

1. Wings level, ball centered, balanced flight.
2. Collective fixed.

6.4.6.4 DATA REQUIREMENTS

1. Stabilize 15 s prior to taking data.
2. Trim $V_o \pm 2$ kn.
3. $H_{P_o} \pm 1000$ ft of the target test altitude.
4. The data system must be capable of recording data with a 10 Hz frequency response.

6.4.6.5 SAFETY CONSIDERATIONS/RISK MANAGEMENT

Observe normal operating limitations. Conduct tests at a safe operating altitude over terrain suitable for forced landings. Maintain proper lookout doctrine. Maintain situational awareness. Consider using a chase aircraft and sole use airspace. Wear parachutes.

A thorough knowledge of airspeed and attitude limitations is essential. Initiate recovery in sufficient time to prevent exceeding a test limit. The use of the control fixture must be thoroughly briefed and practiced.

Observe the response during the maneuver; however, analyze the response and make comments both oral and written after the recovery is complete.

6.4.7 Gust Response

Gust response is a dynamic mode of motion which is treated in a separate category in the discussion and evaluation of flight dynamics. In general terms, the gust response relates to the quality of the ride. This encompasses the combination of instantaneous linear and angular acceleration changes during and immediately following flying through an atmospheric disturbance. Different characteristics are emphasized in the evaluation of the short term and gust responses. Pitch attitude and pitch rate changes are of primary importance for the short term dynamic response. Gust responses could result in pitch rate and attitude changes. The pilot could correct using the short term longitudinal mode but this is just one consideration with the gust response. The gust response evaluation includes determining the difficulty of maintaining a desired flight path in turbulence.

6.4.7.1 EXCITATION METHODS

1. Flight in actual turbulence is the best method of evaluation.
2. Pulse inputs of varying magnitudes and duration can be used to simulate gusts. Using a one inch control deflection for one half second as required by MIL-H-8501A could give excessive response with high hinge offset.
3. Simulated gusts can also be obtained by making inputs directly into the flight control system using the automatic flight control systems.

6.4.7.2 TEST TECHNIQUE

1. Record flight through natural turbulence.
2. If natural turbulence is not available, pulse inputs in the longitudinal axis are substituted. Generally, a one inch aft longitudinal pulse input held for one-half second is acceptable. Following the pulse, maintain controls fixed and record the resulting aircraft response.

6.4.7.3 DATA REQUIRED

V_o, H_{P_o}, V_v, θ, FC, T_o, Q, N_R, cockpit control positions.

Following the excitation, record airspeed, altitude, pitch attitude, pitch rate, and CG normal acceleration as a function of time. Automatic recording systems are required to obtain meaningful test data. AFCS or SAS actuator time histories indicate the system response and show how close the system is to saturation.

6.4.7.4 TEST CRITERIA

1. Wings level, ball centered, balanced flight.
2. Collective fixed.

6.4.7.5 DATA REQUIREMENTS

1. Controls fixed for natural excitation.
2. For pulse excitation:
   a. Trim $V_o \pm 2$ kn.
   b. $H_{P_o} \pm 1000$ ft of the target test altitude.
   c. $\phi \pm 3$ degrees bank angle.
   d. Pulse duration 0.5 s $\pm$ 0.1 s.

6.4.7.6 SAFETY CONSIDERATIONS/RISK MANAGEMENT

Observe normal operating limitations. Conduct tests at a safe operating altitude over terrain suitable for forced landings. Maintain proper lookout procedures. Maintain situational awareness. Consider using a chase aircraft and sole use airspace. Wear parachutes.

A thorough knowledge of airspeed and attitude limitations is essential. Initiate recovery in sufficient time to prevent exceeding a test limit. The use of the control fixture must be thoroughly briefed and practiced.

6.5 DATA REDUCTION

Data reduction requirements for these tests are limited to application of instrument corrections to the recorded data. Airspeed and altitude data are corrected for both instrument and position errors. Automatic data recording systems normally include total instrumentation system corrections in reducing a flight tape (or telemetry) raw data into engineering units. Most quantitative information required for longitudinal flying qualities tests is read from time histories of selected data parameters.

6.5.1 Trimmed Control Positions

Plot longitudinal cyclic position versus calibrated airspeed (Figure 6.20). Normally, pitch attitude, lateral, directional, and collective control positions are presented as well.

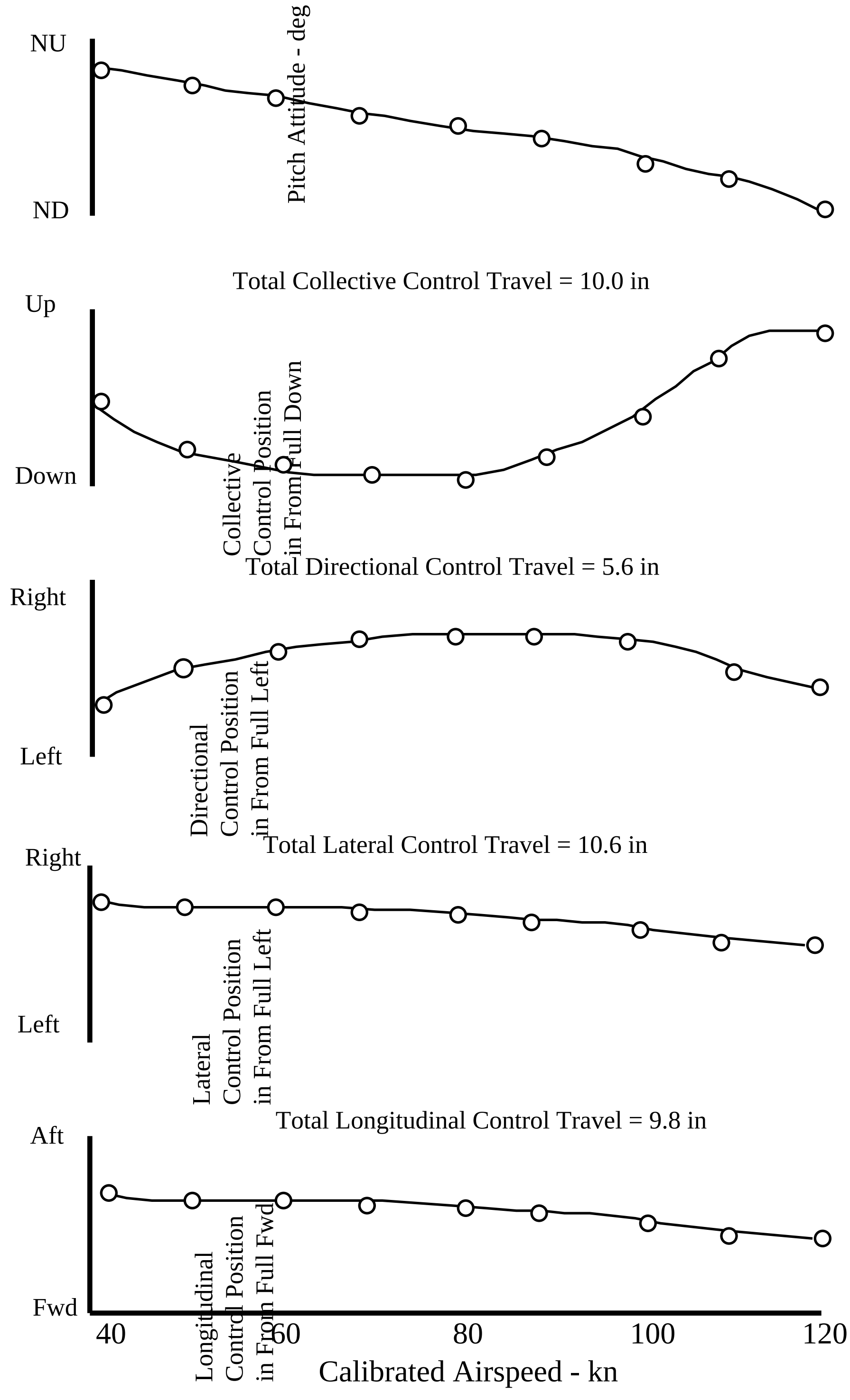


Figure 6.20
Trimmed Control Positions

6.5.2 Static Stability

1. Plot longitudinal cyclic position with calibrated airspeed as shown in Figure 6.21. When percent of control position is plotted on the ordinate, indicate the relationship between inches of control and percent of control.

2. Plot control force versus airspeed as obtained from a cross plot of the control position versus airspeed curve and the plot of control force versus control position obtained in ground mechanical characteristics tests.

3. Include plots of aircraft vertical velocity, etc. against calibrated airspeed. These can be correlated with longitudinal control position and control force variations to help in the explanation of piloting difficulties.

4. Trim conditions are presented and indicated by appropriate symbols.

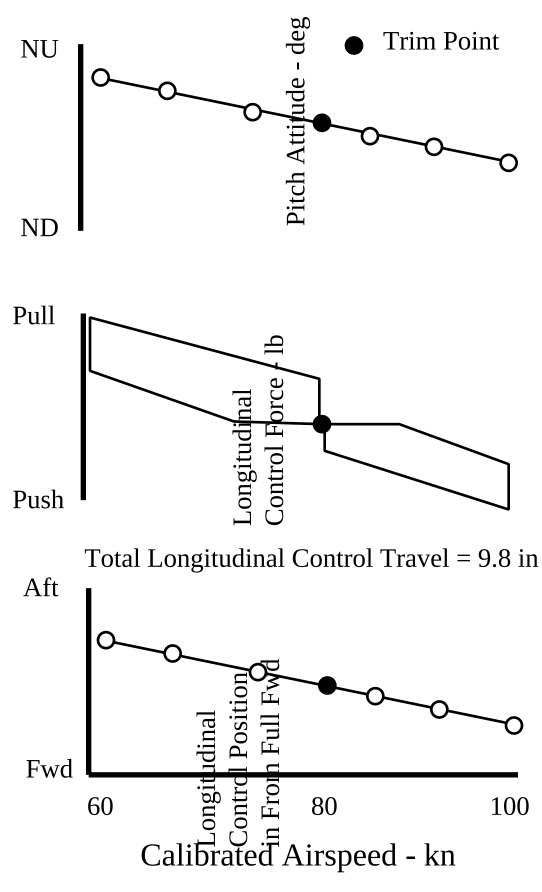


Figure 6.21
Static Longitudinal Stability

6.5.3 Maneuvering Stability

1. Plot longitudinal cyclic displacement versus normal acceleration as shown in Figure 6.22.

2. Plot longitudinal cyclic force versus normal acceleration. The cyclic force data required for a cross plot are obtained from the mechanical characteristic plot of force versus displacement obtained in ground tests.

3. The initial wings level 1.0 g trim point is specifically annotated. The steady turn (each direction) and pull up/push over data may be plotted separately or together. Do not forget to include the breakout including friction force when cross plotting the cyclic forces with normal acceleration.

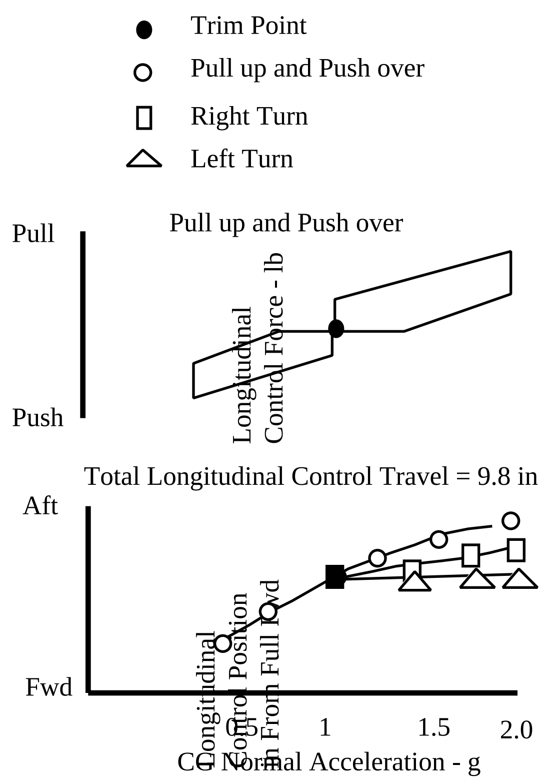


Figure 6.22
Maneuvering Stability

6.5.4 Long Term Dynamic Stability

1. Plot a time history of airspeed as shown in Figure 6.23. Other parameters of interest are pitch attitude (or rate) and the altitude deviations or actual altitude.

2. Tables can be used to summarize the engineering characteristics for several test conditions: period of oscillation, damping ratio (ζ), time to double amplitude (T_d), time to one-half amplitude ($T_{1/2}$), cycles to double amplitude (C_d), cycles to one-half amplitude ($C_{1/2}$), natural frequency (ω_n).

3. Present representative time histories.

4. Clearly annotate entry conditions and configurations.

Period	36 s
Damping Ratio, ζ	0.25
$T_{1/2}$	17 s
$C_{1/2}$	0.4
Damped Frequency, ω_d	0.004 rad/s

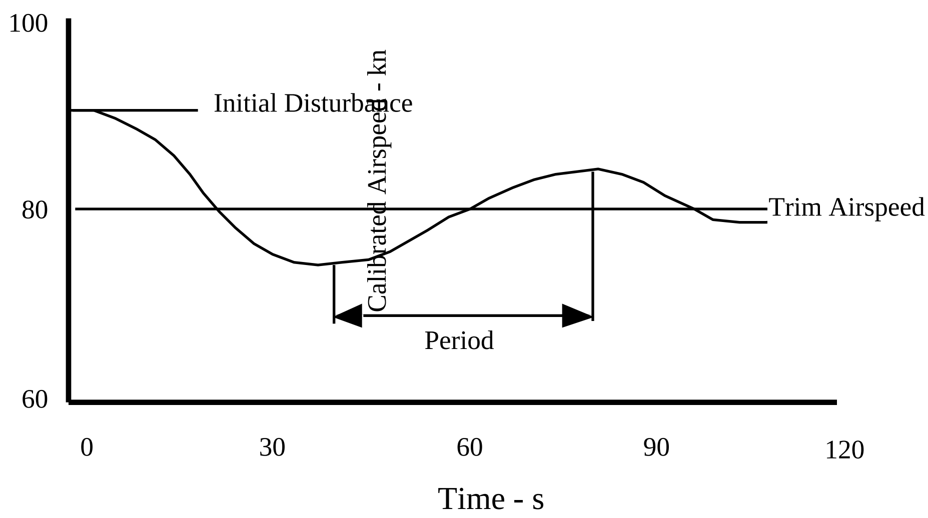


Figure 6.23
Long Term Dynamic Response

6.5.5 Short Term Dynamic Stability

Plot a time history of the pitch attitude response (Figure 6.24). Annotate the plot with the appropriate engineering terms to describe the response: ζ, ω_d, and time constant (τ). Other data parameters which may help with the analysis are the pitch rate and center of gravity normal acceleration traces.

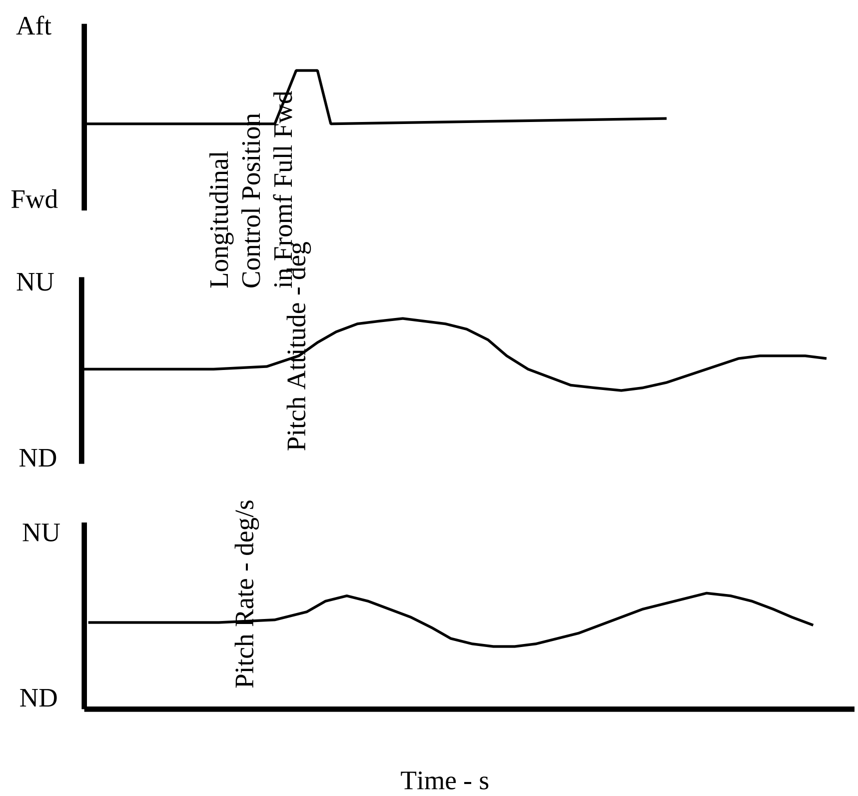


Figure 6.24
Short Term Response

6.5.6 Control Response

1. Representative time histories are presented. An example is provided in Figure 6.19. Proper annotation of the trace is important to help the reader visualize the response. Determine the attitude, rate, and acceleration delay; the maximum rate and

acceleration; the time to 63% of steady state rate; and the attitude change after 1 s. Note the time at which the step input was made and the time at which a recovery was initiated. Present a summary of control response characteristics (Figure 6.25).

2. Tables are helpful in presenting data from various responses.

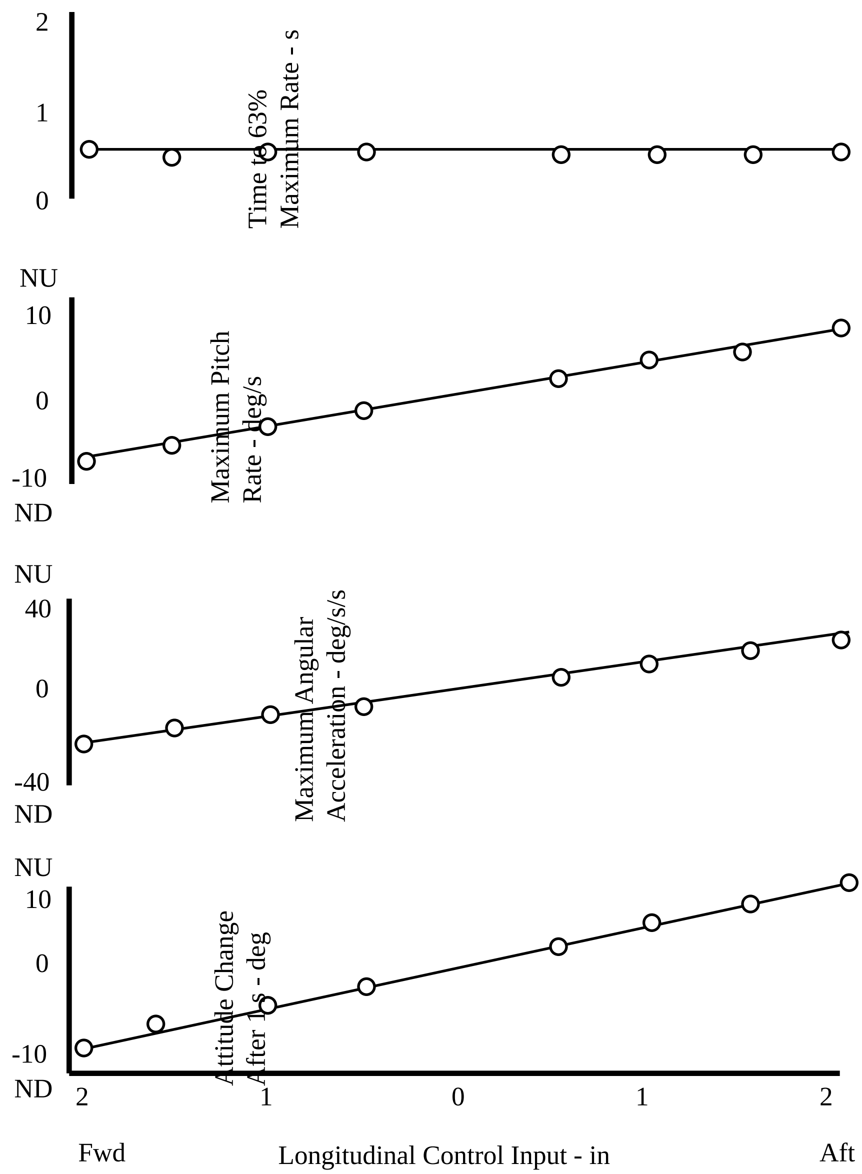


Figure 6.25
Control Response Characteristics

6.5.7 Gust Response

1. Cockpit data of airspeed and attitude changes suffice for qualitative evaluations. The g meter in the cockpit does not display accurately the short duration accelerations felt by the pilot.

2. A time history of the pilot seat acceleration or the aircraft CG acceleration can be useful when shown with the flight control displacements and attitude changes (Figure 6.26).

3. Time histories of the stability system actuator positions may also be included.

4. Qualitative observations are the principal data obtained in the test. Quantitative data are used to reinforce qualitative comments.

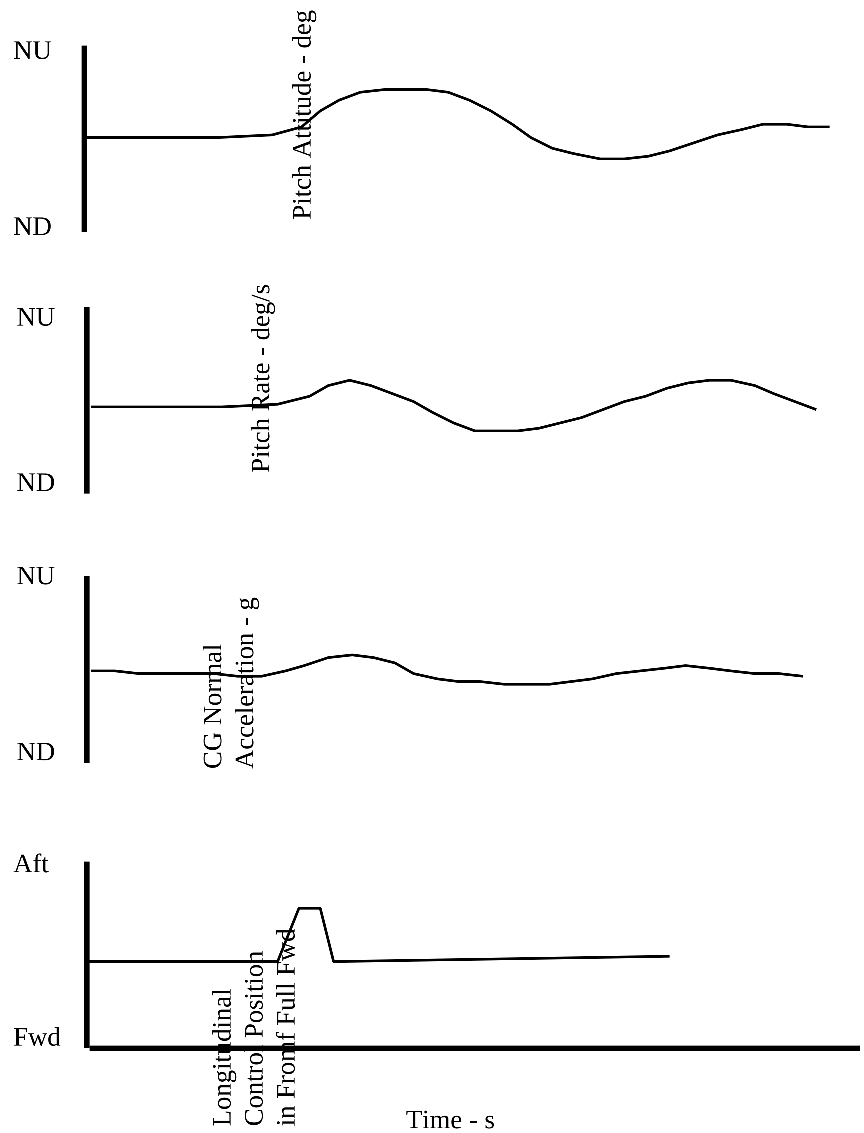


Figure 6.26
Gust Response Characteristics

6.6 DATA ANALYSIS

6.6.1 Trimmed Control Positions

The following considerations are generally included in the analysis of trimmed control position in forward flight data:

1. Were there adequate control margins?
2. Was forward cyclic required with increasing airspeed?
3. Are the control position changes with changes in power excessive?
4. Could forces be trimmed to zero?
5. What were pitch attitude changes with airspeed?
6. Was field-of-view adequate?
7. Were there objectionable vibrations?
8. Was there coupling between the pitch, yaw and roll axes?
9. Could small precise collective changes be made?

6.6.2 Static Stability

The following considerations are generally included in the analysis of static longitudinal stability test data:

1. Positive static stability is indicated by forward longitudinal cyclic position and forces with increasing airspeed.

2. Positive static stability causes the aircraft to tend to return to trim following a disturbance. The force and position provide cues to off trim airspeed. Does the data substantiate the pilot's qualitative opinion concerning ease of trimming the aircraft?

3. Is there a trim speed band within which there are no force or displacement cues for off trim conditions?

4. Discuss gradients (in/kn, lb/kn) and not the strength of the static stability. The longitudinal cyclic position required is not only a function of the pitching moments being generated, but also is influenced by the control system characteristics.

6.6.3 Maneuvering Stability

The following considerations are generally included in the analysis of maneuvering stability test data:

1. Positive maneuvering stability is indicated by an aft longitudinal cyclic displacement with increasing normal acceleration. What is the sign of the stability and how is it indicated?
2. The cyclic forces and displacements provide a physical cue to the pilot of normal acceleration and pitch rate (turn rate). Were these cues available and what is your opinion of their value?
3. Was there any tendency to exceed the g envelope of the aircraft? This might limit the usable g envelope.

6.6.4 Long Term Dynamic Stability

The following considerations are generally included in the analysis of long term dynamic stability test data:

1. How easy or difficult was it to excite the long term response?
2. In engineering terms, how do you describe the aircraft mode of motion? Use the period, damping ratio, time to half amplitude and time to double amplitude to tell the reader what you saw.
3. How does this motion influence the pilot's ability to perform mission tasks? The significance of the long term response is how it degrades or helps the pilot perform a specific task. An evaluation is made of the pilot effort required to suppress or correct the airspeed, attitude, and altitude variations about trim.

6.6.5 Short Term Dynamic Stability

The following considerations are generally included in the analysis of short term dynamic stability test data:

1. Can the pilot make small precise pitch attitude changes?
2. In reference to the time history of the pitch attitude change, the discussion includes the number of pitch oscillations (overshoots), the frequency and magnitude of the cyclic inputs required to obtain the desired performance, the excitation of other modes of motion, and the approximate damping ratio if an estimate (or analysis) can be obtained.
3. Time histories of mission tasks are sometimes helpful in discussing problem areas.

6.6.6 Control Response

The following considerations are generally included in the analysis of longitudinal control response test data:

1. An abundance of quantitative data can be taken from the response time histories. The parameters chosen and presented support your opinion of how well or poorly the aircraft responds to control inputs. Parameters which define the amount of control moment available are: rate control effectiveness; attitude control effectiveness; and steady state angular velocity. Quantities which can be used in discussing the quality of the response or how it got to the observed steady state rate are the: angular accelerations delay time; initial angular acceleration; inflection time; and response time constant. Use these data as an integral part of your discussions.
2. Did the aircraft respond adequately to perform the assigned task?
    a. Were the steady state rates adequate?
    b. Was the response predictable?
    c. Was the response consistent?
    d. Were there any over controlling problems?

6.6.7 Gust Response

The following considerations are generally included in the analysis of gust response test data:

1. How could the ride be described? Jerky, smooth, loping, etc.? Was the ride comfortable or uncomfortable?
2. What were the pitch, roll, and yaw attitude changes and airspeed changes?
3. Did the aircraft tend to return to the original trim condition?
4. How much pilot effort was required to return the aircraft to the trim condition?

6.7 MISSION SUITABILITY

The suitability of the test aircraft for the intended mission is the ultimate reason for conducting any handling qualities test. Each of the specific tests, detailed in the previous sections, provides some information in accessing suitability. Longitudinal trimmed control positions in forward flight provides detailed information about control margins, linearities, discontinuities and gradients which assist the pilot in determining overall suitability. Static longitudinal stability characteristics are good indicators of pilot workload requirements to maintain a desired forward flight speed and the tendencies of the aircraft once disturbed. Maneuvering stability characteristics determine many aspects of the pilot qualitative comments relative to aggressive flight path changes. The long term dynamic stability character in large part determines cruise flying qualities. Short term dynamics in the longitudinal axis are a big factor in how quickly and accurately the pilot can point the aircraft in high gain tasks such as aerial refueling or target tracking. Control response tests quantitatively document the pilots qualitative comments concerning aircraft responsiveness. Gust response evaluation indicates how the crew and passengers like the ride quality.

In addition to the engineering tests, the test pilot investigates how the helicopter is intended to be used and closely duplicates these mission tasks. Generally, knowledge of engineering maneuver shortcomings allows the test team to design operational scenarios which also reflect the poor characteristics. The mission of the aircraft determines the maneuvers required to be flown. Cargo helicopters require close attention to straight and level cruise tasks (long term dynamic stability); while attack aircraft may need special consideration for tracking tasks (short term dynamic stability) and maneuverability (maneuvering stability). Each test program requires a unique set of mission maneuvers.

The most important aspect of mission suitability testing is to keep the operator in mind. How the test vehicle will be used in the field, and who will be operating the aircraft are important factors to consider.

6.8 SPECIFICATION COMPLIANCE

General guidelines for helicopter handling qualities in the hovering and low airspeed flight regime are contained in MIL-H-8501A. Military specifications are only a guide. The following list identifies the paragraph number in MIL-H-8501A and a short description of the requirement.

- 3.2 Longitudinal characteristics
  - 3.2.1 Longitudinal control margin; controls and helicopter shake, vibration, roughness
  - 3.2.2 Hovering turns on a spot
  - 3.2.3 Longitudinal trimmability, control jump
  - 3.2.4 Longitudinal control force gradient
  - 3.2.5 Quick stop, rapid acceleration
  - 3.2.6 Limit control force
  - 3.2.7 Breakout force
  - 3.2.8 Control coupling
  - 3.2.9 Control response
  - 3.2.10 Static longitudinal stability
    - 3.2.10.1 Critical center of gravity
    - 3.2.10.2 Trim change in climbs and descents
  - 3.2.11 Dynamic longitudinal stability
    - 3.2.11.1 Dynamic control response
    - 3.2.11.2 Gust response
  - 3.2.12 Control response, normal acceleration
  - 3.2.13 Control power
  - 3.2.14 Control response damping

Additional requirements are included in the military specification which relate to boosted controls, failure modes, automatic stabilization equipment, and vibrations. Some of these paragraphs may apply to specific helicopters equipped with advanced flight control systems or highly augmented systems.

MIL-H-8501A is old and under most circumstances supplemented with additional detailed specifications listed in the procurement documents. These must be researched thoroughly and carefully to ensure contractual compliance and a complete evaluation.

A new military specification has been under development for years, but to this date not finalized. This new specification may include frequency domain tests to evaluate higher order aircraft flight control systems expected in future helicopters.

6.9 GLOSSARY

6.9.1 Notations

A_{1s}	Lateral cyclic pitch angle, shaft referenced
a_{1s}	Longitudinal flapping angle, shaft referenced
AFCS	Automatic Flight Control System
B	Damping constant
b	Number of blades
B_{1s}	Longitudinal cyclic pitch angle, shaft referenced
b_{1s}	Lateral flapping angle, shaft referenced
$\dot{B}_{1s}$	Time rate of change of longitudinal cyclic pitch angle, shaft referenced
$C_{1/2}$	Cycles to one-half amplitude
C_d	Cycles to double amplitude
CF	Centrifugal force
CG	Center of gravity
D_f	Fuselage drag force
e	Base of natural logarithm, flapping hinge offset
FC	Fuel count
F_q	Pitch input

F_x	Force in the x direction
g	Gravity
H	Rotor hub force, $\perp$ to shaft
h	Height of hub above CG
h'	Longitudinal distance between the rotor shaft and the CG
H_{P_o}	Observed pressure altitude
HQR	Handling Qualities Rating
Hz	Hertz (cycles per second)
in	Inch
i_s	Stabilizer incidence
I_{yy}	Moment of inertia about y axis, pitch moment of inertia
K	Spring constant
kn	Knot
L_t	Tail lift
l_t	Distance from the tail to the CG
M	Net moment about y axis, Pitch moment
m	Mass
M_α	Pitch moment due to angle of attack
$M_{B_{1s}}$	Pitch moment due to longitudinal cyclic pitch angle
M_{CG}	Moment about CG
$M_{CG_{f+t}}$	Pitch moment due to the aerodynamic forces on the fuselage/tail
M_H	Pitch moment due to rotor hub force
$\overline{M}_H$	Average pitch moment due to rotor hub force for b blades
M_q	Pitch moment due to pitch rate
M_{θ_C}	Pitch moment due to collective pitch angle
M_S	Blade mass moment
MSL	Maneuvering stability level
M_u	Pitch moment due to longitudinal velocity
M_w	Pitch moment due to vertical velocity
$M_{\dot{w}}$	Pitch moment due to vertical acceleration
n	Normal acceleration, Normal load factor
N_g	Engine gas generator speed
N_R	Main rotor speed
n_{ss}	Steady state normal acceleration
p	Angular velocity about x axis, Roll rate

PIO	Pilot induced oscillation
Q	Engine torque
q	Angular velocity about y axis, Pitch rate
q_{ss}	Steady state pitch rate
$\ddot{q}$	Second time derivative of pitch rate
$\dot{q}$	Angular acceleration about y axis
R	Resultant aerodynamic force
r	Angular velocity about z axis, Radius along blade, Yaw rate
rad	Radian
s	Second
SAS	Stability Augmentation System
SCAS	Stability Control Augmentation System
T	Thrust
t	Time
$T_{1/2}$	Time to one-half amplitude
T_d	Time to double amplitude
T_o	Observed temperature
TPP	Tip path plane
u	Translational velocity component along x axis
u_0	Initial velocity
$\ddot{u}$	Time rate of change of linear acceleration along x axis
$\dot{u}$	Linear acceleration along x axis
V	Velocity, Free stream velocity, Relative velocity
VAR	Vibration Assessment Rating
V_H	Maximum level flight airspeed
V_{hor}	Velocity in the horizontal plane
V_{NE}	Velocity never exceed
V_o	Observed airspeed
V_v	Vertical velocity
w	Translational velocity component along z axis
w_0	Initial velocity component along z axis
w_{QS}	Quasi-static vertical velocity
$\ddot{w}$	Time rate of change of linear acceleration along z axis
$\dot{w}$	Linear acceleration along z axis
X	Resultant force in x direction

x	Orthogonal direction along longitudinal axis of the aircraft; Distance along x axis
$X_{B_{1s}}$	Longitudinal force due to longitudinal cyclic pitch angle
X_q	Longitudinal force due to pitch rate
X_{θ_C}	Longitudinal force due to collective pitch angle
X_u	Longitudinal force due to longitudinal velocity
X_w	Longitudinal force due to vertical velocity
$\ddot{x}$	Time rate of change of velocity in x direction, Acceleration along x axis
$\dot{x}$	Time rate of change of x, Velocity in x direction
y	Orthogonal direction along lateral axis of the aircraft
Z	Resultant force in z direction
z	Orthogonal direction along vertical axis of the aircraft
$Z_{B_{1s}}$	Vertical force due to longitudinal cyclic pitch angle
Z_q	Vertical force due to pitch rate
Z_{θ_C}	Vertical force due to collective pitch angle
Z_u	Vertical force due to longitudinal velocity
Z_w	Vertical force due to vertical velocity

6.9.2 Greek Symbols

α (alpha)	Angle of attack
α_{BE}	Blade element angle of attack
α_T	Tail angle of attack
$\dot{\alpha}$	Time rate of change of angle of attack
β (beta)	Flapping angle, Sideslip angle
δ (delta)	Control
δ_C	Collective control
δ_{LONG}	Longitudinal control
δ_{th}	Throttle control
$\dot{\delta}$	Rate of change of control
ϕ (phi)	Roll angle
γ (gamma)	Flight path angle
γ_0	Initial flight path angle
$\dot{\gamma}$	Time rate of change of flight path angle
λ (lambda)	Characteristic root

θ (theta)	Pitch angle, shaft angle
θ_C	Collective pitch angle
θ_{TR}	Tail rotor pitch angle
$\ddot{\theta}$	Pitch acceleration
$\dot{\theta}$	Rate of change of pitch angle
$\dot{\theta}_{QS}$	Quasi-static rate of change of pitch angle
τ (tau)	Time constant
ω_d (omega)	Damped frequency
ω_n	Natural frequency
ψ (psi)	Blade azimuth angle, Phase angle, Yaw angle
ζ (zeta)	Damping ratio
Ω (Omega)	Rotor angular velocity

6.10 REFERENCES

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CHAPTER SEVEN

FORWARD FLIGHT LATERAL-DIRECTIONAL STABILITY, CONTROL, AND FLYING QUALITIES

FORWARD FLIGHT LATERAL-DIRECTIONAL STABILITY, CONTROL, AND FLYING QUALITIES

CHAPTER SEVEN

FIGURES

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$$\dot{v} - Y_v\Delta v - Y_r\Delta r + u_0\Delta r - Y_p\Delta p - g\phi = Y_{A_{1s}}\Delta A_{1s} + Y_{\theta_{TR}}\Delta\theta_{TR}$$

$$= Y_{\delta_{LAT}}\Delta\delta_{LAT} + Y_{\delta_{PED}}\Delta\delta_{PED}$$ *eq 7.1* 7.4

$$\dot{p} - L_v\Delta v - L_r\Delta r - L_p\Delta p = L_{A_{1s}}\Delta A_{1s} + L_{\theta_{TR}}\Delta\theta_{TR}$$

$$= L_{\delta_{LAT}}\Delta\delta_{LAT} + L_{\delta_{PED}}\Delta\delta_{PED}$$ *eq 7.2* 7.5

$$\dot{r} - N_v\Delta v - N_p\Delta p - N_r\Delta r = N_{A_{1s}}\Delta A_{1s} + N_{\theta_{TR}}\Delta\theta_{TR}$$

$$= N_{\delta_{LAT}}\Delta\delta_{LAT} + N_{\delta_{PED}}\Delta\delta_{PED}$$ *eq 7.3* 7.5

$$\Delta Y = T\,\Delta b_{1s} + b_{1s}\,\Delta T + \Delta Y_R + \Delta T_{TR} + \Delta Y_{f+t}$$ *eq 7.4* 7.9

$$\Delta L = (Th + \frac{1}{2}\,ebM_S\Omega^2)\Delta b_{1s} + hb_{1s}\Delta T + h\Delta Y_R + h_{TR}\Delta T_{TR} + \Delta L_{f+t}$$ *eq 7.5* 7.9

$$\Delta N = Th'\Delta b_{1s} + h'b_{1s}\Delta T + h'\Delta Y_R - l_t\Delta T_{TR} + \Delta N_{f+t} + \Delta Q_{MR}$$ *eq 7.6* 7.9

$$N_v = \frac{1}{I_{zz}}\left[\,Th'\,\frac{\partial b_{1s}}{\partial v} + h'b_{1s}\frac{\partial T}{\partial v} + h'\,\frac{\partial Y_R}{\partial v} - l_t\,\frac{\partial T_{TR}}{\partial v} + \frac{\partial N_{f+t}}{\partial v} + \frac{\partial Q_{MR}}{\partial v}\right]$$ *eq 7.7* 7.11

$$N_v \approx \frac{1}{I_{zz}}\left[\,-l_t\,\frac{\partial T_{TR}}{\partial v} + \frac{\partial N_{f+t}}{\partial v}\,\right]$$ *eq 7.8* 7.11

$$N_r = \frac{1}{I_{zz}}\left[Th'\frac{\partial b_{1s}}{\partial r} + h'\, b_{1s}\frac{\partial T}{\partial r} + h'\frac{\partial Y_R}{\partial r} - l_t\frac{\partial T_{TR}}{\partial r} + \frac{\partial N_{f+t}}{\partial r} + \frac{\partial Q_{MR}}{\partial r}\right]$$

eq 7.9 7.14

$$N_r \approx \frac{1}{I_{zz}}\left[-l_t\frac{\partial T_{TR}}{\partial r} + \frac{\partial N_{f+t}}{\partial r}\right]$$

eq 7.10 7.15

$$N_p = \frac{1}{I_{zz}}\left[Th'\frac{\partial b_{1s}}{\partial p} + h'b_{1s}\frac{\partial T}{\partial p} + h'\frac{\partial Y_R}{\partial p} - l_t\frac{\partial T_{TR}}{\partial p} + \frac{\partial N_{f+t}}{\partial p} + \frac{\partial Q_{MR}}{\partial p}\right]$$

eq 7.11 7.16

$$N_{\theta_{TR}} = \frac{1}{I_{zz}}\left[Th'\frac{\partial b_{1s}}{\partial \theta_{TR}} + h'b_{1s}\frac{\partial T}{\partial \theta_{TR}} + h'\frac{\partial Y_R}{\partial \theta_{TR}} - l_t\frac{\partial T_{TR}}{\partial \theta_{TR}} + \frac{\partial N_{f+t}}{\partial \theta_{TR}} + \frac{\partial Q_{MR}}{\partial \theta_{TR}}\right]$$

eq 7.12 7.18

$$N_{\theta_{TR}} \approx \frac{1}{I_{zz}}\left[-l_t\frac{\partial T_{TR}}{\partial \theta_{TR}}\right]$$

eq 7.13 7.18

$$N_{\delta_{PED}} = \left(\frac{\partial \theta_{TR}}{\partial \delta_{PED}}\right) N_{\theta_{TR}}$$

eq 7.14 7.18

$$N_{A_{1s}} = \frac{1}{I_{zz}}\left[Th'\frac{\partial b_{1s}}{\partial A_{1s}} + h'b_{1s}\frac{\partial T}{\partial A_{1s}} + h'\frac{\partial Y_R}{\partial A_{1s}} - l_t\frac{\partial T_{TR}}{\partial A_{1s}} + \frac{\partial N_{f+t}}{\partial A_{1s}} + \frac{\partial Q_{MR}}{\partial A_{1s}}\right]$$

eq 7.15 7.20

$$N_{\delta_{LAT}} = \left(\frac{\partial A_{1s}}{\partial \delta_{LAT}}\right) N_{A_{1s}}$$

eq 7.16 7.21

$$L_v = \frac{1}{I_{xx}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial b_{1s}}{\partial v} + hb_{1s}\frac{\partial T}{\partial v} + h\frac{\partial Y_R}{\partial v} + h_{TR}\frac{\partial T_{TR}}{\partial v} + \frac{\partial L_{f+t}}{\partial v}\right]$$

eq 7.17 7.22

$$L_r = \frac{1}{I_{xx}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial b_{1s}}{\partial r} + hb_{1s}\frac{\partial T}{\partial r} + h\frac{\partial Y_R}{\partial r} + h_{TR}\frac{\partial T_{TR}}{\partial r} + \frac{\partial L_{f+t}}{\partial r}\right]$$

eq 7.18 7.24

$$L_p = \frac{1}{I_{xx}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial b_{1s}}{\partial p} + hb_{1s}\frac{\partial T}{\partial p} + h\frac{\partial Y_R}{\partial p} + h_{TR}\frac{\partial T_{TR}}{\partial p} + \frac{\partial L_{f+t}}{\partial p}\right]$$

eq 7.19 7.26

$$L_p \approx \frac{1}{I_{xx}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial b_{1s}}{\partial p} + h_{TR}\frac{\partial T_{TR}}{\partial p}\right]$$

eq 7. 20 7.26

$$L_{A_{1s}} = \frac{1}{I_{xx}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial b_{1s}}{\partial A_{1s}} + hb_{1s}\frac{\partial T}{\partial A_{1s}} + h\frac{\partial Y_R}{\partial A_{1s}} + h_{TR}\frac{\partial T_{TR}}{\partial A_{1s}} + \frac{\partial L_{f+t}}{\partial A_{1s}}\right]$$

eq 7.21 7.28

$$L_{\delta_{LAT}} = \left(\frac{\partial A_{1s}}{\partial \delta_{LAT}}\right)L_{A_{1s}}$$

eq 7.22 7.29

$$L_{\theta_{TR}} = \frac{1}{I_{xx}}\left[h_{TR}\frac{\partial T_{TR}}{\partial \theta_{TR}}\right]$$

eq 7.23 7.30

$$L_{\delta_{PED}} = \left(\frac{\partial\theta_{TR}}{\partial\delta_{PED}}\right) L_{\theta_{TR}} \qquad \textit{eq 7.24} \qquad 7.30$$

$$Y_v = \frac{1}{m}\left[T\frac{\partial b_{1s}}{\partial v} + b_{1s}\frac{\partial T}{\partial v} + \frac{\partial Y_R}{\partial v} + \frac{\partial T_{TR}}{\partial v} + \frac{\partial Y_{f+t}}{\partial v}\right] \qquad \textit{eq 7.25} \qquad 7.32$$

$$Y_r = \frac{1}{m}\left[T\frac{\partial b_{1s}}{\partial r} + b_{1s}\frac{\partial T}{\partial r} + \frac{\partial Y_R}{\partial r} + \frac{\partial T_{TR}}{\partial r} + \frac{\partial Y_{f+t}}{\partial r}\right] \qquad \textit{eq 7.26} \qquad 7.33$$

$$Y_p = \frac{1}{m}\left[T\frac{\partial b_{1s}}{\partial p} + b_{1s}\frac{\partial T}{\partial p} + \frac{\partial Y_R}{\partial p} + \frac{\partial T_{TR}}{\partial p} + \frac{\partial Y_{f+t}}{\partial p}\right] \qquad \textit{eq 7.27} \qquad 7.35$$

$$Y_p \approx \frac{1}{m}\left[T\frac{\partial b_{1s}}{\partial p} + \frac{\partial T_{TR}}{\partial p}\right] \qquad \textit{eq 7.28} \qquad 7.35$$

$$Y_{A_{1s}} = \frac{1}{m}\left[T\frac{\partial b_{1s}}{\partial A_{1s}} + b_{1s}\frac{\partial T}{\partial A_{1s}}\right] \qquad \textit{eq 7.29} \qquad 7.36$$

$$Y_{A_{1s}} \approx g\left[\frac{\partial b_{1s}}{\partial A_{1s}}\right] \qquad \textit{eq 7.30} \qquad 7.36$$

$$Y_{\delta_{LAT}} = \left(\frac{\partial A_{1s}}{\partial\delta_{LAT}}\right) Y_{A_{1s}} \qquad \textit{eq 7.31} \qquad 7.37$$

$$Y_{\theta_{TR}} = \frac{1}{m}\left[\frac{\partial T_{TR}}{\partial\theta_{TR}}\right] \qquad \textit{eq 7.32} \qquad 7.38$$

$$Y_{\delta_{PED}} = \left(\frac{\partial\theta_{TR}}{\partial\delta_{PED}}\right) Y_{\theta_{TR}} \qquad \textit{eq 7.33} \qquad 7.38$$

$$-Y_v\, u_0\Delta\beta \;\; -g\Delta\phi = Y_{A_{1s}}\,\Delta A_{1s} + Y_{\theta_{TR}}\,\Delta\theta_{TR} = Y_{\delta_{LAT}}\,\Delta\delta_{LAT} + Y_{\delta_{PED}}\,\Delta\delta_{PED} \qquad \textit{eq 7.34} \qquad 7.46$$

$$-L_v\, u_0\Delta\beta = L_{A_{1s}}\,\Delta A_{1s} + L_{\theta_{TR}}\,\Delta\theta_{TR} = L_{\delta_{LAT}}\,\Delta\delta_{LAT} + L_{\delta_{PED}}\,\Delta\delta_{PED} \qquad \textit{eq 7.35} \qquad 7.46$$

$$-N_v\, u_0\Delta\beta = N_{A_{1s}}\,\Delta A_{1s} + N_{\theta_{TR}}\,\Delta\theta_{TR} = N_{\delta_{LAT}}\,\Delta\delta_{LAT} + N_{\delta_{PED}}\,\Delta\delta_{PED} \qquad \textit{eq 7.36} \qquad 7.46$$

$$\frac{\Delta\delta_{PED}}{\Delta\beta} = -u_0\left(\frac{L_{\delta_{LAT}}\,N_v - L_v\,N_{\delta_{LAT}}}{L_{\delta_{LAT}}\,N_{\delta_{PED}} - L_{\delta_{PED}}\,N_{\delta_{LAT}}}\right) \qquad \textit{eq 7.37} \qquad 7.50$$

$$\frac{\Delta\delta_{PED}}{\Delta\beta} \approx \left(-\frac{u_0}{N_{\delta_{PED}}}\right) N_v \qquad \textit{eq 7.38} \qquad 7.50$$

$$\frac{\Delta\delta_{LAT}}{\Delta\beta} = -u_0\left(\frac{N_{\delta_{PED}}\,L_v - N_v\,L_{\delta_{PED}}}{L_{\delta_{LAT}}\,N_{\delta_{PED}} - L_{\delta_{PED}}\,N_{\delta_{LAT}}}\right) \qquad \textit{eq 7.39} \qquad 7.50$$

$$\frac{\Delta\delta_{LAT}}{\Delta\beta} \approx \frac{u_0}{L_{\delta_{LAT}}}\left(L_v - L_{\delta_{PED}}\,\frac{N_v}{N_{\delta_{PED}}}\right) \qquad \textit{eq 7.40} \qquad 7.50$$

$$\frac{\Delta\delta_{LAT}}{\Delta\beta} = \left(-\frac{u_0\, L_v}{L_{\delta_{LAT}}}\right) - \left[\frac{L_{\delta_{PED}}}{L_{\delta_{LAT}}}\left(\frac{\Delta\delta_{PED}}{\Delta\beta}\right)\right]$$

eq 7.41 7.50

$$L_{\delta_{PED}} \approx \frac{1}{I_{xx}}\, h_{TR}\left(\frac{\partial T_{TR}}{\partial\delta_{PED}}\right)$$

eq 7.42 7.52

$$-Y_v u_0 \Delta\beta + (u_0 - Y_r)\Delta r - g\phi = Y_{A_{1s}}\Delta A_{1s} + Y_{\theta_{TR}}\Delta\theta_{TR} = Y_{\delta_{LAT}}\Delta\delta_{LAT} + Y_{\delta_{PED}}\Delta\delta_{PED}$$

eq 7.43 7.56

$$-L_v u_0 \Delta\beta - L_r\Delta r = L_{A_{1s}}\Delta A_{1s} + L_{\theta_{TR}}\Delta\theta_{TR} = L_{\delta_{LAT}}\Delta\delta_{LAT} + L_{\delta_{PED}}\Delta\delta_{PED}$$

eq 7.44 7.56

$$-N_v u_0 \Delta\beta - N_r\Delta r = N_{A_{1s}}\Delta A_{1s} + N_{\theta_{TR}}\Delta\theta_{TR} = N_{\delta_{LAT}}\Delta\delta_{LAT} + N_{\delta_{PED}}\Delta\delta_{PED}$$

eq 7.45 7.56

$$L_v u_0 \Delta\beta \qquad + L_r \frac{\Delta r}{u_0\Delta\beta}\, u_0\Delta\beta \qquad + L_{\delta_{LAT}}\Delta\delta_{LAT} = 0$$

eq 7.46 7.60

$$L_v\, u_0\Delta\beta \qquad + L_r \frac{N_v}{\left(-N_r\right)}\, u_0\Delta\beta \qquad + L_{\delta_{LAT}}\Delta\delta_{LAT} = 0$$

eq 7.47 7.61

$$\frac{\Delta\beta}{\Delta\delta_{LAT}} = \frac{1}{u_0} \quad \frac{L_{\delta_{LAT}}}{\left[\left(-L_v\right) - L_r\frac{N_v}{\left(-N_r\right)}\right]}$$

eq 7.48 7.62

$$\frac{\Delta\beta}{\Delta\delta_{LAT}} = \frac{1}{u_0} \quad \frac{\left(-N_r\right) L_{\delta_{LAT}}}{\left(L_v N_r - L_r N_v\right)}$$ *eq 7.49* 7.63

$$\frac{\Delta r}{\Delta\delta_{LAT}} = \frac{\Delta r}{u_0\Delta\beta} \quad \frac{u_0\Delta\beta}{\Delta\delta_{LAT}} = \frac{N_v L_{\delta_{LAT}}}{\left(L_v N_r - L_r N_v\right)}$$ *eq 7.50* 7.63

$$L_v\, u_o\Delta\beta \quad + \quad L_r\Delta r \quad + \quad L_{\delta_{PED}}\Delta\delta_{PED} \quad = \quad 0$$ *eq 7.51* 7.64

$$N_v u_0\Delta\beta \quad + \quad N_r\,\Delta r \;+\; N_{\delta_{PED}}\Delta\delta_{PED} \quad = \quad 0$$ *eq 7.52* 7.64

$$\frac{\left(u_0\Delta\beta\right)}{\Delta r} = \frac{L_r}{\left(-L_v\right)}$$ *eq 7.53* 7.65

$$N_v \frac{L_r}{\left(-L_v\right)}\Delta r \;-\; \left(-N_r\right)\Delta r + N_{\delta_{PED}}\Delta\delta_{PED} \quad = \quad 0$$ *eq 7.54* 7.66

$$\frac{\Delta r}{\Delta\delta_{PED}} = \frac{\left(-L_v\right)N_{\delta_{PED}}}{L_v N_r - L_r N_v}$$ *eq 7.55* 7.66

$$\frac{u_0\Delta\beta}{\Delta\delta_{PED}} = \frac{L_r N_{\delta_{PED}}}{L_v N_r - L_r N_v}$$ *eq 7.56* 7.66

$$\frac{\Delta r}{\Delta\delta_{PED}} = \frac{\left(-L_v\right) N_{\delta_{PED}}\left[1 - \left(\dfrac{L_{\delta_{PED}}}{N_{\delta_{PED}}}\right)\left(\dfrac{N_v}{L_v}\right)\right]}{L_v N_r - L_r N_v}$$ *eq 7.57* 7.69

$$\frac{u_0 \Delta\beta}{\Delta\delta_{PED}} = \frac{L_r N_{\delta_{PED}} \left[1 - \left(\frac{L_{\delta_{PED}}}{N_{\delta_{PED}}} \right) \left(\frac{N_r}{L_r} \right) \right]}{L_v N_r - L_r N_v}$$ *eq 7.58* 7.69

$$W_{BALL} \sin\phi = \frac{W_{BALL}}{g} u_0 \omega \cos\phi = \frac{W_{BALL}}{g} u_0 r$$ *eq 7.59* 7.71

$$\phi \approx \sin\phi = \frac{u_0 r}{g}$$ *eq 7.60* 7.71

$$p = \frac{\left(\frac{\Delta L}{I_{xx}} \right)}{\left(-L_p \right)} \left(1 - e^{-\frac{t}{\tau_R}} \right)$$ *eq 7.61* 7.72

$$\lambda_s = -\frac{1}{\tau_s} = \frac{-\left(L_v N_r - L_r N_v \right)}{-L_v \left(1 - \frac{u_0}{g} N_p \right) - N_v L_p \left(\frac{u_0}{g} \right)}$$ *eq 7.62* 7.75

$$r = r_0 e^{\lambda_s t} = r_0 e^{-\frac{t}{\tau_s}}$$ *eq 7.63* 7.75

$$\Delta\ddot{\psi} + \left(-N_r \right) \Delta\dot{\psi} + \left(N_v u_0 \right) \Delta\psi = 0$$ *eq 7.64* 7.77

$$\frac{\left| \overline{\Delta\phi} \right|}{\left| -\overline{\Delta\beta} \right|} = \frac{\sqrt{(1)^2 + (2)^2}\left(\left| L_v u_0 \right| \right)}{\sqrt{\omega_n^4 + L_p^2 \omega_n^2}}$$ *eq 7.65* 7.79

$$\angle \frac{\Delta\phi}{\left(-\Delta\beta\right)} = \tan^{-1}\left[\frac{-\left(-L_p\right)}{-\omega_n} \right]$$ *eq 7.66* 7.80

$$\angle \frac{\Delta\phi}{\left(-\Delta\beta\right)} = \tan^{-1} \frac{-\left[\frac{\left(-L_p\right)}{\omega_n} - 2\zeta \right]}{-\left[1 + \zeta \frac{\left(-L_p\right)}{\omega_n} \right]}$$ *eq 7.67* 7.81

$$\zeta = \frac{1}{2\omega_n}\left[-N_r - Y_v - \left(\frac{g}{u_0} - N_p \right) \frac{L_v u_0}{\left(\omega_n^2 + L_p^2 \right)} \right]$$ *eq 7.68* 7.81

$$\zeta = \left(-\frac{1}{2} \right)\left(\sqrt{\frac{-gL_v}{\left| L_p \right|^3}} \right)$$ *eq 7.69* 7.89

$$\Delta t = t_2 - t_1 = t_3 - t_2$$ *eq 7.70* 7.110

$$A_1 = p_2 - p_1$$ *eq 7.71* 7.110

$$A_2 = p_3 - p_2$$ *eq 7.72* 7.111

$$\tau = \frac{\Delta t}{\ln_e\left(\frac{A_1}{A_2} \right)}$$ *eq 7.73* 7.111

$$\tau = \frac{(t_4 - t_2)}{\ln_e\left(\frac{\Delta p_A}{\Delta p_B}\right)}$$

eq 7.74 7.111

$$\omega_d = \frac{2\pi}{P}$$

eq 7.75 7.113

$$\omega_n = \frac{\omega_d}{\sqrt{1 - \zeta^2}} = \frac{\frac{2\pi}{P}}{\sqrt{1 - \zeta^2}}$$

eq 7.76 7.113

CHAPTER SEVEN

FORWARD FLIGHT LATERAL-DIRECTIONAL STABILITY, CONTROL, AND FLYING QUALITIES

7.1 INTRODUCTION

This chapter presents the analytic background and flight test techniques associated with the lateral-directional stability, control, and flying qualities of helicopters in powered forward flight above transitional speeds. The objective of the analytic development is to provide an understanding of the factors affecting pilot opinion of the helicopter flying qualities while performing mission tasks.

Discussion of lateral-directional characteristics involves some greater complexity than the previous longitudinal chapter because two rotational axes are involved, including considerable coupling between them. In addition, many of these characteristics are a function of forward speed.

The subject matter is divided into the following topic areas: lateral-directional stability derivatives, trim characteristics, characteristics in static or steady flight conditions, dynamics, control response, and gust response. The chapter concentrates on the single rotor helicopter with some discussion of the lateral-directional characteristics of the tandem.

Lateral control of a helicopter rotor is achieved in a manner identical to longitudinal control. However, because the fuselage roll moment of inertia is less than the pitch moment of inertia, the gearing ratio between the lateral cyclic control and the swashplate is normally less than in the longitudinal control system. In some flight conditions, the thrust vector is tilted laterally to balance side forces on the fuselage developed as the result of lateral velocity and tail rotor thrust. The control gearing, moment of inertia, side force characteristic, and lateral speed stability together provide lateral control characteristics in forward flight different than for the longitudinal axis.

The side force characteristic, analogous to longitudinal drag, is normally a larger contribution to the lateral force equation than drag is in the longitudinal equation. The tail rotor, of a single rotor helicopter, always provides a significant side force and the fuselage lateral flat plate area is greater than the frontal flat plate area. The tandem helicopter does not have a tail rotor, but the fuselage tends to produce higher drag in response to lateral velocity because of its size and shape. Generally, in both the single main rotor and tandem rotor configurations, there are no surfaces generating roll moments which are equivalent to the longitudinal moments produced by the horizontal stabilizer. Therefore, any roll moments generated by the rotor, as it is tilted to balance the lateral force equation, results in a substantial bank angle. However, the tail rotor and/or vertical tail provide forces stabilizing yaw motion analogous to the action of the horizontal tail in restraining pitch motion.

7.2 PURPOSE OF TEST

The purpose of these tests is to evaluate the helicopter forward flight lateral-directional stability, control, and flying qualities. The tests included in the evaluation are:

1. Mechanical Characteristics
2. Trimmed control positions.
3. Static stability.
4. Dynamic stability.
5. Spiral stability.
6. Control response.
7. Gust response.

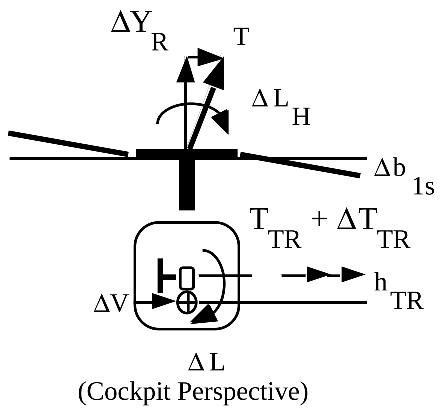
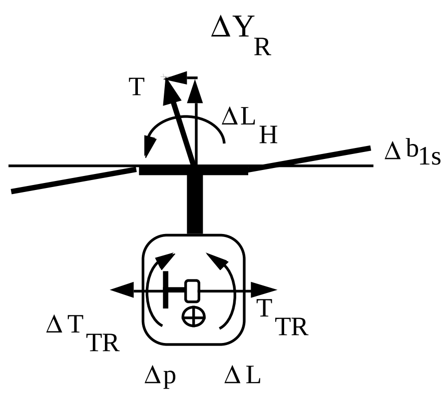
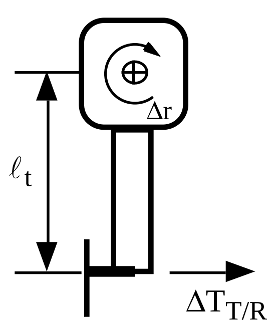
7.3 THEORY

7.3.1 Summary of Quasi-Static Rotor Characteristics

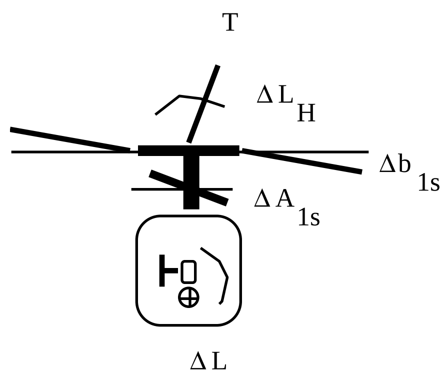
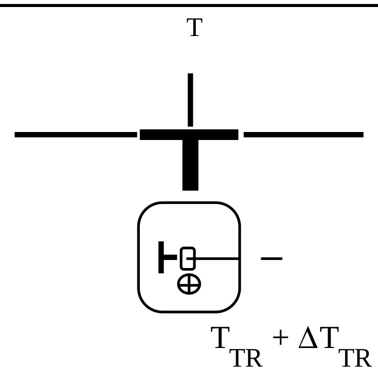
The basic rotor characteristics presented in Chapter 4 indicate a quasi-static approximation for rotor responses which is satisfactory for helicopter stability, control, and flying qualities analysis. Table 7.I presents the summaries of quasi-static lateral rotor response characteristics in forward flight. It is included for reference in discussing the stability derivatives and the helicopter's response. The table summarizes the rotor contributions to stability derivatives by indicating the most significant rotor responses.

Table 7.I

Quasi-Static Rotor Characteristics Affecting Lateral-Directional Derivatives

<table>
<tr><th></th><th>Perturbation</th><th>Rotor Response</th><th>Force and Moments Applied to Aircraft</th></tr>
<tr><td>
</td><td>(a)
Left Lateral Velocity increase,
Δv
(-)</td><td>Rotor tilts away from relative velocity, right tilt, stable, (blowback)
$\frac{\partial b_{1s}}{\partial v}$ (+)</td><td>$T\Delta b_{1s}$ right
ΔY_R right
ΔL_H right
ΔT_{TR} right</td></tr>
<tr><td>
</td><td>(b)
Right Roll Rate
Δp
(+)</td><td>Rotor tilts away from roll
$\frac{\partial b_{1s}}{\partial v}$
(-)</td><td>$T\Delta b_{1s}$ left, provides roll damping
ΔY_R left, increases roll damping
ΔL_H left, increases roll damping
ΔT_{TR} left, increases roll damping</td></tr>
<tr><td>
</td><td>(c)
Yaw rate,
Δr, right
(+)</td><td></td><td>ΔT_{TR}, ℓ_t, right, provides yaw damping moment.</td></tr>
</table>

Quasi-Static Rotor Characteristics Affecting Lateral-Directional Derivatives (cont'd)

	(d) Cyclic pitch, ΔA_{1s}, right (+)	TPP tilts right due to cyclic pitch, $+\dfrac{\partial b_{1s}}{\partial A_{1s}}$	$T\Delta b_{1s}$ right, right moment about CG ΔL_H right, increases control moment about CG
	(e) Tail rotor pitch angle $\Delta\theta_{TR}$ (+)		ΔT_{TR} right, provides left yaw moment (-).

7.3.2 Equations of Motion

The three lateral-directional equations of motion for small disturbances form the basis for the discussions of the individual stability derivatives in the next section. These equations are referred to stability axes and assume near level flight, $\cos\gamma_0 \approx 1$. They do not include couplings with longitudinal motion. Inertial coupling and cross axis acceleration couplings in yaw and roll are not considered. Equations are presented in terms of the main and tail rotor variables $(A_{1s} \text{ and } \theta_{TR})$ and in terms of the cockpit control variables $(\delta_{LAT} \text{ and } \delta_{PED})$. The relationship between the two forms is the gearing ratio between the rotor variables and the cockpit flight control variables. The equilibrium equations for side force, roll moment, and yaw moment are:

$$\dot{v} - Y_v\Delta v - Y_r\Delta r + u_0\Delta r - Y_p\Delta p - g\phi = Y_{A_{1s}}\Delta A_{1s} + Y_{\theta_{TR}}\Delta\theta_{TR}$$

$$= Y_{\delta_{LAT}}\Delta\delta_{LAT} + Y_{\delta_{PED}}\Delta\delta_{PED}$$

eq 7.1

$$\dot{p} - L_v\Delta v - L_r\Delta r - L_p\Delta p = L_{A_{1s}}\Delta A_{1s} + L_{\theta_{TR}}\Delta\theta_{TR}$$
$$= L_{\delta_{LAT}}\Delta\delta_{LAT} + L_{\delta_{PED}}\Delta\delta_{PED} \qquad \textit{eq 7.2}$$

$$\dot{r} - N_v\Delta v - N_p\Delta p - N_r\Delta r = N_{A_{1s}}\Delta A_{1s} + N_{\theta_{TR}}\Delta\theta_{TR}$$
$$= N_{\delta_{LAT}}\Delta\delta_{LAT} + N_{\delta_{PED}}\Delta\delta_{PED} \qquad \textit{eq 7.3}$$

Conventional stability derivative notation is used in which the force derivatives are normalized with respect to mass (m) and the moment derivatives are normalized with respect to moment of inertia (I_{zz}, I_{xx}):

$$Y_{(\,)} = \frac{1}{m}\frac{\partial Y}{\partial(\,)}$$

$$L_{(\,)} = \frac{1}{I_{xx}}\frac{\partial L}{\partial(\,)}$$

$$N_{(\,)} = \frac{1}{I_{zz}}\frac{\partial N}{\partial(\,)}$$

Where:

A_{1s}	- Lateral cyclic pitch angle, shaft referenced
δ_{LAT}	- Lateral control
δ_{PED}	- Pedal control
ϕ	- Roll angle
g	- Gravity
I_{xx}	- Moment of inertia about x axis, roll moment of inertia
I_{zz}	- Moment of inertia about z axis, yaw moment of inertia
L	- Net moment about x axis, Roll moment

$L_{A_{1s}}$	- Roll moment due to lateral cyclic pitch angle
$L_{\delta_{LAT}}$	- Roll moment due to lateral control
$L_{\delta_{PED}}$	- Roll moment due to pedal control
L_p	- Roll moment due to roll rate
$L_{\theta_{TR}}$	- Roll moment due to tail rotor pitch angle
L_r	- Roll moment due to yaw rate
L_v	- Roll moment due to lateral velocity
m	- Mass
N	- Net moment about z axis, Yaw moment
$N_{A_{1s}}$	- Yaw moment due to lateral cyclic pitch angle
$N_{\delta_{LAT}}$	- Yaw moment due to lateral control
$N_{\delta_{PED}}$	- Yaw moment due to pedal control
N_p	- Yaw moment due to roll rate
$N_{\theta_{TR}}$	- Yaw moment due to tail rotor pitch angle
N_r	- Yaw moment due to yaw rate
N_v	- Yaw moment due to lateral velocity
p	- Angular velocity about x axis, Roll rate
$\dot{p}$	- Angular acceleration about x axis
θ_{TR}	- Tail rotor pitch angle
r	- Angular velocity about z axis, Yaw rate
$\dot{r}$	- Angular acceleration about z axis
u_0	- Initial velocity
v	- Translational velocity component along y axis
$\dot{v}$	- Linear acceleration along y axis
Y	- Resultant force in y direction
$Y_{A_{1s}}$	- Side force due to lateral cyclic pitch angle
$Y_{\delta_{LAT}}$	- Side force due to lateral control
$Y_{\delta_{PED}}$	- Side force due to pedal control
Y_p	- Side force due to roll rate
$Y_{\theta_{TR}}$	- Side force due to tail rotor pitch angle
Y_r	- Side force due to yaw rate
Y_v	- Side force due to lateral velocity.

7.3.3 Lateral-Directional Derivatives

The quasi-static tip path plane (TPP) motion presented in Section 7.3.1 is adequate for explaining most helicopter stability and control problems because the rotor blade dynamic responses are rapid and well damped. This quasi-static approximation expresses the TPP variables in terms of the rigid body motions of the helicopter and incorporate their influence on the overall stability derivatives. The derivatives which have the greatest influence on stability and control characteristics are discussed in this section.

Figure 7.1 shows the forces and moments acting on the helicopter which contribute to the side force, roll moment, and yaw moment about the center of gravity (CG). The forces include the lateral thrust component $(\approx T\Delta b_{1s})$,the rotor side force (Y_R) which acts parallel to the TPP, the side force on the fuselage/tail (Y_{f+t}), and tail rotor thrust (T_{TR}). The diagram indicates the roll moment due to the fuselage/tail (L_{f+t}), and due to the offset flapping hinge (L_H). There are roll moments about the CG due to the rotor Y_R force and side component of rotor thrust, both acting through a moment arm (h), and due to the tail rotor thrust acting through the arm h_{TR}.

The torque of the main rotor (Q_{MR}) and the yaw moment due to the fuselage/tail (N_{f+t}) are indicated. Yaw moments are produced by the tail rotor thrust acting through the arm (l_t) and by the main rotor side forces acting through the arm h'. The CG is assumed in the xz plane of the helicopter.

The lateral-directional stability derivatives give the change in the total aerodynamic side force, roll moment, and yaw moment resulting from a change in one of the lateral-directional flight or control variables v, p, r, θ_{TR} (or δ_{PED}), and A_{1s} (or δ_{LAT}) while holding the remaining variables constant. These derivatives are expressed as a sum of

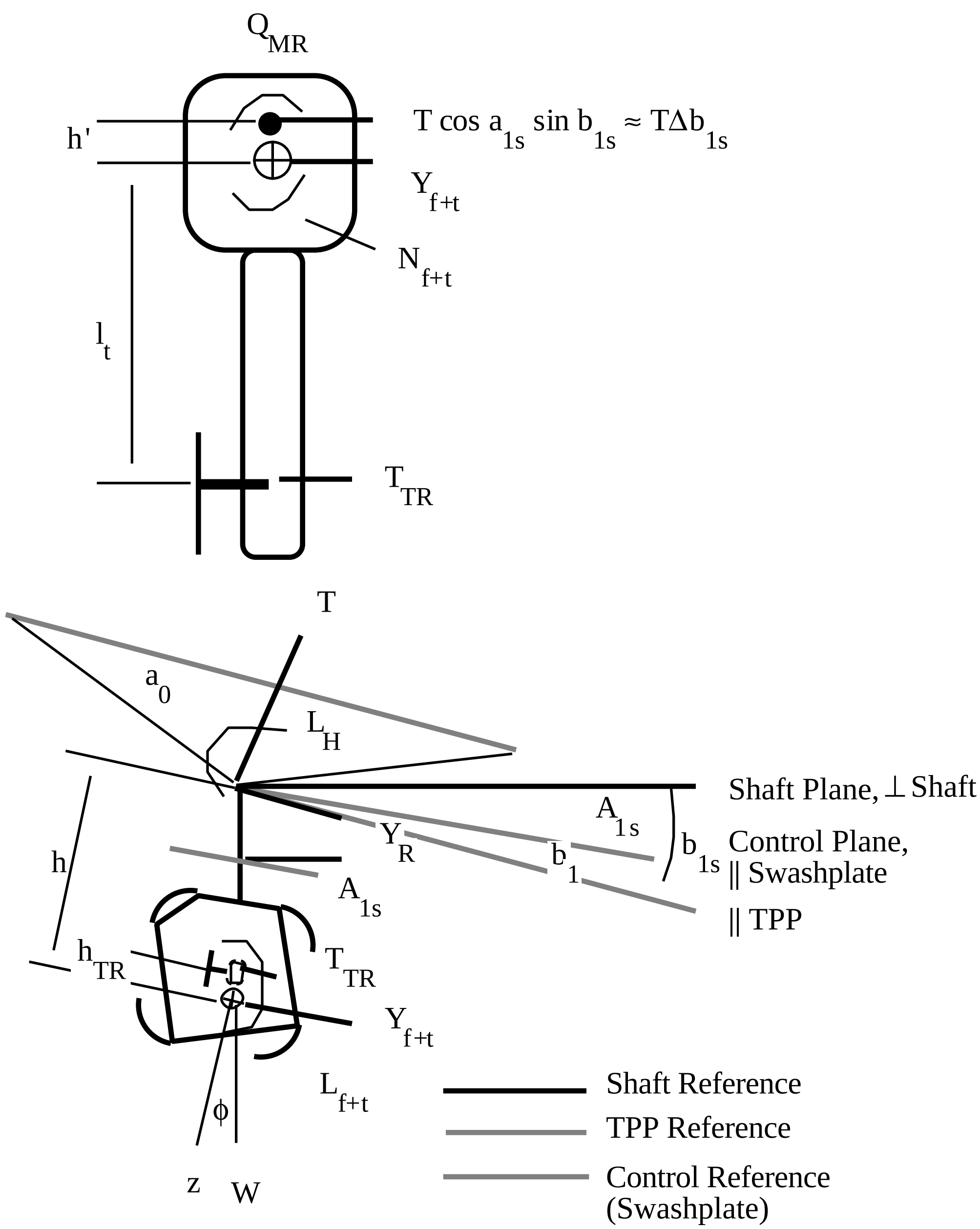


Figure 7.1
Forces and Moments Contributing to Lateral-Directional Stability Derivatives

terms showing the effects of changes in the quantities in Figure 7.1 with respect to the variable of interest. The incremental changes in side force (ΔY), roll moment (ΔL), and yaw moment (ΔN) due to such changes are:

$$\underset{(1)}{\Delta Y = T\,\Delta b_{1s}} + \underset{(2)}{b_{1s}\,\Delta T} + \underset{(3)}{\Delta Y_R} + \underset{(4)}{\Delta T_{TR}} + \underset{(5)}{\Delta Y_{f+t}} \qquad \textit{eq 7.4}$$

$$\Delta L = \underset{(1)}{(Th + \frac{1}{2}\,ebM_S\Omega^2)\Delta b_{1s}} + \underset{(2)}{hb_{1s}\Delta T} + \underset{(3)}{h\Delta Y_R} + \underset{(4)}{h_{TR}\Delta T_{TR}} + \underset{(5)}{\Delta L_{f+t}} \qquad \textit{eq 7.5}$$

$$\Delta N = \underset{(1)}{Th'\Delta b_{1s}} + \underset{(2)}{h'b_{1s}\Delta T} + \underset{(3)}{h'\Delta Y_R} - \underset{(4)}{l_t\Delta T_{TR}} + \underset{(5)}{\Delta N_{f+t}} + \underset{(6)}{\Delta Q_{MR}} \qquad \textit{eq 7.6}$$

Where:

b	- Number of blades
b_{1s}	- Lateral flapping angle, shaft referenced
e	- Flapping hinge offset
h	- Height of hub above CG
h'	- Longitudinal distance between the rotor shaft and the CG
h_{TR}	- Height of the tail rotor above the CG
L	- Net moment about x axis, Roll moment
L_{f+t}	- Roll moment due to the fuselage/tail
l_t	- Distance from the tail to the CG
M_S	- Blade mass moment
N	- Net moment about z axis, Yaw moment
N_{f+t}	- Yaw moment due to the fuselage/tail
Q_{MR}	- Main rotor torque
T	- Thrust

T_{TR} - Tail rotor thrust

Ω - Rotor angular velocity

Y - Resultant force in y direction

Y_{f+t} - Side force due to the fuselage/tail

Y_R - Rotor side force.

The physical explanations for the six types of terms are as follows:

(1) Terms proportional to change in lateral tilt of TPP (Δb_{1s}):

$T\Delta b_{1s}$, increment in right lateral component of rotor thrust.

$Th\Delta b_{1s}$, $Th'\Delta b_{1s}$, roll and yaw moments produced by side force increment due to Δb_{1s}.

$\frac{1}{2}\, ebM_S\Omega^2\Delta b_{1s}$, offset hinge moment corresponding to right TPP tilt.

(2) Terms proportional to change in thrust increment ΔT:

$b_{1s}\Delta T$, increment in lateral component of rotor thrust due to thrust increment ΔT for a trim condition with TPP tilt (b_{1s}).

$hb_{1s}\Delta T$, $h'b_{1s}\Delta T$, roll and yaw moments produced by side force increment due to ΔT.

(3) Terms proportional to the increment in the rotor side force (ΔY_R):

$h\Delta Y_R$, $h'\Delta Y_R$, roll and yaw moment increments produced by side force increment (ΔY_R).

(4) Terms proportional to tail rotor thrust increment (ΔT_{TR}):

$h_{TR}\Delta T_{TR}$, $l_T\Delta T_{TR}$, roll and yaw moment increments due to tail rotor thrust increment, ΔT_{TR}, acting through moment arm h_{TR} and l_t.

(5) Terms proportional to increments in the aerodynamic forces acting on the fuselage/tail.

(6) Increment in yaw moment due to a change in main rotor torque.

The distance of the CG aft of the rotor shaft (h') is usually small for conventional helicopters and is assumed zero to simplify the discussion of the yaw moment derivatives. A further simplification is made by assuming constant collective, thereby neglecting the type (2) terms for the variation in side force, roll moment, and yaw moment produced by thrust changes.

The values of the stability derivatives as a function of airspeed for the BO-105, the OH-6A, and the CH-53D are presented at the end of each subsection.

7.3.3.1 YAW MOMENT DUE TO LATERAL VELOCITY, DIRECTIONAL STABILITY

$$N_v = \frac{1}{I_{zz}}\left[\underset{(1)}{\underset{\Uparrow}{Th'\frac{\partial b_{1s}}{\partial v}}} + \underset{(2)}{\underset{\Uparrow}{h'b_{1s}\frac{\partial T}{\partial v}}} + \underset{(3)}{\underset{\Uparrow}{h'\frac{\partial Y_R}{\partial v}}} - \underset{(4)}{\underset{\Uparrow}{l_t\frac{\partial T_{TR}}{\partial v}}} + \underset{(5)}{\underset{\Uparrow}{\frac{\partial N_{f+t}}{\partial v}}} + \underset{(6)}{\underset{\Uparrow}{\frac{\partial Q_{MR}}{\partial v}}}\right] \qquad \textit{eq 7.7}$$

$$N_v \approx \frac{1}{I_{zz}}\left[\underset{(4)}{\underset{\Uparrow}{-l_t\frac{\partial T_{TR}}{\partial v}}} + \underset{(5)}{\underset{\Uparrow}{\frac{\partial N_{f+t}}{\partial v}}}\right] \qquad \textit{eq 7. 8}$$

Where:

b_{1s} - Lateral flapping angle, shaft referenced

h' - Longitudinal distance between the rotor shaft and the CG

I_{zz} - Moment of inertia about z axis, yaw moment of inertia

l_t	- Distance from the tail to the CG
N_{f+t}	- Yaw moment due to the fuselage/tail
N_v	- Yaw moment due to lateral velocity
Q_{MR}	- Main rotor torque
T	- Thrust
T_{TR}	- Tail rotor thrust
v	- Translational velocity component along y axis
Y_R	- Rotor side force.

Since h' is usually small and there is little change in main rotor thrust with sideslip, the directional stability derivative is primarily due to the tail rotor and fuselage/tail terms.

The fuselage contribution is usually destabilizing. A vertical fin provides a stabilizing moment, and its effectiveness increases approximately linearly with forward speed. The tail rotor term is the largest contributor to the directional stability derivative and does not vary greatly with forward speed.

Several effects can result in nonlinearities causing N_v to vary with sideslip angle. The tail rotor and/or vertical tail in a turbulent flow region due to separation from the fuselage, as indicated in Figure 7.2, causes a reduction in directional stability at small angles.

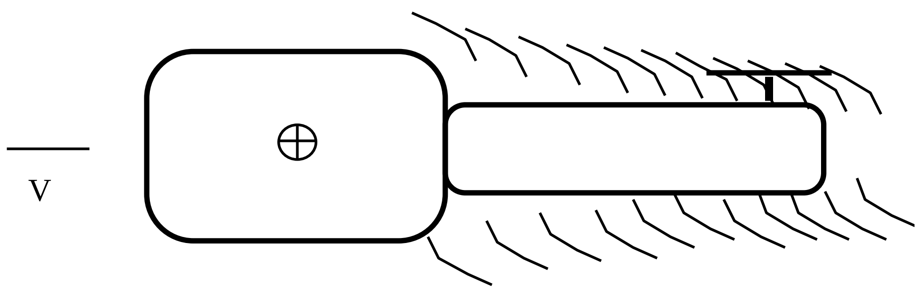


Figure 7.2
Loss of Directional Stability Due to Fuselage/Tail Rotor Interference

The tip vortices of the main rotor are swept aft in forward flight and tend to roll up, producing two trailing vortices similar to those of a fixed wing aircraft. Figure 7.3 indicates the flow field induced by this vortex system in the region of the vertical tail and tail rotor of a conventional helicopter.

The induced flow of the right and left trailing vortices is superimposed on the undisturbed flow, resulting in a diverging flow for the low tail rotor position, and a converging flow for the high tail rotor position. The diagram shows the effect of yawing the helicopter relative to the undisturbed flow. A smaller inclination of the tail rotor is obtained relative to the diverging flow lines in the low tail position resulting in a reduction of directional stability.

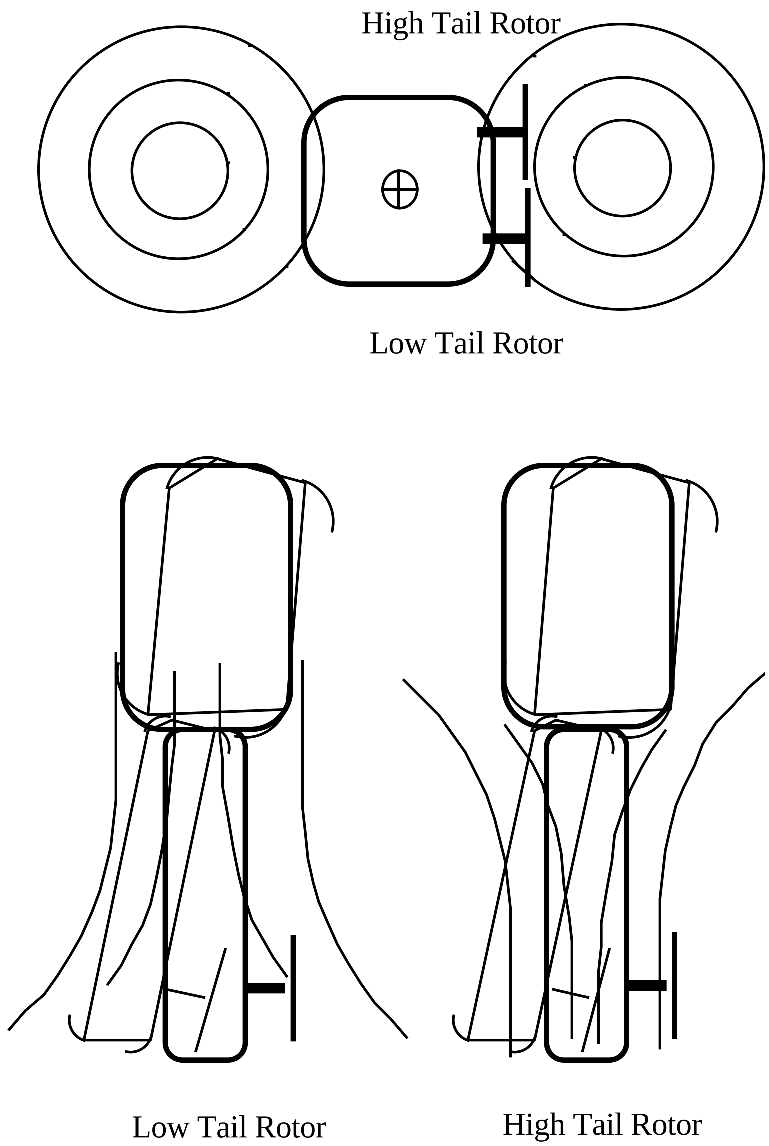


Figure 7.3
Flow Field at Tail Rotor

The trends of N_v with airspeed are shown for three helicopters in Figure 7.4.

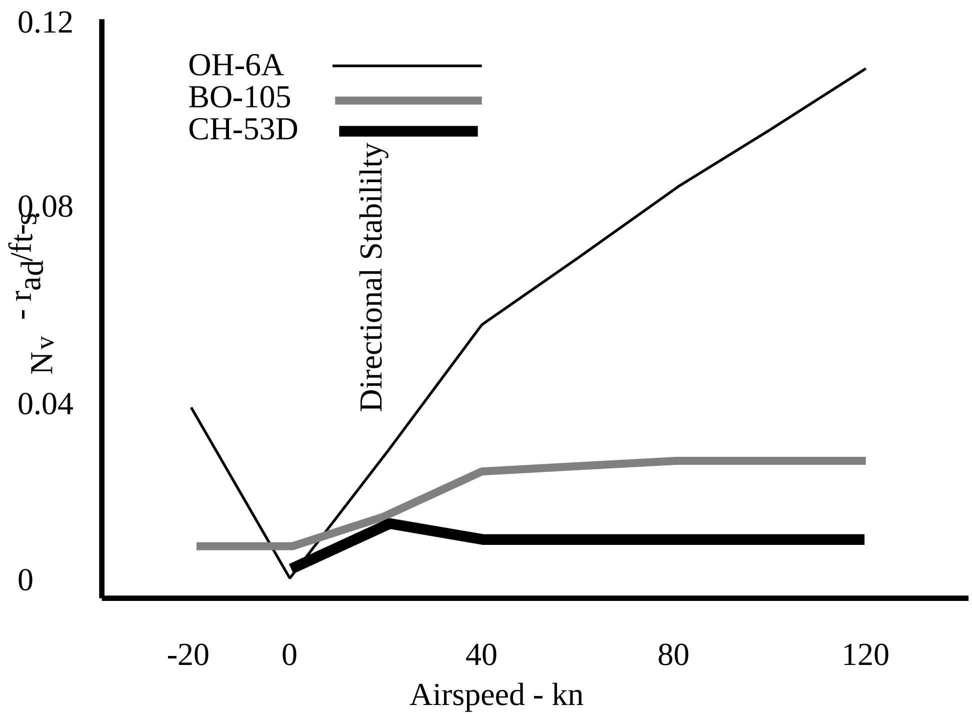


Figure 7.4
Directional Stability Versus Airspeed

7.3.3.2 YAW MOMENT DUE TO YAW RATE, YAW DAMPING

$$N_r = \frac{1}{I_{zz}}\left[Th' \frac{\partial b_{1s}}{\partial r} + h' \, b_{1s} \frac{\partial T}{\partial r} + h' \frac{\partial Y_R}{\partial r} - l_t \frac{\partial T_{TR}}{\partial r} + \frac{\partial N_{f+t}}{\partial r} + \frac{\partial Q_{MR}}{\partial r} \right] \qquad \textit{eq 7.9}$$

$$\begin{array}{cccccc} \Uparrow & \Uparrow & \Uparrow & \Uparrow & \Uparrow & \Uparrow \\ (1) & (2) & (3) & (4) & (5) & (6) \end{array}$$

$$N_r \approx \frac{1}{I_{zz}}\left[\; -l_t \underbrace{\frac{\partial T_{TR}}{\partial r}}_{(4)} + \underbrace{\frac{\partial N_{f+t}}{\partial r}}_{(5)} \;\right] \qquad \textit{eq 7.10}$$

Where:

b_{1s}	- Lateral flapping angle, shaft referenced
h'	- Longitudinal distance between the rotor shaft and the CG
I_{zz}	- Moment of inertia about z axis, yaw moment of inertia
l_t	- Distance from the tail to the CG
N	- Net moment about z axis, Yaw moment
N_{f+t}	- Yaw moment due to the fuselage/tail
Q_{MR}	- Main rotor torque
r	- Angular velocity about z axis, Yaw rate
T	- Thrust
T_{TR}	- Tail rotor thrust
Y_R	- Rotor side force.

Terms (1), (2), and (3) are small since they are multiplied by the small distance h'. Term (6), the main rotor torque variation with yaw rate, is insignificant because the yaw rate change is small compared to the rotor angular velocity (Ω).

Term (4) due to the tail rotor is usually the largest contribution to the yaw damping derivative and is essentially independent of airspeed. A positive yaw rate gives the tail rotor a velocity to the left which increases the effective angles of attack of the tail rotor blades. This gives an increment in tail rotor thrust to the right, damping the yaw motion. The fuselage/tail term (5) contribution to yaw rate damping increases approximately linearly with forward speed.

Typical trends of N_r with airspeed are shown in Figure 7.5.

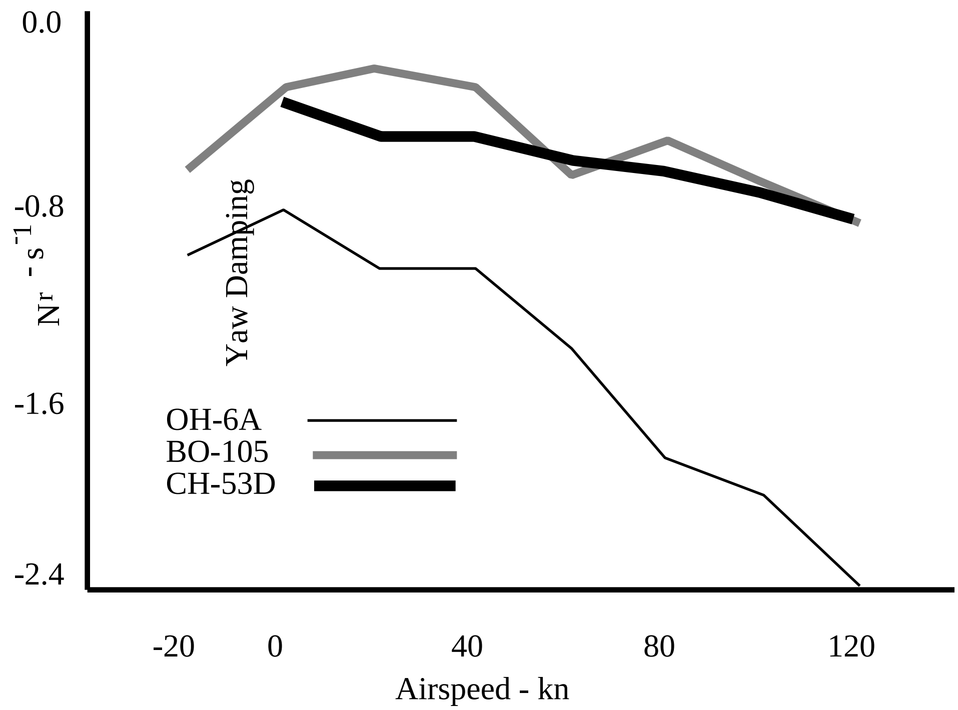


Figure 7.5
Yaw Damping Versus Airspeed

7.3.3.3 YAW MOMENT DUE TO ROLL RATE

$$N_p = \frac{1}{I_{zz}}\left[Th'\frac{\partial b_{1s}}{\partial p} + h'b_{1s}\frac{\partial T}{\partial p} + h'\frac{\partial Y_R}{\partial p} - l_t\frac{\partial T_{TR}}{\partial p} + \frac{\partial N_{f+t}}{\partial p} + \frac{\partial Q_{MR}}{\partial p}\right] \qquad \textit{eq 7.11}$$

$$\Uparrow \quad (1) \qquad \Uparrow \quad (2) \qquad \Uparrow \quad (3) \qquad \Uparrow \quad (4) \qquad \Uparrow \quad (5) \qquad \Uparrow \quad (6)$$

Where:

b_{1s} - Lateral flapping angle, shaft referenced

h' - Longitudinal distance between the rotor shaft and the CG

I_{zz} - Moment of inertia about z axis, yaw moment of inertia

l_t - Distance from the tail to the CG

N_{f+t} - Yaw moment due to the fuselage/tail

N_p - Yaw moment due to roll rate
p - Angular velocity about x axis, Roll rate
Q_{MR} - Main rotor torque
T - Thrust
T_{TR} - Tail rotor thrust
Y_R - Rotor side force.

Since the quantity h' is small, terms (1), (2), and (3) have little effect. High vertical fins and high tail rotor thrust lines might make $\partial N_{f+t} / \partial p$ positive and $\partial T_{TR} / \partial p$ negative. Thus the tail rotor contributions to N_P tends to be positive and is essentially constant since $\partial T_{TR} / \partial p$ is not strongly affected by speed.

Trends of N_p with airspeed are shown in Figure 7.6.

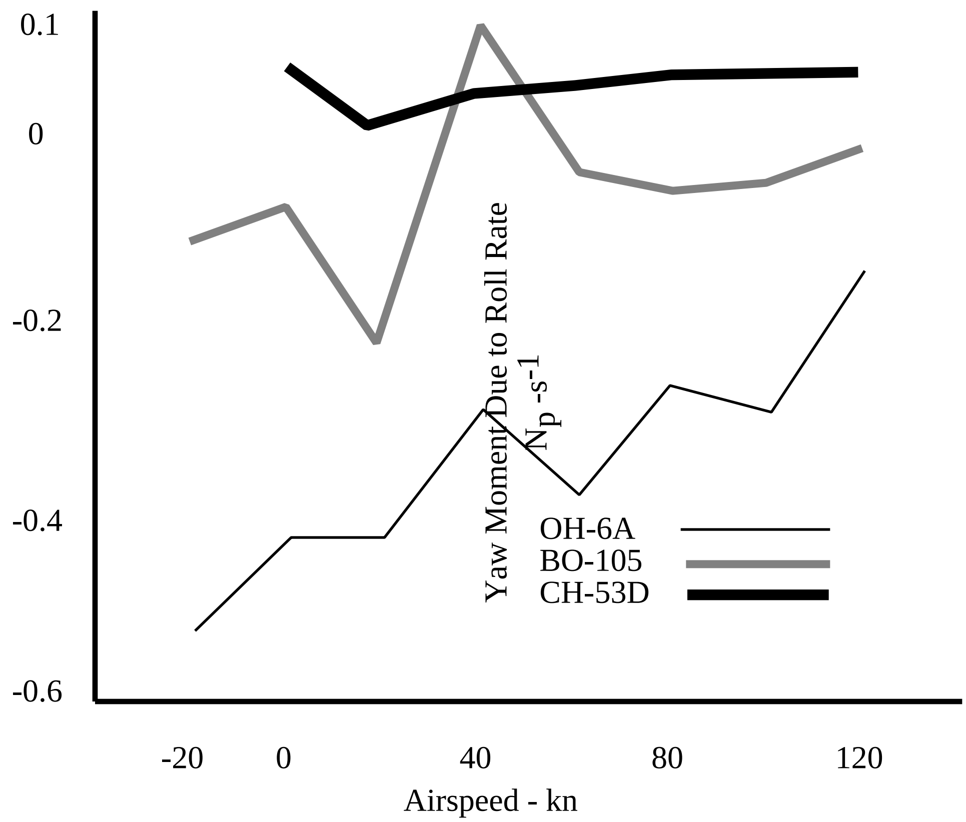


Figure 7.6
Yaw Moment Due to Roll Rate Versus Airspeed

7.3.3.4 YAW MOMENT DUE TO TAIL ROTOR PITCH ANGLE, DIRECTIONAL CONTROL

$$N_{\theta_{TR}} = \frac{1}{I_{zz}}\left[Th'\frac{\partial b_{1s}}{\partial \theta_{TR}} + h'b_{1s}\frac{\partial T}{\partial \theta_{TR}} + h'\frac{\partial Y_R}{\partial \theta_{TR}} - l_t\frac{\partial T_{TR}}{\partial \theta_{TR}} + \frac{\partial N_{f+t}}{\partial \theta_{TR}} + \frac{\partial Q_{MR}}{\partial \theta_{TR}} \right] \qquad \textit{eq 7.12}$$

$$\Uparrow \quad (1) \qquad \Uparrow \quad (2) \qquad \Uparrow \quad (3) \qquad \Uparrow \quad (4) \qquad \Uparrow \quad (5) \qquad \Uparrow \quad (6)$$

$$N_{\theta_{TR}} \approx \frac{1}{I_{zz}}\left[-l_t\frac{\partial T_{TR}}{\partial \theta_{TR}} \right] \qquad \textit{eq 7.13}$$

$$\Uparrow \quad (4)$$

Where:

b_{1s}	- Lateral flapping angle, shaft referenced
h'	- Longitudinal distance between the rotor shaft and the CG
I_{zz}	- Moment of inertia about z axis, yaw moment of inertia
l_t	- Distance from the tail to the CG
N_{f+t}	- Yaw moment due to the fuselage/tail
$N_{\theta_{TR}}$	- Yaw moment due to tail rotor pitch angle
Q_{MR}	- Main rotor torque
θ_{TR}	- Tail rotor pitch angle
T	- Thrust
T_{TR}	- Tail rotor thrust
Y_r	- Rotor side force.

The primary contribution to directional control is the tail rotor thrust due to tail rotor pitch angle assuming: perfect engine governing, no control interconnects, and negligible aerodynamic interference. Control power referred to the cockpit flight controls is obtained by multiplying $N_{\theta_{TR}}$ by the tail rotor pitch angle to pedal control gear ratio:

$$N_{\delta_{PED}} = \left(\frac{\partial \theta_{TR}}{\partial \delta_{PED}}\right) N_{\theta_{TR}} \qquad \textit{eq 7.14}$$

Where:

δ_{PED} - Pedal control

$N_{\delta_{PED}}$ - Yaw moment due to pedal control

$N_{\theta_{TR}}$ - Yaw moment due to tail rotor pitch angle

θ_{TR} - Tail rotor pitch angle.

Directional control is relatively independent of speed because there is little change in $\partial T_{TR}/\partial\theta_{TR}$ with speed as evident in Figure 7.7.

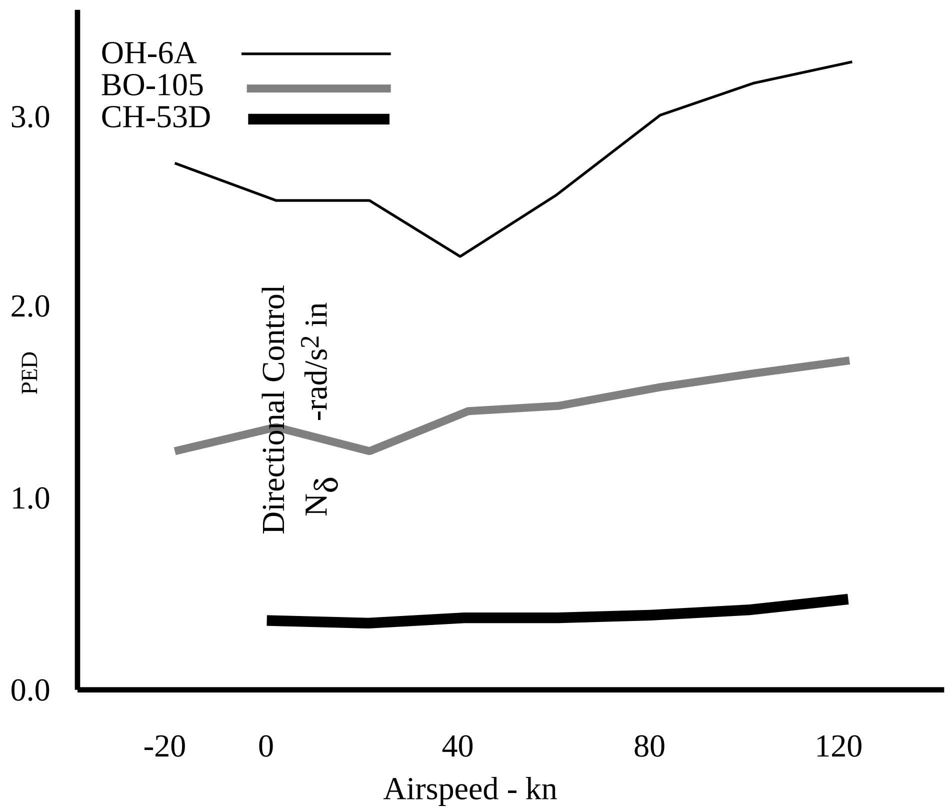


Figure 7.7
Directional Control Versus Airspeed

7.3.3.5 YAW MOMENT DUE TO LATERAL CYCLIC PITCH ANGLE

$$N_{A_{1s}} = \frac{1}{I_{zz}}\left[Th'\frac{\partial b_{1s}}{\partial A_{1s}} + h'b_{1s}\frac{\partial T}{\partial A_{1s}} + h'\frac{\partial Y_R}{\partial A_{1s}} - l_t\frac{\partial T_{TR}}{\partial A_{1s}} + \frac{\partial N_{f+t}}{\partial A_{1s}} + \frac{\partial Q_{MR}}{\partial A_{1s}}\right] \qquad \textit{eq 7.15}$$

$$\Uparrow \qquad \Uparrow \qquad \Uparrow \qquad \Uparrow \qquad \Uparrow \qquad \Uparrow$$
$$(1) \qquad (2) \qquad (3) \qquad (4) \qquad (5) \qquad (6)$$

Where:

A_{1s}	- Lateral cyclic pitch angle, shaft referenced
b_{1s}	- Lateral flapping angle, shaft referenced
h'	- Longitudinal distance between the rotor shaft and the CG
I_{zz}	- Moment of inertia about z axis, yaw moment of inertia
l_t	- Distance from the tail to the CG
$N_{A_{1s}}$	- Yaw moment due to lateral cyclic pitch angle
N_{f+t}	- Yaw moment due to the fuselage/tail
Q_{MR}	- Main rotor torque
T	- Thrust
T_{TR}	- Tail rotor thrust
Y_R	- Rotor side force.

The reasonable contributors to $N_{A_{1s}}$ are the terms $Th'\ \partial b_{1s}/\partial A_{1s}$, $h'\ b_{1s}\ \partial T/\partial A_{1s}$, and $\partial Q_{MR}/\partial A_{1s}$. The partial $\partial b_{1s}/\partial A_{1s}$ is positive, and the sign of $\partial T/\partial A_{1s}$ is most likely dependent on the inherent sideslip angle (negative if inherent sideslip is to the right). In flight observation reveal that $\partial Q_{MR}/\partial A_{1s}$ is negative. The sign of $N_{A_{1s}}$ is dependent on the relative magnitudes of the three contributors and is usually small since two terms are multiplied by h'. Control power referred to the cockpit flight controls is obtained by multiplying $N_{A_{1s}}$ by the lateral cyclic pitch angle to lateral control gear ratio:

$$N_{\delta_{LAT}} = \left(\frac{\partial A_{1s}}{\partial \delta_{LAT}}\right) N_{A_{1s}}$$

eq 7.16

Where:

- A_{1s} - Lateral cyclic pitch angle
- δ_{LAT} - Lateral control
- $N_{A_{1s}}$ - Yaw moment due to lateral cyclic pitch angle
- $N_{\delta_{LAT}}$ - Yaw moment due to lateral control.

Typical variations of $N_{\delta_{LAT}}$ with airspeed are shown in Figure 7.8.

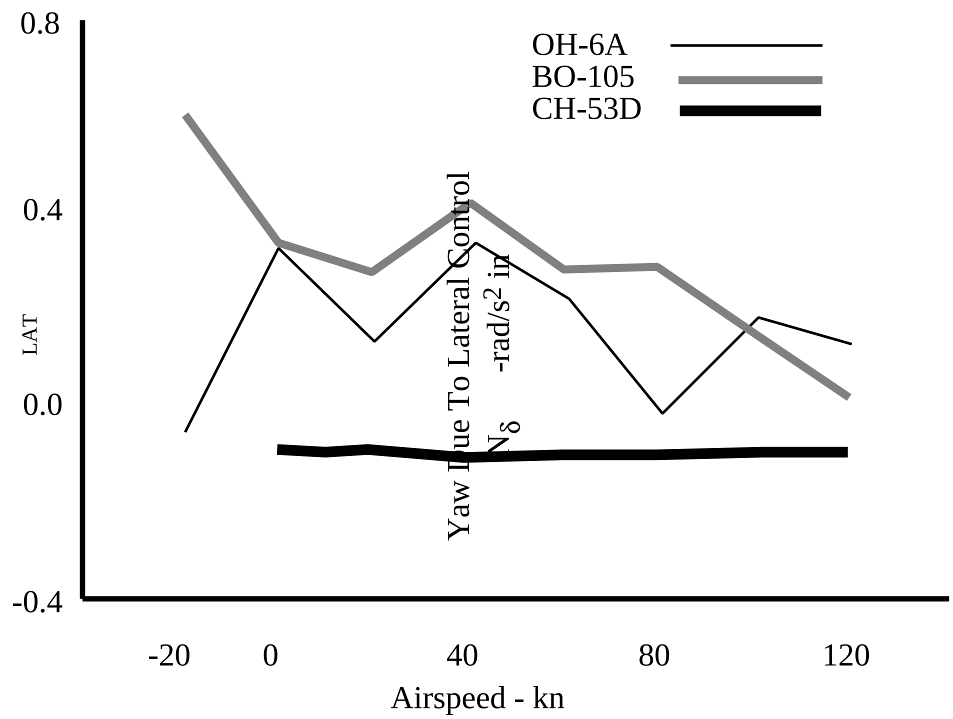


Figure 7.8
Yaw Due to Lateral Cyclic Versus Airspeed

7.3.3.6 ROLL MOMENT DUE TO LATERAL VELOCITY, DIHEDRAL EFFECT

$$L_v = \frac{1}{I_{xx}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial b_{1s}}{\partial v} + hb_{1s}\frac{\partial T}{\partial v} + h\frac{\partial Y_R}{\partial v} + h_{TR}\frac{\partial T_{TR}}{\partial v} + \frac{\partial L_{f+t}}{\partial v}\right]$$

eq 7.17

$$\Uparrow \quad (1) \qquad \Uparrow \quad (2) \qquad \Uparrow \quad (3) \qquad \Uparrow \quad (4) \qquad \Uparrow \quad (5)$$

Where:

b	- Number of blades
b_{1s}	- Lateral flapping angle, shaft referenced
e	- Flapping hinge offset
h	- Height of hub above CG
h_{TR}	- Height of the tail rotor above the CG
I_{xx}	- Moment of inertia about x axis, roll moment of inertia
L_{f+t}	- Roll moment due to the fuselage/tail
L_v	- Roll moment due to lateral velocity
M_S	- Blade mass moment
T	- Thrust
T_{TR}	- Tail rotor thrust
v	- Translational velocity component along y axis
Ω	- Rotor angular velocity
Y_R	- Rotor side force.

Conventional helicopter notation is positive lateral velocity (v) to the right, and a positive roll moment (L) right wing down. However, positive effective dihedral is defined as right wing upward in a right sideslip. Thus, the roll moment due to lateral velocity derivative is negative in the case of a helicopter with positive dihedral effect.

Term (1), roll moment due to rotor blowback, is negative and is usually the largest contributor to L_v. Term (2) is usually negligible while term (3), due to the in plane rotor

Y_R force, makes a small negative contribution.

A right sideslip velocity produces a negative increment in effective tail rotor blade pitch and tail rotor thrust. Thus, term (4) is negative when the tail rotor is above the roll axis and can be of comparable magnitude to term (1). Term (5), due to the fuselage/tail, is positive or negative depending on the location of the vertical tail relative to the roll axis which can develop large side forces in sideslips. Figure 7.9 shows a schematic diagram of the side force on a helicopter fuselage in a sideslip. This force results in a positive roll moment because of the high CG location typical of helicopters and decreases the dihedral effect.

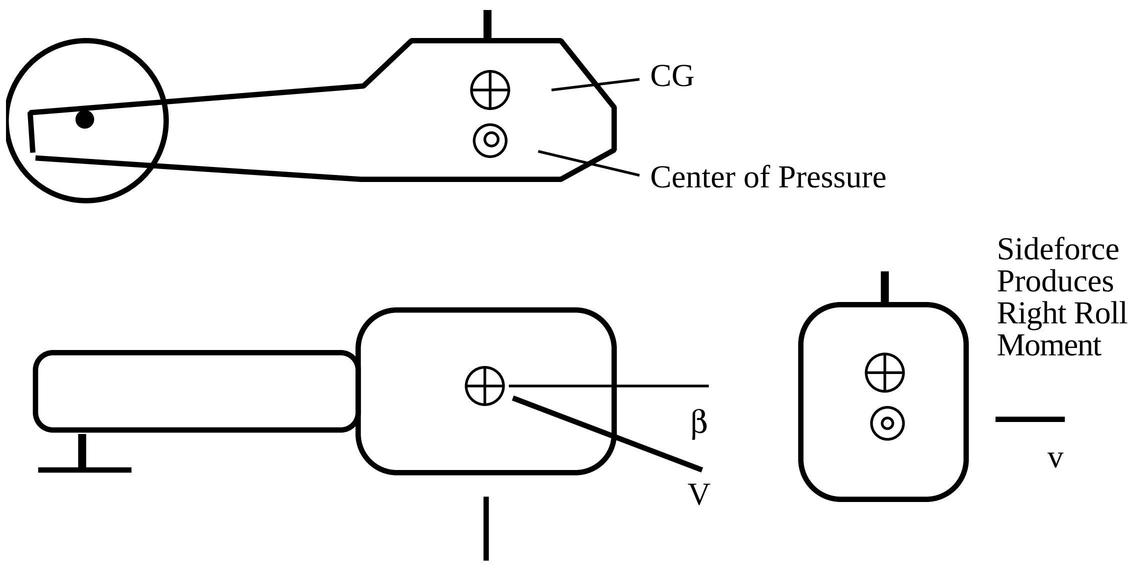


Figure 7.9
Unstable Dihedral Effect Due to Fuselage Side Forces

Generally, the total L_V derivative is negative for single rotor helicopters giving them positive dihedral effect. Typical variations of L_V with airspeed are shown in Figure 7.10.

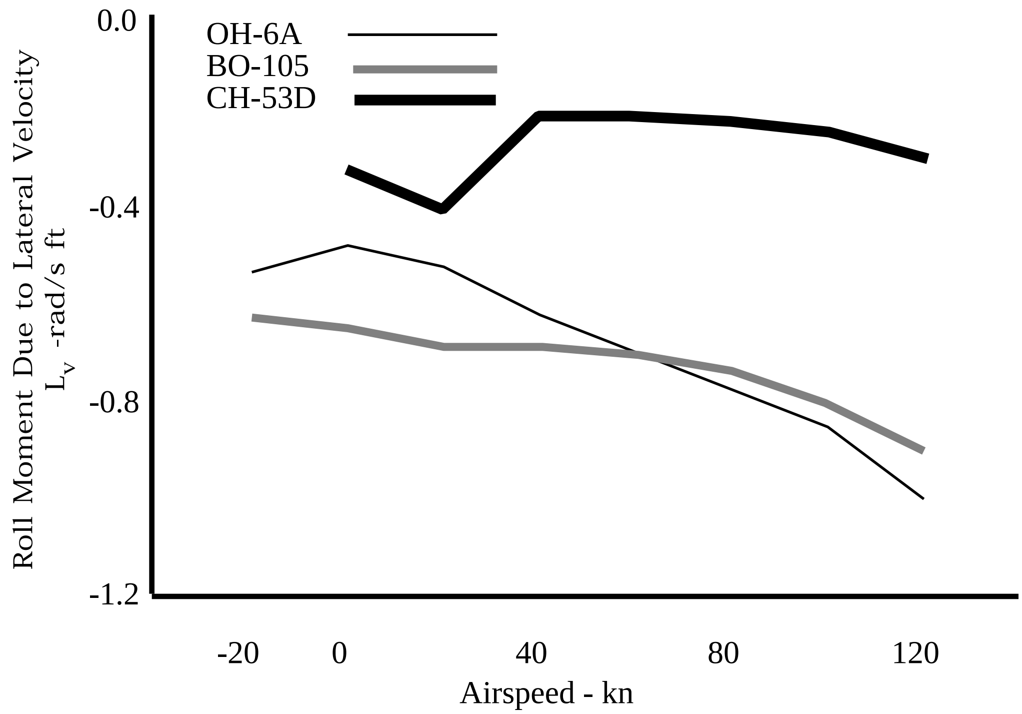


Figure 7.10
Roll Moment Due to Lateral Velocity Versus Airspeed

7.3.3.7 ROLL MOMENT DUE TO YAW RATE

$$L_r = \frac{1}{I_{xx}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial b_{1s}}{\partial r} + hb_{1s}\frac{\partial T}{\partial r} + h\frac{\partial Y_R}{\partial r} + h_{TR}\frac{\partial T_{TR}}{\partial r} + \frac{\partial L_{f+t}}{\partial r}\right]$$

eq 7.18

$$\begin{array}{ccccc} \Uparrow & \Uparrow & \Uparrow & \Uparrow & \Uparrow \\ (1) & (2) & (3) & (4) & (5) \end{array}$$

Where:

b	- Number of blades
b_{1s}	- Lateral flapping angle, shaft referenced

e	- Flapping hinge offset
h	- Height of hub above CG
h_{TR}	- Height of the tail rotor above the CG
I_{xx}	- Moment of inertia about x axis, roll moment of inertia
L_{f+t}	- Roll moment due to the fuselage/tail
L_r	- Roll moment due to yaw rate
M_S	- Blade mass moment
r	- Angular velocity about z axis, Yaw rate
T	- Thrust
T_{TR}	- Tail rotor thrust
Ω	- Rotor angular velocity
Y_R	- Rotor side force.

The partial $\partial b_{1s}/\partial r$ is dependent on longitudinal CG position. The partial $\partial T/\partial r$ is dependent on both longitudinal CG position and the trim value of b_{1s}. The contributions involving these partials are normally not significant.

The fuselage probably does not make any significant contribution except with a large, high vertical tail or if wings are present. The primary contributor for typical helicopter configurations is the tail rotor. Typical variations of L_r with airspeed are shown in Figure 7.11.

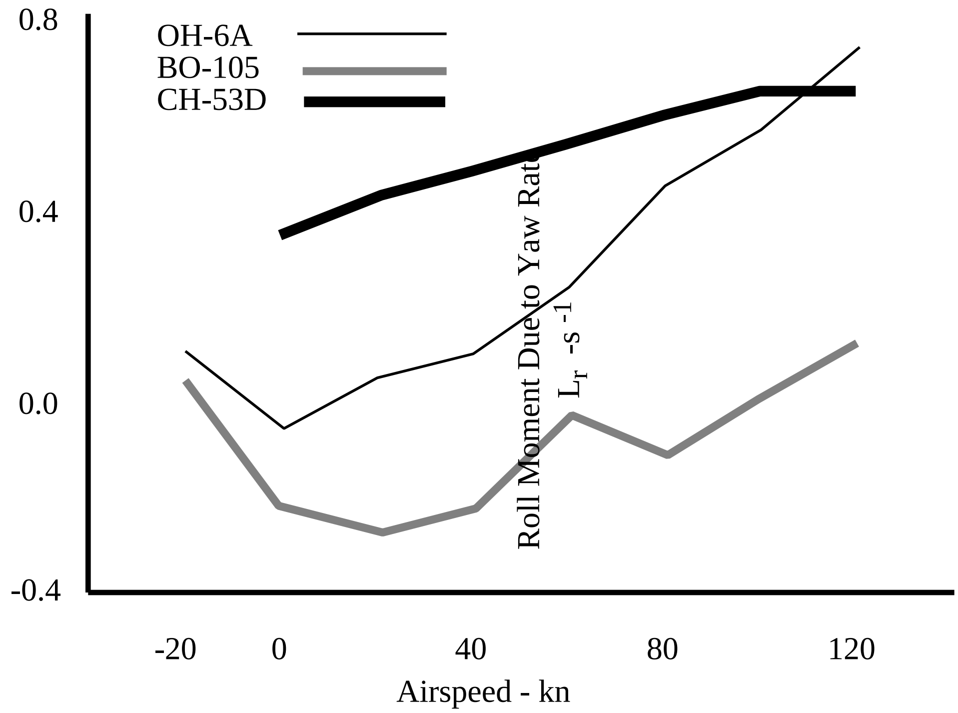


Figure 7.11
Roll Moment Due to Yaw Rate Versus Airspeed

7.3.3.8 ROLL MOMENT DUE TO ROLL RATE, ROLL DAMPING

$$L_p = \frac{1}{I_{xx}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial b_{1s}}{\partial p} + hb_{1s}\frac{\partial T}{\partial p} + h\frac{\partial Y_R}{\partial p} + h_{TR}\frac{\partial T_{TR}}{\partial p} + \frac{\partial L_{f+t}}{\partial p}\right]$$

eq 7.19

$$\Uparrow \quad (1) \qquad \Uparrow \quad (2) \qquad \Uparrow \quad (3) \qquad \Uparrow \quad (4) \qquad \Uparrow \quad (5)$$

$$L_p \approx \frac{1}{I_{xx}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial b_{1s}}{\partial p} + h_{TR}\frac{\partial T_{TR}}{\partial p}\right]$$

eq 7.20

$$\Uparrow \quad (1) \qquad \Uparrow \quad (4)$$

Where:

b	- Number of blades
b_{1s}	- Lateral flapping angle, shaft referenced
e	- Flapping hinge offset
h	- Height of hub above CG
h_{TR}	- Height of the tail rotor above the CG
I_{xx}	- Moment of inertia about x axis, roll moment of inertia
L_{f+t}	- Roll moment due to the fuselage/tail
L_p	- Roll moment due to roll rate
M_S	- Blade mass moment
p	- Angular velocity about x axis, Roll rate
T	- Thrust
T_{TR}	- Tail rotor thrust
Ω	- Rotor angular velocity
Y_R	- Rotor side force.

Term (1) is the largest contributor to roll damping and is due to TPP tilt resulting from flapping introduced by the roll rate. The tail rotor forces due to the lateral velocity induced by the roll are the source for term (4), and are significant depending on the height of the tail rotor above the roll axis. There is little change in thrust due to roll rate, rotor side force due to roll rate, and roll moment due to the fuselage/tail due to roll rate; therefore terms (2), (3), and (5) can be neglected, and Equation 7.19 can be approximated by Equation 7.20. Term (1) is nearly independent of airspeed. The thrust variation with lateral velocity at the tail rotor is also nearly constant with airspeed. However, the lateral velocity at the tail and the roll moment due to tail rotor thrust both depend on h_{TR}. This distance is a function of the roll axis orientation which may vary for different forward flight conditions when the equations of motion are referred to stability axes.

Typical values of L_p as a function of airspeed are shown in Figure 7.12. L_p is essentially independent of airspeed.

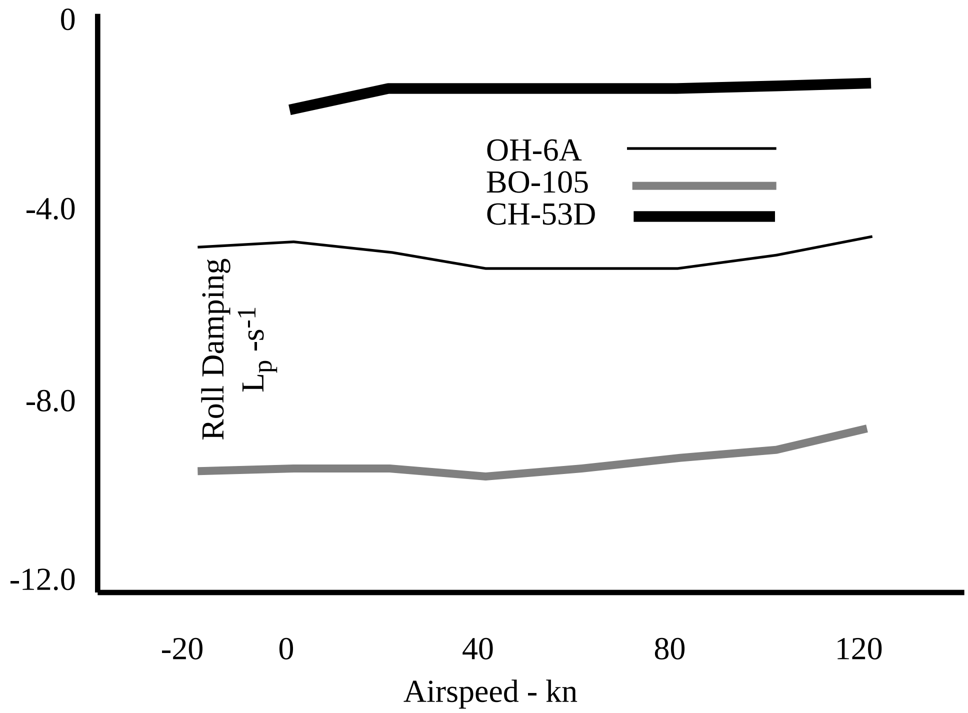


Figure 7.12
Roll Damping Versus Airspeed

7.3.3.9 ROLL MOMENT DUE TO LATERAL CYCLIC PITCH ANGLE, LATERAL CONTROL

$$L_{A_{1s}} = \frac{1}{I_{xx}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial b_{1s}}{\partial A_{1s}} + hb_{1s}\frac{\partial T}{\partial A_{1s}} + h\frac{\partial Y_R}{\partial A_{1s}} + h_{TR}\frac{\partial T_{TR}}{\partial A_{1s}} + \frac{\partial L_{f+t}}{\partial A_{1s}}\right]$$

eq 7.21

$\Uparrow$ (1) $\quad\Uparrow$ (2) $\quad\Uparrow$ (3) $\quad\Uparrow$ (4) $\quad\Uparrow$ (5)

Where:

A_{1s}	- Lateral cyclic pitch angle, shaft referenced
b	- Number of blades
b_{1s}	- Lateral flapping angle, shaft referenced
e	- Flapping hinge offset

h - Height of hub above CG
h_{TR} - Height of the tail rotor above the CG
I_{xx} - Moment of inertia about x axis, roll moment of inertia
$L_{A_{1s}}$ - Roll moment due to lateral cyclic pitch angle
L_{f+t} - Roll moment due to the fuselage/tail
M_S - Blade mass moment
T - Thrust
T_{TR} - Tail rotor thrust
Ω - Rotor angular velocity
Y_R - Rotor side force.

The lateral control derivative is predominantly caused by term (1) which is proportional to the change in TPP tilt with lateral cyclic pitch angle input ($\partial b_{1s}/\partial A_{1s}$). The lateral tilt has little influence on the flow through the rotor disk, its effect primarily being to tilt the TPP laterally relative to the shaft. Thus, $L_{A_{1s}}$ is little affected by the initial trim speed.

The control power referred to the cockpit flight controls is obtained by multiplying $L_{A_{1s}}$ by the lateral cyclic pitch angle to lateral control gear ratio:

$$L_{\delta_{LAT}} = \left(\frac{\partial A_{1s}}{\partial \delta_{LAT}}\right) L_{A_{1s}}$$

eq 7.22

Where:

A_{1s} - Lateral cyclic pitch angle, shaft referenced
δ_{LAT} - Lateral control
$L_{A_{1s}}$ - Roll moment due to lateral cyclic pitch angle
$L_{\delta_{LAT}}$ - Roll moment due to lateral control.

$L_{\delta_{LAT}}$ is independent of airspeed as shown in Figure 7.13.

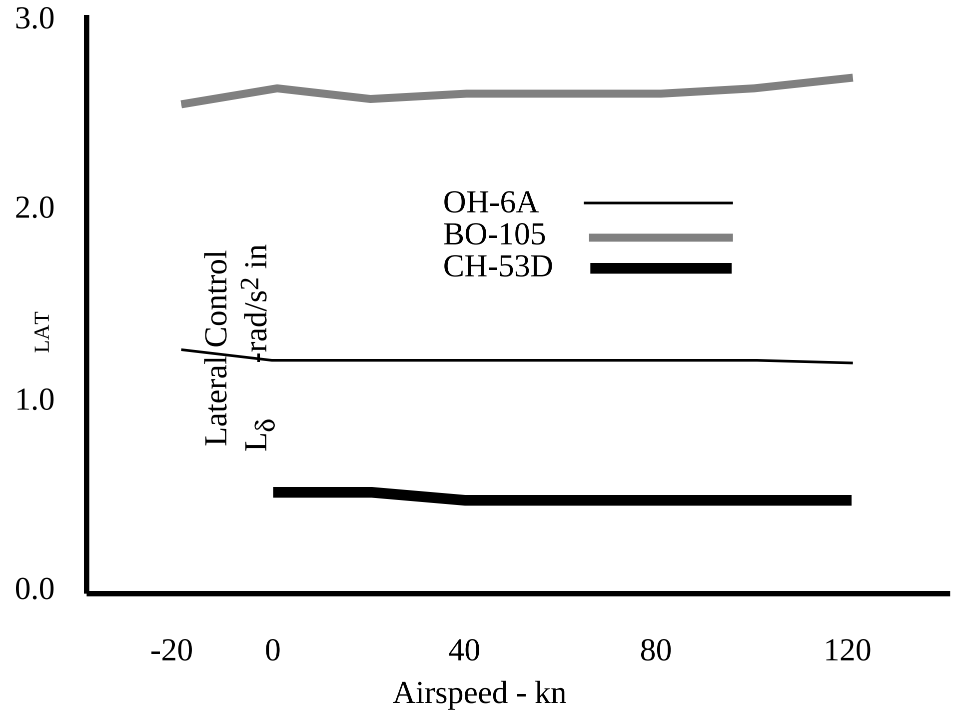


Figure 7.13
Lateral Control Versus Airspeed

7.3.3.10 ROLL MOMENT DUE TO TAIL ROTOR PITCH ANGLE

$$L_{\theta_{TR}} = \frac{1}{I_{xx}}\left[h_{TR}\frac{\partial T_{TR}}{\partial \theta_{TR}}\right] \qquad \text{eq 7.23}$$

Roll moment due to tail rotor pitch angle is due entirely to the change in tail rotor thrust due to tail rotor pitch angle. If the tail rotor thrust line is above the roll axis, right pedal produces a left roll and $L_{\theta_{TR}}$ is negative.

The control power referred to the cockpit flight controls is obtained by multiplying $L_{\theta_{TR}}$ by the tail rotor pitch angle to pedal control gear ratio:

$$L_{\delta_{PED}} = \left(\frac{\partial\theta_{TR}}{\partial\delta_{PED}}\right) L_{\theta_{TR}}$$ *eq 7.24*

Where:

δ_{PED}	- Pedal control
h_{TR}	- Height of the tail rotor above the CG
I_{xx}	- Moment of inertia about x axis, roll moment of inertia
$L_{\delta_{PED}}$	- Roll moment due to pedal control
$L_{\theta_{TR}}$	- Roll moment due to tail rotor pitch angle
θ_{TR}	- Tail rotor pitch angle
T_{TR}	- Tail rotor thrust.

If $\partial T_{TR}/\partial\theta_{TR}$ is independent of airspeed then $L_{\theta_{TR}}$ is essentially independent of airspeed. This is supported by the data in Figure 7.14.

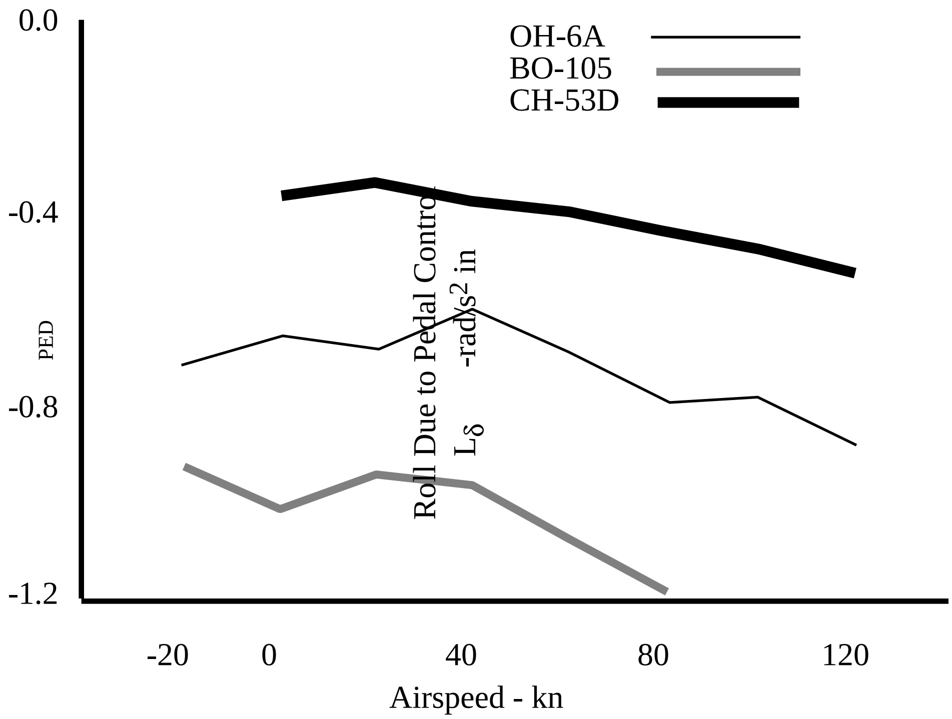


Figure 7.14
Roll Moment Due to Yaw Control Versus Airspeed

7.3.3.11 SIDE FORCE DUE TO LATERAL VELOCITY

$$Y_v = \frac{1}{m}\left[\underset{\substack{\Uparrow \\ (1)}}{T\frac{\partial b_{1s}}{\partial v}} + \underset{\substack{\Uparrow \\ (2)}}{b_{1s}\frac{\partial T}{\partial v}} + \underset{\substack{\Uparrow \\ (3)}}{\frac{\partial Y_R}{\partial v}} + \underset{\substack{\Uparrow \\ (4)}}{\frac{\partial T_{TR}}{\partial v}} + \underset{\substack{\Uparrow \\ (5)}}{\frac{\partial Y_{f+t}}{\partial v}} \right] \qquad \textit{eq 7.25}$$

Where:

b_{1s}	- Lateral flapping angle, shaft referenced
m	- Mass
T	- Thrust
T_{TR}	- Tail rotor thrust
v	- Translational velocity component along y axis
Y_{f+t}	- Side force due to the fuselage/tail
Y_R	- Rotor side force
Y_v	- Side force due to lateral velocity.

Term (2) is dependent on the initial TPP tilt at trim but is usually insignificant. The in plane rotor force contribution (Term (3) is also small. Term (1) due to the increment in lateral tilt of the TPP and term (4) due to the change in tail rotor thrust with lateral velocity are of comparable magnitude, and do not vary much with trim speed. Term (5) is proportional to the product of the dynamic pressure and sideslip angle. Consequently, term (5) varies with trim speed causing a considerable variation in the total derivative through the speed range. The overall variation of Y_v with speed is presented in Figure 7.15.

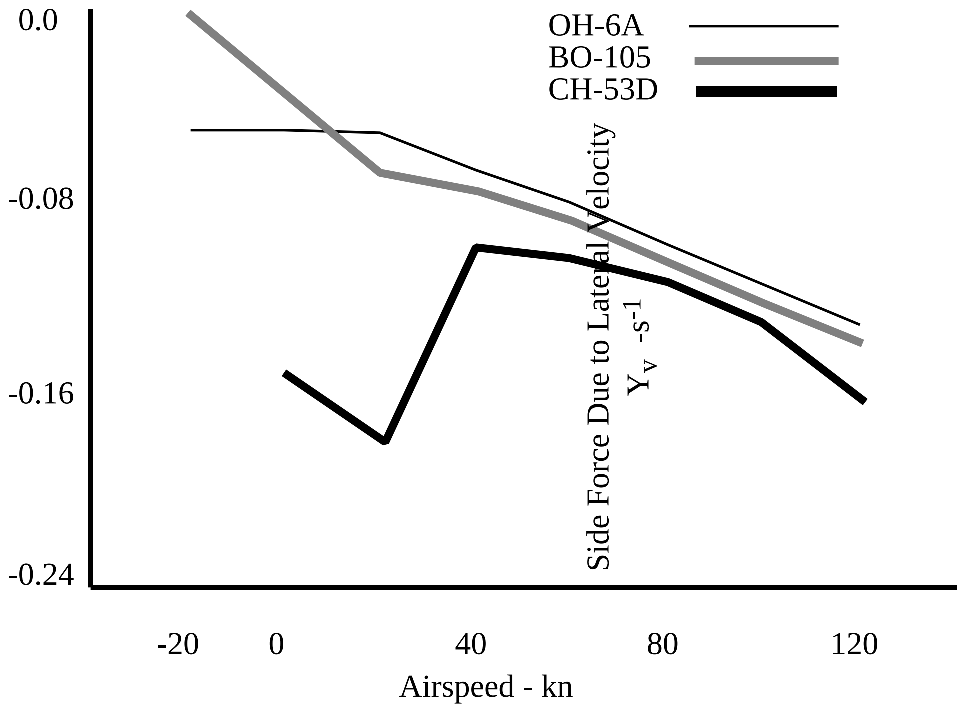


Figure 7.15
Side Force Due to Lateral Velocity Versus Airspeed

7.3.3.12 SIDE FORCE DUE TO YAW RATE

$$Y_r = \frac{1}{m}\left[T\frac{\partial b_{1s}}{\partial r} + b_{1s}\frac{\partial T}{\partial r} + \frac{\partial Y_R}{\partial r} + \frac{\partial T_{TR}}{\partial r} + \frac{\partial Y_{f+t}}{\partial r} \right] \quad \textit{eq 7.26}$$

$$\Uparrow \quad \Uparrow \quad \Uparrow \quad \Uparrow \quad \Uparrow$$
$$(1) \quad (2) \quad (3) \quad (4) \quad (5)$$

Where:

b_{1s}	- Lateral flapping angle, shaft referenced
m	- Mass
r	- Angular velocity about z axis, Yaw rate
T	- Thrust
T_{TR}	- Tail rotor thrust

Y_{f+t} - Side force due to the fuselage/tail

Y_R - Rotor side force

Y_r - Side force due to yaw rate.

In the above expression, the partials $\partial b_{1s} / \partial r$, $\partial T / \partial r$ and $\partial Y_R / \partial r$ are dependent on CG position and normally are not significant quantities. The fuselage may generate side forces particularly if there is a significant vertical tail. Probably the largest contribution to Y_r comes from the tail rotor. Its contribution is dependent on the lateral velocity at the tail rotor as a consequence of the yaw rate.

Typical variations of Y_r with airspeed are shown in Figure 7.16.

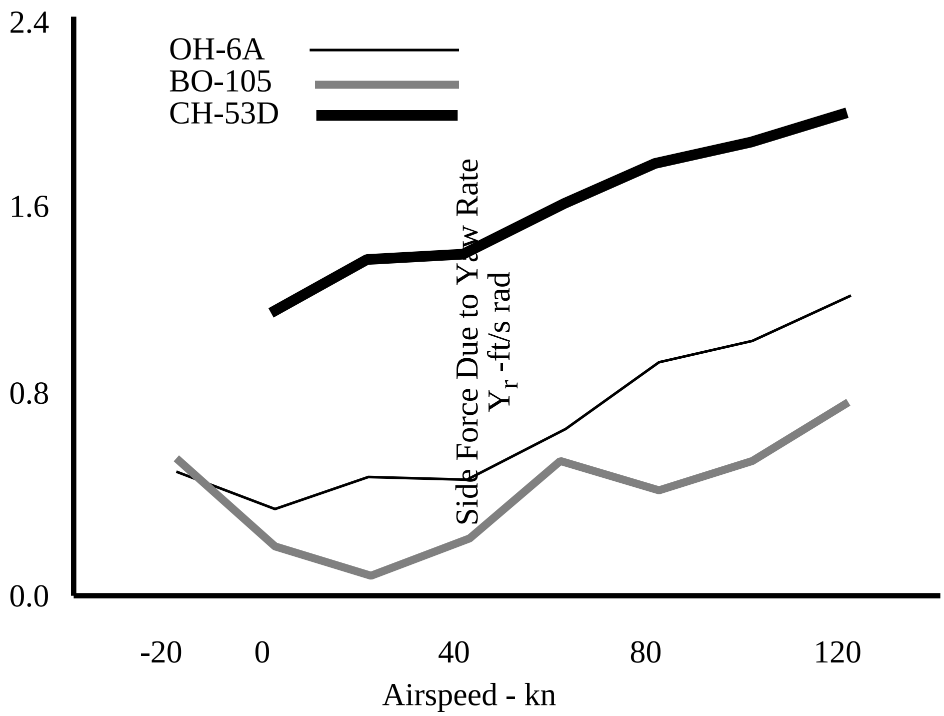


Figure 7.16
Side Force Due to Yaw Rate Versus Airspeed

7.3.3.13 SIDE FORCE DUE TO ROLL RATE

$$Y_p = \frac{1}{m}\left[T\frac{\partial b_{1s}}{\partial p} + b_{1s}\frac{\partial T}{\partial p} + \frac{\partial Y_R}{\partial p} + \frac{\partial T_{TR}}{\partial p} + \frac{\partial Y_{f+t}}{\partial p} \right] \qquad \textit{eq 7.27}$$

The significant quantities in the above expression are $T\,\partial b_{1s}/\partial p$ and $\partial T_{TR}/\partial p$. An approximation for Y_p is:

$$Y_p \approx \frac{1}{m}\left[T\frac{\partial b_{1s}}{\partial p} + \frac{\partial T_{TR}}{\partial p} \right] \qquad \textit{eq 7.28}$$

Where:

b_{1s}	- Lateral flapping angle, shaft referenced
m	- Mass
p	- Angular velocity about x axis, Roll rate
T	- Thrust
T_{TR}	- Tail rotor thrust
Y_{f+t}	- Side force due to the fuselage/tail
Y_p	- Side force due to roll rate
Y_R	- Rotor side force.

The variations of Y_p with airspeed are shown in Figure 7.17.

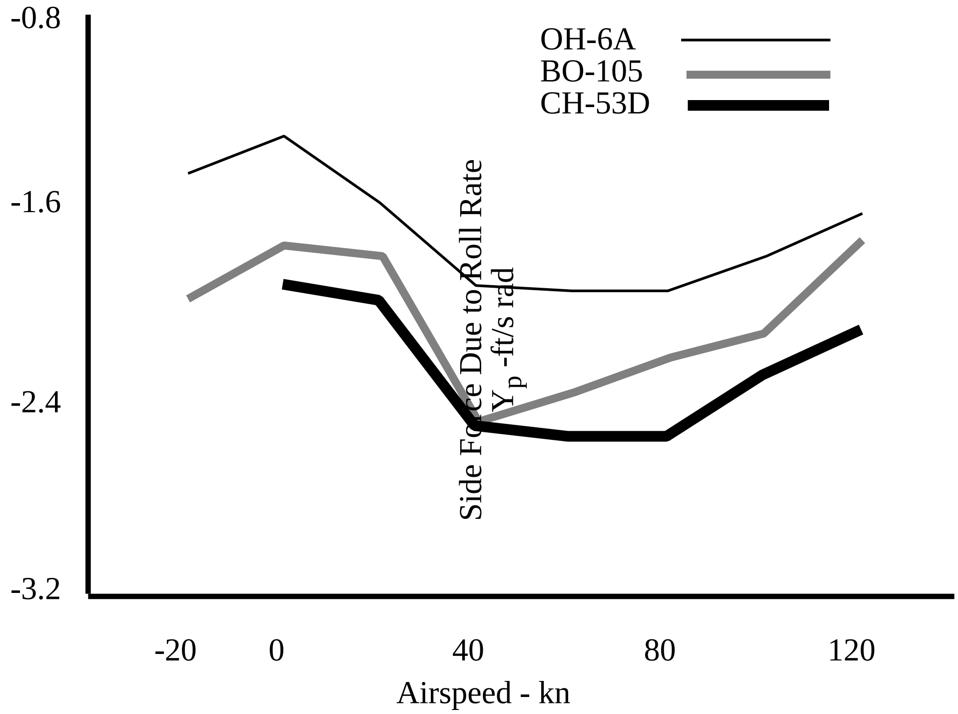


Figure 7.17
Side Force Due to Roll Rate Versus Airspeed

7.3.3.14 SIDE FORCE DUE TO LATERAL CONTROL

$$Y_{A_{1s}} = \frac{1}{m}\left[\underbrace{T \frac{\partial b_{1s}}{\partial A_{1s}}}_{(1)} + \underbrace{b_{1s} \frac{\partial T}{\partial A_{1s}}}_{(2)} \right] \qquad \textit{eq 7.29}$$

Term (1) is the dominate term and since $T \approx mg$:

$$Y_{A_{1s}} \approx g \left[\frac{\partial b_{1s}}{\partial A_{1s}} \right] \qquad \textit{eq 7.30}$$

(1)

The control power referred to the cockpit flight controls is the product of $Y_{A_{1s}}$ and the lateral cyclic pitch angle to lateral control gear ratio:

$$Y_{\delta_{LAT}} = \left(\frac{\partial A_{1s}}{\partial \delta_{LAT}}\right) Y_{A_{1s}}$$

eq 7.31

Where:

A_{1s}	- Lateral cyclic pitch angle, shaft referenced
b_{1s}	- Lateral flapping angle, shaft referenced
δ_{LAT}	- Lateral control
g	- Gravity
m	- Mass
T	- Thrust
$Y_{A_{1s}}$	- Side force due to lateral cyclic pitch angle
$Y_{\delta_{LAT}}$	- Side force due to lateral control.

The side force due to lateral control input is primarily due to the resulting lateral tilt of the TPP and thrust vector. The variations of $Y_{\delta_{LAT}}$ with speed are indicated in Figure 7.18.

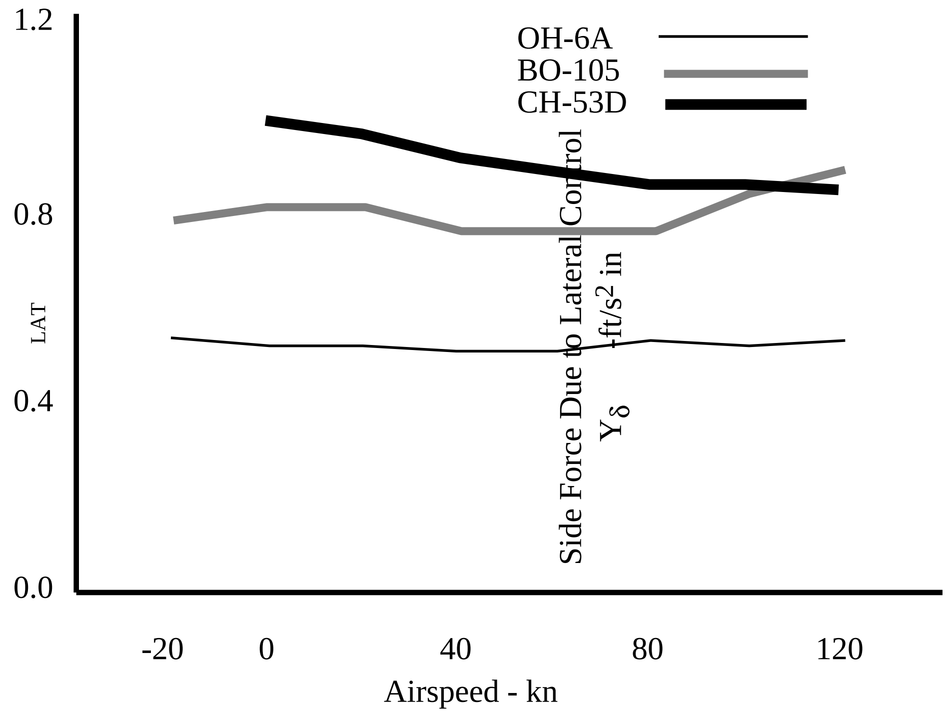


Figure 7.18
Side Force Due to Lateral Control Versus Airspeed

7.3.3.15 SIDE FORCE DUE TO DIRECTIONAL CONTROL

$$Y_{\theta_{TR}} = \frac{1}{m}\left[\frac{\partial T_{TR}}{\partial \theta_{TR}}\right] \qquad \textit{eq 7.32}$$

$$\Uparrow$$

(4)

The control power referred to the cockpit flight controls is the product of $Y_{\theta_{TR}}$ and the tail rotor pitch angle to pedal control gear ratio:

$$Y_{\delta_{PED}} = \left(\frac{\partial \theta_{TR}}{\partial \delta_{PED}}\right) Y_{\theta_{TR}} \qquad \textit{eq 7.33}$$

Where:

δ_{PED}	- Pedal control
m	- Mass
θ_{TR}	- Tail rotor pitch angle
T_{TR}	- Tail rotor thrust
$Y_{\delta_{PED}}$	- Side force due to pedal control
$Y_{\theta_{TR}}$	- Side force due to tail rotor pitch angle.

The tail rotor thrust is the only contribution to this derivative assuming no control coupling and neglecting any interference effects. Variations of $Y_{\delta_{PED}}$ are shown in Figure 7.19.

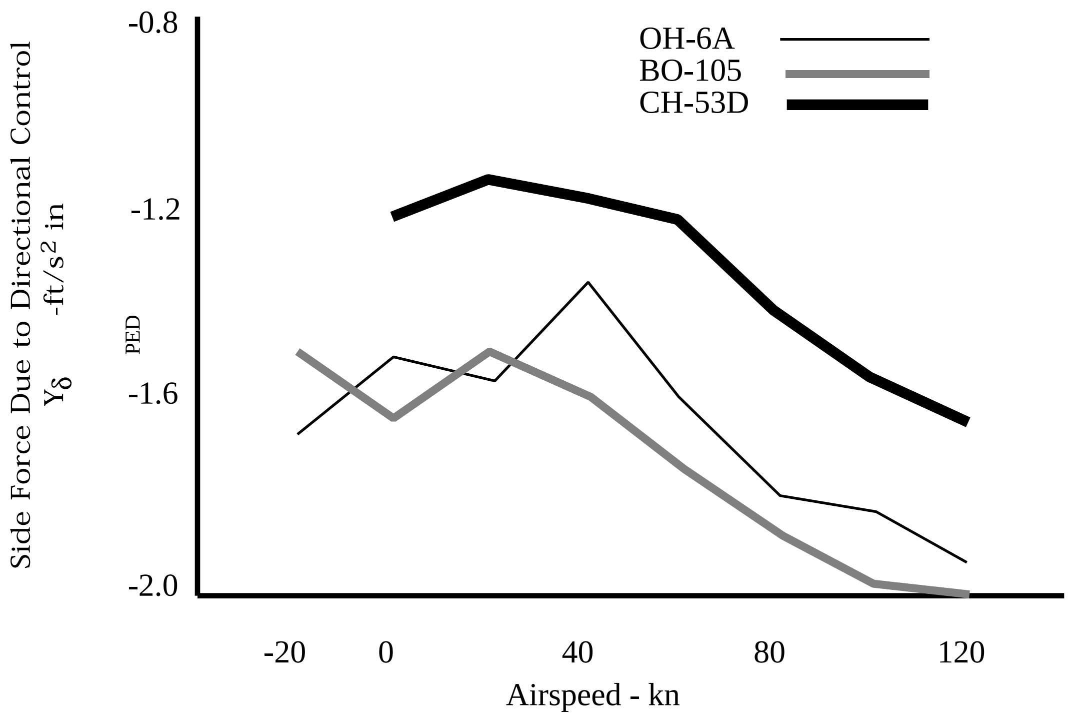


Figure 7.19
Side Force Due to Directional Control Versus Airspeed

7.3.3.16 SUMMARY OF FUSELAGE/TAIL, MAIN ROTOR, AND TAIL ROTOR CONTRIBUTIONS

Table 7.II presents a summary of the fuselage, tail, main rotor, and tail rotor contributions to the lateral-directional derivatives in forward flight.

Table 7.II
Relative Contributions to the Stability Derivatives of the Single Rotor Helicopter

Derivative		Relative Contributions [1]		
Symbol	Usual Sign	Fuselage/Tail	Main Rotor	Tail Rotor
N_v	(+)	B (depends on vertical fin size)		A
N_r	(-)	B		A
N_p	(-)	B (depends on fin size and height)		A (tail rotor height)
$N_{\theta_{TR}}$ $N_{\delta_{PED}}$	(-) (+)			A
$N_{A_{1s}}$ $N_{\delta_{LAT}}$	note [2]		(generally small)	
L_v	(-)	B	A	B (depends on tail rotor height)
L_r	(+)	B (depends on fin size and height)	C	A (depends on tail rotor height)
L_p	(-)	C	A	B (depends on tail rotor height)
$L_{A_{1s}}$ $L_{\delta_{LAT}}$	(+) (+)		A	
$L_{\theta_{TR}}$ $L_{\delta_{PED}}$	(-)(+) (+)(-)			A (depends on tail rotor height)
Y_v	(-)	A	A	A
Y_r	(+)	B (depends on fin size)		A

Relative Contributions to the Stability Derivatives of the Single Rotor Helicopter (cont'd)

Y_p	(-)	B (depends on fin size and height)		A (depends on tail rotor height)
$Y_{A_{1s}}$ $Y_{\delta_{LAT}}$	(+) (+)		A	
$Y_{\theta_{TR}}$ $Y_{\delta_{PED}}$	(+) (-)			A

Notes: (1) A is the highest contributor
(2) Depends on the relative strengths of three factors

7.3.4 Trimmed Control Positions

An important part of helicopter flight testing is determining the trimmed flight control positions throughout the flight envelope. A complete investigation requires a large number of flight tests involving a multidimensional matrix of possible flight conditions. A practical approach is to determine the trimmed control positions when important parameters are changed. Gross weight, altitude, airspeed, rotor speed, as well as powered and autorotational flight are all factors which can have a significant effect on trimmed control positions and control margins. The effect of these parameters is evaluated during rectilinear and turning flight.

The effect of airspeed and torque on trimmed pedal position is considered as an example. Figure 7.20 (a) presents typical plots of pedal position versus airspeed for four different constant rotor torque conditions. At a given airspeed, when a greater torque is applied to the rotor, a greater left pedal displacement is required, producing a greater right tail rotor thrust to maintain equilibrium.

The trimmed flight conditions for different forward velocities along each of the constant torque curves correspond to a variation in descending, level, and climbing flight. Considerable flight testing is required to determine a family of constant torque curves. Consequently, an abbreviated program incorporates a series of level flight airspeeds. This results in a level flight pedal position versus airspeed curve shown on Figure 7.20. Similar plots could be made for constant rates of descent or climb covering the entire speed range to V_{NE}.

The definition of the test corresponding to the plot of pedal position versus airspeed on Figure 7.20 (a) requires specifying other parameters and piloting techniques. These include rotor speed, altitude, gross weight, and longitudinal control for pitch equilibrium in unaccelerated rectilinear flight. There is an option for maintaining side force equilibrium. The pilot can maintain ball centered, level flight so the tail rotor thrust is balanced by a fuselage side force due to flying at a sideslip angle. Figure 7.20 (b) indicates the hypothetical sideslip variation if the pilot maintained ball centered, wings level trim conditions. Measurement of sideslip angle is important during flight tests because it gives information concerning the requirements for navigation systems, gun sight track, and ordnance launches.

An alternative test procedure which provides different results, requires the pilot to maintain zero sideslip at the level flight trim conditions. This results in a bank angle variation with airspeed shown on Figure 7.20 (c) where the horizontal components of the main rotor and tail rotor thrust vectors are balanced.

At the low speed-high torque condition, the left pedal margin is critical. Full left pedal travel might result in insufficient right tail rotor force to obtain yaw equilibrium or make a left pedal turn. The high airspeed-low torque condition is critical for right pedal control. Yaw equilibrium and right turns in high speed autorotational flight might be limited by right pedal control margins. The right pedal control margins are largely dependent on the type of vertical tail used on the helicopter. The vertical tail can be mounted at an angle so it develops a side force proportional to dynamic pressure, reducing the requirement for left pedal for increasing level flight airspeed.

Helicopter flight tests also determine the control positions for steady turns in level flight and in constant power climbs and descents. Turns at various bank angles are investigated, recording the equilibrium control positions. The control margins in turning flight are generally more critical than in rectilinear flight.

An important result of the helicopter flying qualities evaluation is the pilot's comments concerning control displacement, control harmony, and trimmability. The control positions should change smoothly from one trim condition to another and the variation should appear fairly linear to the pilot. Both lateral-directional and longitudinal control positions and forces are considered in discussing control harmony. Large differences between the lateral and longitudinal control forces generally influence pilot opinion.

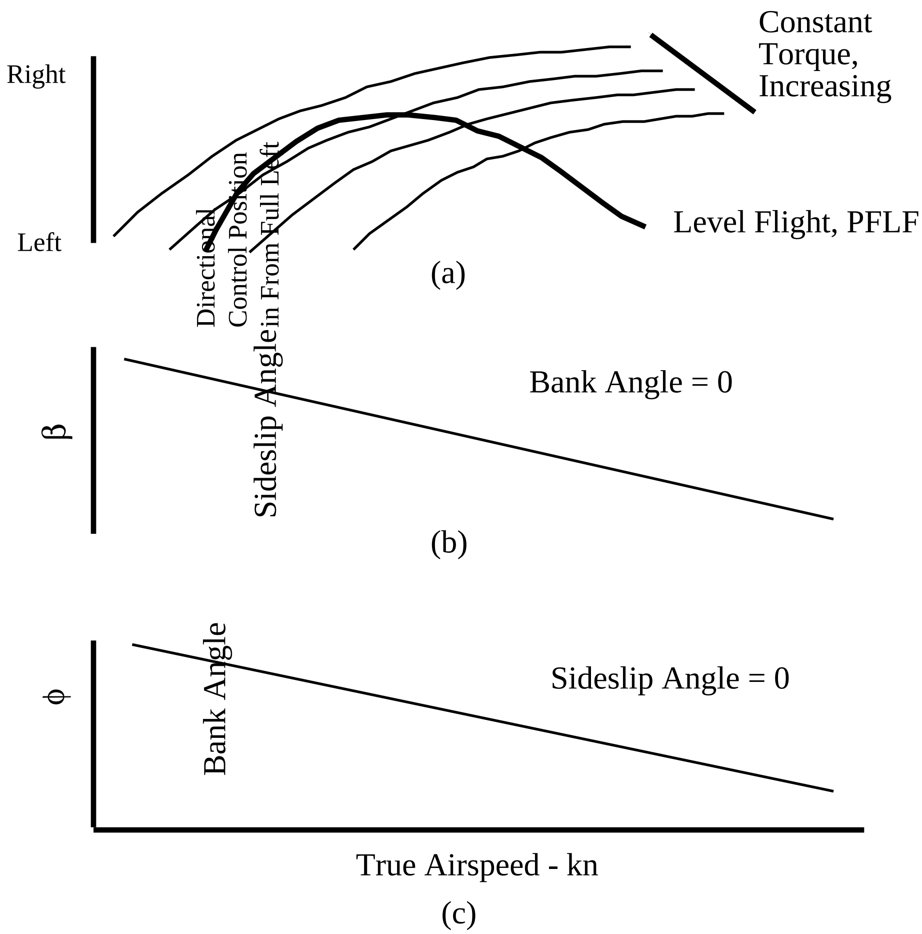


Figure 7.20
Trimmed Pedal Position Versus Airspeed for Rectilinear Flight

7.3.5 Static Stability

The theory of static stability and steady lateral-directional characteristics is simplified by considering only small changes in the variables from an initial trim condition. A wings level initial trim condition is used which involves an inherent sideslip angle.

7.3.5.1 STEADY LATERAL-DIRECTIONAL CHARACTERISTICS

Four different steady conditions are used in flight tests for the evaluation of helicopter lateral-directional flight characteristics. The steady heading sideslip is used to evaluate static lateral-directional stability and side force characteristics through flight control positions and bank angle. The remaining steady conditions obtained in turning flight include pedal only turns, lateral cyclic only turns, and coordinated turns.

7.3.5.2 INHERENT SIDESLIP IN FORWARD FLIGHT

A typical U.S. single rotor helicopter in wings level, coordinated, equilibrium flight is normally flying in a right sideslip. This sideslip is required to achieve a balance between the side force contribution of tail rotor thrust and fuselage-main rotor side forces. The sideslip angle required to achieve the balance in side forces is the inherent sideslip angle. Increasing main rotor torque by going from level to climbing flight, or by increasing gross weight increases the inherent sideslip angle. In addition, a decrease in airspeed at constant main rotor torque may have the same effect; since the tail rotor force is constant, a larger sideslip angle is required at the slower airspeeds to develop the same fuselage side forces.

Important effects associated with the inherent sideslip angle are demonstrated in flight. The static lateral-directional stability can be dissimilar on either side of the inherent sideslip angle. Such a condition results in dissimilar helicopter response to right and left disturbances from trimmed flight.

A second problem area is mission oriented. In terms of dead reckoning navigation, the effects of an inherent sideslip angle can not be separated from the effects of a crosswind. The pilot generally has no way of knowing the sideslip angle. When Doppler ground speed and drift angle are furnished, the combined effects of wind and inherent sideslip are displayed as though caused by crosswind alone.

Thirdly, the variation in inherent sideslip angle with power and airspeed may introduce a significant lateral control task during accelerations and decelerations.

7.3.5.3 AIRSPEED POSITION ERRORS IN SIDESLIPS

Indicated airspeed in sideslips is subject to position error, particularly at slow speeds. Large variations in airspeed position errors with small variations in sideslip angles may present a problem in forward flight. For example, a fictitious helicopter has a 10 kn position error with each 5˚ of sideslip. If this helicopter is disturbed in yaw by a gust or increase in power, the helicopter yaws right and left, and the airspeed never seems to stabilize. Efforts by the pilot to maintain airspeed may keep the helicopter disturbed in yaw, perpetuating the indicated airspeed problems.

Several procedures are used to alleviate the problems of airspeed position errors in helicopter flight tests. Airspeed calibration tests are conducted and corrected airspeed flown in subsequent data flights. The pitot static source can be located on a swivel on the end of a boom so it aligns with the airflow. Another method for avoiding position error due to sideslip in forward flight is use of a trailing bomb towed behind the helicopter.

Omni-directional helicopter airspeed systems are available which are generally free of sideslip position errors. They provide the body axes components of the level flight airspeed, u and v, as well as the total airspeed.

7.3.5.4 STEADY HEADING SIDESLIPS (SHSS)

One of the conventional methods for evaluating helicopter static lateral-directional characteristics is to obtain data at stabilized steady heading sideslip conditions. The tests are conducted at a series of sideslip angles about a specified trim point. Collective pitch is fixed and flight path velocity is maintained by longitudinal cyclic control inputs. The normal initial trim point in forward flight is a steady heading, wings level, balanced condition.

The principal characteristics evaluated in steady heading sideslip tests are the combined effects of dihedral effect, directional static stability, and the side force variation with sideslip. The term *effective dihedral* means “not necessarily caused by dihedral” and is used to refer to the total helicopter roll moment due to sideslip and tail rotor thrust as reflected by the lateral control displacement required to maintain a steady heading sideslip.

The lateral-directional equations of motion reduce to the following for steady rectilinear flight where $\dot{v} = \dot{p} = \dot{r} = p = r = 0$:

$$-Y_v u_0 \Delta\beta - g\Delta\phi = Y_{A_{1s}} \Delta A_{1s} + Y_{\theta_{TR}} \Delta\theta_{TR} = Y_{\delta_{LAT}} \Delta\delta_{LAT} + Y_{\delta_{PED}} \Delta\delta_{PED}$$

eq 7.34

$$-L_v u_0 \Delta\beta = L_{A_{1s}} \Delta A_{1s} + L_{\theta_{TR}} \Delta\theta_{TR} = L_{\delta_{LAT}} \Delta\delta_{LAT} + L_{\delta_{PED}} \Delta\delta_{PED}$$

eq 7.35

$$-N_v u_0 \Delta\beta = N_{A_{1s}} \Delta A_{1s} + N_{\theta_{TR}} \Delta\theta_{TR} = N_{\delta_{LAT}} \Delta\delta_{LAT} + N_{\delta_{PED}} \Delta\delta_{PED}$$

eq 7.36

Where:

A_{1s}	- Lateral cyclic
β	- Sideslip angle
δ_{LAT}	- Lateral control
δ_{PED}	- Pedal control
ϕ	- Roll angle
g	- Gravity
$L_{A_{1s}}$	- Roll moment due to lateral cyclic pitch angle
$L_{\delta_{LAT}}$	- Roll moment due to lateral control
$L_{\delta_{PED}}$	- Roll moment due to pedal control
$L_{\theta_{TR}}$	- Roll moment due to tail rotor pitch angle
L_v	- Roll moment due to lateral velocity
$N_{A_{1s}}$	- Yaw moment due to lateral cyclic pitch angle
$N_{\delta_{LAT}}$	- Yaw moment due to lateral control
$N_{\delta_{PED}}$	- Yaw moment due to pedal control
$N_{\theta_{TR}}$	- Yaw moment due to tail rotor pitch angle
N_v	- Yaw moment due to lateral velocity
θ_{TR}	- Tail rotor pitch angle

u_0	- Initial velocity
$Y_{A_{1s}}$	- Side force due to lateral cyclic pitch angle
$Y_{\delta_{LAT}}$	- Side force due to lateral control
$Y_{\delta_{PED}}$	- Side force due to pedal control
$Y_{\theta_{TR}}$	- Side force due to tail rotor pitch angle
Y_v	- Side force due to lateral velocity.

These three equations represent the static equilibrium of side forces, roll moments, and yaw moments acting on the helicopter. The sideslip angle $(\Delta\beta = \Delta v / \Delta u_0)$, roll angle $(\Delta\phi)$, lateral cyclic pitch angle (ΔA_{1s}), lateral control $(\Delta\delta_{LAT})$, tail rotor pitch angle $(\Delta\theta_{TR})$, and pedal control $(\Delta\delta_{PED})$ are all increments from their trim values. The above equations only apply for steady conditions close to the original trim condition. The dynamic motions involved in going from a trim condition to a stabilized off trim condition are not considered.

Since there are three equations for static lateral-directional equilibrium involving four variables $\Delta\beta$, $\Delta\phi$, $\Delta\delta_{LAT}$, and $\Delta\delta_{PED}$, the pilot can select one variable arbitrarily $(\Delta\beta)$. He manipulates the lateral $(\Delta\delta_{LAT})$ and pedal control $(\Delta\delta_{PED})$ until he achieves a steady heading sideslip condition. The steady sideslip results in the helicopter rolling to the bank angle required for equilibrium. Representative data for stabilized off trim points at a series of sideslip angles are presented in Figure 7.21.

Figure 7.22 presents schematic diagrams helpful in visualizing the combinations of control positions and helicopter attitudes which occur in steady heading sideslips.

According to the linearized small disturbance theory, the plots of the lateral-directional variables obtained in steady heading sideslips versus sideslip angle should fall on straight lines as indicated on Figure 7.22. Thus, the gradients $(\Delta\delta_{LAT}/\Delta\beta$, $\Delta\delta_{PED}/\Delta\beta$, and $\Delta\phi/\Delta\beta)$ are computed from the data obtained at each selected increment in sideslip.

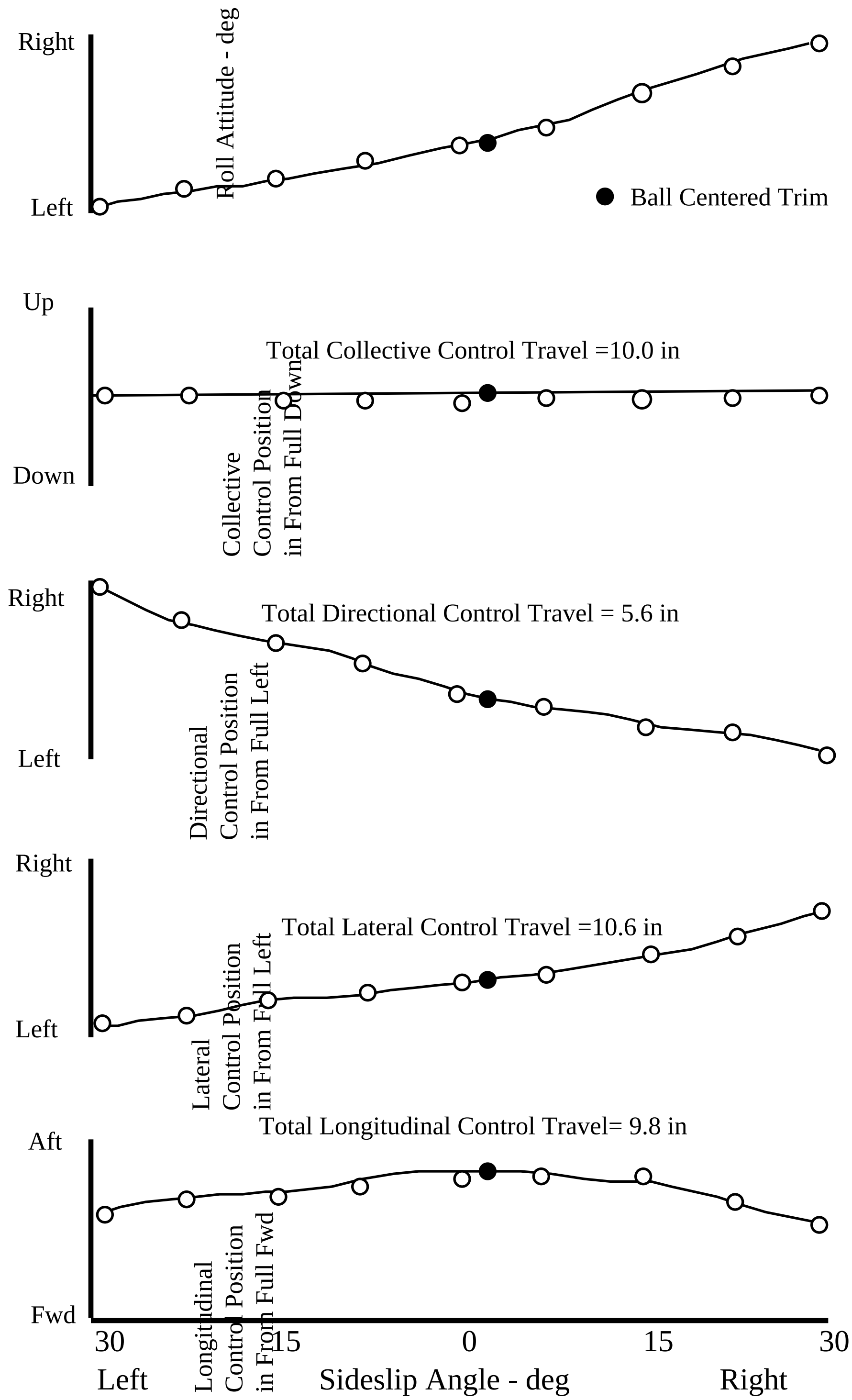


Figure 7.21
Static Lateral-Directional Stability

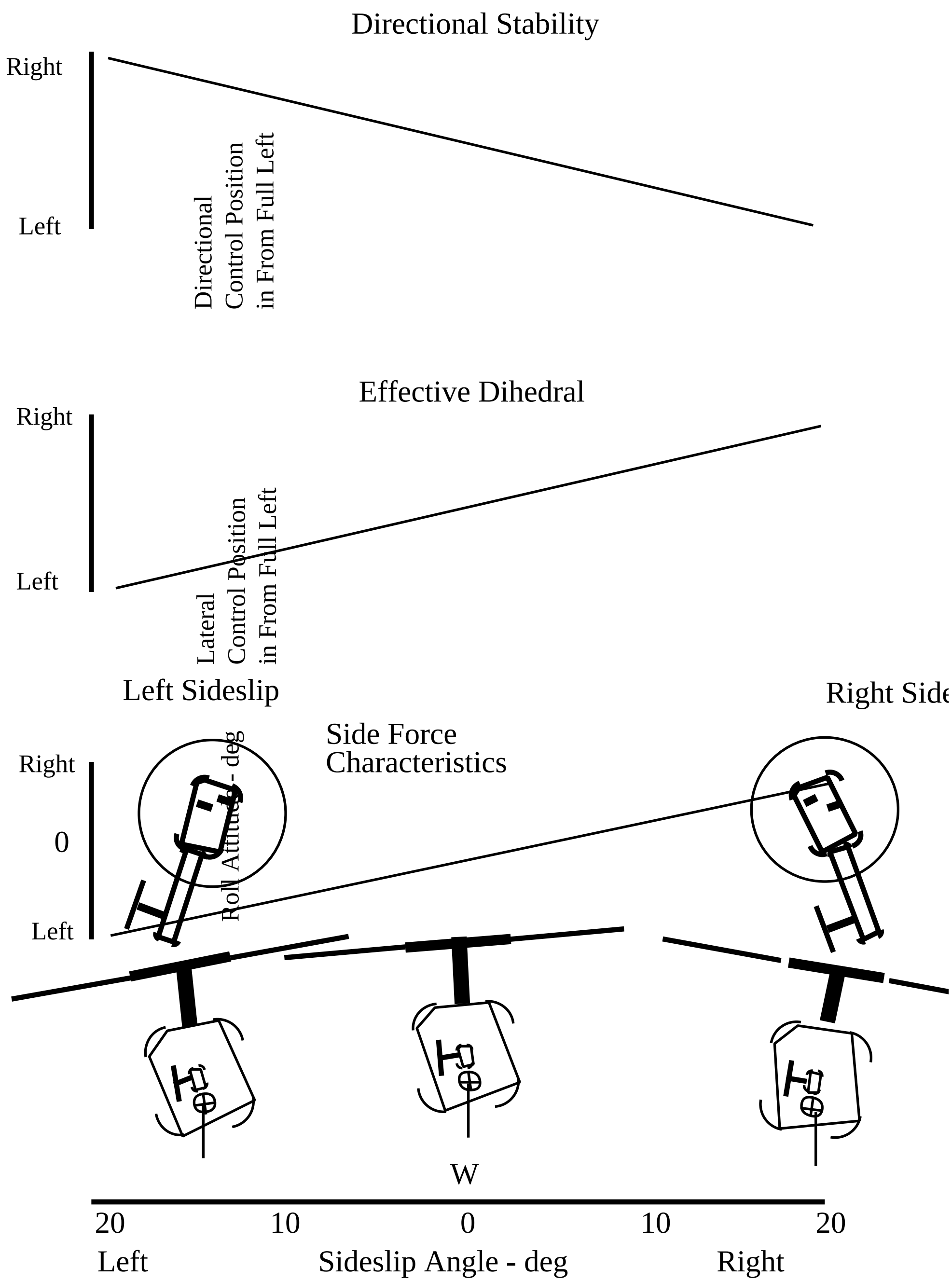


Figure 7.22
Trim Conditions in Steady Sideslips

The roll and yaw moment equations are solved simultaneously for $\Delta\delta_{PED}/\Delta\beta$ and $\Delta\delta_{LAT}/\Delta\beta$ since bank angle ($\Delta\phi$) does not appear in these two equations. The resulting expressions are given below along with the simplified forms obtained assuming yaw moment is not generated by lateral control:

$$\frac{\Delta\delta_{PED}}{\Delta\beta} = -u_0 \left(\frac{L_{\delta_{LAT}} N_v - L_v N_{\delta_{LAT}}}{L_{\delta_{LAT}} N_{\delta_{PED}} - L_{\delta_{PED}} N_{\delta_{LAT}}} \right) \quad \text{eq 7.37}$$

If $N_{\delta_{LAT}}$ is assumed to be zero, Eq 7.37 reduces to:

$$\frac{\Delta\delta_{PED}}{\Delta\beta} \approx \left(-\frac{u_0}{N_{\delta_{PED}}} \right) N_v \quad \text{eq 7.38}$$

$$\frac{\Delta\delta_{LAT}}{\Delta\beta} = -u_0 \left(\frac{N_{\delta_{PED}} L_v - N_v L_{\delta_{PED}}}{L_{\delta_{LAT}} N_{\delta_{PED}} - L_{\delta_{PED}} N_{\delta_{LAT}}} \right) \quad \text{eq 7.39}$$

If $N_{\delta_{LAT}}$ is assumed to be zero, Eq 7.39 reduces to:

$$\frac{\Delta\delta_{LAT}}{\Delta\beta} \approx \frac{u_0}{L_{\delta_{LAT}}} \left(L_v - L_{\delta_{PED}} \frac{N_v}{N_{\delta_{PED}}} \right) \quad \text{eq 7.40}$$

or:

$$\frac{\Delta\delta_{LAT}}{\Delta\beta} = \left(-\frac{u_0 L_v}{L_{\delta_{LAT}}} \right) - \left[\frac{L_{\delta_{PED}}}{L_{\delta_{LAT}}} \left(\frac{\Delta\delta_{PED}}{\Delta\beta} \right) \right] \quad \text{eq 7.41}$$

Where:

β	- Sideslip angle
δ_{LAT}	- Lateral control
δ_{PED}	- Pedal control
$L_{\delta_{LAT}}$	- Roll moment due to lateral control
$L_{\delta_{PED}}$	- Roll moment due to pedal control
L_v	- Roll moment due to lateral velocity
$N_{\delta_{LAT}}$	- Yaw moment due to lateral control
$N_{\delta_{PED}}$	- Yaw moment due to pedal control
N_v	- Yaw moment due to lateral velocity
u_0	- Initial velocity.

Static lateral-directional stability is the sum of the moments generated by an off trim lateral airspeed or sideslip. The moments produced by an off trim sideslip can not be measured directly in flight. The helicopter is maneuvered into an off trim equilibrium flight condition by providing an opposing moment generated by lateral and pedal control. The moment produced by the lateral and pedal control can not be measured directly in flight. However, the lateral and pedal control displacement from trim can be measured. Therefore, static lateral-directional stability is indicated by the gradient of lateral and pedal control position versus sideslip. The gradient of pedal control with sideslip indicates the sign of the directional stability. The gradient of the lateral control with sideslip indicates the sign of the lateral stability or effective dihedral.

Control systems are designed so a right pedal displacement produces a positive or nose right yaw moment ($N_{\delta_{PED}}$ is positive). Consequently, the above expression for $\Delta\delta_{PED}/\Delta\beta$ provides a negative pedal displacement versus sideslip gradient, indicating the yaw moment due to lateral velocity, static directional stability, is positive. A right sideslip produces a nose right yaw moment rotating the nose of the helicopter to the right and reducing the sideslip angle. In a steady heading sideslip, the yaw angle is maintained by the control moment due to left pedal displacement as indicated in Figure 7.22. Although the slope of the pedal control versus sideslip curve indicates the sign of the static directional stability derivative (N_v), it does not indicate the magnitude since the slope is proportional to the yaw moment due to pedal control, which in turn depends on the mechanical ratio of the tail rotor pitch angle to pedal control.

The lateral control displacement to sideslip ratio provides an indication of the roll moment due to sideslip or the effective dihedral of the helicopter. The first term ($-u_0 L_v/L_{\delta_{LAT}}$) in the simplified expression for $\Delta\delta_{LAT}/\Delta\beta$ predominates and is proportional to the roll moment due to lateral velocity. Since the control moment derivative, $L_{\delta_{LAT}}$, is positive (right control deflection gives right wing down roll moment) a positive $\Delta\delta_{LAT}/\Delta\beta$ curve, as shown on Figure 7.22, indicates the L_v derivative is negative corresponding to a right wing up roll in a right sideslip. However, the slope of the curve does not indicate the magnitude of the effective dihedral (L_v), since the slope also depends on the roll moment due to lateral control, which in turn depends on the mechanical ratio of the lateral cyclic pitch angle to lateral control.

The second term in the expression for lateral control versus sideslip also has some influence on the lateral control versus sideslip curve. This second term arises because tail rotor thrust can produce a roll moment if the tail rotor is not located on the roll axis. The $L_{\delta_{PED}}$ derivative for roll moment due to pedal control is expressed as follows:

$$L_{\delta_{PED}} \approx \frac{1}{I_{xx}} h_{TR} \left(\frac{\partial T_{TR}}{\partial \delta_{PED}} \right) \qquad \textit{eq 7.42}$$

Where:

δ_{PED}	- Pedal control
h_{TR}	- Height of the tail rotor above the CG
I_{xx}	- Moment of inertia about x axis, roll moment of inertia
$L_{\delta_{PED}}$	- Roll moment due to pedal control
T_{TR}	- Tail rotor thrust.

The right yaw moment due to a right pedal displacement is produced by a tail rotor thrust to the left or a negative increment in T_{TR}, consequently, $(\partial T_{TR}/\partial \delta_{PED})$ is a negative quantity. For most configurations, the tail rotor is located above the roll axis so h_{TR} is positive. In this case, $L_{\delta_{PED}}$ is negative or a right pedal input produces a left wing down roll moment. Also $(-L_{\delta_{PED}}/L_{\delta_{LAT}})$ is positive since $L_{\delta_{LAT}}$ is positive. On the other hand, $\Delta\delta_{PED}/\Delta\beta$ is usually negative in steady heading sideslips for configurations with positive directional stability. For most configurations then, the roll moment generated in a SHSS by the tail rotor is into the sideslip and must be balanced by cyclic into the sideslip. This additional cyclic requirement contributes to the "effective dihedral" of the vehicle. The term "dihedral effect" refers to the derivative L_v in which the roll due to pilot generated tail rotor inputs is not considered.

The resultant of the aerodynamic forces acting on the helicopter when flying in a steady heading sideslip is in equilibrium with the vertical weight force as depicted in Figure 7.22. The aerodynamic forces are divided into those acting on the main rotor, those acting on the tail rotor, and those acting on the fuselage/tail surface. When the resultant aerodynamic force has a lateral component (a force parallel to the y body axis), equilibrium of the aerodynamic forces with the gravity force is obtained when the helicopter has a bank angle.

When there is no sideslip, there is still a tail rotor force producing a yaw moment to balance the torque on the main rotor. Consequently, the resultant aerodynamic force has a lateral component along the aircraft y body axis to the right which requires a negative bank angle for equilibrium, as depicted in Figure 7.22, for zero sideslip angle. Another way of describing this equilibrium is the left horizontal component of rotor thrust is in equilibrium with the right horizontal component of tail rotor thrust. In this case, the tail rotor thrust acts at the CG which is the case if the tail rotor were on the roll axis.

Left or right side forces act on the fuselage and vertical tail depending on whether the helicopter has a right or left sideslip, respectively. A right sideslip generates a left fuselage side force to balance the right tail rotor force. At the inherent sideslip angle where the side forces are balanced, the helicopter can fly with zero bank angle. In the case shown on Figure 7.22, the inherent sideslip angle is approximately 5°.

A further increase in right sideslip with the corresponding increase in left fuselage side force leads to a resultant aerodynamic side force to the left which requires a right bank angle for equilibrium. Similarly, a left sideslip introduces a side force to the right and requires a left bank angle for equilibrium.

Tail rotor thrust is also changed by lateral velocity which results in a change in the effective angles of attack of the tail rotor blades. However, to maintain yaw moment equilibrium, the pilot introduces a pedal input which changes tail rotor pitch to maintain an approximately constant tail rotor force as the helicopter is placed in steady heading sideslips.

The main rotor is affected by lateral velocity. The principal effect is TPP tilt or blow back resulting in a corresponding lateral tilt of the thrust vector and a roll moment about the CG. However, to maintain roll moment equilibrium in the steady heading sideslip, the pilot makes a lateral control input.

The bank angle versus sideslip variation in steady heading sideslips is determined primarily by the variation of fuselage/tail side forces with sideslip angle. The relationship of bank angle to sideslip is used to make a qualitative determination of the strength of sideforces. The bank angle at which the pilot can feel the sideslip can be used to assist in determining the strength of sideforce.

7.3.5.5 ONE CONTROL ROLLS

During flight evaluation of the Static Lateral Directional flight characteristics of helicopters, stabilized, one control turns are conducted as a way of gaining clues towards the stable or unstable nature of the spiral mode. The following paragraph is included to deal with the unsteady portion, that is to say the portion of a one control turn when angle of bank is being generated $(\dot{p}, \dot{r}, p \neq 0)$. Once steady turning flight is achieved, $(\dot{p} = \dot{r} = \dot{p} = 0)$, paragraph 7.5.5.6 should be used to aid in the analysis of this condition.

Some significant observations can be made immediately following and shortly after a one control input is made to initiate a one control turn. During this transient portion of the maneuver, information is available about adverse/proverse yaw and roll due to pedal as well as the relative strength of the effective dihedral and directional stability. Understand that these are transient maneuvers in which most of the flight variables are changing with time. For this reason, there are no simplified governing equations for these tests and they are therefore conducted in a qualitative fashion and can be reviewed after the flight history plots.

When performing a lateral cyclic only roll, if there is a significant amount of adverse or proverse yaw present in the test vehicle, there will be an immediate yaw response to cyclic application. For adverse yaw, the yaw will be in the direction opposing the turn $(N_{\delta_{lat}} -)$ while for proverse, the response will be into the turn $(N_{\delta_{lat}} +)$. If there is a large amount of yaw due to roll rate (Np), then as the roll rate builds, there could be confusion between this rate coupling and the control coupling, adverse yaw. This ambiguity is exacerbated by short roll mode time constants. Once a sideslip perturbation from inherent is established, the willingness of the sideslip to return towards inherent is considered a measure of the relative strength of directional stability (Nv).

Pedal only rolls are conducted by using the directional pedals to generate a sideslip. Upon initiation of the maneuver, if there is a significant amount of roll due to pedal $(L_{\delta_{ped}})$ present in the test vehicle, there will be an immediate roll response to the pedal application. After the target amount of sideslip has been generated, roll acceleration in response to that sideslip is considered an indication of the relative strength of effective dihedral. Roll acceleration is qualitatively hard to perceive, so roll rate resulting from a given sideslip is often substituted as the measure of dihedral strength. If there is a large amount of roll damping in the test vehicle, the low steady rate reached following the pedal application could mask otherwise strong effective dihedral.

7.3.5.6 STEADY TURNING FLIGHT

By definition, steady turning flight requires a constant roll angle (ϕ) and a constant turn rate about a vertical axis $(\dot{\psi})$. The time derivatives of lateral velocity, roll rate, and yaw rate are zero as in steady heading sideslips $(\dot{v} = \dot{p} = \dot{r} = 0)$. However, the angular velocities about the aircraft body axes are not zero and are approximated by the following expressions in the case of steady level turns:

$p \simeq 0$	roll rate
$q \simeq \dot{\psi} \sin\phi$	pitch rate
$r = \dot{\psi}$	rate of change of yaw attitude ≈ turn rate

Using the above assumptions, the lateral-directional equations of motion for small disturbances reduce to the following for steady turning flight:

$$-Y_v u_0 \Delta\beta + (u_0 - Y_r)\Delta r - g\phi = Y_{A_{1s}} \Delta A_{1s} + Y_{\theta_{TR}} \Delta\theta_{TR} = Y_{\delta_{LAT}} \Delta\delta_{LAT} + Y_{\delta_{PED}} \Delta\delta_{PED}$$

eq 7.43

$$-L_v u_0 \Delta\beta - L_r \Delta r = L_{A_{1s}} \Delta A_{1s} + L_{\theta_{TR}} \Delta\theta_{TR} = L_{\delta_{LAT}} \Delta\delta_{LAT} + L_{\delta_{PED}} \Delta\delta_{PED}$$

eq 7.44

$$-N_v u_0 \Delta\beta - N_r \Delta r = N_{A_{1s}} \Delta A_{1s} + N_{\theta_{TR}} \Delta\theta_{TR} = N_{\delta_{LAT}} \Delta\delta_{LAT} + N_{\delta_{PED}} \Delta\delta_{PED}$$

eq 7.45

Where:

A_{1s}	- Lateral cyclic pitch angle, shaft referenced
β	- Sideslip angle
δ_{LAT}	- Lateral control
δ_{PED}	- Pedal control
ϕ	- Roll angle

g	- Gravity
$L_{A_{1s}}$	- Roll moment due to lateral cyclic pitch angle
$L_{\delta_{LAT}}$	- Roll moment due to lateral control
$L_{\delta_{PED}}$	- Roll moment due to pedal control
$L_{\theta_{TR}}$	- Roll moment due to tail rotor pitch angle
L_r	- Roll moment due to yaw rate
L_v	- Roll moment due to lateral velocity
$N_{A_{1s}}$	- Yaw moment due to lateral cyclic pitch angle
$N_{\delta_{LAT}}$	- Yaw moment due to lateral control
$N_{\delta_{PED}}$	- Yaw moment due to pedal control
$N_{\theta_{TR}}$	- Yaw moment due to tail rotor pitch angle
N_r	- Yaw moment due to yaw rate
N_v	- Yaw moment due to lateral velocity
p	- Angular velocity about x axis, Roll rate
q	- Angular velocity about y axis, Pitch rate
θ_{TR}	- Tail rotor pitch angle
r	- Angular velocity about z axis, Yaw rate
u_0	- Initial velocity
ψ	- Yaw attitude
$Y_{A_{1s}}$	- Side force due to lateral cyclic pitch angle
$Y_{\delta_{LAT}}$	- Side force due to lateral control
$Y_{\delta_{PED}}$	- Side force due to pedal control
$Y_{\theta_{TR}}$	- Side force due to tail rotor pitch angle
Y_r	- Side force due to yaw rate
Y_v	- Side force due to lateral velocity
$\dot{\psi}$	- Rate of change of yaw attitude.

These equations for side force, roll moment, and yaw moment equilibrium are valid for nearly level turns at constant flight path velocity. It is assumed longitudinal control inputs are used to maintain constant speed. The discussion is simplified by assuming lateral control does not produce yaw moments.

The set of variables ($\Delta\beta = \Delta v/u_0$ = sideslip angle, r = roll rate, ϕ = roll angle A_{1s} = lateral cyclic pitch angle, and θ_{TR} = tail rotor pitch angle) which provide a solution for the equations of motion are the incremental changes from an initial trimmed rectilinear condition to a steady off trim turning flight condition. The time history of the variables going from a trimmed rectilinear flight condition to steady turning flight is not considered.

7.3.5.6.1 Cyclic Only Turns

Figure 7.23 depicts a helicopter making an approximately level, cyclic only, right turn. The helicopter was disturbed from an initial condition in which the tail rotor thrust was balanced by a fuselage side force generated by flying at the inherent sideslip angle. As a result there is no initial lateral tilt of the main rotor thrust vector. The diagram shows the changes in forces acting on the helicopter resulting from a steady turn. For simplicity, the rotor thrust is assumed to act along the shaft axis neglecting any rotor side forces due to turning. The diagram corresponds to a helicopter with stable spiral characteristics.

The helicopter is banked to the right so the thrust vector has a horizontal component to the right. This right thrust component balances the sum of the centrifugal force due to turning and the horizontal component of the aerodynamic side force on the fuselage/tail acting to the left.

The resultant of the weight and centrifugal force in Figure 7.23 has a lateral component acting to the right along the y body axis of the helicopter, and the pilot feels a force tending to push him towards the right or lower side of the seat in this uncoordinated turn.

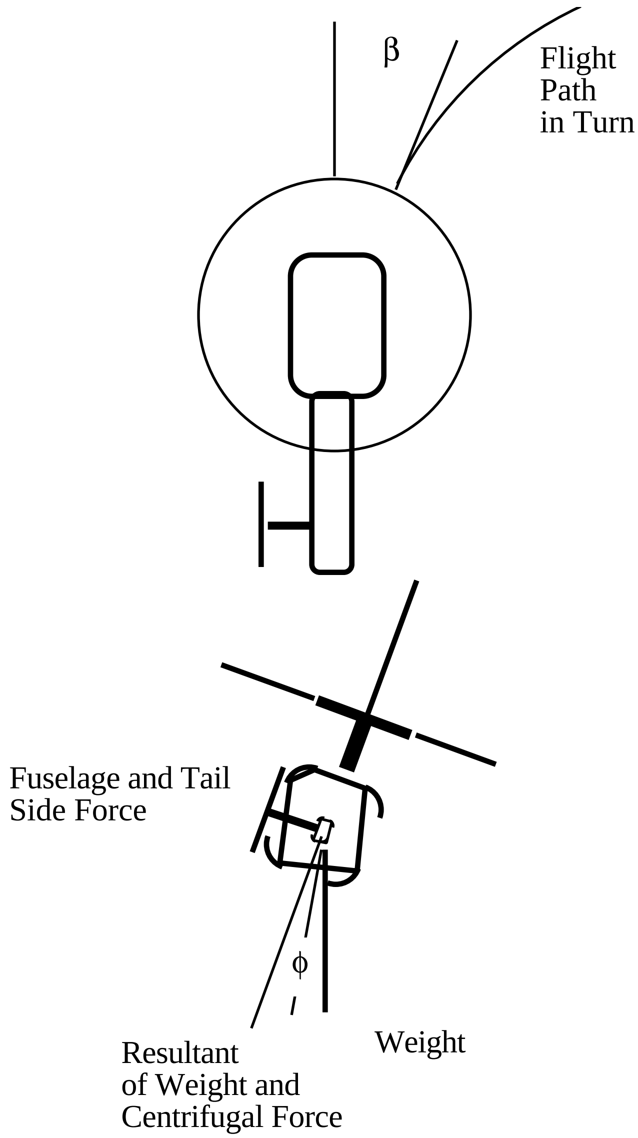


Figure 7.23
Cyclic Only Turn of Directionally and Spirally Stable Helicopter

The side force acts to the left as a result of the right sideslip of the helicopter. When the helicopter has positive directional stability, a right sideslip is required for a right cyclic only turn. A nose right yaw moment due to sideslip balances the nose left damping moment generated by the turn rate to the right. This equilibrium is expressed by the yaw moment equation for a steady turn which reduces to the following form for this cyclic only case:

$$-N_v u_0 \Delta\beta - N_r \Delta r = 0$$

Assumptions:

$\Delta\delta_{PED} = 0$ (No pedal input)

$N_{\delta_{LAT}} = 0$ (No yaw moment due to lateral cyclic)

It follows the yaw rate to lateral velocity ratio is:

$$\frac{\Delta r}{u_0 \Delta\beta} = \frac{N_v}{\left(-N_r\right)}$$

Where:

β	- Sideslip angle
$N_{\delta_{LAT}}$	- Yaw moment due to lateral control
N_r	- Yaw moment due to yaw rate
N_v	- Yaw moment due to lateral velocity
r	- Angular velocity about z axis, Yaw rate
u_0	- Initial velocity.

The yaw moment due to yaw rate is inherently negative so the quantity ($-N_r$) is positive. The ratio of yaw rate to sideslip angle is positive for a helicopter with positive directional stability.

There are roll moments acting on the helicopter shown in Figure 7.23 because of its lateral velocity and yaw rate. These are balanced by the roll moment due to lateral control required for a steady turn. The equilibrium zero roll moment is expressed as follows:

$$L_v u_0 \Delta\beta \qquad + L_r \frac{\Delta r}{u_0 \Delta\beta} u_0 \Delta\beta \qquad + L_{\delta_{LAT}} \Delta\delta_{LAT} = 0 \qquad \textit{eq 7.46}$$

or:

$$\underbrace{L_v u_0 \Delta\beta}_{\substack{\Uparrow \\ \text{Roll moment} \\ \text{due to sideslip}}} \quad + \underbrace{L_r \frac{N_v}{(-N_r)} u_0 \Delta\beta}_{\substack{\Uparrow \\ \text{Roll moment} \\ \text{due to yaw rate}}} \quad + \underbrace{L_{\delta_{LAT}} \Delta\delta_{LAT}}_{\substack{\Uparrow \\ \text{Roll moment} \\ \text{due to lateral control}}} = 0 \qquad \textit{eq 7.47}$$

Where:

β	- Sideslip angle
δ_{LAT}	- Lateral control
$L_{\delta_{LAT}}$	- Roll moment due to lateral control
L_r	- Roll moment due to yaw rate
L_v	- Roll moment due to lateral velocity
$N_{\delta_{LAT}}$	- Yaw moment due to lateral control
N_r	- Yaw moment due to yaw rate
N_v	- Yaw moment due to lateral velocity
r	- Angular velocity about z axis, Yaw rate
u_0	- Initial velocity.

The yaw rate to lateral velocity ratio $\Delta r/u_0\Delta\beta = N_v/(-N_r)$ required for yaw moment equilibrium is introduced explicitly in the second form of the roll moment equilibrium equation.

Helicopter control systems are designed so a right lateral control deflection produces a right wing down roll moment. Thus, the $L_{\delta_{LAT}}$ derivative is positive and the last term gives a positive roll moment for a positive control deflection. On the other hand, existing helicopters generally exhibit positive effective dihedral or negative L_v. Thus, the first term gives a negative roll moment for a positive sideslip.

The N_r derivative is negative corresponding to positive yaw damping, while the N_v derivative is positive for a configuration with positive directional stability. The sign of the roll moment due to yaw rate derivative, L_r, is dependent on the location of the tail rotor and vertical tail. A positive yaw rate results in a left velocity increment at the tail rotor and an incremental tail rotor force to the right. This gives a positive increment in roll moment for a tail rotor above the roll axis corresponding to a positive L_r. The second term in the equation produces a positive roll moment for a configuration with directional stability and a tail rotor and tail combination above the roll axis. This is the roll moment generated by the yaw rate required for yaw moment equilibrium and is proportional to the $\Delta r/u_0\Delta\beta$ ratio in the turn. This moment has the opposite sign to the moment due to the dihedral effect for a configuration with positive directional stability and a positive L_r. The combined moments due to sideslip and yaw rate are balanced by the moment due to lateral control.

The following expression for the sideslip-lateral control ratio in a cyclic only turn is obtained by rearranging the roll moment equilibrium equation:

$$\frac{\Delta\beta}{\Delta\delta_{LAT}} = \frac{1}{u_0} \frac{L_{\delta_{LAT}}}{\left[\left(-L_v\right) - L_r \frac{N_v}{\left(-N_r\right)}\right]} \qquad \textit{eq 7.48}$$

Where:

β	- Sideslip angle
δ_{LAT}	- Lateral control
$L_{\delta_{LAT}}$	- Roll moment due to lateral control
L_r	- Roll moment due to yaw rate
L_v	- Roll moment due to lateral velocity
N_r	- Yaw moment due to yaw rate
N_v	- Yaw moment due to lateral velocity
u_0	- Initial velocity.

The influence of the roll moment due to sideslip, or dihedral effect, is present in the first term in the denominator. The second term in the denominator introduces the effect of the roll moment due to yaw rate. The first term due to dihedral effect is always positive, while the second term due to yaw rate makes a negative contribution to the denominator when L_r and N_v are positive. The second term is more important for configurations with high directional stability which results in high yaw rate to sideslip ratio in cyclic only turns.

The above expression for the ratio of sideslip to lateral control deflection is rearranged giving:

$$\frac{\Delta\beta}{\Delta\delta_{LAT}} = \frac{1}{u_0} \quad \frac{\left(-N_r\right) L_{\delta_{LAT}}}{\left(L_v N_r - L_r N_v\right)} \qquad \textit{eq 7.49}$$

The denominator of the right hand side is the spiral stability parameter $(L_v N_r - L_r N_v)$.

The yaw rate to lateral control ratio in a cyclic only turn follows from the expressions for the $\Delta r / u_0 \Delta\beta$ and $\Delta\beta / \Delta\delta_{LAT}$ ratios:

$$\frac{\Delta r}{\Delta\delta_{LAT}} = \frac{\Delta r}{u_0 \Delta\beta} \quad \frac{u_0 \Delta\beta}{\Delta\delta_{LAT}} = \frac{N_v L_{\delta_{LAT}}}{\left(L_v N_r - L_r N_v\right)} \qquad \textit{eq 7.50}$$

Where:

β	- Sideslip angle
δ_{LAT}	- Lateral control
$L\delta_{LAT}$	- Roll moment due to lateral control
L_r	- Roll moment due to yaw rate
L_v	- Roll moment due to lateral velocity
N_r	- Yaw moment due to yaw rate

N_v - Yaw moment due to lateral velocity

r - Angular velocity about z axis, Yaw rate

u_0 - Initial velocity.

When the spiral stability parameter and directional stability derivative are positive, a right control displacement gives a positive sideslip and yaw rate as shown on Figure 7.24.

7.3.5.6.2 Pedal Only Turns

Pedal only turns, like cyclic only turns, are discussed by considering side force, roll moment, and yaw moment equilibrium in steady turning flight. Bank angle affects side force equilibrium and does not appear in the expressions for roll and yaw moment equilibrium which are rewritten below, with lateral control (δ_{LAT}) equal to zero.

$$\underbrace{L_v u_o \Delta\beta}_{\substack{\text{Roll Moment} \\ \text{due to sideslip}}} + \underbrace{L_r \Delta r}_{\substack{\text{Roll Moment} \\ \text{due to yaw rate}}} + \underbrace{L_{\delta_{PED}} \Delta\delta_{PED}}_{\substack{\text{Roll moment} \\ \text{due to pedal input}}} = 0 \qquad \textit{eq 7.51}$$

and

$$\underbrace{N_v u_0 \Delta\beta}_{\substack{\text{Yaw moment} \\ \text{due to sideslip}}} + \underbrace{N_r \Delta r}_{\substack{\text{Yaw moment} \\ \text{due yaw rate}}} + \underbrace{N_{\delta_{PED}} \Delta\delta_{PED}}_{\substack{\text{Yaw moment due} \\ \text{to right pedal input}}} = 0 \qquad \textit{eq 7.52}$$

Where:

β - Sideslip angle

δ_{PED} - Pedal control

$L_{\delta_{PED}}$ - Roll moment due to pedal control

L_r - Roll moment due to yaw rate

L_v - Roll moment due to lateral velocity

$N_{\delta_{PED}}$ - Yaw moment due to pedal control

N_r - Yaw moment due to yaw rate
N_v - Yaw moment due to lateral velocity
r - Angular velocity about z axis, Yaw rate
u_0 - Initial velocity.

The quantities highlighted by arrows are positive for a conventional helicopter. The sideslip velocity ($u_0\Delta\beta$), yaw rate (Δr) and pedal displacement ($\Delta\delta_{PED}$) in these equations are the incremental changes from an initial trimmed rectilinear flight condition.

The pilot selects one of the three variables in the roll and yaw moment equations; the other variables, or their ratios with respect to the selected variable, are determined from the equilibrium conditions. For example, if he desires a particular turn rate, he adjusts pedal control until the turn rate is established. The pilot accepts the corresponding sideslip determined from the roll and yaw moment equilibrium and the bank angle necessary to achieve side force equilibrium.

There is usually some control moment coupling in helicopters so a lateral control displacement, or pedal displacement, might result in both roll and yaw moments. The yaw moment due to lateral control deflection is generally small and is neglected in considering cyclic only turns. An analogous assumption for pedal only turns neglects the roll moment due to pedal control corresponding to a configuration with the tail rotor on the roll axis. This configuration is considered first, subsequent discussion shows the characteristics obtained when a pedal input causes both roll and yaw moments.

When $L_{\delta_{PED}} = 0$, the roll moment due to sideslip is equal and opposite to the roll moment due to yaw rate. This condition is satisfied when the sideslip to yaw rate ratio is:

$$\frac{\left(u_0\Delta\beta\right)}{\Delta r} = \frac{L_r}{\left(-L_v\right)} \qquad \textit{eq 7.53}$$

where the denominator responsible for the dihedral effect is positive for conventional helicopters. Thus, the sign of $u_0\Delta\beta/\Delta r$ depends on the sign of the roll moment due to yaw rate derivative (L_r). When the sideslip required for roll moment equilibrium given by the above expression is substituted into the equation for yaw moment equilibrium, it becomes:

$$N_v \frac{L_r}{(-L_v)} \Delta r \; - (-N_r)\,\Delta r + N_{\delta_{PED}} \Delta\delta_{PED} \; = \; 0$$

eq 7.54

The yaw rate to pedal deflection in a pedal only turn is:

$$\frac{\Delta r}{\Delta\delta_{PED}} = \frac{(-L_v)N_{\delta_{PED}}}{L_v N_r - L_r N_v}$$

eq 7.55

and the corresponding sideslip to pedal deflection ratio is:

$$\frac{u_0\Delta\beta}{\Delta\delta_{PED}} = \frac{L_r N_{\delta_{PED}}}{L_v N_r - L_r N_v}$$

eq 7.56

Where:

β	- Sideslip angle
δ_{PED}	- Pedal control
L_r	- Roll moment due to yaw rate
L_v	- Roll moment due to lateral velocity
$N_{\delta_{PED}}$	- Yaw moment due to pedal control
N_r	- Yaw moment due to yaw rate
N_v	- Yaw moment due to lateral velocity
r	- Angular velocity about z axis, Yaw rate
u_0	- Initial velocity.

The denominator in the above expressions is the spiral stability parameter (L_vN_r - L_rN_v) which appeared in the corresponding expressions for cyclic only turns. The same expression is obtained in the two cases because the spiral stability of the helicopter depends on the coupling between yaw rate and sideslip and not on the control input.

When the spiral stability parameter is positive (Figure 7.24) a steady right turn is obtained with a right pedal displacement (pedal input is required in the direction of turn). If the pilot required a left pedal input to obtain temporary equilibrium in a right turn, it indicates the spiral stability parameter is negative. Thus, one method used to evaluate the spiral stability is to observe the $\Delta r/\Delta\delta_{PED}$ ratio in pedal only turn.

The yaw moment balance on Figure 7.24 is used to determine the initial tendency of a helicopter to return to, or depart from, a steady turn when disturbed from equilibrium and thus the spiral stability. Assuming the helicopter originally is stabilized in a pedal only turn, if the pedal input is reduced, the corresponding yaw moment is reduced. Consequently, there is a nose left yaw moment tending to increase sideslip. An increase in sideslip results in a side force to the left, decreasing the turn rate and a roll moment due to dihedral effect decreasing the bank angle. Consequently, where (L_vN_r -L_rN_v) > 0, the initial tendency of the helicopter when disturbed is to return to its original steady turn condition.

The configuration in Figure 7.24 has positive directional stability as well as spiral stability, and the side force at the vertical tail is assumed to act above the roll axis so L_r is positive. The helicopter is banked to the right giving a right horizontal thrust component to balance the horizontal component of the aerodynamic side force and the centrifugal force due to turning. The positive roll moment due to yaw rate is balanced by the negative dihedral effect. Since the roll moment due to lateral velocity derivative (L_v) is negative, the helicopter has a positive or right sideslip which is obtained with a nose left yaw angle relative to the turning flight path.

There is a negative yaw damping moment tending to damp the positive yaw rate due to turning toward the right. This is partially compensated by the positive yaw moment due to right sideslip.

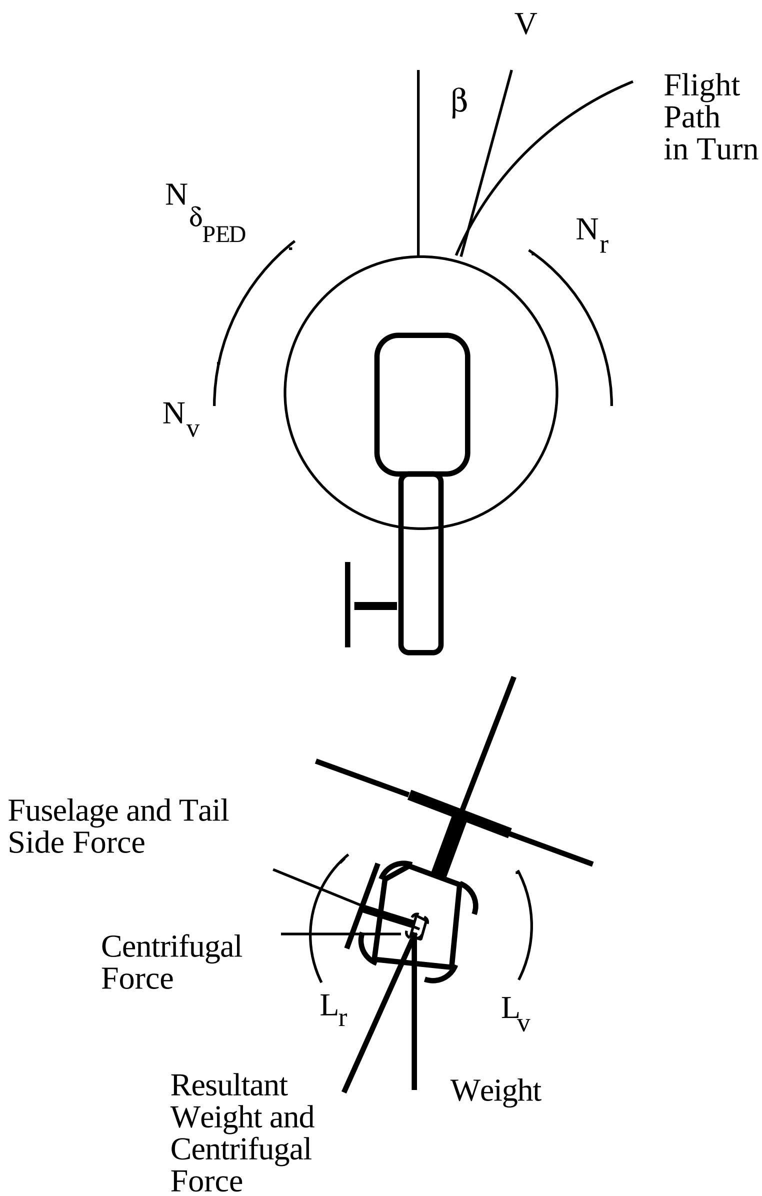


Figure 7.24
Pedal Only Turn of Directionally and Spirally Stable Helicopter

The yaw moment due to a right pedal input is required to obtain yaw moment equilibrium. If the spiral stability is negative, a left pedal displacement is required to hold the right turn. In this case, a reduction in left pedal causes a nose right yaw moment increment and a reduction in sideslip. There is a reduction in dihedral effect roll moment and left side force, both of which cause a steeper bank and tighter turn.

The preceding discussion of a pedal only turn is simplified by assuming a pedal input produces only a yaw moment and no roll moment. The effect of a roll moment due to a pedal input is the same as a moment of equal magnitude produced by lateral control deflection. Consequently, the result obtained for the cyclic only turn is used for determining the contribution of $L_{\delta_{PED}}\Delta\delta_{PED}$ in the pedal only turn. The expression for $\Delta r/\Delta\delta_0$ and $u_0\Delta\beta/\Delta\delta_{PED}$ in the pedal only turn are modified as follows when both the roll and yaw moment due to pedal input are included.

$$\frac{\Delta r}{\Delta\delta_{PED}} = \frac{\left(-L_v\right) N_{\delta_{PED}}\left[1 - \left(\frac{L_{\delta_{PED}}}{N_{\delta_{PED}}}\right)\left(\frac{N_v}{L_v}\right)\right]}{L_vN_r - L_rN_v}$$

eq 7.57

$$\frac{u_0\Delta\beta}{\Delta\delta_{PED}} = \frac{L_r N_{\delta_{PED}}\left[1 - \left(\frac{L_{\delta_{PED}}}{N_{\delta_{PED}}}\right)\left(\frac{N_r}{L_r}\right)\right]}{L_vN_r - L_rN_v}$$

eq 7.58

Where:

β	- Sideslip angle
δ_{PED}	- Pedal control
$L_{\delta_{PED}}$	- Roll moment due to pedal control
L_r	- Roll moment due to yaw rate
L_v	- Roll moment due to lateral velocity
$N_{\delta_{PED}}$	- Yaw moment due to pedal control
$N_{\delta_{PED}}$	- Yaw moment due to pedal control
N_r	- Yaw moment due to yaw rate

N_v - Yaw moment due to lateral velocity
r - Angular velocity about z axis, Yaw rate
u_0 - Initial velocity.

The term in square brackets in the numerator of the expression for $\Delta r / \Delta\delta_{PED}$ is generally positive. Consequently, the sign of the $\Delta r / \Delta\delta_{PED}$ ratio in a pedal only turn remains an indication of the sign of the spiral stability parameter.

The square bracket term in the numerator of the expression for $u_0 \Delta\beta / \Delta\delta_{PED}$ can be positive or negative. Thus the sign of the $u_0 \Delta\beta / \Delta\delta_{PED}$ ratio in a pedal only turn may not reflect the sign of the roll moment due to yaw rate derivative (L_r).

7.3.5.6.3 Coordinated Turns

Figure 7.25 depicts a simplified case of coordinated turning flight in which the helicopter is turning about a vertical axis with the nose on the horizon at a roll angle (ϕ). The ball centered flight condition is illustrated for a right turn (nose-on view).

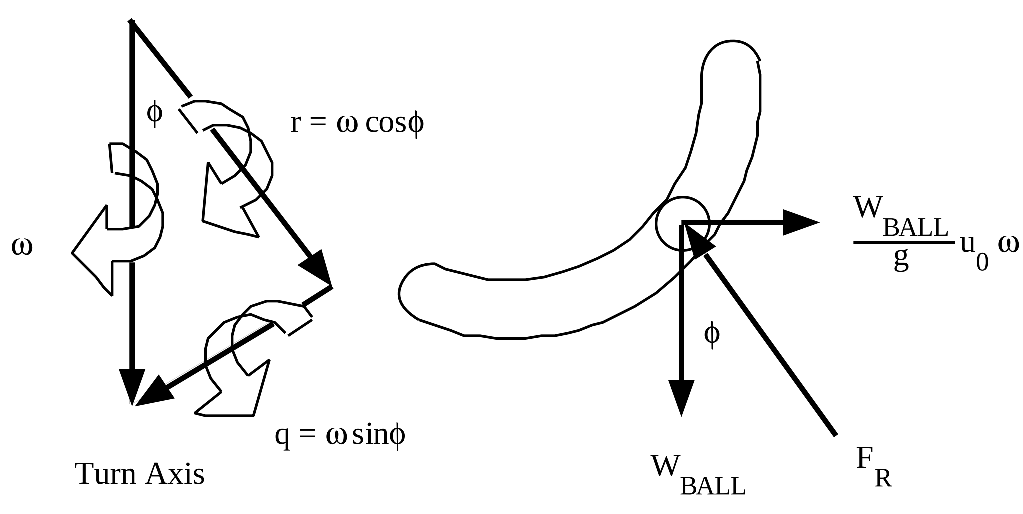


Figure 7.25
Simplified Coordinated Turning Flight

The resolution of turn angular velocity (ω) along the helicopter axes gives a pitch rate ($q = \omega \sin \phi$), a yaw rate ($r = \omega \cos \phi$), and no roll rate.

Three forces are shown on a hypothetical ball used for turn coordination. They are, the weight force (W_{BALL}), the centrifugal force due to the curved flight path in the turn ($W_{BALL}/g\ u_0\ \omega$), and the reaction force (F_R) from the ball race.

The race reaction force is in the z direction in the ball centered condition, while the y components of the weight and centrifugal forces are balanced for equilibrium:

$$W_{BALL} \sin\phi = \frac{W_{BALL}}{g} u_0 \omega \cos\phi = \frac{W_{BALL}}{g} u_0 r \qquad \textit{eq 7.59}$$

$$\phi \approx \sin\phi = \frac{u_0 r}{g} \qquad \textit{eq 7.60}$$

Where:

ϕ	- Roll angle
g	- Gravity
r	- Angular velocity about z axis, Yaw rate
u_0	- Initial velocity
ω	- Turn angular velocity
W_{BALL}	- Weight of the ball.

In discussing cyclic only and pedal only turns, it was shown that side force equilibrium is obtained by flying at the proper bank angle. In the case of coordinated turns, the bank angle is related to the yaw rate by the above expression. In this case, the pedal and lateral control inputs are adjusted until side force equilibrium is obtained at the proper bank angle.

7.3.6 Dynamic Stability

Lateral-directional dynamics are discussed neglecting coupling with longitudinal dynamics. The lateral-directional dynamics of the unaugmented helicopter is represented by fourth order equations of motion leading to a characteristic equation with four roots. Two of these roots are usually real and correspond to the first order roll and spiral mode responses of the helicopter. The other two roots are usually complex, corresponding to the lateral-directional oscillation (LDO) mode. The roll rate response of the helicopter is not changed greatly as forward speed increases. However, the directional stability which develops has a large effect on the other two modes. As speed increases from hover, the yaw motion couples with the oscillatory lateral velocity to roll motion, producing a mode that is very similar to the Dutch-roll mode of the conventional fixed wing airplane. Approximate equations of motions used to analyze these modes are summarized in Table 7.III.

7.3.6.1 ROLL MODE

The helicopter roll rate response with offset flapping hinges or hingeless rotors is rapid compared with those for the yaw degree of freedom. Because of this difference in response times, there is little coupling of the rapid roll response of the helicopter with the slower yaw response.

The initial roll response following the application of a step roll moment (ΔL) is the typical first order response.

$$p = \frac{\left(\frac{\Delta L}{I_{xx}}\right)}{\left(-L_p\right)} \left(1 - e^{-\frac{t}{\tau_R}}\right)$$

eq 7.61

Where:

$$\tau_R \approx -\frac{1}{L_p}$$

And:

e	- Base of natural logarithm
I_{xx}	- Moment of inertia about x axis, roll moment of inertia
ΔL	- Net moment about x axis, Roll moment
L_p	- Roll moment due to roll rate
p	- Angular velocity about x axis, Roll rate
t	- Time

τ_R - Roll mode time constant.

The inherent or augmented roll damping derivative (L_p), is negative giving an exponentially damped roll rate response. This roll rate response mode does not vary greatly with trim airspeed because the roll damping derivative remains approximately constant.

The preceding approximate formulas are not valid when the effective damping derivative is small, for example when there is a stronger effect of yaw and sideslip motions on the roll response mode.

7.3.6.2 SPIRAL STABILITY

The long term lateral-directional dynamics or spiral stability in forward flight affects pilot workload in steady turns. It also affects workload in straight and level trimmed flight. An approximate form of the lateral-directional equations of motion in Table 7.III is used in discussing the spiral mode and relating it to steady turns. The discussion is for pedal control inputs, but the resultant character spiral mode is independent of the way the mode is excited. The yaw moment due to pedal displacement is included on the right hand side of the yaw equation only. The roll moments and side forces due to pedal inputs usually are of secondary importance and are neglected to simplify the following discussion.

Table 7.III

Lateral-Directional Equations of Motion for Small Disturbances from Trim

Full Equations Neglecting Coupling with Longitudinal Motion

$$\Delta\dot{v} - Y_v\Delta v + (u_0 - Y_r)\Delta r - Y_p\Delta p - g\Delta\phi = Y_{\delta_{LAT}}\Delta\delta_{LAT} + Y_{\delta_{PED}}\Delta\delta_{PED}$$

$$\Delta\dot{p} - L_v\Delta v - L_r\Delta r - L_p\Delta p = L_{\delta_{LAT}}\Delta\delta_{LAT} + L_{\delta_{PED}}\Delta\delta_{PED}$$

$$\Delta\dot{r} - N_v\Delta v - N_r\Delta r - N_p\Delta p = N_{\delta_{LAT}}\Delta\delta_{LAT} + N_{\delta_{PED}}\Delta\delta_{PED}$$

$$\Delta p = \Delta\dot{\phi} - \Delta\dot{\psi}\sin\theta \approx \Delta\dot{\phi}$$

Simplified Equation for Treating the Roll Mode

(Assumes large inherent or augmented roll damping derivative)

$$\dot{p} - L_p p = L_{\delta_{LAT}}\Delta\delta_{LAT} + L_{\delta_{PED}}\Delta\delta_{PED} + L_v\Delta v + L_r\Delta r = \frac{\Delta L}{I_{xx}}$$

Simplified Equations for Treating the Spiral Mode (Pedal only input)

$$(u_0\,\Delta r) - g\,\Delta\phi = 0$$

$$- L_v\Delta v - L_r\Delta r - L_p\Delta p = 0$$

$$\Delta\dot{r} - N_v\Delta v - N_r\Delta r - N_p\Delta p = N_{\delta_{PED}}\Delta\delta_{PED}$$

Simplified Equations for Treating the LDO Mode

$$\Delta\dot{v} - Y_v\Delta v + (u_0 - Y_r)\Delta r - g\Delta\phi = Y_{\delta_{LAT}}\Delta\delta_{LAT} + Y_{\delta_{PED}}\Delta\delta_{PED}$$

$$\Delta\ddot{\phi} - L_v\Delta v - L_p\Delta\dot{\phi} = L_{\delta_{LAT}}\Delta\delta_{LAT} + L_{\delta_{PED}}\Delta\delta_{PED}$$

$$\Delta\dot{r} - N_v\Delta v - N_r\Delta r = N_{\delta_{LAT}}\Delta\delta_{LAT} + N_{\delta_{PED}}\Delta\delta_{PED}$$

The simplification of the equations of motion are in part based on the assumption the spiral mode response is much slower than the other lateral-directional responses of the helicopter. The spiral root and corresponding time constant is:

$$\lambda_s = -\frac{1}{\tau_s} = \frac{-\left(L_v N_r - L_r N_v\right)}{-L_v\left(1 - \frac{u_0}{g} N_p\right) - N_v L_p \left(\frac{u_0}{g}\right)}$$

eq 7.62

Where:

δ_{PED}	- Pedal control
g	- Gravity
L_p	- Roll moment due to roll rate
L_r	- Roll moment due to yaw rate
λ_s	- Spiral mode root
L_v	- Roll moment due to lateral velocity
$N_{\delta_{PED}}$	- Yaw moment due to pedal control
N_p	- Yaw moment due to roll rate
N_r	- Yaw moment due to yaw rate
N_v	- Yaw moment due to lateral velocity
r	- Angular velocity about z axis, Yaw rate
$\dot{r}$	- Angular acceleration about z axis
τ_s	- Spiral mode time constant
u_0	- Initial velocity
v	- Translational velocity component along y axis.

The time solution for the first order spiral mode response is:

$$r = r_0\, e^{\lambda_s t} = r_0\, e^{-\frac{t}{\tau_s}}$$

eq 7.63

Where:

e	- Base of natural logarithm
λ_s	- Spiral mode root
r	- Angular velocity about z axis, Yaw rate
r_0	- Initial angular velocity about z axis, Yaw rate
t	- Time
τ_s	Spiral mode time constant.

An aperiodic divergent spiral mode response is obtained when the spiral root, λ_s, is positive. The denominator in the expression for the spiral root is positive with configurations having positive directional stability ($N_v > 0$). In this case, the sign of the spiral root depends upon the sign of the spiral stability parameter ($L_vN_r - L_rN_v$) in the numerator. This same parameter was discussed in connection with steady turning flight. The product L_vN_r is typically positive for configurations with positive yaw damping and positive dihedral effect. Configurations with high tail rotor and vertical tails have a positive roll moment due to yaw rate derivative (L_r); therefore, the spiral stability parameter is negative with large, positive directional stability producing a divergent spiral mode. The discussion in Section 7.3.5.6.1 and 7.3.5.6.2 indicated how the sign of the spiral stability root is estimated in steady turns. The spiral mode is observed by introducing a bank angle, then carefully returning the controls to their respective level flight trim positions. The time-to-one-half ($T_{1/2}$) or time-to-double (T_2) bank amplitude is then determined.

7.3.6.3 LATERAL-DIRECTIONAL OSCILLATION

The LDO mode of the helicopter changes character going from hovering to forward flight. In hovering flight, where the helicopter has little directional stability, an oscillatory mode is obtained which primarily involves lateral velocity and roll motions. In forward flight, the directional stability of the typical helicopter results in the oscillatory roll/yaw mode. Generally, the frequency of the LDO is determined by coupling between the lateral velocity and yaw motions, while the coupling with the roll motion has a significant effect on its damping. The amplitude and phase of the roll to sideslip ratio in the LDO mode may influence the pilot's evaluation of lateral-directional handling qualities. To simplify understanding of the LDO motion, a particular case without roll motion in the mode is considered first. This is then generalized to include roll motions.

The simplified equations approximating the LDO mode, given in Table 7.III, are rearranged in Table 7.IV. The right hand side control input (forcing function) terms are set to zero to model the free response characteristics. Considering small disturbances close to an initial trim condition, the yaw rate (r) is equal to the rate of change of yaw attitude ($\dot{\psi}$) and the roll rate (p) is equal to the rate of change of bank angle ($\dot{\phi}$).

The simplest approximation for the LDO response is obtained by assuming the helicopter is restrained by a vertical pivot, so the lateral velocity relative to non-rotating axes and the roll angle are zero ($v_{NR} = 0$ and $\phi = 0$). In this case, $v = -u_0\psi$ and the yaw moment equilibrium equation becomes:

$$\Delta\ddot{\psi} + (-N_r)\Delta\dot{\psi} + (N_v u_0)\Delta\psi = 0 \qquad \textit{eq 7.64}$$

Where:

N_r	- Yaw moment due to yaw rate
N_v	- Yaw moment due to lateral velocity
u_0	- Initial velocity along x axis
ψ	- Yaw attitude
$\ddot{\psi}$	- Yaw attitude acceleration
$\dot{\psi}$	- Rate of change of yaw attitude.

This is the equation of motion for a second order spring mass damper system where the spring restraint is furnished by the directional stability derivative, $(k/m) = (N_v u_0)$, and the damping coefficient is due to the yaw damping derivative, $(B/m) = (-N_r)$. Approximations for the natural frequency and damping ratio are presented in Table 7.IV.

A more realistic description of the LDO is obtained when no restraint is assumed on lateral velocity. However, when considering a configuration with small or negligible dihedral effect, the roll angle is still assumed zero. This case is represented by the two equations on Table 7.IV for coupled yaw and sideslip motion which express, respectively, the equilibrium of yaw moments and side forces. The LDO mode corresponding to these

Table 7.IV
Approximate Equations for the Lateral-Directional Oscillation

Simplified Equations for the LDO

$$\Delta\dot{v} - Y_v\Delta v + \left(u_0 - Y_r\right)\Delta\dot{\psi} - g\Delta\phi = 0$$

$$\Delta\ddot{\phi} - L_v\Delta v - L_p\Delta\dot{\phi} = 0$$

$$\Delta\ddot{\psi} - N_r\Delta\dot{\psi} - N_v\Delta v - N_p\Delta\dot{\phi} = 0$$

$$\Delta\ddot{\psi} + \left(-N_r\right)\Delta\dot{\psi} + \left(N_v u_0\right)\Delta\psi - N_v\Delta v_{NR} - N_p\Delta\dot{\phi} = 0$$

(Yaw equation referred to non-rotating axes)

Helicopter Assumed Pivoted at the CG ($\phi = v_{NR} = 0$)

$$\Delta\ddot{\psi} + \left(-N_r\right)\Delta\dot{\psi} + \left(N_v u_0\right)\Delta\psi = 0$$

$$\omega_n \approx \sqrt{N_v u_0}$$

$$\zeta \approx \frac{1}{2\omega_n}\left(-N_r\right)$$

Coupled Equations for Yaw and Sideslip Motions ($\phi = 0$)

$$\left(\Delta\ddot{\psi}\ N_r\ \Delta\dot{\psi}\right) - N_v\ \Delta v = 0$$

$$\Delta\dot{v} - Y_v\Delta v + \left(u_0 - Y_r\right)\Delta\dot{\psi} = 0$$

Separate ψ and v Equations for Coupled Motion ($\phi = 0$)

$$\Delta\ddot{\psi} + \left(-N_r - Y_v\right)\Delta\dot{\psi} + \left[\ N_v u_0 + \left(Y_v N_r - N_v Y_r\right)\right]\ \Delta\psi = 0$$

$$\Delta\ddot{v} + \left(-N_r - Y_v\right)\Delta\dot{v} + \left[\ N_v u_0 + \left(Y_v N_r - N_v Y_r\right)\right]\ \Delta v = 0$$

$$\omega_n \approx \sqrt{N_v u_0\left(Y_v N_r - N_v Y_r\right)}$$

$$\zeta \approx \frac{1}{2\omega_n}\left(-N_r - Y_v\right)$$

$$\omega = \omega_n\sqrt{1 - \zeta^2}$$

assumptions is similar to the short period longitudinal mode in forward flight. Yaw motion and lateral velocity in the lateral-directional case correspond to pitch motion and vertical velocity in the longitudinal case.

The two coupled equations for yaw and sideslip motions are combined to obtain separate second order equations for ψ and v, which together also represent the coupled yaw and sideslip motions.

The separate ψ and v equations for coupled motion are included on Table 7.IV. The initial conditions for these equations in a single variable must be consistent with those used for the original coupled equations. The equations can be interpreted in terms of an equivalent spring mass damper system with the approximation for natural frequency and damping ratio given in Table 7.IV.

A comparison of these results with the results obtained assuming yaw motion of the helicopter pivoted at the CG shows the addition of lateral velocity changes the effective spring restraint stiffness by the increment (Y_vN_r -N_vY_r). For most single rotor helicopters, this quantity is small compared to the spring stiffness term due to the directional stability (u_0N_v). The damping of the oscillation is affected to some extent by the lateral velocity. The damping ratio for coupled LDO is proportional to the sum of the yaw moment due to yaw rate derivative (N_r) and the side force due to lateral velocity derivative (Y_v). The effect due to Y_v is the lesser effect for single rotor helicopters.

The roll to sideslip ratio for an undamped oscillation:

$$\frac{\left|\,\overline{\Delta\phi}\,\right|}{\left|\,-\overline{\Delta\beta}\,\right|} = \frac{\sqrt{(1)^2 + (2)^2}\left(\left|\,L_v u_0\right|\right)}{\sqrt{\omega_n^4 + L_p^2\omega_n^2}} \qquad \textit{eq 7.65}$$

And the phase angle ($\angle$) between the roll and sideslip:

$$\angle \frac{\Delta\phi}{(-\Delta\beta)} = \tan^{-1}\left[\frac{-(-L_p)}{-\omega_n}\right]$$

eq 7.66

Where:

(1) $$\frac{-\omega_n(-L_v u_0)}{\left(\omega_n^4 + L_p^2\omega_n^2\right)}$$

(2) $$\frac{-(-L_p\omega_n)(-L_v u_0)}{\left(\omega_n^4 + L_p^2\omega_n^2\right)}$$

β	- Sideslip angle
e	- Base of natural logarithm
ϕ	- Roll angle
i	- Imaginary index
L_p	- Roll moment due to roll rate
L_v	- Roll moment due to lateral velocity
t	- Time
u_0	- Initial velocity along x axis
ω_n	- Natural frequency.

The ratio of $\Delta\phi$ is presented relative to $(-\Delta\beta)$ rather than to $(\Delta\beta)$ because $(-\Delta\beta) \approx \Delta\psi$ tends to produce a positive roll moment via the dihedral effect. The expression for phase angle for sinusoidal motion is approximately correct for lightly damped LDO modes.

When the damping ratio is larger a better estimate for the phase angle is obtained by a similar derivatation using $\Delta\phi = \Delta\phi\,(- \omega_n + i\omega_n\,(1 - \zeta^2\,t)^{0.5})$:

$$\angle \frac{\Delta\phi}{\left(-\Delta\beta\right)} = \tan^{-1} \frac{-\left[\frac{\left(-L_p\right)}{\omega_n} - 2\zeta\right]}{-\left[1 + \zeta\frac{\left(-L_p\right)}{\omega_n}\right]}$$

eq 7.67

Where:

β	- Sideslip angle
ϕ	- Roll angle
L_p	- Roll moment due to roll rate
ω_n	- Natural frequency
ζ	- Damping ratio.

The equation given on Table 7.IV for the helicopter pivoted about the CG was obtained by neglecting the lateral velocity term and roll rate term in the yaw equation of motion. The change in response characteristics obtained when these terms are included is found using approximate expressions obtained for Δv_{NR} and $\Delta\phi$ in a neutrally stable LDO. These approximations make it possible to obtain expression for the neglected terms as linear functions of $\Delta\psi$ and $\Delta\dot{\psi}$; thus, determining how the neglected terms affect the response characteristics. The corrections have little effect on the LDO mode frequencies for typical single rotor helicopters whose tail rotors and vertical stabilizers provide adequate directional stability. However, an appreciable correction to the LDO mode damping ratio is required as a result of the roll motion:

$$\zeta = \frac{1}{2\omega_n}\left[-N_r - Y_v - \left(\frac{g}{u_0} - N_p\right)\frac{L_v u_0}{\left(\omega_n^2 + L_p^2\right)}\right]$$

eq 7.68

Where:

g	- Gravity
L_p	- Roll moment due to roll rate
L_v	- Roll moment due to lateral velocity
N_p	- Yaw moment due to roll rate
N_r	- Yaw moment due to yaw rate
u_0	- Initial velocity along x axis
ω_n	- Natural frequency
Y_v	- Side force due to lateral velocity
ζ	- Damping ratio.

The first two terms within the brackets of the above expression give the damping ratio when roll in neglected, while the third term gives the correction due to the roll response. The yaw moment due to roll rate derivative (N_p) obtained when the equations are rearranged to eliminate inertia couplings is usually a somewhat larger negative number. Thus, a positive sign is obtained for the (g/u_0 -N_p) factor in the third term. The second factor in the third term is equal to the real part or in phase part of the $\Delta\phi/(-\Delta\beta)$ ratio which is usually negative. Thus, the correction due to the roll response is negative, tending to reduce the LDO mode damping ratio. This correction is larger when the dihedral effect is increased and is reduced by an increase in roll damping.

Several approximate formulas are presented in this section as an aid to understanding the affect of various parameters on the response characteristics of the LDO mode of the single rotor helicopter. Table 7.V summarizes the LDO mode data for five unaugmented helicopters using the approximate equations. These results were computed from data in reference 7.1 for a 100 kn trim condition. Frequency and damping ratios obtained from the approximate formulas are compared with those obtained in reference 7.1 for coupled six degree of freedom analysis.

Table 7.V
LDO Mode Characteristics 100 kn Trim Speed

Condition	Parameter	Helicopter Type				
		OH-6A	BO-105	AH-1G	UH-1H	CH-53D
Yaw motion only	ω_n	3.88	2.19	1.60	2.61	1.20
	ζ	0.26	0.15	0.37	0.33	0.30
Coupled yaw and sideslip motion ($\phi = 0$)	ω_n	3.90	2.20	1.65	2.65	1.23
	ζ	0.27	0.18	0.40	0.36	0.35
Approximate equation for damping ratio including roll correction	ζ	0.24	0.17	0.39	0.33	0.26
Coupled six degree-of-freedom analysis	ω_n	3.93	2.26	1.60	2.67	1.25
	ζ	0.25	0.14	0.38	0.33	0.22
Roll to sideslip ratio	$\dfrac{\lvert\overline{\Delta\phi}\rvert}{\lvert -\overline{\Delta\beta}\rvert}$	0.41	0.67	0.13	0.40	1.37
	$\angle \dfrac{\Delta\phi}{(-\Delta\beta)}$	-149.7°	-114.9°	-186.3°	-194.7°	-158.0°

7.3.7 Control Response

The primary complication associated with the helicopter response to either lateral or pedal control inputs is the coupled motion in both roll and yaw. This is due to the control moment coupling ($N_{\delta_{LAT}}$ and $L_{\theta_{TR}}$) and the coupled dynamic response.

The other factors affecting control response are the mechanical characteristics of the flight control system including the mechanical mixing, the gearings between pilot controls and the swashplate or tail rotor, and response due to flight control system augmentation. In general, control response refers to the complete response of the helicopter to pilot lateral or pedal control inputs.

Typically, helicopters are subjected to considerable inter-axis coupling (from longitudinal to lateral-directional, not just roll to yaw). Hence, the control responses in roll or yaw is dependent on how the pilot chooses to regulate coupled longitudinal motions. The question is how much such undesired motions affect task performance and/or workload, and how dependent the answer is on pilot aggressiveness. The aggressiveness is of interest because the number of dynamic modes excited by a control input is a function of the frequency content of the excitation. This is of interest for Automatic Flight Control System (AFCS) operation both ON and OFF. The AFCS may include rate dampers and devices to cancel the coupling from the control input. Inherently included in the test pilot's task is an assessment of workload created by any requirement to use coordinated control inputs such as pedal inputs accompanying the lateral control to minimize the excitation of unwanted motions during a lateral-directional task.

The coordinated control workload is a function of the size, direction, and timing of the pedal input required to accompany the lateral cyclic to accomplish particular tasks. Associated with this consideration is how natural the coordinated inputs are to the pilot. Helicopter pilots generally are adept at using their feet in maneuvering tasks and there is a spectrum of coordinated inputs which require relatively little extra mental or physical workload, while outside this spectrum, significant workload increases occur. For example, pedal input at the same time and in the same direction as a lateral control input may feel natural. Coordinating for proverse yaw or for a delayed yaw is more difficult.

The overall response to any control input can be divided into the initial response and the longer term response. For a lateral cyclic input, the initial response is composed of the desired roll motion and the undesirable aspect, yaw moment due to lateral control ($N_{\delta_{LAT}}$, adverse or proverse). For a given $N_{\delta_{LAT}}$, the level of the undesirable motion is a function of the magnitude and abruptness of the lateral control input (δ_{LAT}). This yaw may confuse the pilot in a directional control task since the lateral input is generally used to effect a heading change or lateral pipper placement. Another undesired by product is a yaw moment due to roll rate (N_p) which occurs soon after the lateral control input, but later than the $N_{\delta_{LAT}}$. The particular combination of these two yaw motions have a significant effect on the predictability of directional response initiated by a lateral cyclic input and on the ability of the pilot to suppress these motions using pedal inputs. For pedal only turns, yaw motions may cause some undesired roll motions on top of the commanded roll via dihedral effect. The test pilot must assess the workload such distortion produces for particular piloting tasks.

The longer term response is dominated by the characteristics of the LDO mode, primarily damping ratio, and by the effects of the pilot's attempts to control residual undesired motions while performing a closed loop task. Furthermore, the spiral mode may significantly affect the steady state lateral cyclic or pedal inputs for a desired bank angle.

The discussion applies equally for helicopters with an AFCS where the specific purpose of augmentation: control shaping (modify the effective $N_{\delta_{LAT}}$ or $L_{\delta_{PED}}$) or stability augmentation (yaw damper or roll damper) is to remove the undesirable distortions which interfere with task performance. Even with augmentation, the outcome may be less than perfect. The test pilot assesses the augmentation gains/shaping as well as the effects of AFCS failure on mission performance.

7.3.8 Tandem Rotor Characteristics

7.3.8.1 STABILITY DERIVATIVES

Figure 7.26 indicates the principal effects contributing to the lateral-directional derivatives of the tandem helicopter recognized early in the development of this configuration. They are presented on the basis of the individual rotor characteristics discussed in Chapter 4.

The diagrams on Figure 7.26 are simplified by not picturing the front and rear rotors; however, the thrust vectors are shown which correspond to the TPP tilt due to disturbances from trim conditions by an incremental lateral velocity, roll rate, or yaw rate.

A lateral velocity to the left causes both the front and rear thrust vectors to tilt to the right. This tilt is another example of the stable speed response of the isolated rotor tilting away from the relative wind. The roll moment due to sideslip or effective dihedral is because the rotors are above the CG and a right tilt produces a right roll moment.

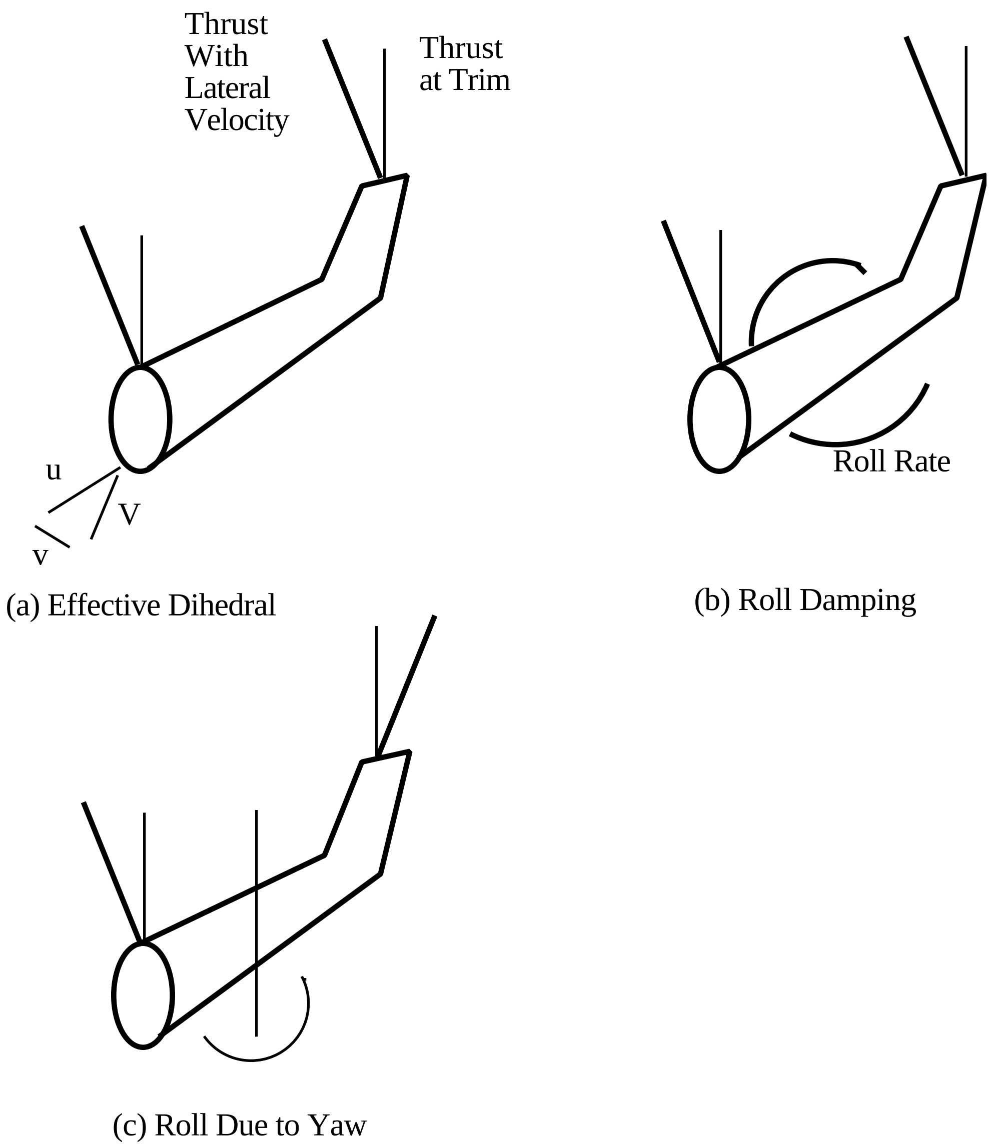


Figure 7.26
Source of Dihedral Effect, Roll Damping, and Roll Due to Yaw for the Tandem Helicopter

Aerodynamic forces on the aft pylon, fuselage, and tail of typical tandems make significant contributions to the effective dihedral. The large effective dihedral due to these non-rotor forces which occur in high speed powered flight tends to cause an LDO mode instability.

The tandem helicopter has a side force due to the lateral velocity derivative resulting from the sum of the side force contributions by the two rotors and the fuselage/tail. The lateral tilt of a rotor due to lateral velocity is proportional to the average blade angle of attack. If the CG of the tandem is forward of the midpoint between the two rotors, the forward rotor thrust is greater than the thrust of the rear rotor to trim the helicopter. Consequently, the average blade angle of attack is greater for the forward rotor than for the rear rotor; therefore, the forward rotor has a larger lateral tilt and side force than the rear rotor in response to an increment in lateral velocity. The side forces on the two rotors produce a net unstable yaw moment about the forward CG in the direction tending to rotate the tandem away from the lateral velocity.

The tail rotor of the single rotor helicopter tends to give that configuration inherent directional stability which is increased by the addition of a vertical tail located toward the rear of the tail boom where it has a long moment arm generating yaw moments. Vertical tails added to tandems are less effective because of the comparatively short moment arms from the CG, usually located near the middle of the helicopter. As a result, marginal directional stability is often a major deficiency of unaugmented tandem helicopters. This problem is accentuated in nose up autorotational flight where aerodynamic flow disturbances and interferences reduce the contributions of the aft pylon and vertical tail to directional stability.

A roll velocity of the helicopter to the left, indicated on Figure 7.26, results in a tilt of the TPPs to the right. There are small relative lateral velocities at the rotors due to roll rate about the CG and the vertical moment arms of the two rotors. These lateral velocities result in a small increment in the TPP tilt due to roll rate. If one rotor is higher than the other, this effect results in a small yaw moment due to roll rate.

The third diagram on Figure 7.26 indicates the rotor tilt produced by a yaw velocity about the CG. The tilt in the thrust vectors give thrust components opposing the yaw motion and are responsible for the yaw damping.

If the rear rotor is higher than the front rotor, as indicated in Figure 7.26, the lateral component of the rear rotor thrust due to yaw rate produces a greater roll moment about the CG than the front rotor, and a net roll moment due to yaw velocity results. The aerodynamic forces due to yaw rate acting on the aft pylon and vertical surfaces mounted above the CG also make a positive contribution to the roll moment due to yaw rate derivative.

Roll control of the tandem helicopter is achieved by applying lateral cyclic pitch to the front and rear rotors and is analogous to the single rotor helicopter. The resulting tilt of the forward and aft TPP introduce rotor side forces and offset hinge roll moments.

Yaw control is accomplished by differential lateral cyclic pitch introduced by pedal inputs. The tandem helicopter can employ various combinations of lateral cyclic and differential cyclic of the two rotors to make turns about any desired turning axis. However, if the resulting motion causes different lateral velocities at the forward and aft rotors, a thrust difference can develop between the rotors causing cross coupling with the helicopter longitudinal responses.

7.3.8.2 DYNAMIC RESPONSE MODES OF THE UNAUGMENTED TANDEM

In hovering flight there is little coupling between yaw and lateral motions of the tandem helicopter. The yaw rate response of the hovering tandem is more sluggish than the single rotor helicopter both because of the tandem's large yaw moment of inertia and the low yaw damping provided by the thrust vector tilt of the forward and aft rotors in response to their lateral velocities generated by yaw rate.

The spiral mode is usually a slowly convergent mode for the unaugmented tandem helicopter. The tandem helicopter is usually free of spiral instability because of its low directional stability. Consequently, when the helicopter is disturbed in bank angle, the predominant effect of any sideslip angle which develops in the subsequent motion is to produce dihedral effect roll moments tending to bring the bank angle to the wing's level trim condition rather than producing yaw moments leading to an ever steepening inward spiraling flight path.

The roll rate response mode of the tandem is quite similar to the single rotor helicopter. Coupling effects are small and roll motion predominates in this first order subsidence mode in both hovering and forward flight. Tandem helicopters tend to have somewhat longer roll rate response times than single rotor helicopters with similar rotor configurations because of their larger roll moments of inertia.

The LDO mode of the tandem helicopter in hovering flight generally involves little yaw motion and is primarily coupled lateral translation and roll motion. It is analogous to the longitudinal oscillatory mode of the hovering single rotor helicopter with lateral velocity replacing longitudinal velocity and roll angle replacing pitch angle.

In the LDO a roll angle gives a lateral component of thrust tending to cause lateral motion of the helicopter. The rotor blowback in response to a right translational velocity gives a left roll moment which is approximately balanced by a right roll damping moment due to the rotor response to the left roll rate. Thus when the helicopter has a right lateral displacement, the corresponding left roll angle and left tilt of the thrust vectors provides a restoring force tending to bring it back to the equilibrium position. This effective restoring force is the source of the LDO mode in hovering flight. A more exact consideration of the requirements for roll moment equilibrium shows the moment due to rate of change of roll rate causes a phase shift in the roll response. As a result, the horizontal thrust force due to roll angle is given a component in phase with the lateral translational velocity of the helicopter which produces negative damping of the motion. This is the reason for the divergent tendency of the LDO of both the tandem and single rotor helicopter in hovering flight and results in a negative damping ratio of approximately:

$$\zeta = \left(-\frac{1}{2}\right)\left(\sqrt{\frac{-gL_v}{\left|L_p\right|^3}}\right)$$

eq 7.69

Where:

- g - Gravity
- L_p - Roll moment due to roll rate
- L_v - Roll moment due to lateral velocity
- ζ - Damping ratio.

The character of the LDO mode of the single rotor helicopter changes significantly going into forward flight because of coupling with yaw motion. The yaw motion becomes dominant for most single rotor helicopters and the frequency of the LDO for this configuration is primarily determined by the yaw stiffness provided by the directional stability derivative.

The LDO mode of the tandem helicopter is changed less than the single rotor helicopter going into forward flight because of the small directional stability derivative. The coupling of yaw motion remains comparatively small. The phase shift due to roll moments are still destabilizing. However, in low speed forward flight, aerodynamic forces on the rotor pylons and fixed surface add to the roll damping derivative and tend to stabilize the motion.

The small yaw motion which is present in the LDO mode of the tandem can contribute to the instability of the mode. The destabilizing effect on this oscillatory mode is dependent on the directional stability derivative. A more divergent LDO mode is found in high speed flight where the tandem's directional stability is usually negative. This is the most significant deficiency of the lateral-directional handling qualities of the unaugmented tandem helicopter. Although negative directional stability increases the instability of the LDO mode, it does not generally result in an aperiodic divergence. The yaw motion is a secondary effect in this mode and the effective stiffness of the tandem's LDO mode is largely determined by the dihedral mechanism rather than by the directional stability.

7.4 TEST METHODS AND TECHNIQUES

7.4.1 Trimmed Control Positions

The purpose of this test is to evaluate the variation in control positions with changes in power, airspeed, and bank angle; control margins; the ease of trimming the flight control forces to zero; and the qualitative importance of trim changes to the piloting task. The entire airspeed envelope is evaluated to V_{NE} and the complete power range is evaluated from autorotation to maximum power. For level flight, the helicopter is established in trimmed, wings level, unaccelerated flight, and the data are recorded. The airspeed is varied approximately 10 kn and the test is repeated.

For climbs and descents, one or more mission representative airspeeds are used. The helicopter is stabilized on airspeed and the power is varied from autorotation to maximum available. Airspeed remains constant while the power is varied incrementally.

At least 5 power increments are used. The climb and descent test is repeated at another airspeed. Usually, the airspeed for maximum rate of climb, or minimum rate of descent is evaluated. If large control displacements are observed while making a power change, the test can be conducted open loop with all controls fixed except for power. The resulting attitude and rate changes show the helicopter response which the pilot must counteract to maintain wings level, unaccelerated flight conditions. The results of the open loop test are used to relate the possible helicopter excursions when performing constant airspeed climbs and descents.

"If large excursions are seen in the pedal positions while making power changes especially in full power climbs and minimum power descents, turns at these conditions should be considered to determine if any additional margin of control is required. For example, in US built helos, low power descents require right pedal and a right turn at this condition may require more right pedal. The reverse would be true in high power climb."

7.4.1.1 TEST TECHNIQUE

1. Establish wings level, ball centered, unaccelerated, trimmed flight. Record the initial trim conditions.
2. Assign a Handling Qualities Rating (HQR) and Vibration Assessment Rating (VAR) to each data point.
3. Use the force trim and AFCS in a mission representative manner.
4. Gather data in steady rectilinear flight, varying airspeed; steady turning flight, varying bank angle while maintaining altitude and airspeed; and climb and descent, varying power while maintaining constant airspeed.

7.4.1.1.1 Level Flight

1. This data can be obtained concurrently with level flight performance testing.
2. Investigate entire airspeed profile (30 kn to V_H).

7.4.1.1.2 Diving Flight

1. If V_{NE} is greater than V_H, fix the collective at maximum power and dive the aircraft to obtain control position data at the higher airspeeds.
2. Control margins are particularly important.

7.4.1.1.3 Steady Turns

1. Establish wings level, ball centered, unaccelerated, trimmed flight at a mission representative airspeed. Record the initial trim conditions.
2. Vary roll attitude, establishing trimmed incremental bank angles both left and right. Adjust collective as necessary to maintain altitude and airspeed.
3. Incrementally increase bank angle to the test limit.
4. Repeat the bank angle series at another mission representative airspeed as desired.

7.4.1.1.4 Climbs and Descents (Closed Loop)

1. Establish wings level, unaccelerated flight at the desired trim airspeed with collective control position at incremental power settings.
2. Vary power incrementally from autorotation to the maximum available.
3. Normally five increments define a satisfactory curve.
4. Altitude range, ±1000 ft about target.
5. If near a margin in high power climb or auto, check that a turn doesn't decrease the margin.

7.4.1.2 DATA REQUIRED

Control positions, V_o, H_{P_o}, T_o, V_V (climb and descent) θ, ϕ, β, Q, N_R, fuel counts (FC), HQR, and VAR.

7.4.1.3 TEST CRITERIA

1. Stabilized, wings level (level flight), ball centered flight.
2. No vertical velocity (level flight).
3. Constant vertical velocity (climb and descent).
4. All control forces trimmed to zero.

7.4.1.4 DATA REQUIREMENTS

1. Stabilize 15 s prior to recording data.
2. Record 10 s data.
3. H_{P_o} ± 1000 ft of the target test altitude.

4. V_v level flight, ±10 fpm; climb and descent, stabilized ± 25 fpm.
5. N_R ± 1%.
6. V_o ± 1 kn.
7. $\phi \pm 2^\circ$.

7.4.1.5 SAFETY CONSIDERATIONS/RISK MANAGEMENT

Observe normal operating limitations, including acceleration and bank angle limitations. Use incremental build up procedures when approaching any envelope boundary. Conduct tests at a safe operating altitude over terrain suitable for forced landings. Maintain proper lookout procedures. Consider using a chase aircraft and sole use airspace.

7.4.2 Static Lateral-Directional Stability (SHSS)

The purpose of this test is to evaluate the static lateral-directional stability or the change in the total lateral-directional restoring moments generated by off trim lateral velocity (sideslip angles). Static stability is indicated by the initial tendency of a helicopter to return to or depart from equilibrium if disturbed. Positive static stability is indicated by the tendency to return toward the original trimmed condition.

Directional static stability is indicated by the variation of pedal control with sideslip. Positive static directional stability is indicated by increasing left pedal control displacement with increasing right sideslip. Since the forces and moments can not be measured directly in flight, the purpose of this test is to determine the control forces and positions necessary to balance the directional moments caused by sideslip variations from the inherent trim sideslip. Cockpit control position versus sideslip angle indicates the sign but not the magnitude of static stability since the slope in proportional to the yaw moment due to pedal control, which in turn depends on the mechanical ratio of tail rotor pitch angle to pedal control.

Lateral static stability (effective dihedral) is indicated by the variation of lateral control with sideslip. Positive effective dihedral is indicated by increasing right lateral control displacement with increasing right sideslip. Since the forces and moments can not be measured directly in flight, the purpose of this test is to determine the control forces and positions necessary to balance the roll moments caused by sideslip variations from the inherent trim sideslip. Cockpit control position versus sideslip angle indicates the sign but not the magnitude of static stability since the slope is proportional to the roll moment due to lateral control, which in turn depends on the mechanical ratio of the lateral cyclic pitch angle to lateral control.

Side force is indicated by the variation in bank angle with sideslip. The lateral component of the gravity vector provides a cue to the pilot that he is in out of balanced flight. The sideslip angle and the corresponding bank angle at which the pilot proprioceptively recognizes the out of balanced flight condition is a measure of the strength of the side force cue. Ideally, a small sideslip angle produces a side force cue to the pilot that he is in out of balanced flight. Conclusions are drawn on the aircraft characteristics of sideforce and not sideforce cues to the pilot.

The test is accomplished by establishing a wings level, ball centered, unaccelerated trim condition. Without changing the collective position, trim, or rotor speed, the sideslip is varied and stabilized at incremental sideslips. The sideslip around trim is most important. Small 2 to 3˚ increments are used around trim. As sideslip increases from trim, larger 5˚ increments are used. At each stabilized flight condition, lateral and directional controls are used to maintain steady, non-turning flight. The longitudinal control is used to maintain constant boom airspeed. Collective is maintained constant. A stabilized climb or descent is accepted and altitude is maintained within ± 1000 ft of the test altitude. If power must be changed to return to the test altitude, record all engine parameters and collective control position to assist in returning to trim. Do not retrim the collective. When the test altitude is reestablished, return the collective control to trim.

7.4.2.1 TEST TECHNIQUE

1. Stabilize at the initial trim airspeed in ball-centered, wings level, unaccelerated flight and reduce all control forces to zero. Do not retrim control forces or adjust engine controls during the test.
2. Record trim conditions.
3. Smoothly yaw the helicopter to the desired sideslip angle for the next data point while simultaneously adding lateral cyclic to prevent the helicopter from turning. Generally 5˚ sideslip variations are acceptable; however, around the inherent trim sideslip, use 2 to 3˚ sideslip variations. Use longitudinal control to maintain trim airspeed as

indicated by the boom airspeed. Selecting a prominent landmark along the original flight path is helpful in assuring a straight flight path.

4. Stabilize at each sideslip increment for 15 s and record data for approximately 10 s. Make qualitative comments on the control force, control displacement, and side force (bank angle) cues at the off trim conditions. Note the sideslip angle at which the pilot proprioceptively is aware of the side force and out of balanced flight condition.

5. Collect the data at incremental sideslip angles both left and right covering the allowable range. Obtain data at zero bank angle and ball centered, and zero sideslip angle.

6. Good in flight visibility with a defined horizon and long straight line land features decrease the pilot effort required to conduct this test. Strong winds at the test altitude make the task more difficult.

7. If it is necessary to adjust altitude to remain within the test band, note the collective position, torque, and N_g, before moving the collective. Do not retrim any control forces. The test is continued after the collective control is returned to the original trim position.

8. Determine if airspeed errors need to be accounted for by yawing to a given sideslip and observing airspeed. If a change is noted, yaw the aircraft back to trim. If the airspeed returns to trim, fly the off airspeed when in a slip as you have an airspeed error in sideslip. If the airspeed did not return to trim airspeed when sideslip was removed, you must make adjustments in longitudinal control in the face of sideslip to hold the original airspeed.

7.4.2.2 DATA REQUIRED

Control positions, control force, V_o, V_v, ϕ, β, θ, Q, N_R, FC, H_{P_o}, T_o.

7.4.2.3 TEST CRITERIA

1. Wings level, ball centered, unaccelerated flight at trim.
2. Constant heading, no turning.
3. Collective fixed.
4. No retrimming.

7.4.2.4 DATA REQUIREMENTS

1. Stabilize 15 s prior to taking data.
2. Record 10 s stabilized data.
3. $V_o \pm 1$ kn.
4. Stabilized sideslip, $\beta \pm 0.5^\circ$.
5. Steady heading, $\psi \pm 1^\circ$.
6. $H_{P_o} \pm 1000$ ft of the target test altitude.
7. $N_R \pm 0.5\%$.

7.4.2.5 SAFETY CONSIDERATIONS/RISK MANAGEMENT

Observe normal operating limitations. Use incremental build up procedures when approaching any envelope boundary. Maintain proper lookout procedures. Consider using a chase aircraft and using sole use airspace. In most cases the sideslip envelope for the helicopter as a function of calibrated airspeed must be obtained from the airframe manufacturer or other competent authority (Figure 7.27). These limits are normally not published in documents readily available to the test team. A well written safety of flight release may be the source for this data.

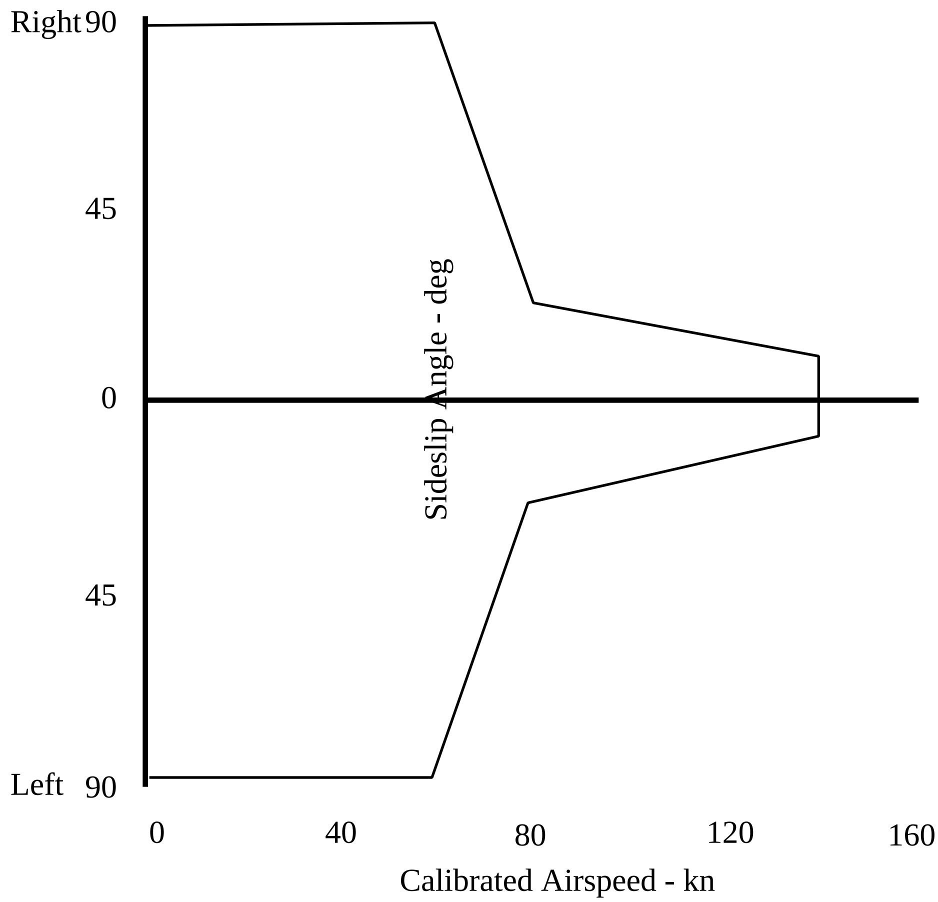


Figure 7.27
Example Sideslip Envelope

7.4.3 Turns On One Control (TOOC)

7.4.3.1 CYCLIC ONLY TURNS

Lateral control only turns provide an evaluation of the yaw due to lateral control, adverse/proverse yaw characteristics. As the lateral control is applied, observe the turn needle, heading indicator, sideslip indicator, and outside the cockpit. If the helicopter has adverse yaw, the indications show an initial turn in the opposite direction to the lateral control. If the adverse yaw is weak, a delay in turn rate may be observed, rather than a turn in the opposite direction. If the helicopter has proverse yaw, the initial turn rate in the direction of lateral control is greater than the sustained turn rate.

Lateral control only turns reveal the relative magnitude of the directional static stability. Assuming adverse/proverse yaw is present, if the sideslip angle during the steady roll remains at the inherent sideslip or quickly returns toward the inherent sideslip angle, the helicopter exhibits relatively strong directional stability. If the sideslip angle does not return toward the inherent sideslip, the directional stability is relatively weak.

Lateral control only turns provide a secondary indication of the spiral stability. Once a stabilized bank angle is established, the lateral control deflection from trim provides an indication of the character of the spiral mode. If the control is displaced into the bank angle, the spiral is positive or convergent. If the control is displaced away from the bank angle, the spiral mode is negative or divergent.

7.4.3.1.1 Test Technique

1. Trim the helicopter at desired airspeed in ball-centered, wings level, unaccelerated flight. Record the trim conditions including inherent sideslip angle.
2. Smoothly apply lateral cyclic to establish the desired roll rate and ultimately the roll attitude. Do not move directional controls.
3. Observe the turn needle, heading indicator, sideslip indicator, and outside the cockpit while displacing the lateral cyclic.
4. Observe the turn needle, heading indicator, sideslip indicator, and outside the cockpit when the bank angle and turn rate are steady.
5. Repeat for both left and right turns at desired bank angles.
6. Repeat for faster and slower rate of control application in an attempt to generate sideslip.
7. Assign an HQR for capturing and maintaining bank angle or turn rate.

7.4.3.1.2 Data Required

Control positions, initial and final lateral control position, V_o, V_v, ϕ, β, θ, Q, N_R, FC, H_{P_o}, T_o, HQR.

7.4.3.1.3 Test Criteria

1. Wings level, ball centered, unaccelerated flight at trim.
2. Collective and pedal control fixed.
3. No retrimming.

7.4.3.1.4 Data Requirements

1. Stabilize 15 s at trim.
2. Record 10 s stabilized data.
3. $V_o \pm 1$ kn.
4. Stabilized bank angle, $\phi \pm 1^\circ$.
5. $H_{P_o} \pm 1000$ ft of the target test altitude.
6. $N_R \pm 0.5\%$.

7.4.3.1.5 Safety Considerations/Risk Management

Lateral cyclic only turn tests are generally low risk. Observe normal flight test precautions and operating limits. No additional considerations peculiar to this test method are required.

7.4.3.2 PEDAL ONLY TURNS

Directional control only turns reveal the relative magnitude of the effective dihedral. If a relatively small sideslip angle results in a large roll rate in the opposite direction, the helicopter exhibits relatively strong effective dihedral. If a large sideslip angle is required to generate a roll in the opposite direction, the helicopter exhibits relatively weak positive effective dihedral. If a sideslip angle results in a roll in the direction of the sideslip, the helicopter exhibits negative effective dihedral.

7.4.3.2.1 Test Technique

1. Trim the helicopter at the desired airspeed in ball-centered, wings level, unaccelerated flight. Record trim conditions.
2. Smoothly apply directional control to change sideslip angle. Do not move the lateral control.
3. Observe the roll response during application of directional control.
4. When bank angle and turn rate are steady, record the directional control displacement and sideslip angle.
5. Repeat for both left and right directional control displacements and desired bank angles.

7.4.3.2.2 Data Required

Control positions, initial and final pedal control position, V_o, V_v, ϕ, β, θ, Q, N_R, FC, H_{P_o}, T_o, HQR.

7.4.3.2.3 Test Criteria

1. Wings level, ball centered, unaccelerated flight at trim.
2. Collective and lateral control fixed.
3. No retrimming.

7.4.3.2.4 Data Requirements

1. Stabilize 15 s prior to taking data.
2. Record 10 s stabilized data.
3. $V_o \pm 1$ kn.
4. Stabilized bank angle, $\phi \pm 1^\circ$.
5. $H_{P_o} \pm 1000$ ft of the target test altitude.
6. $N_R \pm 0.5\%$.

7.4.3.2.5 Safety Considerations/Risk Management

Directional only turn tests are generally low risk. Observe normal flight test precautions and operating limits. No additional considerations peculiar to this test method are required.

7.4.4 Spiral Stability

The purpose of this test is to determine the helicopter response to bank angle deviations from trim.

7.4.4.1 TEST TECHNIQUE

During turns on one control tests, a look into the spiral was accomplished. For cyclic only turns, the position of the control when the angle of bank was steady was an indication of the spiral mode. Likewise, the pedal position during pedal only turns when the bank angle was steady, also gave an indication of the spiral. The following test will quantify the spiral mode.

1. Trim the helicopter in wings level, steady heading, coordinated flight at the desired test conditions.
2. Use lateral cyclic to establish the desired bank angle, smoothly return the lateral cyclic to trim, and observe the helicopter response. Since the spiral mode is usually weak, the response is easily contaminated if the controls are not returned exactly to trim.
3. Use a range of bank angles from 5˚ to approximately 20˚ unless significantly larger bank angles are routinely experienced in accomplishing the mission of the helicopter.
4. If the effective dihedral of the helicopter is sufficient to achieve the desired bank angle, the alternate method uses directional controls to develop a sideslip angle. Hold sideslip until the helicopter rolls to the desired bank angle. For tail rotor helicopters, the rate of directional control application should not create a roll moment due to change in tail rotor thrust. Lateral control is fixed throughout the evaluation at the initial trim position. Since the spiral mode is usually weak, the response is easily contaminated if the controls are not returned exactly to trim.
5. Record a time history of the response or record the bank angle at selected time intervals to generate a plot of bank angle as a function of time.
6. Repeat steps 1 through 7 as required to evaluate increasingly larger bank angles both left and right.

7.4.4.2 DATA REQUIRED

Control positions, initial and final lateral or directional control positions, V_o, V_v, ϕ, β, θ, Q, N_R, FC, H_{P_o}, T_o.

Hand recorded or automatic time history of bank angle versus time.

7.4.4.3 TEST CRITERIA

1. Wings level, ball centered, unaccelerated flight at trim.
2. Controls fixed except for the excitation control.
3. Excitation control returned exactly to trim.
4. No retrimming.

7.4.4.4 DATA REQUIREMENTS

1. Stabilize 15 s prior to returning excitation control to trim.
2. $V_o \pm 1$ kn.
3. $H_{P_o} \pm 1000$ ft of the target test altitude.
4. $N_R \pm 0.5\%$.

7.4.4.5 SAFETY CONSIDERATIONS/RISK MANAGEMENT

Spiral stability tests are generally low risk. Observe normal flight test precautions and operating limits. No additional considerations peculiar to this test method are required.

7.4.5 Lateral-Directional Oscillation

The purpose of this test is to document the variation of bank angle and sideslip during the LDO. The LDO is a nuisance mode of motion and when excited is suppressed by the pilot or AFCS. Tests include a qualitative determination of the ease of exciting and the difficulty of suppressing the LDO. The test basically consists of recording the transient responses of the helicopter resulting from a disturbance from a trim introduced by turbulence or a pilot input.

7.4.5.1 EXCITATION METHODS

1. No conscious pilot inputs. Imperfect trim conditions or atmospheric disturbance may excite an aperiodic divergent mode or lightly damped oscillatory mode. If no significant response is obtained, it could indicate a desirable helicopter characteristic particularly for flight in turbulence.
2. Pilot excitation.

   a. Release from steady-heading sideslip. This is generally the preferred technique because it closely resembles conditions encountered in flight. The technique requires that the cyclic and directional controls are simultaneously returned to the trim conditions. Control system mechanical characteristics often complicate this requirement. These difficulties can be overcome with use of control fixtures.

b. Release from pedal driven LDO. After trimming the helicopter at the desired conditions, drive the helicopter in yaw with a sinusoidal pedal input. Identify the damped frequency of the yaw oscillation. After exciting the yaw oscillation, stop the sinusoidal input with the directional control at the trim position and observe (record) the helicopter response.

c. Pedal doublet. At the frequency determined from pedal driven LDO, apply a doublet and observe the helicopter response after the input is stopped.

d. Lateral cyclic or collective pulse. Secondary test techniques used to create a transient sideslip angle include a lateral cyclic pulse or a collective pulse input. The lateral cyclic pulse technique is applicable where the adverse/proverse yaw characteristics are appreciable and AFCS provides no heading hold or coordinated turn feature. The collective pulse technique is applicable for tail rotor configured helicopter.

7.4.5.2 TEST TECHNIQUE

1. Stabilize at the desired trim airspeed and reduce all control forces to zero. Subsequently, do not retrim control forces. Assure you can return and maintain the controls at the trim position after exciting the response.
2. Record trim condition.
3. First determine if a LDO results from a natural disturbance. With the controls either fixed or free, note the open loop helicopter response. If no helicopter response is observed, use an artificial excitation.
4. Excite the helicopter response using one of the artificial methods.
5. If the helicopter is operationally flown hands off, obtain controls free responses following excitation. Controls are released at trim so the subsequent response indicates the effect of control motions.
6. Record the resulting motion using cockpit data and automatic recording system.
7. The effects of various degraded AFCS modes should be considered and investigated if warranted.

7.4.5.3 DATA REQUIRED

β, ϕ, θ, V_o, H_{P_o}, V_V, FC, T_o, Q, N_R, and cockpit control positions.

Time history of the helicopter response to excitation. Attitudes (θ, ϕ, and β), rates (q, p, and r), and control positions. Estimate the ϕ/β ratio, period, and damping.

7.4.5.4 TEST CRITERIA

1. Ball centered, balanced flight at trim.
2. Longitudinal, lateral, and pedal controls fixed (at trim) and/or free.
3. Collective fixed.

7.4.5.5 DATA REQUIREMENTS

1. Trim $V_o \pm 2$ kn.
2. $H_{P_o} \pm 1000$ ft of the target test altitude.
3. $N_R \pm 0.5\%$.

7.4.5.6 SAFETY CONSIDERATIONS/RISK MANAGEMENT

Observe normal operating limitations. Knowledge of airspeed and attitude limitations is essential. Initiate recovery in sufficient time to prevent exceeding any limits. Maintain proper lookout procedures. Consider using a chase aircraft, sole use airspace, and wearing parachutes.

Sideslip limits for the particular test airspeed can be exceeded. Degraded AFCS functions, if investigated, can exhibit surprising oscillatory responses and must be approached cautiously.

7.4.6 Control Response

Static and maneuvering stability tests are primarily concerned with evaluating the stability characteristics which keep the helicopter at a trim condition and the cues provided to the pilot of an off-trim condition. Control response testing involves evaluating the helicopter response generated from cyclic, pedal, and collective inputs. Response to control input depends on mechanical characteristics, static stability, dynamic stability, and the control characteristics of the helicopter. The tests are conducted by applying step control inputs of increasing magnitude against a fixture and recording a time history of the helicopter response.

7.4.6.1 TEST TECHNIQUE

1. Stabilize in wings level, coordinated flight at the desired trim conditions. The collective should be held constant during control response tests.
2. Set the control fixture for the desired displacement magnitude, direction, and axis. The control input direction and size is verbally and visually (if able) verified between the pilot and copilot/engineer. Start with small displacements.
3. The voice procedures are: “Data ON, standby for a (displacement magnitude), (direction of input), (control) input on three; thousand one, thousand two, thousand three.” For example: “Data ON, standby for a one inch, right lateral input on three; thousand one, thousand two, thousand three.”
4. An automatic data recording system is required to record accurately helicopter responses and is activated prior to the countdown to record the initial trim conditions.
5. After the count down, a step input is made against the fixture. The input control is held rigidly against the fixture while maintaining all other controls fixed. When making cyclic inputs, a crisp quick input is obtained using only wrist action and not the entire arm. When making pedal inputs, relax the other foot so the input is made by one foot only.
6. Do not attempt to decouple responses in other axes. The response lasts only for a couple of seconds and attempts to decouple usually result in contaminating the response.
7. Hold the input control fixed and make recovery when a steady state rate is obtained, or a predetermined flight limit is reached. Recover when you have the data you need or a limit is approached.
8. Restabilize at the trim conditions and repeat the procedure until the desired range of control input is achieved or a predetermined flight limit is reached.
9. Use incremental buildup in step input size. Do not overpower the person holding your fixture.
10. Perform appropriate lateral-directional tasks for the test helicopter's mission to assess the mission suitability of the control response. Test control inputs outlined in the

previous steps document control response characteristics. However, the qualitative evaluation of mission tasks produces the bottom line whether the control responses are enhancing or detracting to mission performance and corresponding pilot workload.

7.4.6.2 DATA REQUIRED

V_o, H_{P_o}, FC, T_o, Q, N_R, cockpit control positions, control input size and direction.

Automatic recording systems are required for data collection. A time history of the control position, attitude (ϕ, ψ, and β), rate (p and r), and acceleration ($\dot{p}$ and $\dot{r}$) are the minimum essential data elements.

7.4.6.3 TEST CRITERIA

1. Wings level, ball centered, balanced flight at trim.
2. Collective and off axis controls fixed.

7.4.6.4 DATA REQUIREMENTS

1. Stabilize 15 s at trim prior to input.
2. Trim $V_o \pm 2$ kn.
3. $H_{P_o} \pm 1000$ ft of the target test altitude.
4. $N_R \pm 0.5\%$.

7.4.6.5 SAFETY CONSIDERATIONS/RISK MANAGEMENT

Observe normal operating limitations. Maintain proper lookout doctrine. Maintain situational awareness. Consider using a chase aircraft, sole use airspace, and wearing a parachute.

A thorough knowledge of airspeed and attitude limitations is essential. Initiate recovery in sufficient time to prevent exceeding a test limit. Use a control fixture to obtain consistent inputs. Brief and practice use of the control fixture. Brief and rehearse recovery procedures.

Aircraft attitude limits could be exceeded during these tests. Exercise care when approaching limit conditions. Cautious, incremental buildup is important.

7.4.7 Gust Response

The lateral-directional gust response is assessed in terms of effect on tasks performance and ride quality which may in turn contribute to crew fatigue or distraction. Pertinent factors may include the initial motions induced by the gusts, the degree of residual upset requiring pilot recovery, and the induced dynamic motions the pilot might have to subdue.

The initial upset likely triggers one or more of the dynamic modes of motion. Explicit testing for the dynamic modes is conducted under carefully controlled conditions using test inputs designed to excite dynamic modes selectively to permit documentation. However, in the majority of operational circumstances, turbulence may excite more dynamic modes than selective test inputs. Hence, the pilot may contend with both short term and long term gust motions. In both cases, since the gust induced motions are a hindrance to a particular piloting task, the time for the disturbance to reach negligible levels is the pilot's key concern. Adequate total damping in any motion is the key factor. The quicker the undesired motions disappear, the lower the requirement for pilot intervention, and the lower the pilot's workload. However, there are tasks when the actual number of overshoots in the undesired motions becomes a dominant factor. The relative importance of short term versus longer term gust induced motions is primarily determined by the specific nature of a given piloting task.

Because of the simultaneous mode excitation produced by turbulence, the subsequent interaction between longitudinal and lateral-directional disturbances may result in a residual upset from the original trim from which the pilot must recover after the oscillatory dynamic modes are suppressed.

7.4.7.1 TEST TECHNIQUE

1. Record flight through natural turbulence at various speeds with various AFCS modes activated. With sufficient control response test results, the input size for the artificial gust input is determined to be that input size which will generate 0.2 radian/sec pitch rate within 2 seconds, or the input size which will develop a normal acceleration of 1.5 g within 3 seconds, or 1 inch, whichever is less.

2. If natural turbulence is not available, use pulse lateral, directional or collective inputs. Generally a one inch pulse input held for one-half second is considered acceptable. Use caution and buildup since a one inch control deflection for one half second could provide excessive response. Following the pulse, maintain controls fixed and record the resulting helicopter response.

7.4.7.2 DATA REQUIRED

V_o, H_{P_o}, V_v, θ, ϕ, β, FC, T_o, Q, N_R, cockpit control positions.

Following the excitation, record airspeed, altitude, attitudes, rates, and CG normal acceleration as a function of time. Automatic recording systems are required to obtain meaningful test data. AFCS actuator time histories indicate the system response characteristics.

7.4.7.3 TEST CRITERIA

1. Wings level, ball centered, balanced flight at trim.
2. Collective fixed.
3. Other cockpit controls fixed and/or free.

7.4.7.4 DATA REQUIREMENTS

1. Controls fixed for natural excitation.
2. For pulse excitation:
    - a. Trim $V_o \pm 2$ kn.
    - b. $H_{P_o} \pm 1000$ ft of the target test altitude.
    - c. $\phi \pm 3$ degrees bank angle.
    - d. Pulse duration 0.5 s ± 0.1 s.

7.4.7.5 SAFETY CONSIDERATIONS/RISK MANAGEMENT

If these tests are conducted well within an established flight envelope, consider normal operating limitations. Maintain situational awareness and proper lookout procedures. This is particularly important when control fixtures are involved. Consider using a chase, sole use airspace, and wearing a parachute. A thorough knowledge of airspeed and attitude limitations is essential. Initiate recovery in sufficient time to prevent exceeding limits. Brief and practice use of the control fixture.

Approach flight into known turbulence cautiously. Comparison with known chase aircraft gust response characteristics may be of some benefit. Consider saturation of AFCS and AFCS capabilities to deal with the turbulence.

7.5 DATA REDUCTION

Data reduction requires the application of instrument corrections to the hand recorded data. Airspeed and altitude data are corrected for both instrument and position errors. Automatic data recording systems normally include total instrumentation system corrections in the process used to change a flight tape (or telemetry) raw data into engineering units. Most quantitative information required for lateral-directional flying qualities tests is read from time histories of selected data parameters. The specific data elements as well as general definitions are included in each test technique section.

Specific data reduction procedures are discussed for the two most common types of dynamic time history responses. Roll control response time histories may resemble a classical first order response, while the LDO is typically a lightly damped second order response. These two procedures encompass most of the commonly expected test results. Non-classical responses may require characterization by other data reduction techniques.

7.5.1 First Order Systems

A high quality time history of the response of interest is required to determine the first order time constant (τ). Additional data are read directly from the time history. The time constant is usually used to describe a first order convergent response as shown in Figure 7.28.

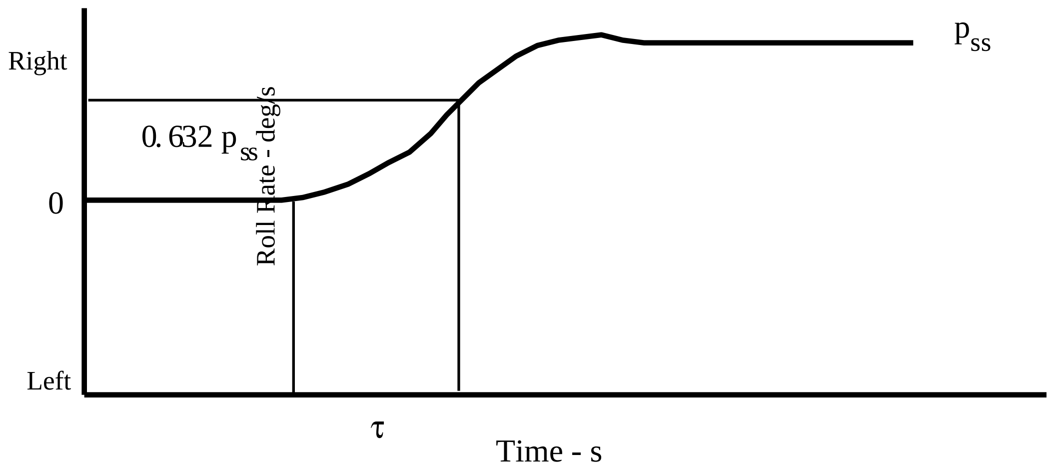


Figure 7.28
First Order Response

Using a time history of a first order convergent exponential response such as the example shown in Figure 7.29, measure the indicted values.

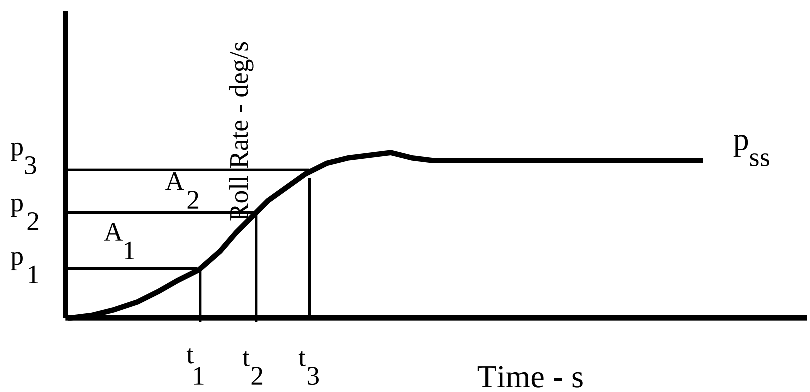


Figure 7.29
Computation Technique for τ

The following equations define the values required to calculate the time constant which describes this type response:

$$\Delta t = t_2 - t_1 = t_3 - t_2 \qquad \textit{eq 7.70}$$

$$A_1 = p_2 - p_1 \qquad \textit{eq 7.71}$$

$$A_2 = p_3 - p_2 \qquad \textit{eq 7.72}$$

$$\tau = \frac{\Delta t}{\ln_e\left(\frac{A_1}{A_2}\right)} \qquad \textit{eq 7.73}$$

Where:

A	- Rate response value
$\ln_e$	- Natural logarithm
p	- Roll rate
τ	- First order response time constant
t	- Time.

Another method of calculating τ involves the use of semi-log graph paper. Using a high quality rate response construct and measure the values shown in Figure 7.30 (a). Plot these values on semi-log paper as shown in the example in Figure 7.30 (b).

Measure the values depicted in Figure 7.30 (b) and use in the following equation to calculate τ:

$$\tau = \frac{\left(t_4 - t_2\right)}{\ln_e\left(\frac{\Delta p_A}{\Delta p_B}\right)} \qquad \textit{eq 7.74}$$

Where:

$\ln_e$	- Natural logarithm
p	- Roll rate
τ	- First order response time constant
t	- Time.

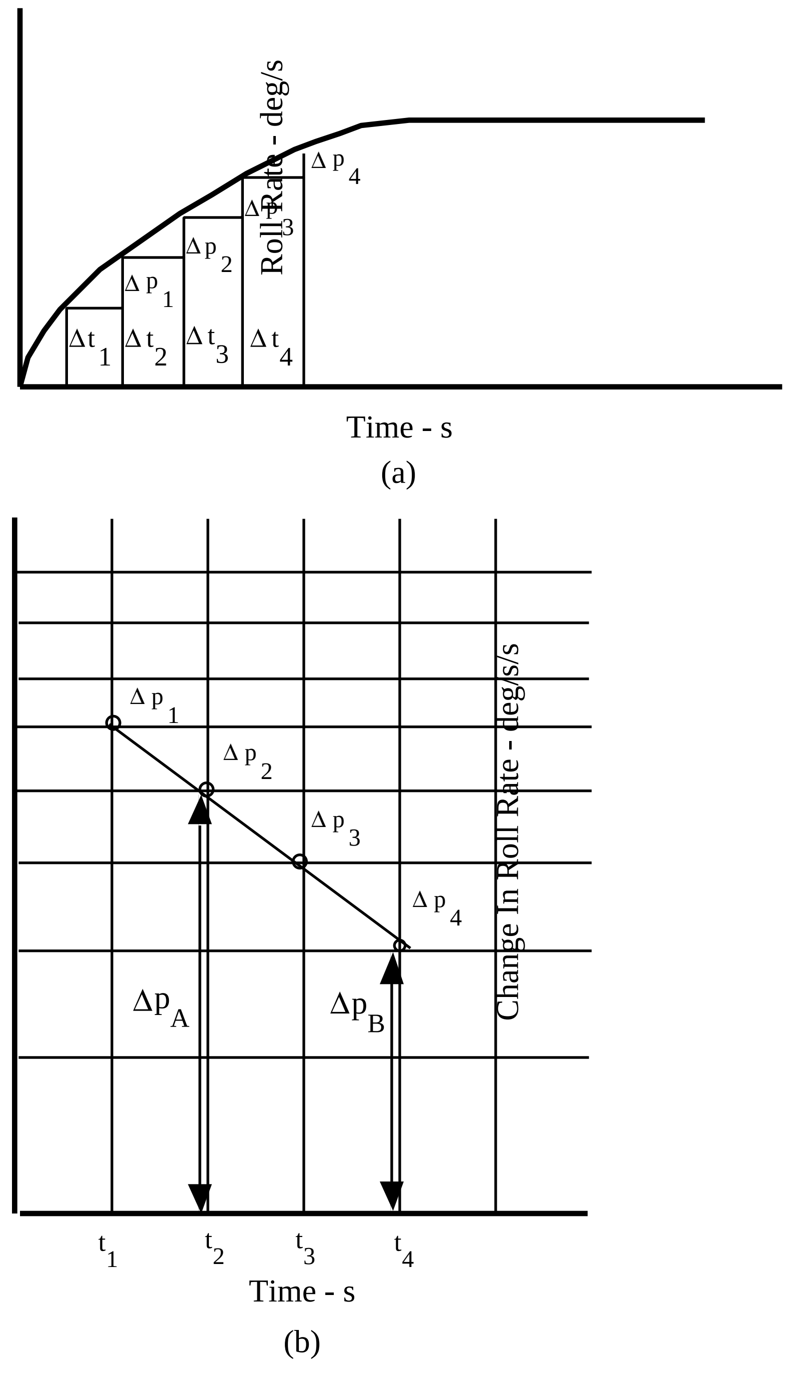


Figure 7.30
Graphic Technique for Determining τ

7.5.2 Second Order Systems

High quality traces of the rate and/or attitude responses are required to determine the second order characteristic values. All forcing functions must be zero. Second order responses commonly observed in lateral-directional flying qualities tests are usually lightly damped and can be either oscillatory convergent or divergent. The damping ratio for these types of responses normally fall within the $-0.5<\zeta<+0.5$ range. To determine the damping ratio of these characteristic responses construct a diagram similar to that in Figure 7.31 and measure the appropriate quantities.

The following equation is used to compute the damped frequency in radians per second:

$$\omega_d = \frac{2\pi}{P}$$ *eq 7.75*

Where:

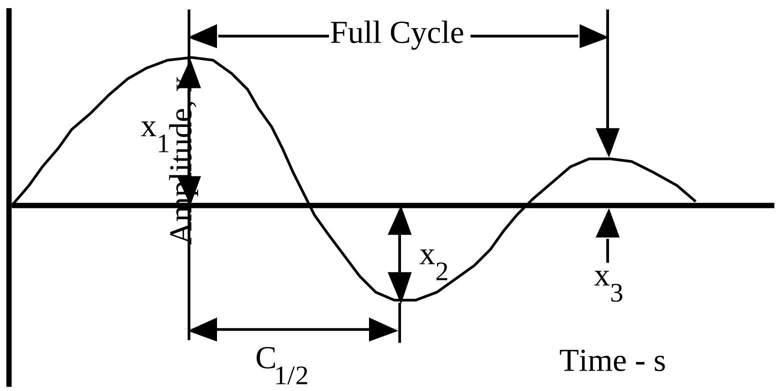
P	- Period
π	- Mathematical constant
ω_d	- Damped frequency.

The following equation is used to compute the natural frequency in radians per second:

$$\omega_n = \frac{\omega_d}{\sqrt{1 - \zeta^2}} = \frac{\frac{2\pi}{P}}{\sqrt{1 - \zeta^2}}$$ *eq 7.76*

Where:

P	- Period
π	- Mathematical constant
ω_d	- Damped frequency
ω_n	- Natural frequency
ζ	- Damping ratio.



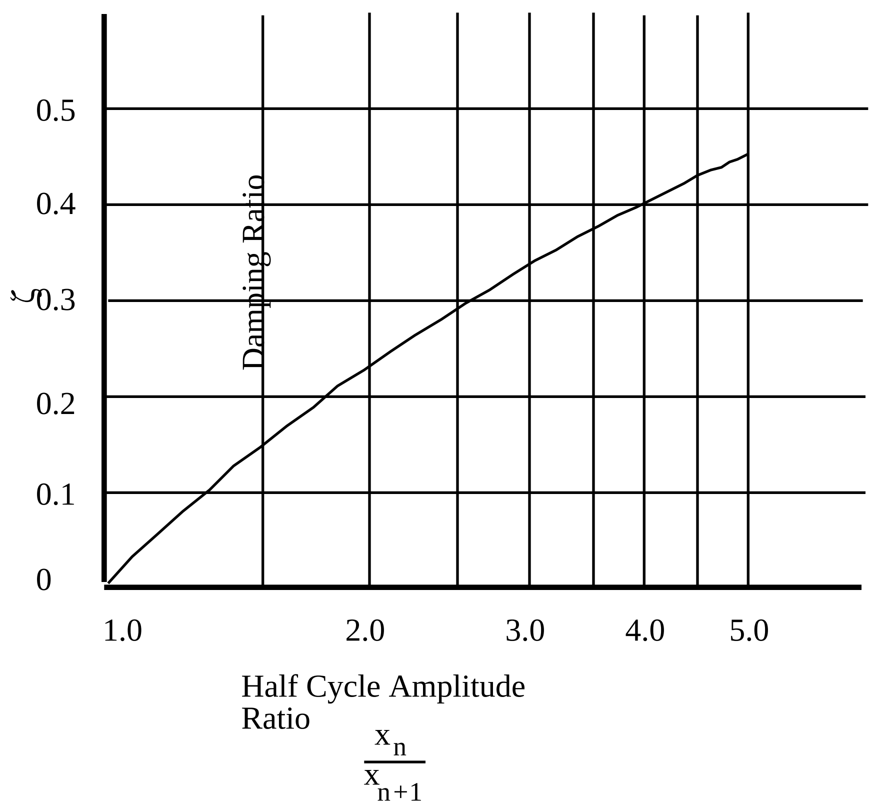


Figure 7.31
Determining Damping Ratio of a Lightly Damped Second Order Response

The bank angle to sideslip ratio is easily obtained by constructing a diagram similar to the example in Figure 7.32 and measuring the width of the oscillation decay envelopes (or oscillation increase envelope if divergent). Compare rates or angles, don't mix the two.

These measurements can be taken from reconstructions of either attitude or rate traces. The three values: P (Period), ζ (Damping ratio), and ϕ/β (Roll to yaw ratio) describe the helicopter's classical LDO.

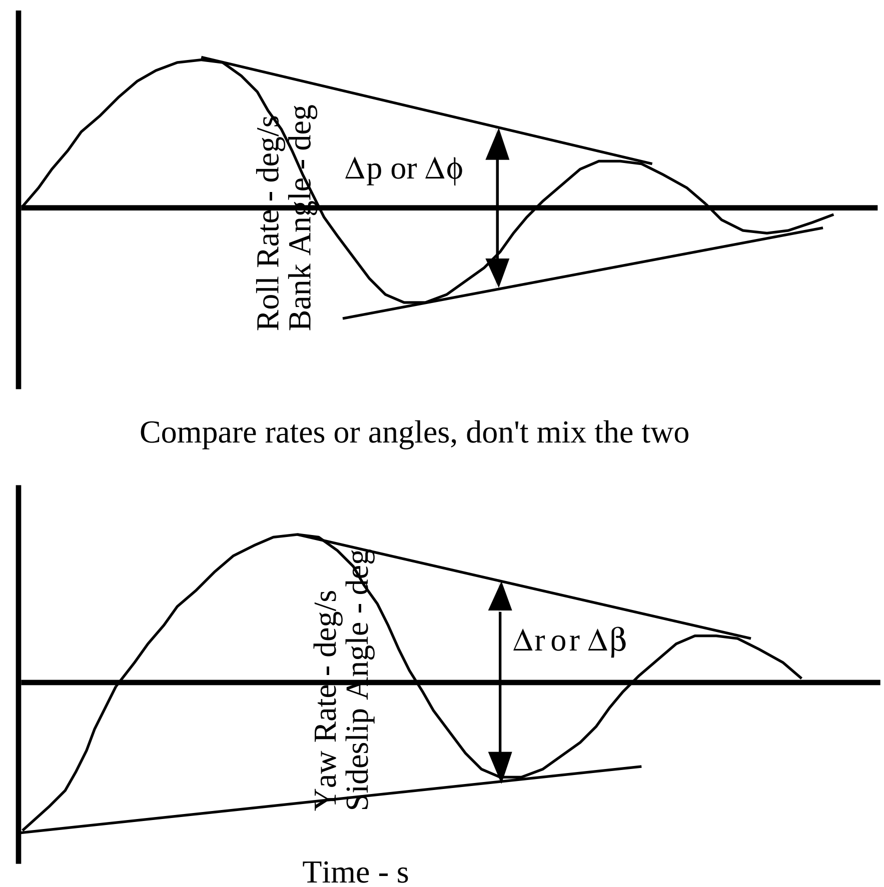


Figure 7.32
Determining Roll to Sideslip Ratio

7.5.3 Trimmed Control Positions

1. Plot lateral and directional control positions, inherent sideslip angle, and collective position with calibrated airspeed as in Figure 7.33.
2. For climbs and descents plot lateral and directional control positions, inherent sideslip angle with collective position (or torque) for a constant airspeed.
3. For steady turning flight plot lateral and directional control positions, sideslip angle, and collective position with varying bank angles.

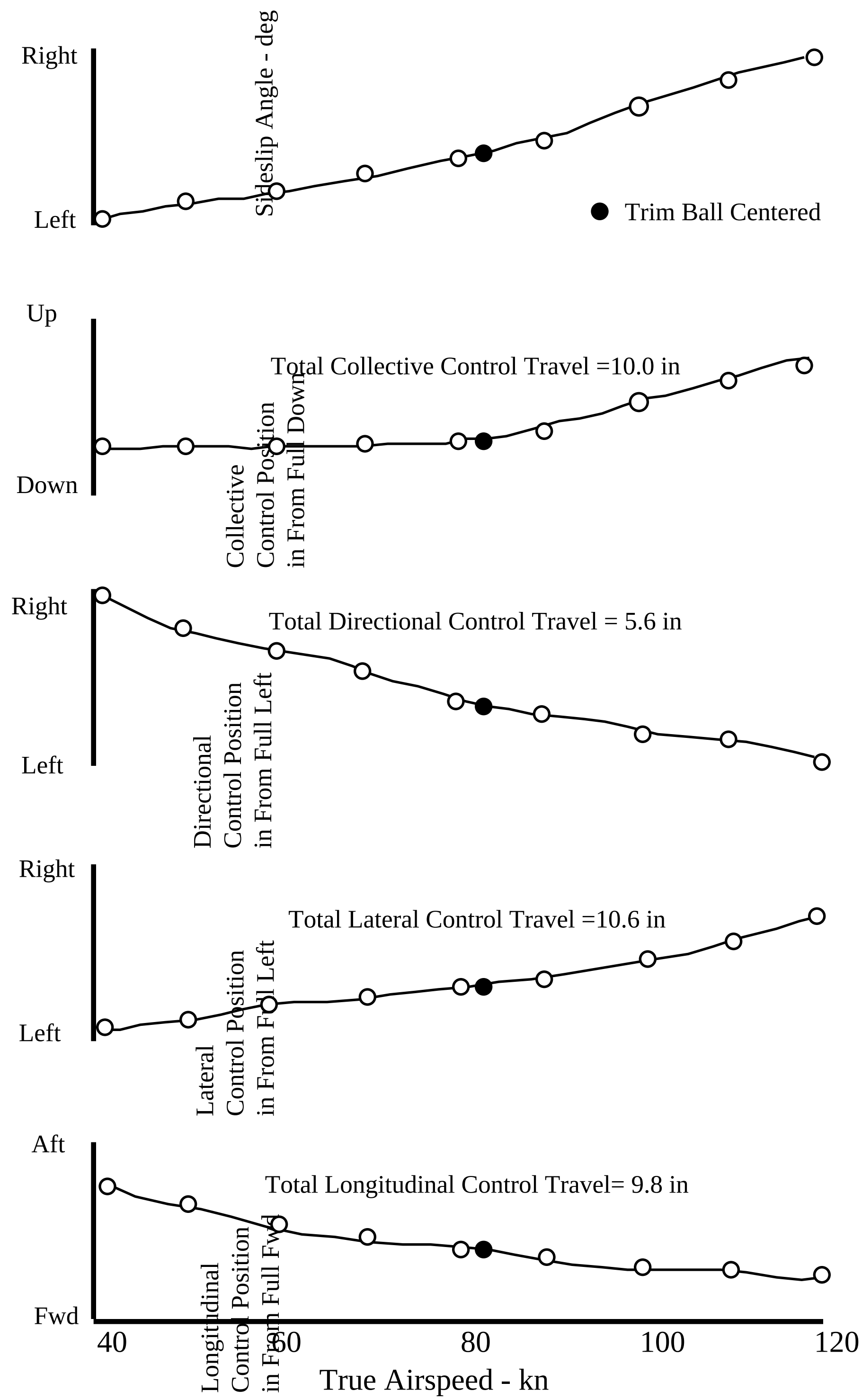


Figure 7.33
Trimmed Control Positions

7.5.4 Static Lateral-Directional Stability (SHSS)

Static lateral-directional data obtained from SHSS are presented in graphic form (Figure 7.34), plotting helicopter control positions, and roll attitude versus sideslip angles for a given airspeed. If pitch attitude or vertical velocities are significant they are presented. Either zero sideslip or ball centered is the trim point, depending on whether engineering results or mission relative results are being presented. In either case, the trim point should be annotated.

7.5.5 Cyclic Only Turns

Cyclic only turn data are generally presented in narrative form to support the discussion of static lateral-directional stability. Include a discussion of the pilot workload in performing cyclic only turns. Assess and report on the workload associated with capturing and maintaining bank angle. Assess the workload associated with maintaining ball centered. Report on the yaw due to lateral control, adverse/proverse yaw, and what effect, if any, this had on the workload associated with capturing and maintaining a bank angle. The behavior of BETA during the initial part of the roll should be discussed as an indication of the strength of directional stability and related to the results of pedal requirements found in steady heading sideslip.

7.5.6 Pedal Only Turns

Pedal only turn data are generally presented in narrative form to support the discussion of static lateral-directional stability. Assess and report on the workload associated with capturing and maintaining bank angle. The behavior of the aircraft in roll due to pedal input and the presence of BETA should be addressed and related to the strength of effective dihedral and cyclic requirements found in steady heading sideslips.

7.5.7 Spiral Stability

Spiral stability data are presented with a properly annotated time history as shown in Figure 7.35. Report the spiral stability characteristics as convergent, divergent or neutral. Include time to half/double amplitude ($T_{1/2}$, T_d). Results for left versus right bank angles may not be the same.

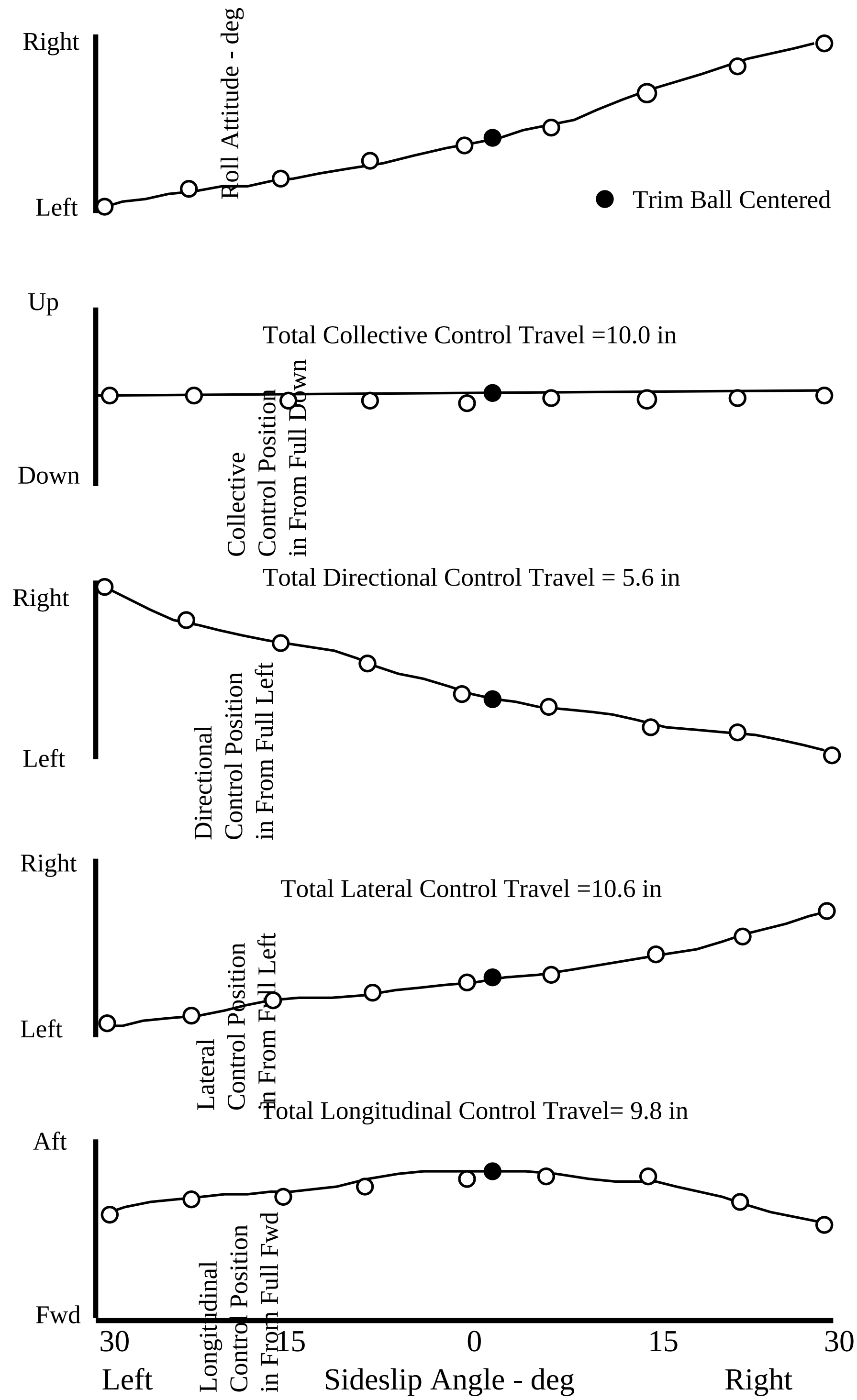


Figure 7.34
Static Lateral-Directional Stability

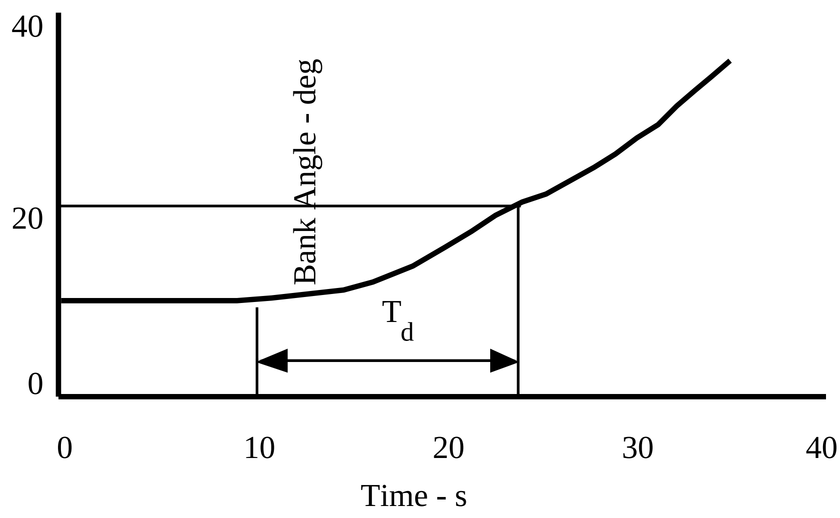


Figure 7.35
Spiral Stability Time History

7.5.8 Lateral-Directional Oscillation

1. Present an annotated time history of the helicopter LDO.

2. Use Tables to summarize the engineering characteristics for several test conditions (period, time to half/double amplitude, damping ratio, roll to yaw ratio). These tables are useful in comparing characteristics at different helicopter conditions (gross weight, center-of-gravity, altitude, airspeed) and configurations.

3. A qualitative assessment is made of pilot effort required to suppress or correct the attitude and sideslip deviations about trim.

7.5.9 Control Response

Present representative time histories (Figure 7.36). Proper annotation is important to help the reader visualize the response. Note the time at which the step input was made and the time at which recovery was initiated. Tables are helpful in presenting the engineering data of the various responses, as well as response data for various flight conditions and airframe configurations.

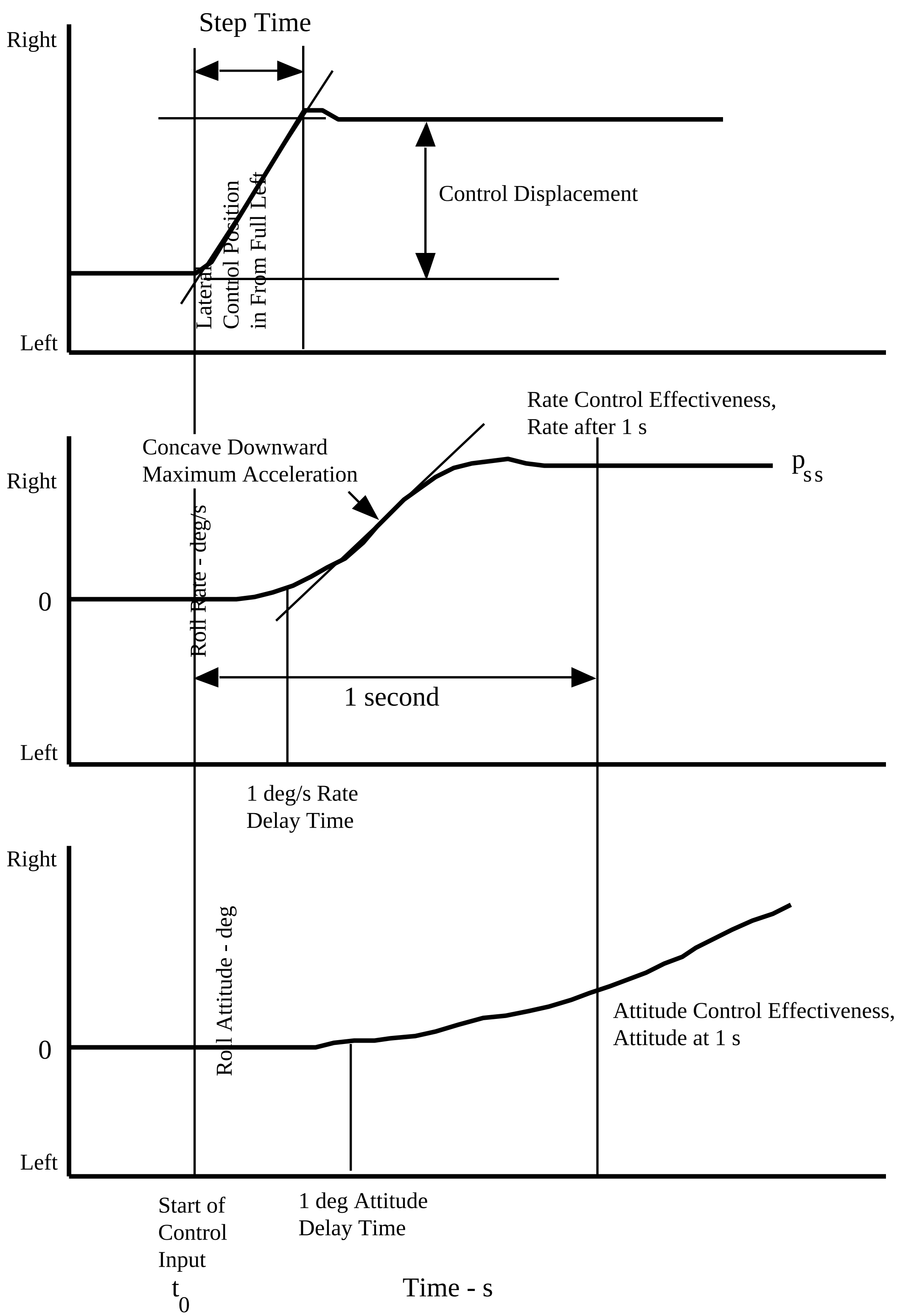


Figure 7.36
Control Response Characteristics

7.5.10 Gust Response

The following data are presented for gust response:

1. Cockpit data of attitude changes.
2. A time history of helicopter attitudes, rates, stability system actuator activity, and required pilot intervention is presented as supporting data for qualitative comments.
3. Qualitative observations are the principal data obtained in the test.

7.6 DATA ANALYSIS

The following paragraphs include items that should be considered in the data analysis of the specific tests. Although these lists are not all inclusive, these factors provide guides for the analysis.

7.6.1 Trimmed Control Positions

1. Were the control margins adequate?
2. Was lateral cyclic required with increasing airspeed/power?
3. Were the control position changes with power changes excessive?
4. Were the forces trimmed to zero?
5. What were inherent sideslip angle changes with airspeed/power?
6. Was there coupling between pitch, roll, and yaw axes?

7.6.2 Static Lateral-Directional Stability

1. Positive static directional stability is indicated by the requirement for left pedal displacement to generate a right sideslip angle (and right pedal displacement to generate a left sideslip angle).
2. Positive dihedral effect is indicated by the requirement for lateral control displacement in the same direction as the sideslip angle during steady heading sideslips.
3. Comment on side forces in terms of bank angle. Did bank angle (side force) provide cues to out of balance flight conditions? At what sideslip angle was the pilot aware of the side force cue to out of balanced flight?
4. Were there any pitot static problems (airspeed or vertical velocity) in sideslips.

7.6.2.1 TURNS ON ONE CONTROL

1. What does the lateral control only turns reveal about the relative magnitude of the directional stability of the helicopter? If the sideslip angle during the steady state turn returns to the inherent sideslip angle, the helicopter exhibits strong directional stability. Is adverse/proverse yaw present? Does it cause an increase in pilot workload for coordinated turns.
2. In addition to testing for directional stability, a limited examination of the spiral mode of the helicopter can be obtained during lateral cyclic only and pedal only turns. The requirement for lateral cyclic displacement into the turn after a steady state turn rate is established is indicative of a convergent spiral mode. If cyclic or pedal position during steady state turn is the same as trim, neutral spiral mode is indicated. A requirement for cyclic or pedal displacement away from the turn during steady state turns is indicative of a divergent spiral mode. Cyclic or pedal displacement during turns on one control is only one portion of a spiral mode evaluation.
3. What does the directional control only turns reveal about the relative magnitude of the effective dihedral of the helicopter? If a relatively small sideslip angle results in a large roll rate in the opposite direction, the helicopter exhibits strong effective dihedral.

7.6.3 Spiral Stability

1. Did the spiral stability characteristics vary with airspeed and power?
2. What is the impact of the spiral stability characteristics on mission performance and pilot workload?

7.6.4 Lateral-Directional Oscillation

1. How easily is the LDO excited by atmospheric turbulence?
2. Is the LDO excited by normal pilot control motions?
3. What is the ϕ/β ratio?
4. How well damped is the LDO?
5. What is the frequency of the LDO?
6. Is the LDO disturbing or distracting to the pilot, crew members, or passengers?

7. If the helicopter can be flown with various AFCS modes selected, are the characteristics of the LDO appreciably different?

8. Can the pilot damp the LDO or does pilot interaction further aggravate the situation?

9. What are the pilot workload implications of this nuisance mode?

7.6.5 Control Response

1. Quantitative data are taken from the response time histories. The parameters chosen and presented support pilot opinion of the helicopter response to control inputs. Parameters which define the control moment available are the rate control effectiveness, attitude control effectiveness, and the steady-state angular velocity. Quantities which are used in discussing the quality of the response are the angular accelerations delay time, initial angular acceleration, inflection time, and response time constant. Use these data as an integral part of your discussions.

2. Did the helicopter respond adequately to perform the assigned task?
    a. Were the steady-state rates adequate?
    b. Was the response predictable?
    c. Was the response consistent?
    d. Were there any over controlling problems?
    e. Were off axis responses observed?

7.6.6 Gust Response

1. Discuss the effects of varying levels of turbulence on task performance. Did workload increase? Describe the ride quality.

2. What were the pitch, roll and yaw attitude changes? Airspeed changes?

3. Did the helicopter tend to return to the original trim condition?

4. How much pilot effort was required to return the helicopter to the trim condition?

5. Were any of the helicopter modes of motion excited?

7.7 MISSION SUITABILITY

The suitability of the test helicopter for the intended mission is the ultimate reason for conducting any handling qualities test. Each of the specific tests provides some information in accessing that suitability. Lateral-directional trimmed control positions provides information about control margins, linearities, discontinuities, and gradients which assist the pilot in determining overall suitability. Static lateral-directional stability characteristics are good indicators of pilot workload required to maintain a desired forward flight trim condition and the motion of the helicopter once disturbed. Spiral stability characteristics is an indicator of the pilot workload required to maintain a trimmed wings level condition. Short term lateral-directional dynamics are a big factor in how much difficulty the pilot will have maneuvering the helicopter in roll and yaw during high gain tasks. Coupled lateral-directional short term dynamics are most often found objectionable. Control response tests quantitatively document the pilots qualitative comments concerning helicopter responsiveness.

In addition to the engineering tests discussed the test pilot should investigate how the helicopter is intended to be used and duplicate mission tasks. Generally, knowledge of engineering shortcomings allow the test team to design operational scenarios which reflect the poor characteristics. Each test program and helicopter require a unique set of mission maneuvers. The most important aspect of mission suitability testing is to keep the operator in mind. Consider how the test helicopter is used in the field and who will be operating the helicopter.

7.8 SPECIFICATION COMPLIANCE

General guidelines for lateral-directional helicopter handling qualities in forward flight are contained in applicable paragraphs of MIL-H-8501A. Read the specification carefully to understand each requirement. Remember the military specifications are only a guide which leave some room for interpretation for individual systems and situations. The following list identifies the MIL-H-8501A paragraph number and a short description of the requirement.

- 3.3 Directional and Lateral characteristics.
  - 3.3.8 Directional control for autorotation
  - 3.3.9 Directional stability and effective dihedral
    - 3.3.9.1 Turns on one control
    - 3.3.9.2 Adverse yaw
  - 3.3.10 Trimmability, control jump
  - 3.3.11 Control force gradients

3.3.12 Limit forces
3.3.13 Breakout forces
3.3.14 Control coupling
3.3.15 Lateral control response, over control
3.3.16 Lateral and directional control response
3.3.17 Lateral trim with power effects
3.5.9 Automatic stabilization system
3.5.10 Total control system freeplay
3.5.11 Mechanical coupling
  3.5.11.1 Mixing
3.6.1 Instrument flight characteristics
  3.6.1.1 Control power
  3.6.1.2 Lateral directional oscillation IFR
3.6.2 Lateral directional stability IFR
3.7.1 Vibrations
3.7.2 Vibrations
3.7.3 Mechanical Instability

Additional requirements are included in the military specification relating to boosted controls, failure modes, automatic stabilization equipment, and vibrations. Some of these paragraphs may apply to specific helicopters equipped with an AFCS.

7.9 GLOSSARY

7.9.1 Notations

A	Rate response value
A_{1s}	Lateral cyclic pitch angle, shaft referenced
AFCS	Automatic Flight Control System
B	Damping constant
b	Number of blades
b_{1s}	Lateral flapping angle, shaft referenced
$C_{1/2}$	Cycles to one-half amplitude
CG	Center of gravity
e	Base of natural logarithm, flapping hinge offset

FC	Fuel count
F_R	Reaction force from ball race
ft	Foot
g	Gravity
GCA	Ground controlled approach
h	Height of hub above CG
h'	Longitudinal distance between the rotor shaft and the CG
H_{P_o}	Observed pressure altitude
HQR	Handling Qualities Rating
h_{TR}	Height of the tail rotor above the CG
i	Imaginary index
I_{xx}	Moment of inertia about x axis, roll moment of inertia
I_{xz}	Product of inertia about x z axes
I_{zz}	Moment of inertia about z axis, yaw moment of inertia
kn	Knot
L	Net moment about x axis, Roll moment, Lift, Length
$L_{A_{1s}}$	Roll moment due to lateral cyclic pitch angle
$L_{\delta_{LAT}}$	Roll moment due to lateral control
LDO	Lateral-directional oscillation
$L_{\delta_{PED}}$	Roll moment due to pedal control
L_{f+t}	Roll moment due to the fuselage/tail
L_H	Roll moment due to rotor hub forces
ln_e	Natural logarithm
L_p	Roll moment due to roll rate
$L_{\theta_{TR}}$	Roll moment due to tail rotor pitch angle
L_r	Roll moment due to yaw rate
l_t	Distance from the tail to the CG
L_v	Roll moment due to lateral velocity
$L_{v_{PED}}$	Pedal contribution to roll moment due to lateral velocity
m	Mass
M_S	Blade mass moment
N	Net moment about z axis, Yaw moment
$N_{A_{1s}}$	Yaw moment due to lateral cyclic pitch angle

$N_{\delta_{LAT}}$	Yaw moment due to lateral control
$N_{\delta_{PED}}$	Yaw moment due to pedal control
N_{f+t}	Yaw moment due to the fuselage/tail
N_g	Engine gas generator speed
N_p	Yaw moment due to roll rate
$N_{\theta_{TR}}$	Yaw moment due to tail rotor pitch angle
N_R	Main rotor speed
N_r	Yaw moment due to yaw rate
N_v	Yaw moment due to lateral velocity
P	Period
p	Angular velocity about x axis, Roll rate
p_0	Initial angular velocity about x axis, Roll rate
PFLF	Power for level flight
p_{ss}	Steady state angular velocity about x axis, roll rate
$\dot{p}$	Angular acceleration about x axis
Q	Engine torque
q	Angular velocity about y axis, Pitch rate
Q_{MR}	Main rotor torque
r	Angular velocity about z axis, Yaw rate
r_0	Initial angular velocity about z axis, Yaw rate
rad	Radian
$\dot{r}$	Angular acceleration about z axis
s	Second
SHSS	Steady heading sideslip
T	Thrust
t	Time
t_0	Initial time
$T_{1/2}$	Time to one-half amplitude
$T_{1/N}$	Time to decay to 1/N of maximum amplitude
T_d	Time to double amplitude
T_o	Observed temperature
TOOC	Turns on one control
TPP	Tip path plane

T_{TR}	Tail rotor thrust
u	Translational velocity component along x axis
u_0	Initial velocity along x axis
u_{NR}	Translational velocity component along the non-rotating x axis system
V	Velocity, Free stream velocity, Relative velocity
v	Translational velocity component along y axis
VAR	Vibration Assessment Rating
V_H	Maximum level flight airspeed
V_{NE}	Velocity never exceed
v_{NR}	Translational velocity component along the non-rotating y axis system
V_o	Observed airspeed
V_v	Vertical velocity
$\ddot{v}$	Time rate of change of linear acceleration along y axis
$\dot{v}$	Linear acceleration along y axis
W	Weight
W_{BALL}	Weight of the ball
x	Orthogonal direction along longitudinal axis of the aircraft
$\dot{x}$	Time rate of change of x
Y	Resultant force in y direction
y	Orthogonal direction along lateral axis of the aircraft
$Y_{A_{1s}}$	Side force due to lateral cyclic pitch angle
$Y_{\delta_{LAT}}$	Side force due to lateral control
$Y_{\delta_{PED}}$	Side force due to pedal control
Y_{f+t}	Side force due to the fuselage/tail
Y_p	Side force due to roll rate
$Y_{\theta_{TR}}$	Side force due to tail rotor pitch angle
Y_R	Rotor side force
Y_r	Side force due to yaw rate
Y_v	Side force due to lateral velocity

Y_{v_R}	Main rotor contribution to side force due to lateral velocity
$Y_{v_{TR}}$	Tail rotor contribution to side force due to lateral velocity
z	Orthogonal direction along vertical axis of the aircraft

7.9.2 Greek Symbols

β (beta)	Sideslip angle
δ (delta)	Control
δ_{LAT}	Lateral control
δ_{PED}	Pedal control
ϕ (phi)	Roll angle
$\ddot{\phi}$	Roll angle acceleration
$\dot{\phi}$	Rate of change of roll angle
γ (gamma)	Flight path angle
γ_0	Initial flight path angle
λ_s (lambda)	Spiral mode root
π (pi)	Mathematical constant
θ (theta)	Pitch angle
θ_{TR}	Tail rotor pitch angle
τ (tau)	Time constant
τ_R	Roll mode time constant
τ_s	Spiral mode time constant
Ω (Omega)	Rotor angular velocity
ω (omega)	Turn angular velocity
ω_d	Damped frequency
ω_n	Natural frequency
ψ (psi)	Yaw attitude

$\ddot{\psi}$	Yaw attitude acceleration
$\dot{\psi}$	Rate of change of yaw attitude
ζ (zeta)	Damping ratio

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CHAPTER EIGHT

HOVER AND LOW AIRSPEED STABILITY, CONTROL, AND FLYING QUALITIES

CHAPTER EIGHT

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EQUATIONS

Equation		PAGE
$\dot{u} - X_u\Delta u - X_q\Delta q + g\Delta\theta = X_{B_{1s}}\Delta B_{1s} = X_{\delta_{LONG}}\Delta\delta_{LONG}$	*eq 8.1*	8.4
$\dot{q} - M_u\Delta u - M_q\Delta q = M_{B_{1s}}\Delta B_{1s} = M_{\delta LONG}\Delta\delta_{LONG}$	*eq 8.2*	8.5
$\dot{w} - Z_w\Delta w = Z_{\theta_C}\Delta\theta_C = Z_{\delta_C}\Delta\delta_C$	*eq 8.3*	8.5
$\dot{v} - Y_v\Delta v - Y_p\Delta p = Y_{A_{1s}}\Delta A_{1s} + Y_{\theta_{TR}}\Delta\theta_{TR} = Y_{\delta_{LAT}}\Delta\delta_{LAT} + Y_{\delta_{PED}}\Delta\delta_{PED}$	*eq 8.4*	8.6
$\dot{p} - L_v\Delta v - L_p\Delta p = L_{A_{1s}}\Delta A_{1s} + L_{\theta_{TR}}\Delta\theta_{TR} = L_{\delta_{LAT}}\Delta\delta_{LAT} + L_{\delta_{PED}}\Delta\delta_{PED}$	*eq 8.5*	8.6
$\begin{aligned}\dot{r} - N_v\Delta v - N_p\Delta p - N_r\Delta r &= N_{A_{1s}}\Delta A_{1s} + N_{\theta_{TR}}\Delta\theta_{TR} \\ &= N_{\delta_{LAT}}\Delta\delta_{LAT} + N_{\delta_{PED}}\Delta\delta_{PED}\end{aligned}$	*eq 8.6*	8.7
$M_u u + M_{\delta_{LONG}}\delta_{LONG} = 0$	*eq 8.7*	8.27
$\dfrac{\Delta\delta_{LONG}}{\Delta u} = -\dfrac{M_u}{M_{\delta_{LONG}}} = -\dfrac{\text{Speed Stability Derivative}}{\text{Sensitivity}}$	*eq 8.8*	8.27
$\text{Period} = 2\pi\sqrt{\dfrac{L}{g}}$	*eq 8.9*	8.33

$$u \cong \dot{x} \cong L\dot{\theta}$$ *eq 8.10* 8.33

$$\Delta M = 0 = \frac{\partial M}{\partial u} u + \frac{\partial M}{\partial \dot{\theta}} \dot{\theta}$$ *eq 8.11* 8.35

$$\dot{\theta} = \frac{\frac{\partial M}{\partial u}}{\frac{\partial M}{\partial \dot{\theta}}} u = -\frac{M_u}{M_{\dot{\theta}}} u$$ *eq 8.12* 8.35

$$L_{Equiv} = \frac{u}{\dot{\theta}} = -\frac{M_{\dot{\theta}}}{M_u}$$ *eq 8.13* 8.35

$$\text{Period} = 2\pi\sqrt{\frac{L_{Equiv}}{g}} = 2\pi\sqrt{\frac{M_{\dot{\theta}}}{gM_u}}$$ *eq 8.14* 8.35

$$\dot{q} - M_q q = M_{B_{1s}} B_{1s} = M_{\delta_{LONG}} \delta_{LONG}$$ *eq 8.15* 8.37

$$\dot{p} - L_p p = L_{A_{1s}} A_{1s} + L_{\theta_{TR}} \theta_{TR} = L_{\delta_{LAT}} \delta_{LAT} + L_{\delta_{PED}} \delta_{PED} \approx L_{\delta_{LAT}} \delta_{LAT}$$ *eq 8.16* 8.37

$$\dot{u} = \frac{B}{m} u = \frac{1}{m} \frac{\partial X}{\partial \delta} \delta$$ *eq 8.17* 8.39

$$\dot{u} - X_u u = X_\delta \delta$$ *eq 8.18* 8.39

$$\tau = -\frac{1}{X_u}$$ *eq 8.19* 8.39

$$q(t) = -\frac{M_{B_{1s}} B_{1s}}{M_q} \left(1 - e^{M_q t}\right)$$ *eq 8.20* 8.40

$$\dot{q}(t) = M_{B_{1s}} B_{1s} e^{M_q t}$$ *eq 8.21* 8.40

$$\theta (t) = \int_0^t q \, dt = - \frac{M_{B_{1s}} B_{1s}}{M_q} \left[t + \frac{1}{M_q} \left(1 - e^{M_q t} \right) \right]$$ *eq 8.22* 8.40

$$\dot{q} (0) = M_{B_{1s}} B_{1s}$$ *eq 8.23* 8.42

$$q_{ss} = - \frac{M_{B_{1s}} B_{1s}}{M_q}$$ *eq 8.24* 8.42

$$\tau = - \frac{1}{M_q}$$ *eq 8.25* 8.42

CHAPTER EIGHT

HOVER AND LOW AIRSPEED STABILITY, CONTROL, AND FLYING QUALITIES

8.1 INTRODUCTION

This chapter discusses stability, control, and flying qualities in the hover and low airspeed flight regime. The helicopter possess the unique capability of vertical and low airspeed flight. The greatest tactical advantage of helicopter employment relies on this unique capability. Therefore, a large part of the helicopter mission is conducted in the hover and low airspeed regime. Understanding, testing, and documenting the flying qualities in this flight regime is critical to the success of the helicopter and the test program.

8.2 PURPOSE OF TEST

The purpose of these tests is to evaluate the hover and low airspeed stability, control, and flying qualities of the helicopter. The tests included in the evaluation are:

1. Trim control positions.
2. Critical azimuth.
3. Turn on a spot.
4. Static stability.
5. Long term dynamic stability.
6. Control response.
7. Mission maneuvers.

8.3 THEORY

The linear analysis of the helicopter response to control inputs and other disturbances provides a valuable indication of important handling qualities characteristics. A number of the stability derivatives are zero or negligible in the hovering regime resulting in a simplification of the equations of motion and a decoupling of some of the degrees of freedom. The linearized equations used in the discussion of low airspeed stability and control are the basis for the theory presented in this chapter.

8.3.1 Summary of Quasi-Static Rotor Characteristics

The rotor characteristics are presented in Chapter 4 along with rotor response to various inputs. Those discussions indicated a quasi-static approximation for rotor response is satisfactory for most discussions of helicopter stability, control, and flying qualities.

Table 8.I repeats the summaries of quasi-static rotor response characteristics given in Chapter 4 for hovering flight. The table indicates the most significant quasi-static responses of the rotor to specified input conditions and summarizes the effects responsible for particular rotor contributions to stability derivatives.

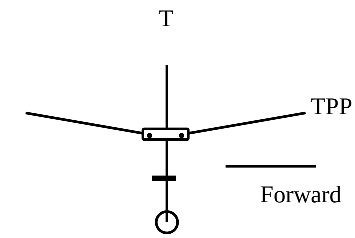
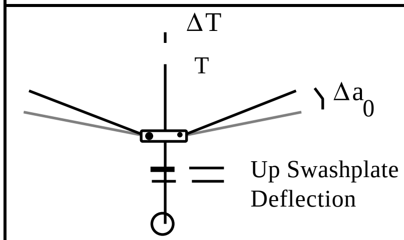
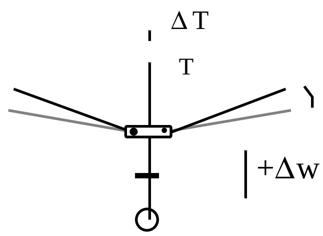
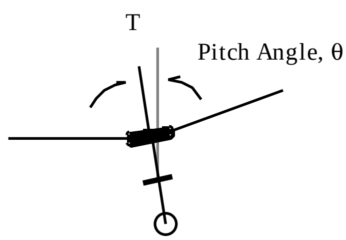
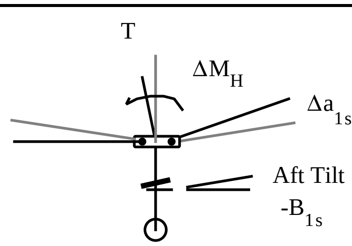
The diagrams on Table 8.I show only longitudinal inputs and responses. Similar diagrams are applicable for lateral inputs and responses with pitch angle (θ) replaced by roll angle (ϕ), aft longitudinal cyclic pitch angle $\left(-B_{1s}\right)$ replaced by right lateral cyclic pitch angle (A_{1s}), aft longitudinal flapping angle (a_{1s}) replaced by right lateral flapping angle (b_{1s}), pitch rate (q) replaced by roll rate (p), and translational velocity component along x axis (u) replaced by translational velocity component along y axis (v).

8.3.2 Linearized Equations of Motion

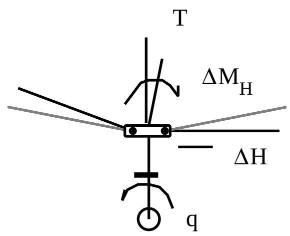
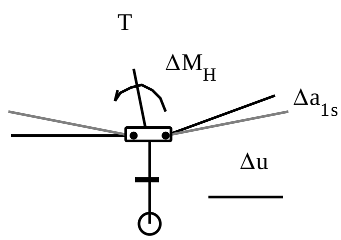
The set of linearized longitudinal and lateral-directional equations of motion which are the basis for discussing hovering and low airspeed flight dynamics are presented below. Coupling terms between these two sets of equations are omitted. Some qualitative discussions of the coupling effects are included in this chapter and further considered in Chapter 9. Equations are presented in terms of the main and tail rotor variables (A_{1s}, B_{1s}, θ_C, and θ_{TR}) and in terms of the cockpit control variables (δ_{LONG}, δ_{LAT}, δ_C, and δ_{PED}). The relationship between the two forms is the gearing ratio between the rotor variables and the cockpit flight control variables.

Table 8.I

Quasi-Static Rotor Characteristics in Hovering Flight

	Perturbation	Rotor Response	Rotor Force Applied to Aircraft
	(a) Nominal hover Nominal collective No cyclic	Steady state.	Thrust (T) acts along shaft axis through CG
	(b) Collective Increment, $+\Delta\theta_C$	Increased coning. $+\dfrac{\partial a_0}{\partial \theta_C}$	Thrust increase, $+\Delta T$
	(c) Downward rotor velocity or upward gust $+\Delta w$	Increased blade angle of attack. Increased coning. $+\dfrac{\partial a_0}{\partial w}$	Thrust increase, $+\Delta T$
	(d) Shaft pitch angle, $+\Delta\theta$	TPP remains perpendicular to swashplate.	Thrust remains along shaft. No moment increase.
	(e) Cyclic pitch, $+\Delta B_{1s}$	TPP tilts due to cyclic pitch, nose up $-\dfrac{\partial a_{1s}}{\partial B_{1s}}$	Thrust vector tilts relative to shaft producing moment about CG. Offset hinge moment due to rotor tilt.

Quasi-Static Rotor Characteristics in Hovering Flight (cont'd)

	(f) Pitch rate, $+\Delta q$	Rotor lags shaft by angle proportional to pitch rate $-\frac{\partial a_{1s}}{\partial q}$	Longitudinal thrust component damps pitch rate. H force reduces pitch damping. Offset hinge moment due to rotor tilt damps pitch rate.
	(g) Airspeed increase, $+\Delta u$	Rotor tilts away from relative velocity, nose up, stable (blowback) $+\frac{\partial a_{1s}}{\partial u}$	Thrust vector tilts back producing nose up pitching moment. Nose up offset hinge moment due to aft rotor tilt. Aft rotor H force.

The lateral-directional equations of motion are simplified by assuming the I_{xz} product of inertia to equal zero. If I_{xz} is not zero, the same form of the equations of motion is obtained using primed derivatives which correct for the effect of inertia coupling.

8.3.2.1 LONGITUDINAL

The longitudinal derivatives X_w, X_{θ_C}, Z_u, Z_q, $Z_{B_{1s}}$, M_w, and M_{θ_C} are set equal to zero since first order contributions to these derivatives (which are proportional to u_0) are all zero in the hovering regime. In this case, the variables u and q do not appear in the Z force equilibrium equation since it is decoupled from those for pitching moment and X force equilibrium. Also, X_q is usually small, having a negligible effect on dynamic motions. Alternative forms are indicated for the control derivatives in terms of longitudinal cyclic pitch angle (B_{1s}) or longitudinal control (δ_{LONG}) for the X force and pitching moment equations. Alternative forms are indicated as well in terms of collective pitch angle (θ_C) or collective control (δ_C) for the Z force equation.

$$\dot{u} - X_u\Delta u - X_q\Delta q + g\Delta\theta = X_{B_{1s}}\Delta B_{1s} = X_{\delta_{LONG}}\Delta\delta_{LONG} \qquad \textit{eq 8.1}$$

$$\dot{q} - M_u \Delta u - M_q \Delta q = M_{B_{1s}} \Delta B_{1s} = M_{\delta_{LONG}} \Delta \delta_{LONG}$$

eq 8.2

$$\dot{w} - Z_w \Delta w = Z_{\theta_C} \Delta \theta_C = Z_{\delta_C} \Delta \delta_C$$

eq 8.3

Conventional stability derivative notation is used in which the force derivatives are normalized with respect to mass (m) and the moment derivatives are normalized with respect to moment of inertia (I_{yy}):

$$X_{(\,)} = \frac{1}{m} \frac{\partial X}{\partial (\,)}$$

$$Z_{(\,)} = \frac{1}{m} \frac{\partial Z}{\partial (\,)}$$

$$M_{(\,)} = \frac{1}{I_{yy}} \frac{\partial M}{\partial (\,)}$$

Where:

B_{1s}	- Longitudinal cyclic pitch angle, shaft referenced
δ_C	- Collective control
δ_{LONG}	- Longitudinal control
g	- Gravity
$M_{B_{1s}}$	- Pitch moment due to longitudinal cyclic pitch angle
$M_{\delta_{LONG}}$	- Pitch moment due to longitudinal control
M_q	- Pitch moment due to pitch rate
M_u	- Pitch moment due to longitudinal velocity
q	- Angular velocity about y axis, Pitch rate
θ	- Pitch angle
θ_C	- Collective pitch angle
$\dot{q}$	- Angular acceleration about y axis
u	- Translational velocity component along x axis

$\dot{u}$	- Linear acceleration along x axis
w	- Translational velocity component along z axis
$\dot{w}$	- Linear acceleration along z axis
$X_{B_{1s}}$	- Longitudinal force due to longitudinal cyclic pitch angle
$X_{\delta_{LONG}}$	- Longitudinal force due to longitudinal control
X_q	- Longitudinal force due to pitch rate
X_u	- Longitudinal force due to longitudinal velocity
Z_{δ_C}	- Vertical force due to collective control
Z_{θ_C}	- Vertical force due to collective pitch angle
Z_w	- Vertical force due to vertical velocity.

8.3.2.2 LATERAL-DIRECTIONAL

The lateral-directional derivatives Y_u, Y_r, and L_r are set equal to zero because they are proportional to initial trim velocity. The side force and rolling moment equations do not depend on yaw rate in this approximation and are decoupled from the yawing moment equation. However, tail rotor thrust inputs could introduce rolling moments and side forces. Again, alternate forms are indicated for control forces and moments expressed in terms of lateral cyclic pitch angle (A_{1s}) and tail rotor pitch angle (θ_{TR}), or in terms of lateral control (δ_{LAT}) and pedal control (δ_{PED}).

Control coupling depends on the center of gravity (CG) location, tail rotor height, and control system. It is convenient to simplify some of the discussion by assuming pedal inputs produce only yawing moment and lateral control inputs only rolling moments.

$$\dot{v} - Y_v\Delta v - Y_p\Delta p = Y_{A_{1s}}\Delta A_{1s} + Y_{\theta_{TR}}\Delta\theta_{TR} = Y_{\delta_{LAT}}\Delta\delta_{LAT} + Y_{\delta_{PED}}\Delta\delta_{PED} \qquad \textit{eq 8.4}$$

$$\dot{p} - L_v\Delta v - L_p\Delta p = L_{A_{1s}}\Delta A_{1s} + L_{\theta_{TR}}\Delta\theta_{TR} = L_{\delta_{LAT}}\Delta\delta_{LAT} + L_{\delta_{PED}}\Delta\delta_{PED} \qquad \textit{eq 8.5}$$

$$\dot{r} - N_v\Delta v - N_p\Delta p - N_r\Delta r = N_{A_{1s}}\Delta A_{1s} + N_{\theta_{TR}}\Delta\theta_{TR}$$

$$= N_{\delta_{LAT}}\Delta\delta_{LAT} + N_{\delta_{PED}}\Delta\delta_{PED} \qquad \textit{eq 8.6}$$

Conventional stability derivative notation is used in which the force derivatives are normalized with respect to mass (m) and the moment derivatives are normalized with respect to moment of inertia (I_{zz}, I_{xx}):

$$Y_{(\,)} = \frac{1}{m}\frac{\partial Y}{\partial(\,)}$$

$$L_{(\,)} = \frac{1}{I_{xx}}\frac{\partial L}{\partial(\,)}$$

$$N_{(\,)} = \frac{1}{I_{zz}}\frac{\partial N}{\partial(\,)}$$

Where:

A_{1s}	- Lateral cyclic pitch angle, shaft referenced
δ_{LAT}	- Lateral control
δ_{PED}	- Pedal control
$L_{A_{1s}}$	- Roll moment due to lateral cyclic pitch angle
$L_{\delta_{LAT}}$	- Roll moment due to lateral control
$L_{\delta_{PED}}$	- Roll moment due to pedal control
L_p	- Roll moment due to roll rate
$L_{\theta_{TR}}$	- Roll moment due to tail rotor pitch angle
L_v	- Roll moment due to lateral velocity
$N_{A_{1s}}$	- Yaw moment due to lateral cyclic pitch angle
$N_{\delta_{LAT}}$	- Yaw moment due to lateral control
$N_{\delta_{PED}}$	- Yaw moment due to pedal control
N_p	- Yaw moment due to roll rate
$N_{\theta_{TR}}$	- Yaw moment due to tail rotor pitch angle

N_r	- Yaw moment due to yaw rate
N_v	- Yaw moment due to lateral velocity
p	- Angular velocity about x axis, Roll rate
$\dot{p}$	- Angular acceleration about x axis
θ_{TR}	- Tail rotor pitch angle
r	- Angular velocity about z axis, Yaw rate
$\dot{r}$	- Angular acceleration about z axis
v	- Translational velocity component along y axis
$\dot{v}$	- Linear acceleration along y axis
$Y_{A_{1s}}$	- Side force due to lateral cyclic pitch angle
$Y_{\delta_{LAT}}$	- Side force due to lateral control
$Y_{\delta_{PED}}$	- Side force due to pedal control
Y_p	- Side force due to roll rate
$Y_{\theta_{TR}}$	- Side force due to tail rotor pitch angle
Y_v	- Side force due to lateral velocity.

8.3.3 Stability Derivatives

There are a number of nonlinear aerodynamic phenomena which influence the flying characteristics of a helicopter in hovering flight. For example, changes in the downwash impingement on the fuselage, or movement of rotor tip trailing vortices in relationship to the tail rotor, both result in forces and moments being applied to the helicopter.

However, in considering flight dynamics, it is convenient to use the linear equations of motion presented above. These contain constant derivatives corresponding to the initial trim conditions. Such equations give a satisfactory indication of the dynamic characteristics which are important from the handling qualities standpoint although they may not explain all trim changes and response peculiarities. Forces and moments due to the main rotor and the tail rotor make the dominant contributions to the derivatives in the linear equations of motion.

8.3.4 Z Force Equation Derivatives

Table 8.I (b) indicates a main rotor collective pitch input results in increased rotor thrust and coning. This is the source of the vertical force due to collective pitch angle derivative (Z_{θ_C}) which is negative because the z axis is directed downward. The magnitude of the corresponding collective control derivative (Z_{δ_C}) is determined by the gearing used in the control system. The downward motion of the helicopter shown in Table 8.I (c) produces positive increments in effective blade angles of attack and increased thrust. This is the source of the Z_w damping derivative which like Z_{θ_C} is negative because of the z axis sign convention.

8.3.5 X Force and Pitching Moment Equation Derivatives

The largest contributions to the derivatives in the X force and pitching moment equations result from the aft tilt of the thrust vector and tip path plane (TPP) relative to the shaft. Table 8.I (e) indicates aft longitudinal cyclic pitch results in an aft thrust component responsible for the X force due to longitudinal cyclic pitch angle derivative ($X_{B_{1s}}$), and the corresponding X force due to longitudinal control derivative ($X_{\delta_{LONG}}$). The pitch moment about the CG due to the aft thrust component and offset hinge moment both contribute to the pitch moment due to longitudinal cyclic pitch angle derivative ($M_{B_{1s}}$). This corresponds to the pitch moment due to longitudinal control derivative ($M_{\delta_{LONG}}$) when referred to cockpit controls. The usual sign convention results in $X_{B_{1s}}$ and $X_{\delta_{LONG}}$ being positive while $M_{B_{1s}}$ and $M_{\delta_{LONG}}$ are negative.

Table 8.I (f) depicts the TPP lag when the shaft and hub have a nose up pitch rate. The resulting nose down pitch moment about the CG due to the forward thrust component and the offset hinge moment provide the principal contributions to the pitch moment due to pitch rate derivative (M_q). However, an H force due to blade induced drag acts in a direction tending to reduce the magnitude of M_q.

Table 8.I (g) shows the aft TPP tilt or blowback due to an increase in velocity (Δu). This results in an aft thrust component and offset hinge moment in large part responsible for the speed stability derivative (M_u) in hovering flight. The fuselage profile drag does not

contribute to M_u for a zero airspeed trim condition. However, the blade profile drag due to the combination of speed caused by shaft rotation and aircraft translational velocity result in contributions to the X_u and M_u derivatives.

8.3.6 Side Force and Rolling Moment Equation Derivatives

Rotor forces and moments contribute to the derivatives in the Y force and rolling moment equations analogous to those in the X force and pitching moment equations. The analogous derivative pairs are X_u and Y_v, X_q and Y_p, $X_{B_{1s}}$ and $Y_{A_{1s}}$, $X_{\delta_{LONG}}$ and $Y_{\delta_{LAT}}$, M_u and L_v, M_q and L_p, $M_{B_{1s}}$ and $L_{A_{1s}}$, and $M_{\delta_{LONG}}$ and $L_{\delta_{LAT}}$.

Tail rotor force increments also contribute to the Y force and rolling moment derivatives. First order tail rotor effects are due to changes in the axial thrust of the tail rotor rather than to changes in the direction of the tail rotor thrust. Lateral velocity of the helicopter (Δv) produces a relative axial velocity at the tail rotor analogous to the relative axial velocity at the main rotor produced by a downward velocity of the helicopter (Δw). The change in tail rotor thrust due to Δv is a source of the side force due to lateral velocity derivative (Y_v). This tail rotor thrust increment contributes to the rolling moment due to lateral velocity derivative (L_v) depending on the height of the tail rotor relative to the aircraft CG.

Aircraft roll rate also induces axial velocity at the tail rotor if the tail rotor is displaced from the roll axis. The resulting tail rotor thrust increment effects the roll moment due to roll rate derivative (L_p), as well as the side force due to roll rate derivative (Y_p).

The side force due to tail rotor pitch angle derivative $(Y_{\theta_{TR}})$ results from the change in tail rotor thrust with blade pitch angle, and is analogous to the Z_{θ_C} derivative of the main rotor. There is a corresponding rolling moment due to tail rotor pitch derivative $(L_{\theta_{TR}})$ whose magnitude depends on the displacement of the tail rotor from the roll axis. Lateral control derivatives can be referred to lateral cyclic pitch angle (A_{1s}) and tail rotor pitch angle (θ_{TR}), or to the cockpit lateral control (δ_{LAT}) and pedal control (δ_{PED}).

8.3.7 Yawing Moment Equation Derivatives

When the main rotor thrust vector is aligned with the CG at the initial zero airspeed trim condition, the derivatives in the yawing moment equation are primarily due to the tail rotor. The tail rotor thrust increment due to lateral velocity produces a yawing moment about the CG, and contributes to the yaw moment due to lateral velocity derivative (N_v). Similarly, a yaw rate about the CG results in an axial velocity at the tail rotor and a tail rotor thrust increment which produces a yawing moment about the CG. This is the primary source of the yaw moment due to yaw rate derivative (N_r) in the case of the single rotor helicopter.

If the tail rotor is displaced from the roll axis, a roll rate causes an axial velocity change at the tail rotor and a corresponding tail rotor thrust increment. This results in an increase in the yaw moment due to roll rate derivative (N_p), which introduces coupling between the yawing and rolling moment equations.

The main rotor force increments which effect the derivatives in the Y force equation also effect the derivatives in the yawing moment equation when the main rotor is displaced from the CG. Under these conditions, there is a yaw moment due to lateral cyclic pitch angle derivative ($N_{A_{1s}}$). However, the predominant yawing moments used for directional control in hovering flight are introduced by changes in tail rotor pitch via pedal control producing yaw moment due to tail rotor pitch angle ($N_{\theta_{TR}}$). Directional control derivatives can be referred to lateral cyclic pitch angle (A_{1s}) and tail rotor pitch angle (θ_{TR}), or to the cockpit lateral control (δ_{LAT}) and pedal control (δ_{PED}).

8.3.8 Trim and Static Stability

A number of aerodynamic effects influence trimmed control positions in hovering and low airspeed flight both in and out of ground effect (IGE, OGE). The important effects are discussed to provide background concerning possible sources of difficulty in obtaining and holding trimmed flight conditions. It is difficult to generalize the relative importance of these and other aerodynamic effects encountered with various helicopter configurations.

8.3.8.1 HOVER ATTITUDE

Trim changes in low airspeed flight are referred to the zero airspeed hovering condition. The hovering helicopter shown in Figure 8.1 does not have any flight path velocity, yet the aircraft is trimmed with the left side down. This example illustrates the roll attitude required so the horizontal components of the force vectors sum to zero.

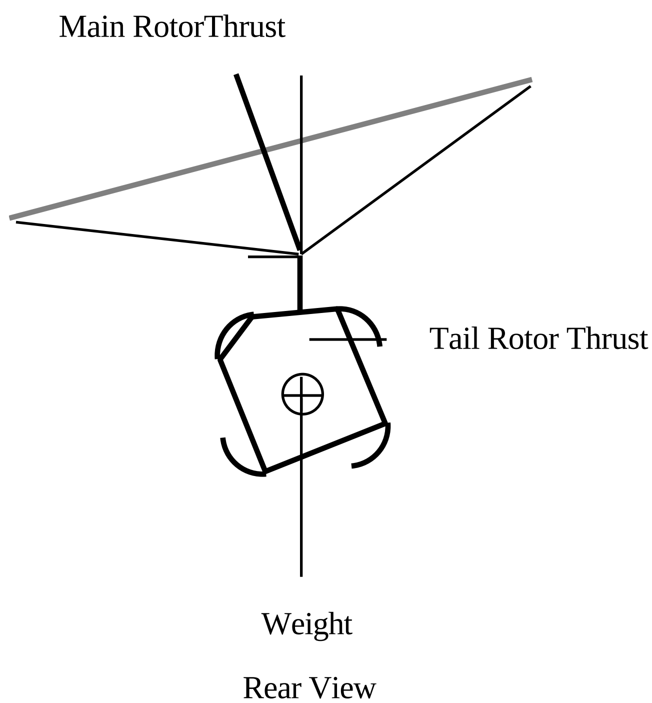


Figure 8.1
Forces on a Single Rotor Helicopter in Zero Wind Equilibrium Hover

The tail rotor thrust of a helicopter with a counterclockwise rotating main rotor is directed towards the right to counteract the clockwise torque imparted to the fuselage. Consequently, the aircraft drifts to the right unless the rotor is tilted to the left. A general zero airspeed equilibrium hover is shown in Figure 8.2 for a configuration with blade flapping hinge offset and the tail rotor above the CG.

In Figure 8.2, the rotor tilts to the left relative to the shaft to develop a rolling moment about the CG to balance the rolling moment due to the tail rotor. The presence of flapping hinge offset reduces the flapping angle required to obtain rolling moment equilibrium. A left lateral main rotor thrust component is required, as shown in Figure 8.1, to prevent sideways drift due to tail rotor thrust.

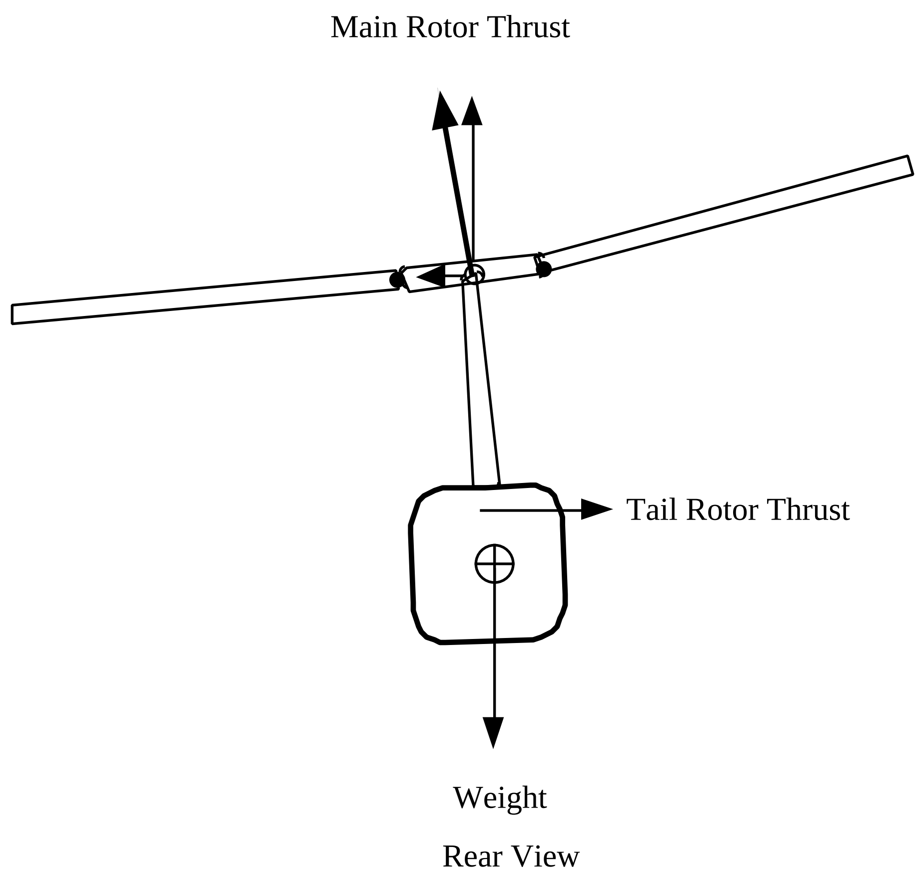


Figure 8.2
Equilibrium Hover - General Case

The longitudinal and lateral control positions required for zero airspeed hover equilibrium are influenced by moments due to aerodynamic loads on the tail rotor, tail boom, and stabilizer produced by rotor downwash and by ground effects.

8.3.8.2 TRIM AND POWER CHANGES DURING TRANSLATIONAL FLIGHT

When a helicopter hovering OGE translates into low airspeed flight, adjustments are required in cyclic pitch, collective pitch, engine power, and tail rotor thrust to maintain trim. Cyclic pitch is used to compensate for rotor blowback and to keep the TPP approximately horizontal and the main rotor thrust vector approximately vertical. The small increase in parasite drag at low translational velocity is balanced by a small inclination of the thrust vector.

The speed change results in an increase in thrust if the collective pitch is held fixed. After lowering the collective pitch to maintain constant thrust and prevent the helicopter from climbing, there is a corresponding decrease in the blade section induced drag and the torque required to drive the rotor. Consequently, engine power is reduced to prevent an increase in rotor speed. Less tail rotor thrust is required to counter the decreased main rotor torque. Pedal inputs are used to keep this translational velocity to yaw coupling from changing the heading of the aircraft. The tail rotor pitch adjustments made by the pilot depend on whether the translational velocity is forward, aft, right, or left, since these motions have different effects on tail rotor thrust as well as different power requirements.

The effect of trim and power changes in translational flight result from changes in the average downwash velocity. A qualitative argument, according to the vortex theory, states the downward velocity at the rotor plane results from a superposition of the velocities induced by the system of vortices which are generated by the rotor. In forward flight, the higher velocity of the rotor relative to the undisturbed air mass causes the wake vortices to move away from the rotor more rapidly and, therefore, induce smaller velocities at the rotor.

Figure 8.3 depicts flow conditions at a typical blade section of a rotor in hovering or low airspeed flight. The blade section is shown rotating in a horizontal plane. The downward perpendicular flow component (u_P) is entirely due to the induced velocities caused by the vorticity in the wake of the rotor. The transverse velocity component is approximately equal to the velocity due to shaft rotation. The resultant of these two components is the total relative velocity at the blade section which has a inflow angle (φ) relative to the horizontal rotor disk.

The effective angle of attack (α_{eff}) of the blade section is the difference between the average blade pitch angle and the section inflow angle. The cyclic pitch used to prevent rotor blowback is comparatively small at low translational velocities and is omitted for simplicity. Average lift on the section is proportional to its α_{eff}. When the aircraft attains a small translational velocity, the effective angle of attack becomes larger since u_P and φ become smaller with the reduction in downwash velocity induced by the wake vortices. Thus, rotor thrust increases at constant collective. Therefore, the collective control is lowered to keep the main rotor thrust constant.

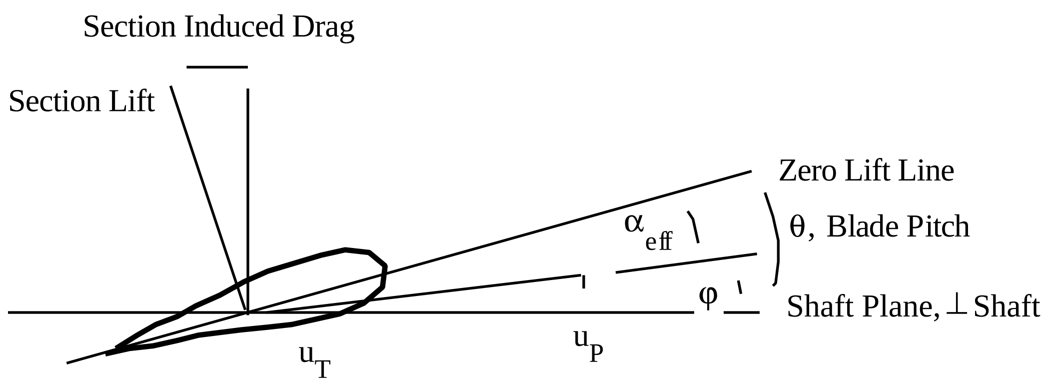


Figure 8.3
Flow Conditions at a Rotor Blade Section in Low Speed Flight

The section lift is perpendicular to the section relative velocity and its horizontal component in the plane of rotation is the section induced drag. When φ becomes smaller in translational flight, the lift vector tilts forward giving a smaller section induced drag. The section profile drag remains approximately the same in hovering and low airspeed translational flight. Consequently, the aerodynamic torque due to the combined effect of blade profile and induced drag is smaller at low translational speeds and the engine torque and tail rotor reaction torque is reduced to trim the aircraft.

8.3.8.3 ROTOR BLOWBACK AND NONUNIFORM INFLOW DISTRIBUTION OVER ROTOR DISK

To trim the helicopter when going into forward flight, the pilot uses forward control displacement which applies longitudinal cyclic pitch for overcoming the aft flapping or blowback of the rotor. Similarly, in flight to the right, he must hold right control to compensate for rotor blowback which, in this case, tilts the TPP to the left. Both of these effects are due to the higher dynamic pressure encountered by the advancing blade in translational flight. The compensating control inputs are direct (forward or aft cyclic control is used to control longitudinal flapping and right or left cyclic control is used to control lateral flapping).

The TPP tilts because of the nonuniform inflow distribution over the rotor disk which results from translational flight. Coupled control inputs are required to overcome this tilt and maintain trim. Figure 8.4 indicates how blade coning causes an effective variation in inflow over the rotor disk when the aircraft is in translational flight. The relative velocity (V) has an upward component at the forward blade position giving an increase in blade angle of attack (α) but it has a downward component of flow at the aft blade position giving a decrease in α. The resulting flapping response takes place with an approximate 90˚ phase lag so the blade reaches a maximum flapping angle while rotating aft on the left side of the aircraft. A decreased load on the aft blade causes the blade to flap down as it rotates forward on the right hand side of the aircraft. The net effect of a forward translational velocity increment is the TPP tilts to the right unless trimmed out by the pilot with left control displacement. A similar effect is obtained in sideward flight. A right velocity produces an increase in α_{eff} on the blade due to coning when on the right hand side of the aircraft. This causes the blade to flap up as it moves over the nose. Forward control displacement is required to compensate for this coupling effect caused by right sideward flight.

As the translational velocity of the helicopter is increased, the velocities induced by the trailing vortex system results in dissymmetries of the inflow at the TPP. This results in first harmonic blade loads tending to tilt the TPP similar to the loads caused by coning. The effects due to induced velocity variations are larger than those due to coning in the transition speed range. The induced velocity distribution is reasonably symmetric over the TPP of a hovering helicopter. However, as the translational speed is increased, the trailing vortex system below the helicopter is skewed aft. The successive rings of the helicopter tip vortex wake are closer to the aft than to the forward portion of the rotor disk and the downward flow induced at the aft edge of the rotor disk is larger than at the front edge. Although the nonuniform induced velocity distribution is quite complex, its principal effect on trim control position can be explained using the simple linear induced velocity distribution shown in Figure 8.5.

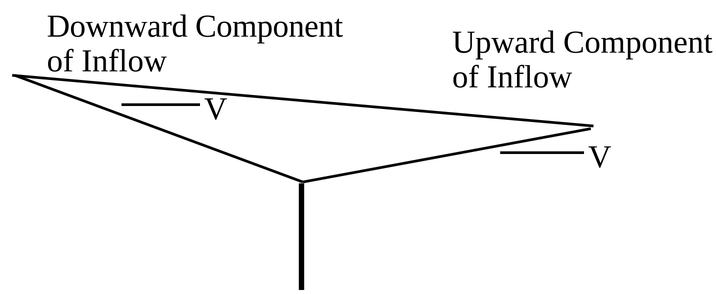


Figure 8.4
Effective Inflow Variation
in Translational Flight Due to Coning

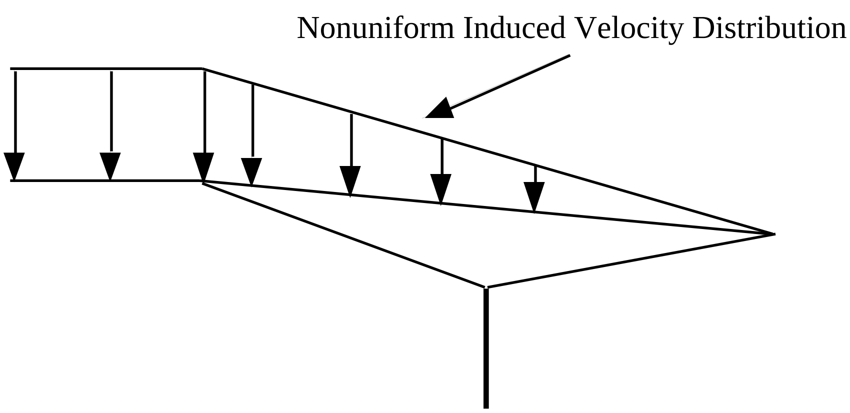


Figure 8.5
Fore and Aft Inflow Variation in Translation Flight

The downward induced velocity at the forward edge of the rotor is less than the average which means the nonuniform distribution produces an upflow at the front of the rotor disk. In the aft portion of the disk, the downward induced velocity is greater than the average, producing a downward component of inflow. Thus, the nonuniform induced velocity distribution in Figure 8.5 produces an inflow variation similar to the one caused by coning in Figure 8.4. This produces an upload on a blade in the forward position and causes the blade to flap up on the left hand side of the helicopter. Consequently, left control movement is required to prevent the TPP from tilting when there is a fore and aft induced velocity variation.

The critical nonuniform induced velocity conditions occur in the transition region. At higher speeds the TPP and thrust vector are inclined forward giving a forward horizontal force component to balance the higher parasite drag of the aircraft. Consequently, a component of translational velocity adds to the total downward inflow and the variation of the induced velocity over the rotor disk is a smaller percentage of the total downward flow through the rotor disk.

8.3.8.4 DOWNWASH IMPINGEMENT ON FUSELAGE, TAIL, AND HORIZONTAL STABILIZER

When the pilot goes from one stabilized condition to another in hovering flight, low airspeed forward or aft flight, climb, or descent, changes occur in the fuselage attitude, TPP tilt relative to the shaft, and the direction of the relative velocity vector. Corresponding shifts in the main rotor wake position change the downwash impingement on the fuselage, tail boom, and horizontal stabilizer. The aerodynamic moments applied to the aircraft as a result of changes in downwash impingement might cause large and discontinuous control position changes. These effects are very dependent on the particular configuration involved.

A fixed stabilizer located in an aft position could be clear of the rotor downwash in hovering flight. However, in low airspeed forward flight, the downwash trails further aft where it can influence the stabilizer, causing an abrupt nose up pitching moment and requiring forward cyclic control to trim the aircraft. If the stabilizer is located at a forward position, a download acts on the stabilizer due to downwash impingement both in hovering and low airspeed forward flight. However, in aft flight, the downwash moves forward producing a nose down pitching moment requiring aft cyclic control to maintain trim. A high stabilizer position in the T-tail design eliminates most trim problems due to interference but could be subject to downwash impingement in forward or climbing flight.

8.3.8.5 TAIL ROTOR THRUST CHANGES IN SIDEWARD FLIGHT

Figure 8.6 depicts some of the factors causing tail rotor thrust changes in low airspeed sideward flight and the compensating pilot control positions required to maintain trim. Left pedal is forward in a hover to provide tail rotor pitch and thrust to the right which balances the main rotor torque. No cyclic control displacement is shown while hovering. The tail rotor is at the same height as the CG. In this case, the thrust vector is aligned with the shaft and the aircraft is rolled to the left so the tail rotor thrust is balanced by the lateral component of the weight force.

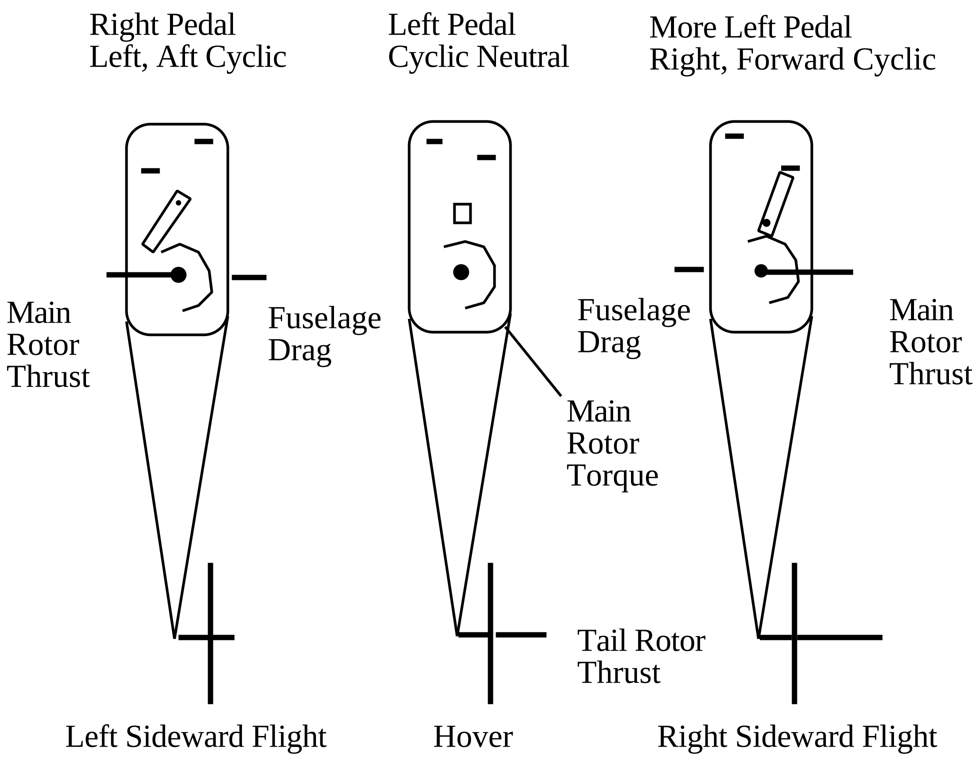


Figure 8.6
Tail Rotor Thrust Changes in Low Speed Sideward Flight

Lateral velocity produces changes in the α_{eff} and thrust of the tail rotor corresponding to the main rotor changes obtained with vertical velocity. The tail rotor thrust change is in the same direction as the change in relative sideward velocity, creating a damping force similar to the vertical damping force generated by the main rotor when given a vertical velocity. Both of these rotor damping forces can be generated without forward motion of the aircraft.

A right sideward velocity causes a relative velocity to the left of the tail rotor and incremental tail rotor thrust to the left. This left tail rotor thrust produces a stable yawing moment, yawing the aircraft to the right into the relative wind. A left pedal input is required to increase tail rotor pitch and tail rotor thrust to the right to prevent this yawing motion. Left sideward velocity causes an increase in tail rotor thrust to the right which is balanced by right pedal input.

However, several factors lead to unsymmetrical directional control for sideward flight to the left or right. A reduction in main rotor power required and torque is obtained with translational velocity to the right or left. Therefore, less tail rotor thrust is required to react main rotor torque and results in a right pedal increment in both cases.

Another source of control dissymmetry for sideward flight is the possibility of the tail rotor entering the vortex ring state. This occurs in left sideward flight with a high enough relative velocity to the right to exceed the induced velocity of the tail rotor to the left. In this case, an abrupt loss in tail rotor thrust is obtained and a left pedal input is required to obtain trim.

Figure 8.6 indicates changes in trimmed cyclic control positions as well as changes in pedal position in going into sideward flight. Right cyclic control deflection is shown for right sideward flight. The lateral cyclic compensates for the lateral blowback of the rotor which tilts the TPP to the left. Similarly, left cyclic control displacement is required in left sideward flight to prevent the TPP tilting to the right. Some change in trimmed roll angle is required so lateral components of the gravity force balance fuselage or tail boom drag caused by sideward velocity.

In right sideward flight, there is an upward flow increment for a blade on the right hand side because of coning and lower induced velocity at the right edge of the rotor disk. These same effects cause a downward flow on the blades rotating through the left hand side The resulting increase in blade lift on the right hand side and decrease on the left hand side causes the blades to flap up at the front of the disk and down at the aft. Forward control displacement is used to compensate for this flapping. The opposite effect in left sideward flight is compensated by aft cyclic displacement.

8.3.8.6 INTERFERENCE OF MAIN ROTOR TIP VORTICES WITH TAIL ROTOR AND FIXED STABILIZER SURFACES

Tail rotor thrust changes occur when the aircraft attitude changes due to variations in its CG and flight path. Corresponding changes in control position are required to maintain trim. The moments due to the rotor downwash are not only caused by the general change in flow direction at the stabilizer and tail rotor, but also by the strong local velocity fields induced by vortices carried with the flow.

If the trailing vortex passes below and parallel to the tail rotor the induced velocity on the tail rotor is to the left. This decreases the α_{eff} of the blades of a conventional tail rotor whose thrust is directed to the right. Thus, the tail rotor thrust is increased with a left pedal input to maintain yawing moment equilibrium. If the trailing vortex passes above the tail rotor, the direction of the induced velocity at the tail rotor is to the right increasing the right tail rotor thrust. Right pedal input is required to trim the aircraft.

In general, the effects of blade tip trailing vortices are irregular and are difficult to predict or explain. The interaction is dependent on the direction of rotation of the tail rotor and whether it is a tractor or pusher type. In any case, the test pilot should document any combination of control positions and maneuvers leading to objectionable changes in trim.

8.3.8.7 REDUCTION OF POWER REQUIRED IGE

A change in aerodynamic flow pattern around the helicopter takes place when the helicopter transitions from OGE to IGE. When operating IGE, the downwash flow direction changes near the ground. The downwash region expands in diameter as it moves downward and the flow moves outward parallel to the ground. The induced velocity through the rotor is reduced because of the ground boundary conditions. The reduction in induced velocity produces an increase in thrust. To counter the increase in thrust, the pilot lowers the collective. A corresponding pedal input is made to counter the reduced torque on the fuselage. There is usually a change in the pressures acting on the bottom of the fuselage IGE. These usually do not have a significant effect on the required thrust but can introduce pitching moments affecting the trim of the aircraft.

8.3.8.8 INFLUENCE OF GROUND VORTICES ON TRIM CONTROL POSITIONS

Helicopters require less induced power when operating IGE than OGE. Consequently, less tail rotor thrust is required to react main rotor torque while flying IGE. A second source of trim change during IGE operations is the ground vortex formed by the downwash of the main rotor. Figure 8.7 depicts flight situations where loads induced by ground vortices produce trim changes.

The downwash for the hovering helicopter shown in Figure 8.7 curves outward when it approaches the ground where the vertical velocity is zero. The boundary of the downwash region is defined by the helical paths of the trailing vortices originating at the tips of the individual blades. These tip vortices are modeled by a system of horizontal vortex rings in the case of a lightly loaded rotor.

Figure 8.7 also shows a sequence of flow conditions which are obtained as the helicopter goes into forward flight. The blade tip vortex rings which are carried downward with the downwash start to form a ground vortex whose forward edge is at a stagnation point near the ground and ahead of the rotor disk. Starting from this stagnation point, the ground vortex curves around the right and the left sides of the helicopter and extends aft on both sides into the wake of the aircraft. Successive blade tip trailing vortex rings are stretched and distorted as they are carried downward from the TPP towards the ground. The front sections of these rings tend to strengthen the ground vortex, while the aft sections are carried by the flow into the remote wake behind the helicopter where their influence becomes negligible.

At a given forward speed, the ground vortex has a steady position relative to the helicopter. As the speed is increased, the ground vortex moves closer to the helicopter and induces velocities through the rotor disk shown schematically in Figure 8.7. The resulting blade loads effect the collective and lateral cyclic control positions required for trim.

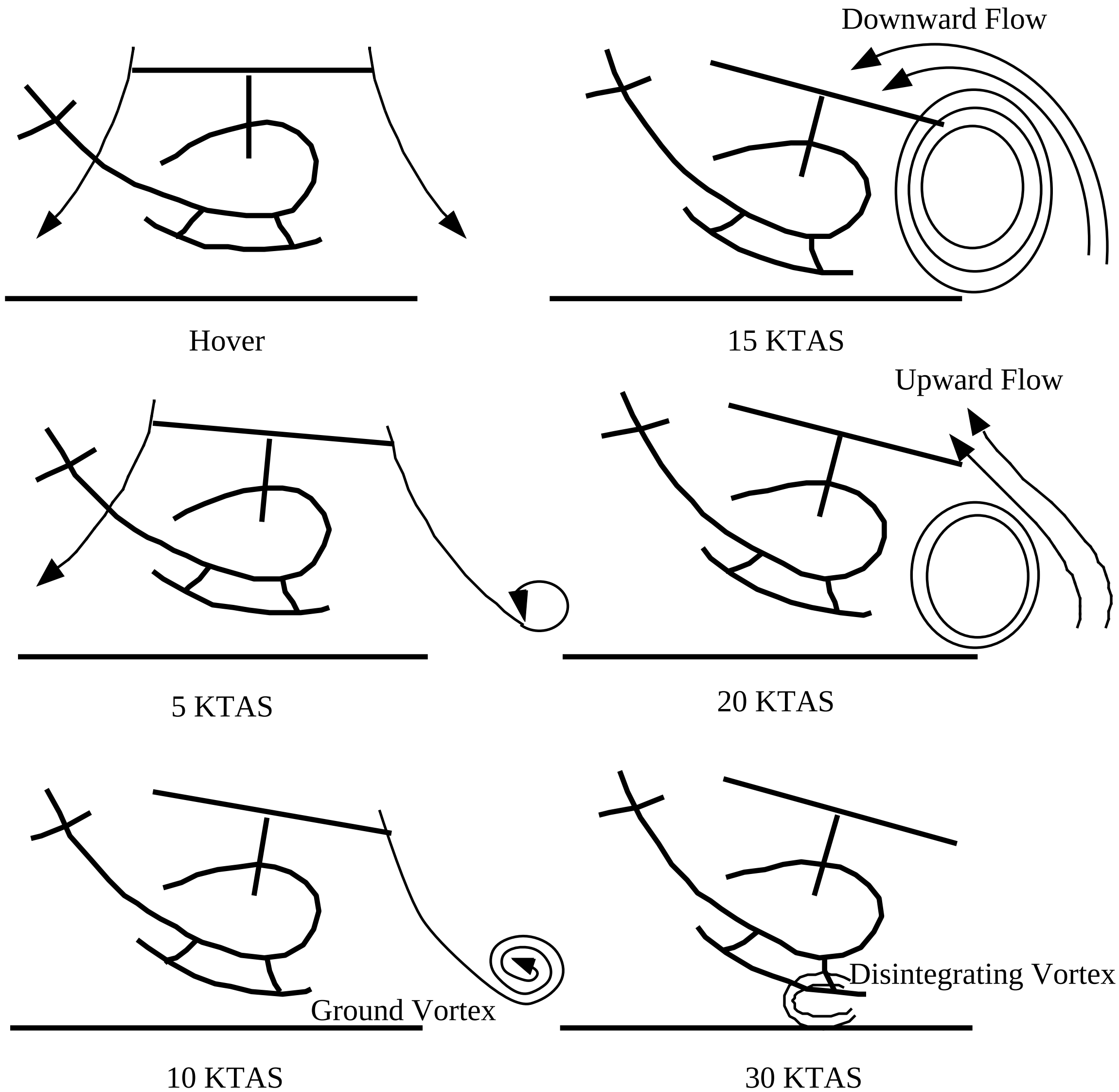


Figure 8.7
Effect of Ground Vortex on Inflow Patterns

The required induced power is reduced in going from hovering to forward flight at heights where ground effect is negligible. However, the curve for a rotor height of 0.3 rotor diameter on Figure 8.8 indicates an increase in required power during the transition to forward flight until a forward speed of 25 kn is attained. The downflow through the TPP induced by the ground vortex as the speed is increased reduces the effective angles of attack of the main rotor blades requiring an increase in collective pitch to maintain the same rotor thrust. Furthermore, the vector addition of the downflow tilts the relative velocity vector at each blade section downward and the section lift vector aft. Consequently, there is an increase in blade section induced drag and required engine torque when the collective pitch is adjusted to maintain a constant thrust. When the helicopter flies over the ground vortex at about 35 kn, its strength is dissipated and the effects due to ground vortex induced velocities become small at higher forward speeds.

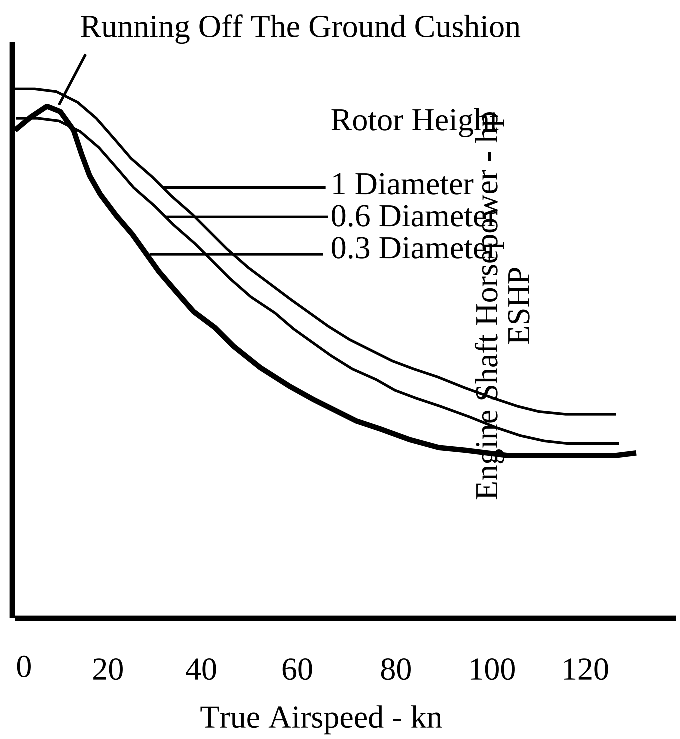


Figure 8.8
Ground Effect in Forward Flight

Left cyclic control displacement is required for lateral trim during the transition from hovering to forward flight when performed OGE. The lateral flapping angle is a function of the advance ratio. Increased forward speed produces increased lateral flapping and right tilt of the TPP. Left cyclic control introduces a compensating cyclic pitch change.

However, when a helicopter flying IGE goes into forward flight, the downward induced velocities produced by the ground vortex over the forward part of the TPP tend to balance the decreases in this region caused by the aft skewing of the trailing vortex system. Consequently, a more uniform downwash distribution is obtained and less left cyclic control displacement is required to maintain trim as forward speed is increased. Figure 8.9

indicates a sudden left control displacement is required close to the speed at which the helicopter flies over the ground vortex because there is a sharp reduction in the velocities induced by the ground vortex at the rotor.

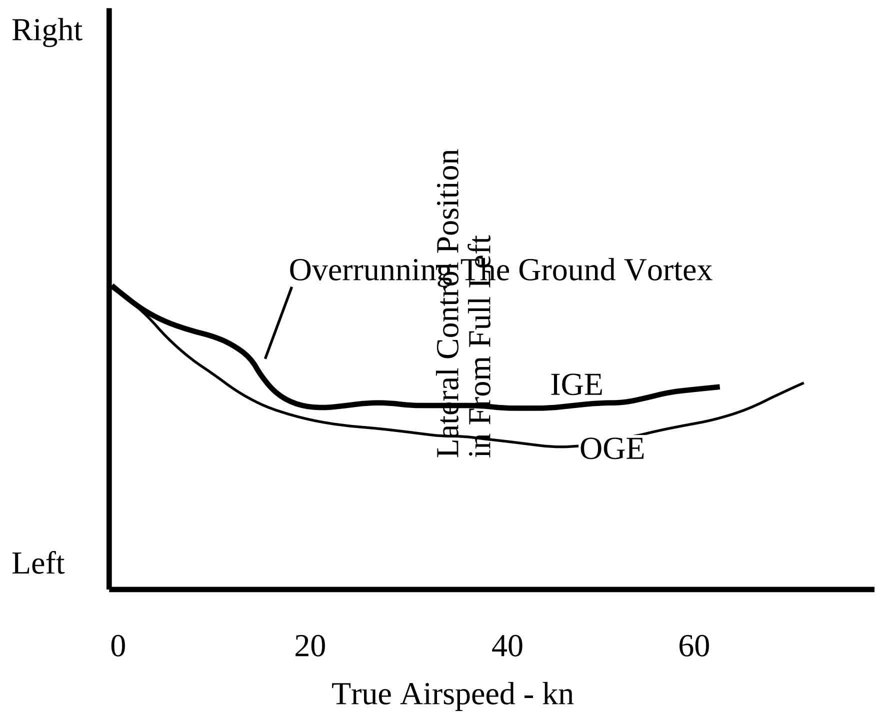


Figure 8.9
Effect of Ground Vortex on Lateral Trim Control Position

The same type of ground vortex is generated in rearward flight as in forward flight except it is located aft of the aircraft instead of ahead of it. The ground vortex obtained in rearward flight induces downward velocities at the horizontal stabilizer and result in a nose up pitching tendency. Forward cyclic control displacement is required to maintain trim. If the aircraft had sideward as well as rearward velocity, the ground vortex would induce velocities perpendicular to the plane of the tail rotor. These could result in right or left yawing depending on the direction of the side velocity component of the helicopter.

Another potential source of interference in rearward flight is the main rotor blades interference caused by the wake of the tail rotor. This effect depends on the height of the tail rotor and the pitch attitude of the aircraft.

8.3.8.9 STATIC STABILITY

Low airspeed static stability tests are influenced by the various aerodynamic phenomena previously discussed for trim control positions. These effects can result in nonlinear variations in control position with airspeed which are both difficult to predict and explain. A further difficulty in low airspeed static stability testing is finding a satisfactory method for measuring the airspeed at the test conditions. It is difficult to obtain accurate results using a ground pace vehicle since most test conditions involve a climb rate as well as a horizontal velocity. Furthermore, self-contained airspeed indicators accurate at low airspeeds are available but not installed in many helicopters.

Figure 8.10 provides typical results from a low airspeed longitudinal stability test where the longitudinal control position variation with forward speed is used as an indication of static stability.

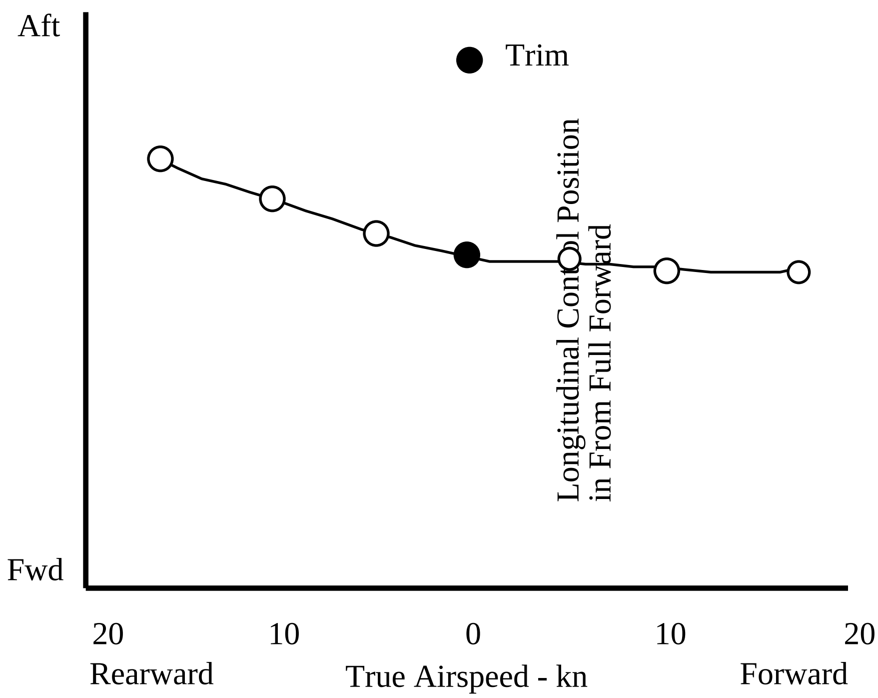


Figure 8.10
Typical Static Stability for a Hovering Single Rotor Helicopter

The aircraft was originally trimmed at zero airspeed. The pilot used longitudinal control to stabilize the aircraft at a series of forward and aft airspeeds. Testing was conducted with power held constant. Inflow velocity changes introduced during static stability flight testing tend to cause variations in the thrust developed for a fixed collective setting. Depending upon the aircraft power management system, this variation in thrust produces various combinations of rotor speed and engine output power with the collective fixed. Although changes occur, the collective fixed method is used to eliminate one additional variable.

The pilot suppresses lateral and directional responses when performing static longitudinal stability tests as he does during longitudinal maneuvers. Similarly, longitudinal control displacement is used to suppress pitching motion in the investigation of low airspeed lateral and directional stability similar to steady heading sideslips.

The pitching moment equation for hovering flight reduces to the following for a stabilized flight condition:

$$M_u u + M_{\delta_{LONG}} \delta_{LONG} = 0$$

eq 8.7

The static longitudinal stability as indicated by the longitudinal control position variation with airspeed shown below is proportional to M_u:

$$\frac{\Delta\delta_{LONG}}{\Delta u} = - \frac{M_u}{M_{\delta_{LONG}}} = - \frac{\text{Speed Stability Derivative}}{\text{Sensitivity}}$$

eq 8.8

Where:

δ_{LONG}	- Longitudinal control
$M\delta_{LONG}$	- Pitch moment due to longitudinal control
M_u	- Pitch moment due to longitudinal velocity
u	- Translational velocity component along x axis.

M_u is entirely due to the pitching moment produced by rotor blowback in the case of an idealized hovering helicopter. However, changes in downwash impingement on the fuselage and the interaction effects involving the blade tip vortices can cause large shifts in the control positions required to stabilize when speed is changed.

In forward flight, the variation in control position with velocity depends on the lift and pitching moment variation with angle of attack. A speed change from a trimmed forward flight condition produces a lift increment and an angle of attack adjustment is required to stabilize the aircraft. The corresponding pitching moment due to angle of attack change is balanced by a longitudinal cyclic pitch input and this effects the observed forward flight static stability.

The situation is simplified near a zero airspeed hover where the vertical forces due to speed change are negligible so little coupling is obtained between forward and vertical motions. However, the reduction in the average induced velocity of the rotor, obtained as the speed is increased, causes the aircraft to develop greater lift and a vertical velocity.

A plot of control position versus airspeed, such as shown in Figure 8.10, is useful in considering handling qualities even if the causes for the nonlinearities in the curve are unknown. In this example, the data indicate substantial forward speed changes are experienced with little forward cyclic control required. Conversely, small rearward speed changes require significant aft displacement of the cyclic control.

Often control position versus airspeed plots are not obtained for the low airspeed range because of the difficulties in airspeed measurement. However, it is important to obtain at least a qualitative evaluation of static stability for all directions of flight from a zero airspeed hovering condition. These data are particularly valuable for determining the effect of inadvertent relative wind change caused by ambient winds and gusts, particularly when hovering over a spot. A head-on gust results in the aircraft experiencing an airspeed greater than trim. In the example given, very little forward longitudinal control is required to compensate for a 5 kn speed change. The slope is almost neutral or flat. A neutral slope means that if the wind did change, as on a light gusty day, the aircraft could be stabilized without large force or displacement requirements. Tests have indicated precision hover handling qualities in gusty weather are degraded by an increase in the speed stability derivative.

8.3.9 Turn on a Spot

A turn over a fixed spot on the ground is a frequently required operational task and is evaluated in flying qualities tests. Some differences may be noted when comparing these results with qualitative and quantitative results obtained during sideward and rearward flight evaluations.

8.3.9.1 DIRECTIONAL CONTROL

One significant difference between sideward flight and a turn on the spot is the power required. In sideward flight, a significant amount of collective to yaw coupling is caused by the change in power required and main rotor torque associated with the lateral velocity change. There is not such a significant main rotor torque change during a turn on a spot. The collective to yaw coupling is considerably less for a turn on a spot.

The directional control requirements of a single rotor helicopter are normally of primary interest during a turn on a spot in a wind. The most important characteristics are revealed by the shape of the control position curve and the control margin at the critical azimuth. Three example plots of directional control variation with sideslip angle are presented in Figure 8.11.

Figure 8.11 (a) depicts a fairly typical pattern with control reversals occurring at a relative wind of 90˚ and 270˚. Maximum left pedal is required at 90˚ when the relative wind from the right decreases the α_{eff} and thrust of the tail rotor. Maximum right pedal is required at 270˚ where the wind increases the tail rotor thrust. The significance of the reversals from a handling qualities standpoint is the requirement for the pilot to use nonlinear pedal displacement in controlling the aircraft. In addition, should the aircraft turn past the azimuth corresponding to the reversal, the pilot must operate in an unstable control gradient region. Operations on either side of the reversal are easier than operations at the point of reversal.

Critical azimuth is the heading of the relative wind which causes a minimum control margin in any axis. Control margin is defined as the moment available to maneuver the aircraft beyond that required to hold the azimuth position; however, it is satisfactory to report the control position margin as long as the data are properly labeled and discussed. A critical azimuth exists for all controls, even the collective if total θ_C available is a problem. In Figure 8.11 (a), the critical azimuth is 90˚ (right cross wind).

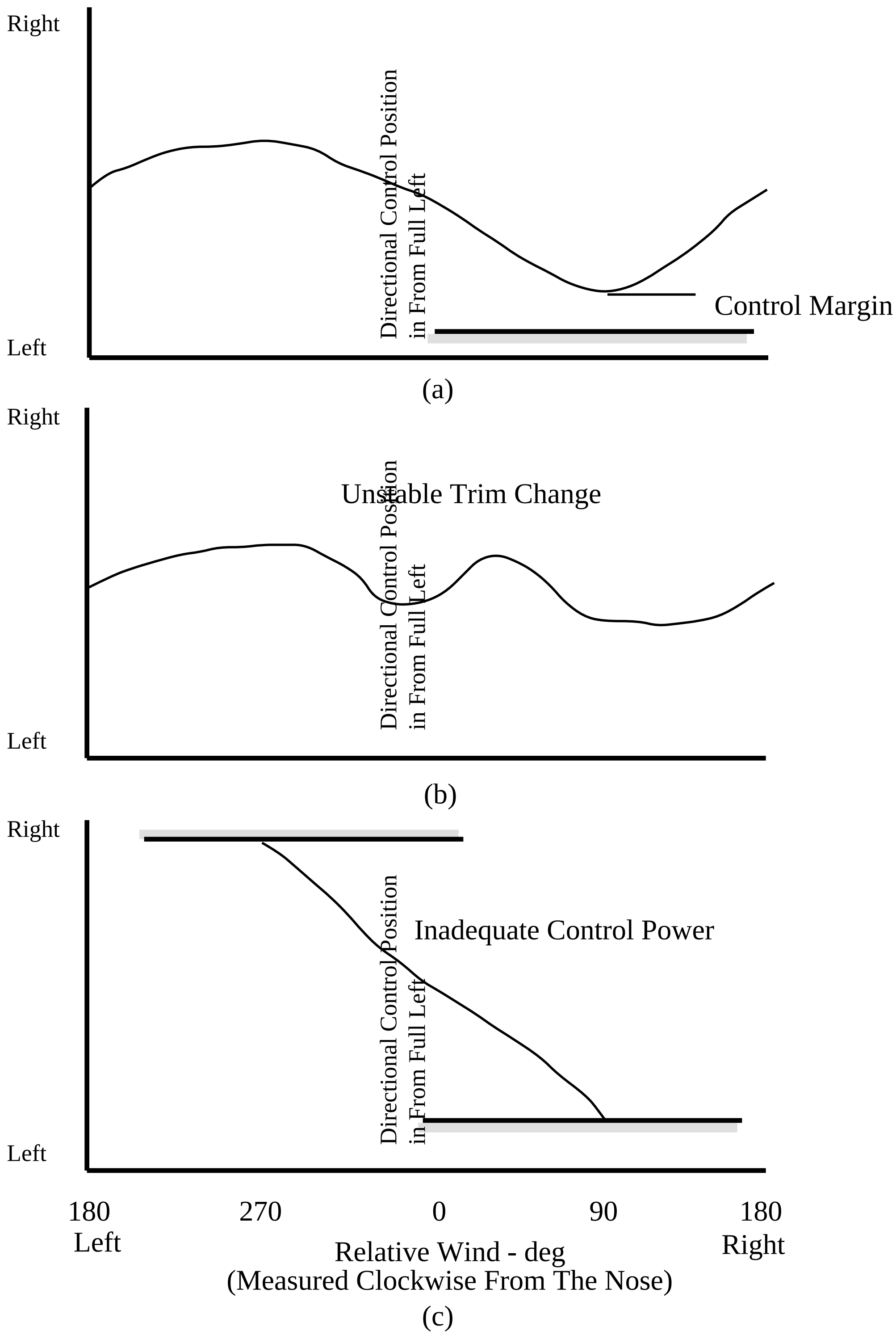


Figure 8.11
Directional Control Position Characteristics During a Turn on a Spot for a Single Rotor Helicopter

In Figure 8.11 (b), an undesirable situation exists because of the unstable control position pattern required when headed into the wind. Right pedal is required to hold small heading changes to the right. This might be caused by aerodynamic interference by the fuselage or distortion of the flow field near the tail rotor by the rotor tip vortices. The aircraft must be operated significantly out of the wind line before the pilot workload is reduced. Such a characteristic is occasionally observed and the test pilot should be alert for such an instability near the zero azimuth wind line.

In Figure 8.11 (c), the utility of the aircraft is limited by the directional control power and static directional stability of the aircraft. Full pedal is required prior to reaching a 90˚ turn away from the wind line. A critical azimuth then exists at about 80˚ and 285˚. The critical azimuth of Figure 8.11 (a) occurs at an azimuth of about 90˚, but in this case a control displacement margin remains (approximately 10%).

A serious control problem can occur in turns on a spot in winds with enough velocity to cause the tail rotor to go into the vortex ring state. A nose right yaw rate further increases the relative velocity at the tail rotor and the possibility of obtaining a vortex ring state of the tail rotor. The loss of tail rotor thrust at the onset of the tail rotor vortex ring state increases the right yaw rate.

8.3.9.2 BANK AND HEIGHT CONTROL

During a turn on a spot, the lateral side force of the fuselage, tail rotor, and main rotor develop as a function of the ambient wind and rate of turn. Lateral control is required to overcome the tilt of the TPP due to the lateral speed stability characteristic. In addition, lateral control is required to stabilize lateral motion of the aircraft. The aircraft is banked into the lateral relative wind introduced during the turn to achieve side force equilibrium by balancing the contributions of the fuselage/tail rotor with a horizontal component of the thrust vector. If the aircraft is yawed 90˚ with respect to the wind with the fuselage level, the aircraft accelerates downwind. The importance of the side force characteristic on handling qualities becomes obvious during such a yawing maneuver. If the aircraft has small side force characteristics, it is easy to maintain position over the ground during a 90˚ turn from the wind line. Large side force characteristics, require increased pilot effort to maintain position over a spot.

Height control is significant during operations requiring large bank angles. The orientation of the thrust vector to the vertical is a source of difficulty. Any change in rotor thrust introduces both a vertical and a lateral thrust component. When power is added to maintain height, the aircraft accelerates laterally. This lateral acceleration is instinctively canceled by the pilot through a decrease in bank angle. The decrease in bank angle reorients the thrust vector to a more vertical position. This vertical position results in an increase in the vertical thrust component, possibly causing over control in height.

8.3.9.3 LONGITUDINAL CONTROL

A helicopter turning over a spot in a wind experiences a continuous change in the velocities along the longitudinal and lateral axes. A turn away from the wind line decreases the velocity along the longitudinal axis. After a 90˚ heading change, there is no longitudinal velocity; however, the lateral velocity has increased. The longitudinal control required to make the turn from the wind line through 90˚ is a function of the longitudinal stability of the aircraft and the pitch coupling due to lateral velocity.

The forward control displacement to compensate for rotor blowback and fuselage drag when hovering headed into the wind is no longer required after a 90˚ turn. However, a longitudinal control displacement is used to maintain trim in the presence of the variation in induced drag obtained at low relative velocities. The induced velocity is largest on the downwind side of the TPP which decreases the angle of attack of the blades passing through this region. In a left turn from a head wind, the downwind side of the rotor disk is on the left hand side and a decrease in blade angle of attack here tends to make the blade flap downward when going over the tail boom and tilt the TPP aft. Consequently, forward control displacement is used to maintain trim. On the other hand, an aft control is required to overcome this coupling effect in a right turn.

Continuing a turn over a spot past 90˚ toward 180˚ out of the wind, places the aircraft in a rearward flight condition. The longitudinal control normally trims further aft following a turn downwind in a single rotor helicopter. Significant changes in control position can be expected since the change in relative velocity in the maneuver is twice the wind speed. Large trim changes may be required in abrupt 180˚ changes near the ground due to interactions with the ground vortex system.

8.3.10 Long Term Dynamic Stability

The dominant response in the long term for most helicopters is longitudinal. Therefore, the following discussion centers on the longitudinal axis. The lateral response is similar in nature to the longitudinal if the yaw motion is constrained. The specific characteristics of the lateral motion are governed by the lateral inertia and aerodynamic characteristics analogous to the longitudinal response discussed below.

The discussion of the longitudinal long term response is simplified by assuming coupling with lateral-directional motion is suppressed by lateral cyclic and directional control inputs. Lateral-directional responses are suppressed while investigating the longitudinal long period mode to simplify the interpretation of test results.

The simple pendulum is used as a model for examining the typical lightly damped long term mode of the hovering helicopter while neglecting the helicopter's moment of inertia. Figure 8.12 (a) shows an oscillating pendulum of length L, and Figure 8.12 (b) shows the corresponding motion of the helicopter.

The support string for the pendulum mass is pivoted at point P and the tension force in the support string is approximately equal to the weight of the pendulum $(T \cong mg)$. The x component of the tension force tends to return the mass, m, to the neutral position The period for the pendulum is given by the formula:

$$\text{Period} = 2\pi\sqrt{\frac{L}{g}} \qquad \textit{eq 8.9}$$

During small oscillations of the pendulum, the fore and aft velocity is proportional to the pitch rate ($\dot{\theta}$) about the pivot:

$$u \cong \dot{x} \cong L\dot{\theta} \qquad \textit{eq 8.10}$$

Where:

g	- Gravity
L	- Length
π	- Mathematical constant
$\dot{\theta}$	- Time rate of change in θ
u	- Translational velocity component along x axis
$\dot{x}$	- Time rate of change of x.

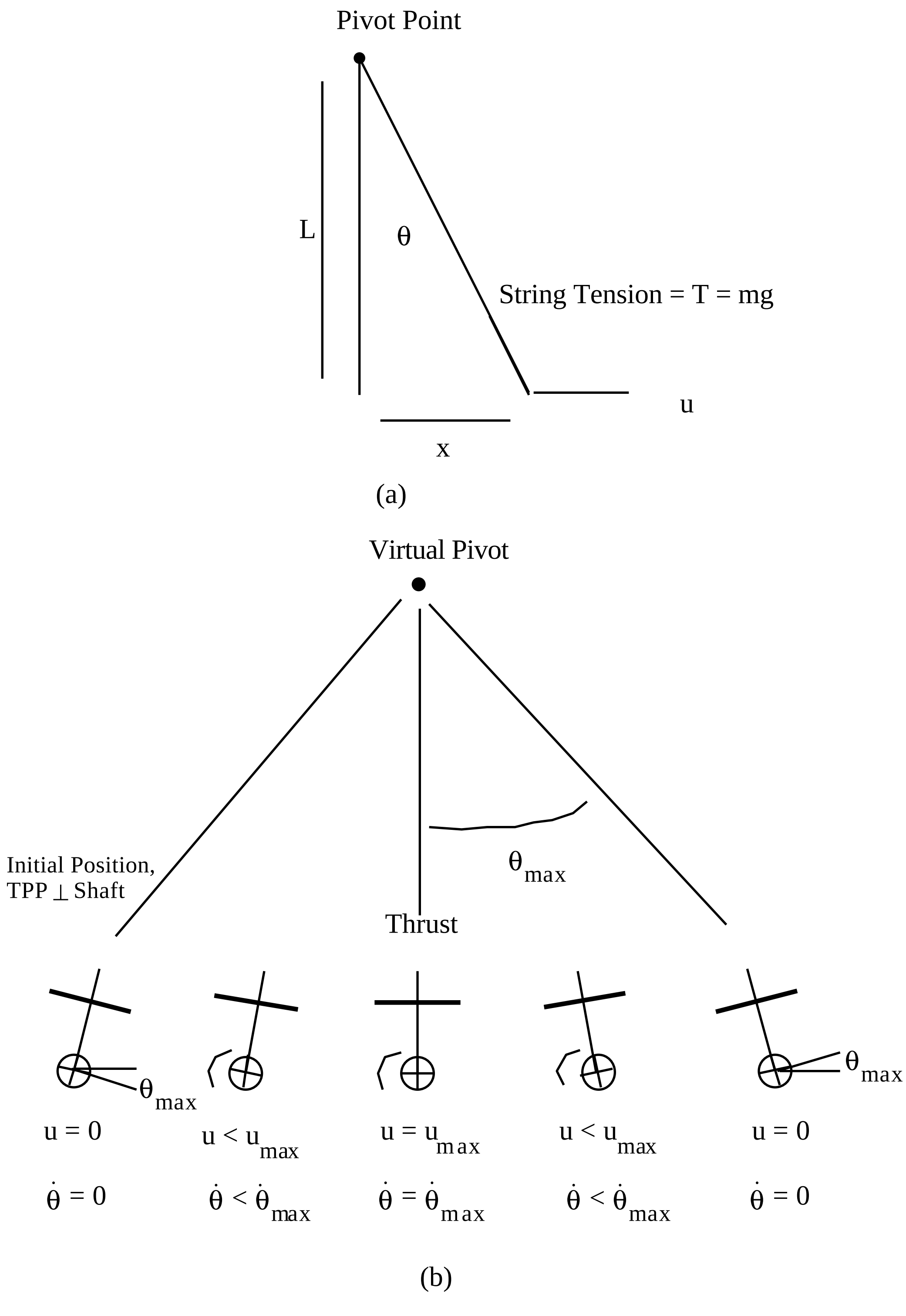


Figure 8.12
Analogy of Simple Pendulum and Helicopter Long Period

In Figure 8.12 (b), the string tension force is replaced by the rotor thrust force (T = mg) acting parallel to the shaft towards a virtual pivot point. At the initial position, the helicopter has a nose down pitch angle $\theta = -\theta_{max}$ and the pitch rate $(\dot{\theta})$ and forward velocity (u) are both zero. The shaft is tilted forward and the TPP is perpendicular to the shaft. This gives a forward tilt of the thrust vector, causing the helicopter to accelerate to the right.

When the helicopter develops a forward velocity, the TPP tilts aft. This produces a nose up pitching moment. However, if the helicopter has zero moment of inertia, any pitching moment causes an infinite nose up pitching acceleration.

Consequently, a nose up pitch rate develops instantaneously and the forward rotor tilt due to lag exactly cancels the aft tilt due to the forward speed increment. The TPP remains perpendicular to the shaft as indicated in the diagram and the pitch rate at any forward speed makes the total pitching moment on the helicopter equal to zero. That is:

$$\Delta M = 0 = \frac{\partial M}{\partial u} u + \frac{\partial M}{\partial \dot{\theta}} \dot{\theta} \qquad \textit{eq 8.11}$$

$$\dot{\theta} = \frac{\frac{\partial M}{\partial u}}{\frac{\partial M}{\partial \dot{\theta}}} u = -\frac{M_u}{M_{\dot{\theta}}} u \qquad \textit{eq 8.12}$$

The above linear relationship holds during long period oscillations for a configuration assumed to have zero moment of inertia. This long period motion can be described as the oscillation about a virtual pivot point of a pendulum with equivalent length:

$$L_{Equiv} = \frac{u}{\dot{\theta}} = -\frac{M_{\dot{\theta}}}{M_u} \qquad \textit{eq 8.13}$$

The period for this long term oscillation is obtained from the analogy with the simple pendulum:

$$\text{Period} = 2\pi\sqrt{\frac{L_{Equiv}}{g}} = 2\pi\sqrt{\frac{M_{\dot{\theta}}}{gM_u}} \qquad \textit{eq 8.14}$$

Where:

g	- Gravity
	- L_{equiv} Equivalent length
M	- Net moment about y axis, Pitch moment
$M_{\dot{\theta}}$	- Pitch moment due to time rate of change of pitch angle
M_u	- Pitch moment due to longitudinal velocity
π	- Mathematical constant
θ	- Pitch angle
$\dot{\theta}$	- Time rate of change in θ
u	- Translational velocity component along x axis.

In sinusoidal oscillations, the moment of inertia about the pitch axis is small at the low frequency of the long term oscillation. This is the basis for assuming it negligible compared to the aerodynamic pitching moment in the preceding discussion. However, a more accurate analysis, including non-zero moment of inertia, shows the rotor tilt due to velocity and pitch rate do not exactly balance. The residual pitching moments usually cause an unstable long period mode of the basic helicopter in hovering flight.

8.3.11 First Order Representation of Vertical Velocity, Pitch Rate, Roll Rate, and Yaw Rate Responses

The first order equations used to approximate the initial vertical velocity OGE, pitch rate, roll rate, and yaw rate responses in hovering flight are presented in Table 8.II.

Table 8.II
First Order System Parameters

	Normalized Damping	Time Constant	Control Variable	Control Sensitivity
Mass Damper Model	$\frac{B}{m} = -X_u$	$\frac{1}{(B/m)} = -\frac{1}{X_u}$	δ	X_δ
Vertical Velocity Response Mode	$-Z_w$	$\frac{1}{(-Z_w)}$	θ_C or δ_C	Z_{θ_C}
Pitch Rate Response Mode	$-M_q$	$\frac{1}{(-M_q)}$	B_{1s} or δ_{LONG}	$M_{B_{1s}}$ or $M_{\delta_{LONG}}$
Yaw Rate Response Mode	$-N_r$	$\frac{1}{(-N_r)}$	θ_{TR} or δ_{PED}	$N_{\theta_{TR}}$ or $N_{\delta_{PED}}$
Roll Rate Response Mode	$-L_p$	$\frac{1}{(-L_p)}$	A_{1s} or δ_{LAT}	$L_{A_{1s}}$ or $L_{\delta_{LAT}}$

The longitudinal velocity develops more slowly than the pitch rate in the initial response to a longitudinal control input, and the lateral velocity develops more slowly than the roll rate in response to a lateral control input. Consequently, there is a period of time during which the pitching moment due to longitudinal velocity has a negligible effect on the pitching moment equilibrium, and similarly there is a period when the moment due to lateral velocity has a negligible effect on the rolling moment equilibrium. When the terms due to linear velocity changes are neglected, the pitching and rolling moment equilibrium equations for hovering and low airspeed flight become:

$$\dot{q} - M_q q = M_{B_{1s}} B_{1s} = M_{\delta_{LONG}} \delta_{LONG} \qquad \textit{eq 8.15}$$

$$\dot{p} - L_p p = L_{A_{1s}} A_{1s} + L_{\theta_{TR}} \theta_{TR} = L_{\delta_{LAT}} \delta_{LAT} + L_{\delta_{PED}} \delta_{PED} \approx L_{\delta_{LAT}} \delta_{LAT} \qquad \textit{eq 8.16}$$

Where:

A_{1s}	- Lateral cyclic pitch angle, shaft referenced
B_{1s}	- Longitudinal cyclic pitch angle, shaft referenced
δ_{LAT}	- Lateral control
δ_{LONG}	- Longitudinal control
δ_{PED}	- Pedal control
$L_{A_{1s}}$	- Roll moment due to lateral cyclic pitch angle
$L_{\delta_{LAT}}$	- Roll moment due to lateral control
$L_{\delta_{PED}}$	- Roll moment due to pedal control
L_p	- Roll moment due to roll rate
$L_{\theta_{TR}}$	- Roll moment due to tail rotor pitch angle
$M_{B_{1s}}$	- Pitch moment due to longitudinal cyclic pitch angle
$M_{\delta_{LONG}}$	- Pitch moment due to longitudinal control
M_q	- Pitch moment due to pitch rate
p	- Angular velocity about x axis, Roll rate
$\dot{p}$	- Angular acceleration about x axis
q	- Angular velocity about y axis, Pitch rate
θ_{TR}	- Tail rotor pitch angle
$\dot{q}$	- Angular acceleration about y axis.

The rolling moment due to pedal input is assumed zero as a simple approximation. These first order equations for pitch rate and roll rate, as well as the equations for vertical velocity and yaw rate, in hovering flight are represented by an equivalent mass-damper system as shown schematically in Figure 8.13.

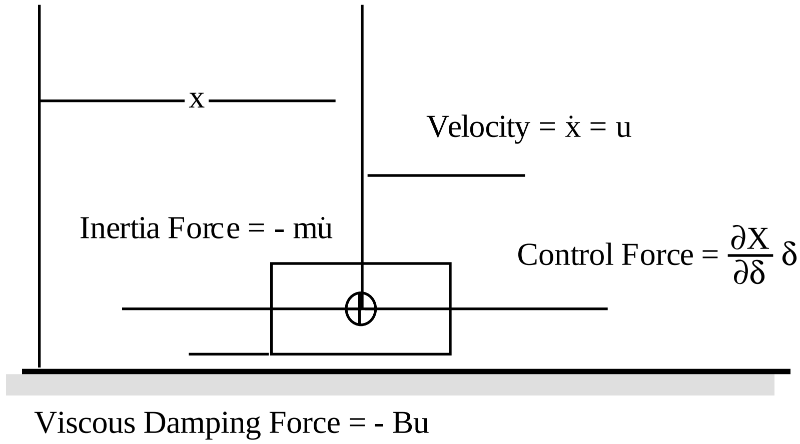


Figure 8.13
Mass-Damper Model for a First Order System

The model consists of a block of mass (m) sliding along a planar surface on an oil film. The diagram shows a damping force opposing the velocity, an inertia force opposing the acceleration, and a force proportional to a control input (δ) in the x direction. The equation of motion for the system which expresses the equilibrium of forces is normalized by dividing by m. It is given both in terms of system constants and in terms of normalized derivatives as follows:

$$\dot{u} = \frac{B}{m}\, u = \frac{1}{m}\, \frac{\partial X}{\partial \delta}\, \delta \qquad \textit{eq 8.17}$$

$$\dot{u} - X_u\, u = X_\delta\, \delta \qquad \textit{eq 8.18}$$

$$\tau = -\frac{1}{X_u} \qquad \textit{eq 8.19}$$

Where:

B	- Damping constant
δ	- Control
m	- Mass
τ	- Time constant

u	- Translational velocity component along x axis
$\dot{u}$	- Linear acceleration along x axis
X	- Resultant force in x direction
X_u	- Longitudinal force due to longitudinal velocity.

Common names are used when referring to some of the derivatives. The acceleration per unit control input is the control sensitivity and the acceleration per unit velocity is referred to as damping. The real root obtained with a first order system is equal to the damping, and the negative of its inverse is equal to the system time constant. Table 8.II shows the corresponding first order system and mass-damper parameters. The pitch rate response is used as an example in the following discussion.

The solution of the first order pitch rate equation for a step aft cyclic input ($-B_{1s}$) applied at time t = 0 gives the pitch variation with time:

$$q(t) = -\frac{M_{B_{1s}} B_{1s}}{M_q}\left(1 - e^{M_q t}\right)$$

eq 8.20

The corresponding pitch acceleration is:

$$\dot{q}(t) = M_{B_{1s}} B_{1s} e^{M_q t}$$

eq 8.21

and the pitch attitude is:

$$\theta(t) = \int_0^t q\,dt = -\frac{M_{B_{1s}} B_{1s}}{M_q}\left[t + \frac{1}{M_q}\left(1 - e^{M_q t}\right)\right]$$

eq 8.22

Where:

B_{1s}	- Longitudinal cyclic pitch angle, shaft referenced
e	- Base of natural logarithm
M	- Net moment about y axis, Pitch moment
$M_{B_{1s}}$	- Pitch moment due to longitudinal cyclic pitch angle
M_q	- Pitch moment due to pitch rate
q	- Angular velocity about y axis, Pitch rate

θ	- Pitch angle
$\dot{q}$	- Angular acceleration about y axis
t	- Time.

The trends in $q(t)$, $\dot{q}(t)$ and $\theta(t)$ for a step cyclic input are presented in Figure 8.14.

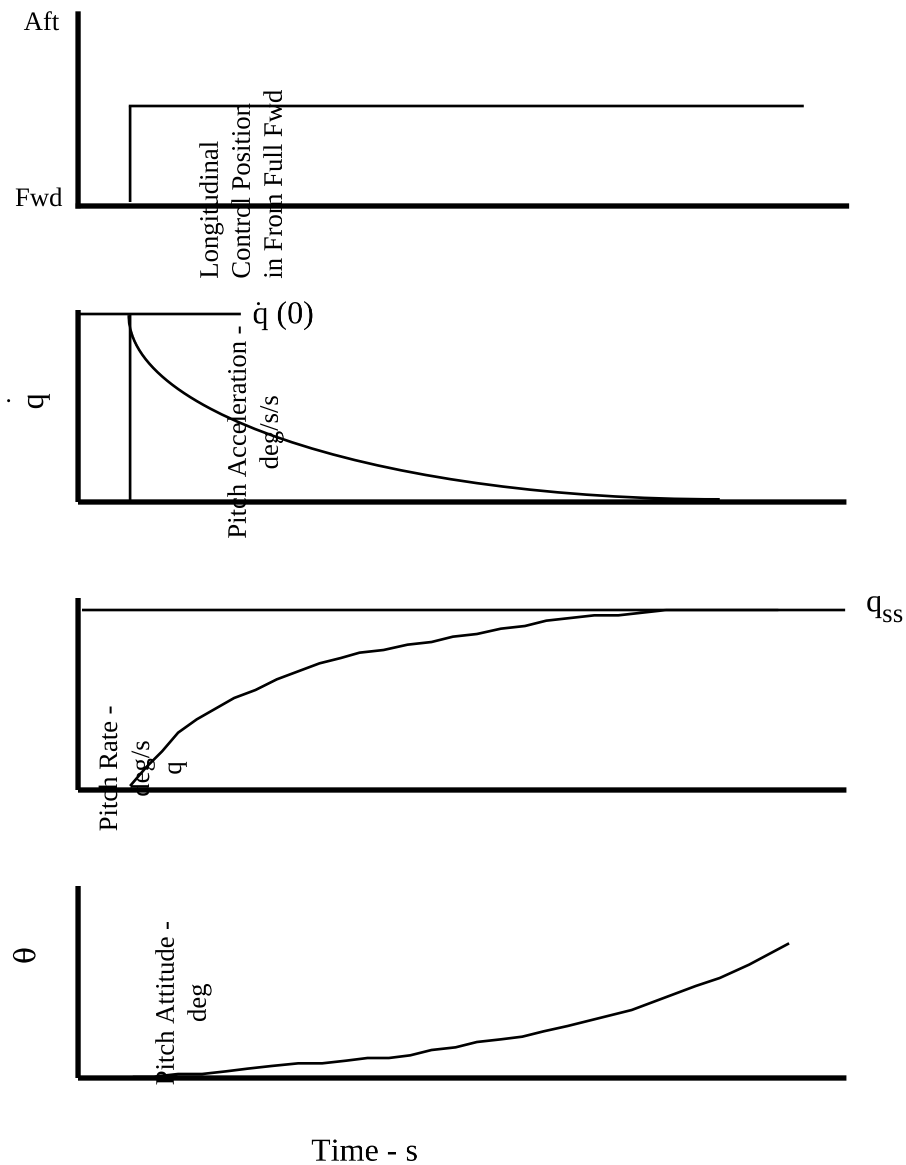


Figure 8.14
Idealized First Order Pitch Response to Aft Longitudinal Cyclic Step Input

The pitch rate and pitch acceleration time histories provide considerable insight to the handling characteristics of the helicopter as measured by the quantities $\dot{q}(t)$, q_{SS}, and τ. It follows from the pitch acceleration and pitch rate equations given above:

$$\dot{q}(0) = M_{B_{1s}} B_{1s} \qquad \textit{eq 8.23}$$

$$q_{ss} = -\frac{M_{B_{1s}} B_{1s}}{M_q} \qquad \textit{eq 8.24}$$

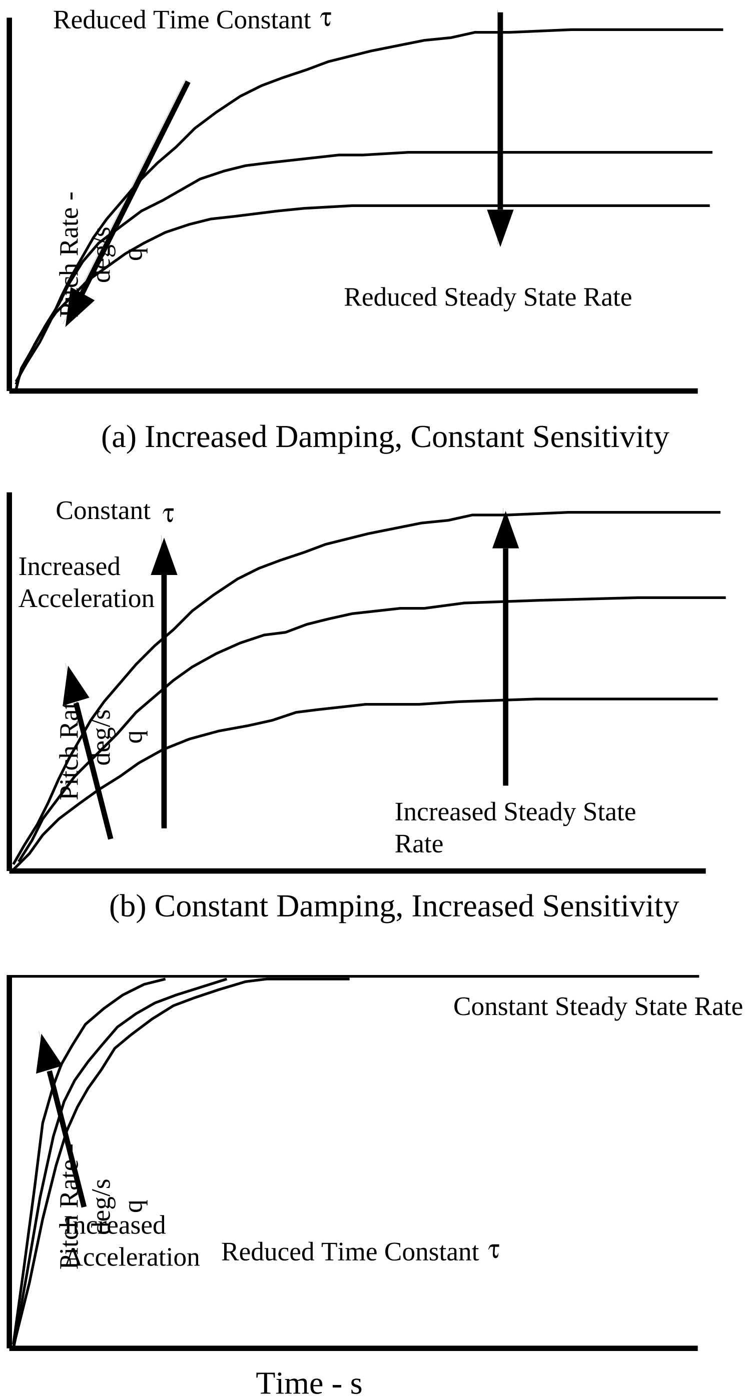
$$\tau = -\frac{1}{M_q} \qquad \textit{eq 8.25}$$

Where:

B_{1s}	- Longitudinal cyclic pitch angle, shaft referenced
$M_{B_{1s}}$	- Pitch moment due to longitudinal cyclic pitch angle
M_q	- Pitch moment due to pitch rate
q_{ss}	- Steady state pitch rate
$\dot{q}$	- Angular acceleration about y axis
τ	- Time constant.

The initial pitch acceleration is dependent on size of the longitudinal cyclic pitch angle (B_{1s}) and the control sensitivity $\left(M_{B_{1s}}\right)$. The steady state pitch rate is dependent on the ratio of the control sensitivity to the pitch damping $\left(M_q\right)$ and the size of the control input. The time constant (the time for $\dot{q}$ to diminish to 0.368 $\dot{q}(0)$, or for q to rise to 0.632 q_{SS}) provides information on the rate damping. The time constant, τ, is inversely proportional to damping, increased damping means reduced time constant. For this ideal response to a step input, it is a simple matter to measure $\dot{q}(t)$, q_{SS}, and B_{1S}; and determine the size of the derivatives, M_q and $M_{B_{1S}}$. This is an ideal response since the pilot cannot physically make a true step input. Also, the measured data are affected by other variables such as transport lags which are not considered in this discussion.

The effects of changes in the damping (M_q) and sensitivity ($M_{B_{1s}}$) on the pitch rate response obtained with step cyclic inputs of the same size are shown in Figure 8.15.



(c) Increased Damping, Increased Sensitivity

Figure 8.15
Effect of Sensitivity and Damping on Initial Pitch Rate Response

For a constant $M_{B_{1s}}$, Figure 8.15 (a) shows increased damping (M_q) reduces the steady state rate and shortens the time constant.

For a constant M_q, Figure 8.15 (b) shows increased control sensitivity ($M_{B_{1s}}$) increases the initial pitch acceleration and steady state rate, while the time constant is unchanged.

For a constant ratio of control sensitivity to damping ($M_{B_{1s}} / M_q$), Figure 8.15 (c) shows increased damping and sensitivity increases the initial pitch acceleration and shortens the time constant, while the steady state rate remains constant.

8.3.12 Control Response

The discussion in the preceding section used a first order system approximation for initial helicopter responses. The effect of control sensitivity and damping on the responses was presented. This section relates these results to the concepts of control effectiveness and response quality.

Attitude control effectiveness is the change in aircraft attitude achieved per time period (usually one second) following a unit step control displacement. Rate control effectiveness is the angular rate achieved per time period (usually one second) following a unit step control displacement. The concept of control effectiveness is illustrated using the pitch responses in Figure 8.16, for two different helicopters, with different levels of both damping and sensitivity. The rate control effectiveness of the two configurations is equal at a time of 1.25 s, and the attitude control effectiveness is equal at 2 s.

The response quality is determined by the pilot's judgment of the overall response which is influenced to a large extent by:

1. Control sensitivity.
2. Damping.
3. Response lags.

The two configurations, in Figure 8.16, composed of completely different combinations of damping and sensitivity, have far different response qualities although their rate and attitude control effectiveness are equal. The characteristics listed above are important not only for the pitch response but relate to all helicopter handling qualities.

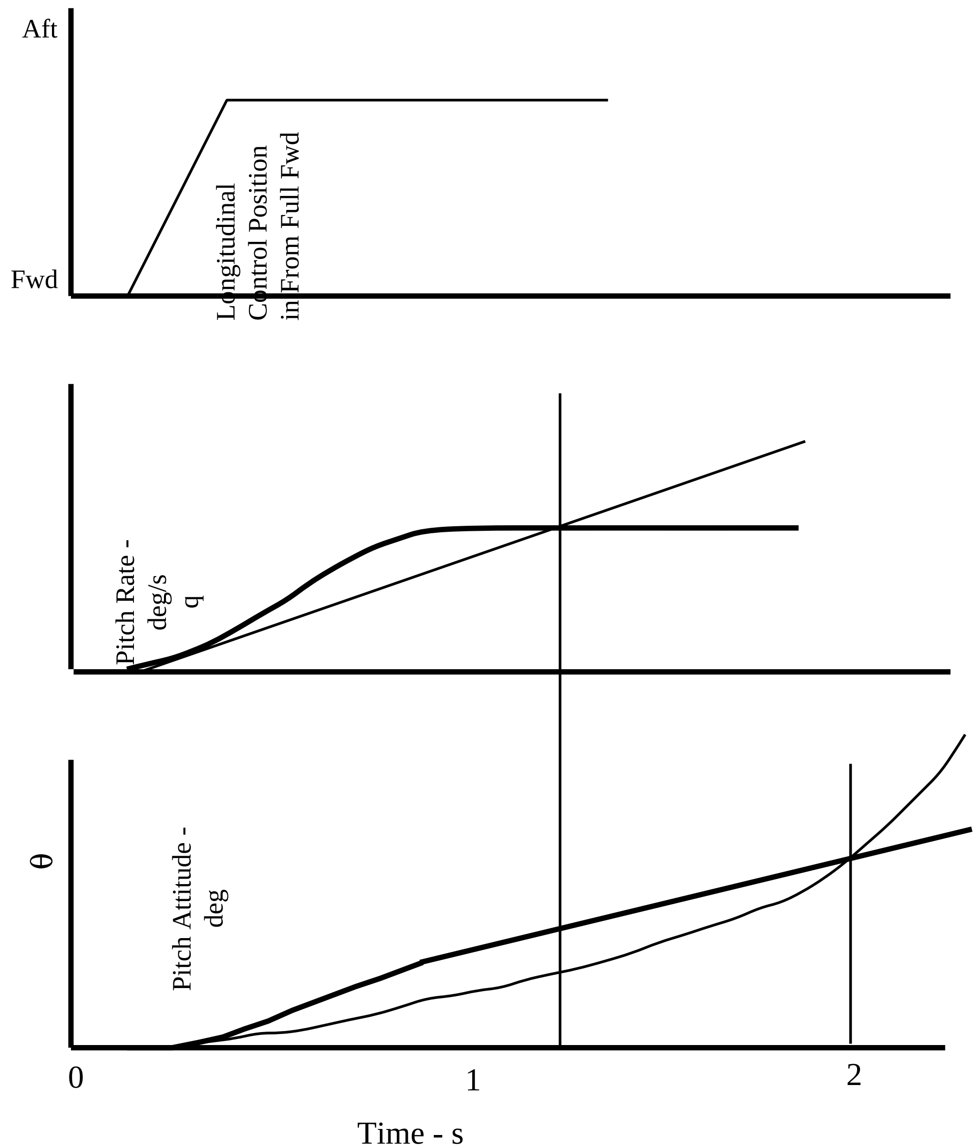


Figure 8.16
Comparison of Control Effectiveness and Response Qualities

Their importance is apparent in all closed loop handling qualities evaluations. A pilot's Handling Qualities Rating (HQR) reflects the impact of the aircraft's closed loop response characteristics on task accomplishment. Several characteristics contributing to desirable response quality are listed below.

1. Pilots want a vehicle responsive enough to achieve some level of attitude change within a certain time after control input (without being overly responsive).
2. Pilots want a reasonable steady state rate for a given size input.
3. Pilots want a predictable steady state response for a given control input.
4. Pilots want a reasonable initial response (acceleration in the desired direction) shortly after control input.

Item 1 above is related to the attitude control effectiveness which is related to both sensitivity ($M_{B_{1s}}$) and damping (M_q). Item 2 indicates the need for an acceptable ratio of sensitivity to damping. Item 3 addresses the time to steady rate and is associated with damping. Item 4 addresses the levels of initial acceleration and is related to control sensitivity.

8.3.13 Vertical Response IGE

Ground effect is a term used as a catch-all for many interesting and important phenomena resulting from changes in the downwash flow field when the helicopter is close to the ground. The term is used to refer to the reduction in power required when the helicopter descends into ground effect. The reduction in power required for a given thrust is reflected by a requirement to decrease collective pitch as altitude decreases IGE. This collective variation with hover height can be considered a form of vertical static stability.

The reduction in power required produced by descent into ground effect requires a collective reduction as the horizontal surface is approached. This is often thought of as only a performance characteristic. The manner of collective reduction is a flying qualities consideration whenever operating IGE.

A plot of this vertical static stability, presented in Figure 8.17, indicates there is only one collective position for a given hover height at constant wind. Although the curve is nonlinear, there are no discontinuities or reversals in slope. If the collective is lowered from position A to B, the aircraft eventually assumes a corresponding wheel height.

Depending upon the vertical damping (Z_w), the helicopter should exhibit a well damped vertical oscillatory response about the new hover height. Figure 8.18 provides an example response corresponding to the aircraft descending from A to B in Figure 8.17.

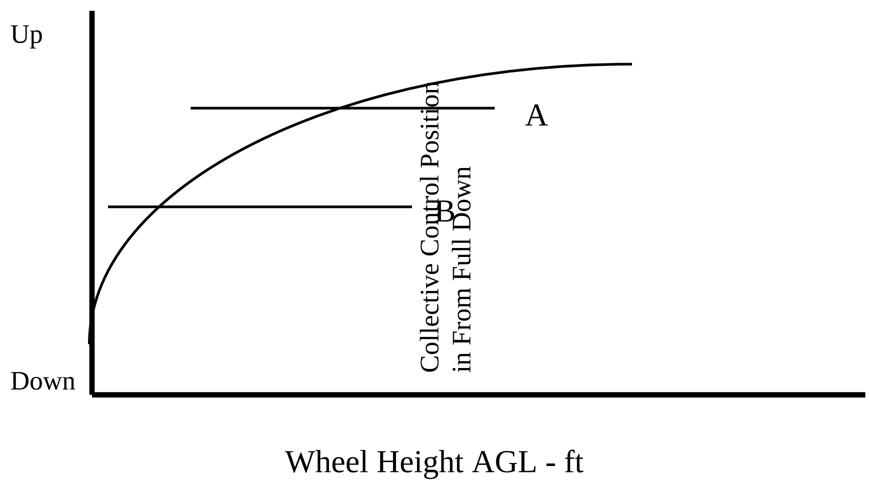


Figure 8.17
Vertical Static Stability

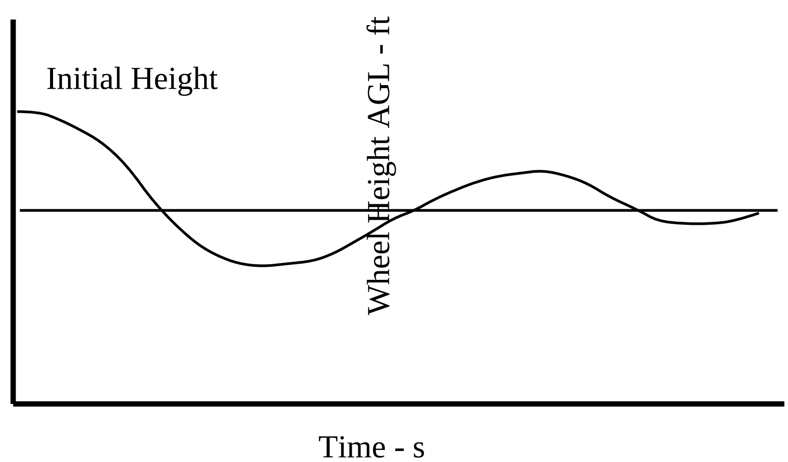


Figure 8.18
Vertical Oscillation IGE After a Step Collective Adjustment Down

In addition to the increase in thrust descending IGE, a substantial portion of the IGE vertical static stability is supplied by the pressure pattern under the fuselage. This phenomenon is so strong on some aircraft a considerable pitching moment is developed as a result. This moment must be countered with longitudinal control input to zero drift and avoid undesirable landing attitudes. This pressure pattern characteristic may provide desirable or undesirable results varying with gross weight, power, wind, and ground speed. The pattern may be unstable and introduce a random disturbance in pitch. A wide variety of ground effects characteristics are possible.

Tandem rotor helicopters have a particular problem in pitch control during vertical descents into ground effect. The forward and aft rotors normally do not experience the same increase in thrust during a decrease in hover height. Thus, as the aircraft descends into ground effect, the thrust of the individual rotors change in an unequal fashion producing a longitudinal pitching moment away from the more effective rotor. As the tandem helicopter descends into ground effect, the pilot may not perceive the actual vertical motion of the aircraft CG. He may see the combined vertical and pitching response as the vertical motion of the aircraft.

8.4 TEST METHODS AND TECHNIQUES

8.4.1 General

The tests generally included in an evaluation of low airspeed handling qualities are: trim control positions in low airspeed flight, critical azimuth determination, turn on a spot, static stability, long term dynamic response, control response, and departures and approaches.

Most low airspeed tests require calm winds to quantify accurately the aircraft handling characteristics. A test limit of 3 to 5 kn true wind speed is the norm. When winds are present, the velocity is vectorially included in the data presentation to establish aircraft operating limits. During periods of steady wind conditions, turn on a spot data are collected and various mission maneuvers are qualitatively and quantitatively evaluated.

Low airspeed velocity is measured in a variety of ways. The most common method is formation flight with a ground pace vehicle. Other methods include use of low airspeed sensing systems, Doppler ground speed systems, and inertial systems. All these systems are calibrated prior to test and all, except the low airspeed systems, depend on adjustment for the local wind speed and direction.

Qualitative evaluation of the helicopter is accomplished with a full and complete understanding of the intended mission. HQRs are assigned with respect to pilot workload, adequate and desired performance, and the test conditions. For instance, an attack helicopter pilot may need to maintain aircraft heading within 3° to fire effectively a particular weapon system. The attack pilot heading control criteria may be too strict for the utility mission. These judgments require operational experience and influence the HQRs assigned to a particular maneuver.

8.4.2 Trimmed Control Positions

The purpose of this test is to determine the control margins and handling qualities in simulated crosswind, downwind, and low airspeed forward flight. The qualitative evaluation of trim changes is an integral part of the test. The test is accomplished by flying formation on a ground pace vehicle having a calibrated airspeed system. The helicopter is flown at a constant height AGL. Winds should be less than 3 kn. Quantitative data are collected in forward, rearward, and sideward trimmed IGE flight. The entire low airspeed envelope of the helicopter is investigated. Sideward and rearward flight to the maximum allowable velocity, as determined by control position and control power margins; and forward flight to approximately 50 kn are conducted. Airspeed increments of approximately 5 kn are used. A ground observer records the ambient wind information and relays this information to the crew of the pace vehicle. The crew of the pace vehicle uses the ambient wind information along with runway heading to determine aircraft heading and pace vehicle speed for each test point.

8.4.2.1 TEST TECHNIQUE

1. Establish a zero drift, unaccelerated, trimmed hover on the test heading. Record the initial trim conditions.
2. The pace vehicle establishes the next test speed. The aircraft stabilizes in formation on the pace vehicle. The standoff distance is sufficiently close to detect small, relative motions; but far enough to avoid downwash interference.

3. When all relative motion is stopped, the flight control forces trimmed to zero, and the helicopter on heading at the correct altitude, data are recorded.
4. Assign an HQR and Vibration Assessment Rating (VAR) to each data point.
5. The pace vehicle establishes the next test speed and the process is repeated.
6. Use the force trim and Automatic Flight Control System (AFCS) in a mission representative manner.
7. Fly all test points for the same direction (forward, rearward, sideward) in the same geographical direction.

8.4.2.2 DATA REQUIRED

Cockpit. Run number, control positions, heading, airspeed from low airspeed system (if available), ground speed (if available), fuel counts (FC), Q, N_R, θ, ϕ, HQR, and VAR.

Ground station. Run number, wind direction, wind speed, T_o, and H_{P_o}.

8.4.2.3 TEST CRITERIA

1. No relative motion between helicopter and pace vehicle.
2. Stabilized, unaccelerated flight.
3. No vertical climb/descent.
4. Constant height AGL.
5. Minimum control movement.
6. All control forces trimmed to zero.

8.4.2.4 DATA REQUIREMENTS

1. Stabilize 15 s prior to recording data.
2. Record 10 s data.
3. Altitude ± 5 ft.
4. Vertical speed 0.
5. N_R ± 1%.
6. Heading ± 2°.

8.4.2.5 SAFETY CONSIDERATIONS/RISK MANAGEMENT

Trimmed control positions in low airspeed flight require operating close to the ground and pace vehicle at relatively high speed. The helicopter can experience large pitch or roll attitudes reducing field of view. Three different groups of people are involved: the helicopter crew, the pace vehicle crew, and the ground wind observer operating in close proximity. Terminating from high speed sideward, rearward, and forward flight close to the ground must be briefed and practiced. Emergency procedures for engine failure, engine control failure, and flight control system failures must be briefed. Procedures for terminating a data point must be briefed. All participants must maintain spatial awareness and conduct test operations clear of airport hazards and traffic.

8.4.3 Critical Azimuth

The purpose of this test is an evaluation of the control margins and pilot workload with changes in airspeed and relative wind azimuth. The qualitative evaluation of trim changes and pilot workload is an integral part of this test. Data are collected simultaneously with trimmed control positions in low airspeed flight. Of specific interest are the control margins and pilot workload to accomplish the task of hovering the helicopter within predetermined tolerances at various wind speeds and directions. The techniques, data, criteria, and safety considerations for trimmed control positions apply to critical azimuth. Critical azimuth data are gathered in relative wind azimuth increments of 30˚ to 45˚; unless a region of particular interest is identified and then as little as 15˚ increments are common. Critical azimuth data are gathered for at least one airspeed, but a range of speeds can be used.

8.4.4 Turn on a Spot

In conducting this test, a steady hovering condition is established heading into a wind of known velocity. The aircraft is then turned to the right or left through a small heading angle. Records are taken of control positions, attitudes, and rates after the aircraft is stabilized on the new heading. Altitude and position over the ground are maintained constant. Tests are continued at small increments in heading (15˚) until data are obtained for a full 360˚ turn. Several different ambient wind speeds are used to document fully the handling characteristics. The test team must be prepared to take advantage of all ambient wind for data collection opportunities. A ground wind observer is used to record wind information.

Constant yaw rate turns to the right, or the left, are used to obtain pilot evaluations of handling qualities in turns over a spot. A continuous record of this maneuver, considering the piloting task and performance criteria, indicates how well the aircraft is controlled. Pilot workload is determined by examining the control patterns required to perform the turn.

8.4.4.1 TEST TECHNIQUE

1. Establish a zero drift, unaccelerated, trimmed hover on the test heading. Record the initial trim conditions.
2. Establish a zero drift, unaccelerated, trimmed hover on the new test heading.
3. Trim the flight control forces to zero, stabilize on heading at the correct altitude, and record data.
4. Minimize control movement during data recording.
5. Assign an HQR and VAR to each data point.
6. Use the force trim and AFCS in a mission representative manner.
7. Complete the 360˚ turn.

8.4.4.2 DATA REQUIRED

Cockpit. Run number, control positions, heading, airspeed from low airspeed system (if available), fuel counts (FC), Q, N_R, θ, ϕ, HQR, and VAR.

Ground station. Run number, wind direction, wind speed, T_o, and H_{P_o}.

8.4.4.3 TEST CRITERIA

1. Stabilized, unaccelerated, zero drift flight.
2. No vertical climb/descent.
3. Constant height AGL.
4. Minimum control movement.
5. All control forces trimmed to zero.

8.4.4.4 DATA REQUIREMENTS

1. Stabilize 15 s prior to recording data.
2. Record 10 s data.
3. Altitude ± 5 ft.
4. Vertical speed 0.
5. N_R ± 1%.
6. Heading ± 2˚.

8.4.5 Static Stability

The purpose of this test is to evaluate the static stability or total restoring moment generated by off trim conditions in the longitudinal, lateral, and directional axes. Data are collected during forward, rearward, and sideward collective fixed flight about zero trimmed airspeed. A 10 kn variation about trim is used. Data are collected with the collective fixed at the zero airspeed value. Static stability is indicated by the initial tendency of the aircraft to return to or depart from equilibrium if disturbed.

Static stability is indicated by pilot control displacement and/or control force required to generate balancing pitching, rolling, or yawing moments when the aircraft is stabilized at off trim conditions. Cyclic displacement in the direction of movement is required to hold an off trim condition in an aircraft possessing positive longitudinal or lateral static stability. Positive directional stability is indicated by directional control displacement opposite to the direction of sideward flight (increasing left pedal with increasing right sideward flight). No variation in control positions with speed is obtained if the aircraft is neutrally stable. Cyclic displacement opposite the direction of movement is required to hold an off trim condition in an aircraft with negative longitudinal and lateral static stability. Directional control displacement in the direction of sideward flight indicates negative directional static stability (right pedal applied with right lateral flight). Calm wind conditions are required to accomplish this test.

The test technique for evaluating the low airspeed static stability of the helicopter are broken into two portions, longitudinal and lateral/directional.

8.4.5.1 TEST TECHNIQUE

1. Longitudinal static stability.

   a. Establish a zero drift, unaccelerated, trimmed hover on the test heading. Record the initial trim conditions.

   b. Accelerate approximately 2 to 3 kn forward, collective fixed at the zero airspeed trim position, do not retrim, stabilize airspeed, and record data.

   c. Repeat step b to achieve a 10 kn band.

   d. Repeat steps a, b, and c for rearward flight.

2. Lateral and directional static stability.

   a. Establish a zero drift, unaccelerated, trimmed hover on the test heading. Record the initial trim conditions.

   b. Accelerate the aircraft sideways 2 to 3 kn, collective fixed at the zero airspeed trim positions, do not retrim, stabilize airspeed, ensuring no forward or aft drift and relative wind azimuth equal 90˚ or 270˚. Record data.

   c. Repeat step b until total velocity is approximately 10 kn.

   d. Repeat steps a, b, and c for lateral flight in the other direction.

8.4.5.2 DATA REQUIRED

Cockpit. Run number, control positions, heading, airspeed from low airspeed system (if available), ground speed (if available), fuel counts (FC), Q, N_R, θ, ϕ, HQR, and VAR.

Ground station. Run number, wind direction, wind speed, T_o, and H_{P_o}.

8.4.5.3 TEST CRITERIA

1. Stabilized, unaccelerated flight.
2. Constant collective.
3. Minimum control movement.
4. Do not retrim.

8.4.5.4 DATA REQUIREMENTS

1. Stabilize 15 s prior to recording data.
2. Record 10 s data.
3. Heading ± 2˚.

8.4.6 Long Term Dynamic Stability

The purpose of this test is to evaluate the long term response (change in attitude, airspeed, and altitude) of the aircraft as a function of time. This is a nuisance mode generally suppressed by the pilot or AFCS. The tests should include a qualitative determination of the difficulty of suppressing the long term response and the ease of excitation. The test basically consists of recording the response of the aircraft to off trim conditions resulting from turbulence or disturbances introduced by the pilot. Calm atmospheric conditions are required to accomplish this test.

Static stability influences the low airspeed long term dynamic stability of the helicopter. Negative static stability results in an aperiodic divergent response. Positive static stability causes an oscillatory response which may be convergent, or divergent, depending on damping. Higher damping results in fewer overshoots and a more convergent oscillation. Also, angle of attack variations are more significant for the long term helicopter mode than for the airplane phugoid mode.

Often the low airspeed long term response can be excited without conscious input. With the controls fixed and no pilot inputs, an imperfect trim condition, or atmospheric disturbance, may excite an aperiodic divergent mode or a lightly damped oscillatory mode. This is an unsatisfactory technique if no response is obtained, but indicates a desirable aircraft characteristic.

If maintaining control fixed fails to produce a long term response, use the longitudinal control to increase, or decrease, the pitch attitude of the hovering helicopter. Then smoothly return the control to the original trim position and record the aircraft motion. An off trim attitude variation of 5 to 10˚ is used to excite the motion. Natural disturbances which result in a long term response are desirable but these responses are usually contaminated by another disturbance before the motion is completed. This makes quantifying the mode of motion difficult.

Artificial excitation using the longitudinal control to generate an off trim pitch attitude condition is used to stimulate the response and obtain time histories from which engineering data are obtained. The excitation method chosen should result in an aircraft response similar to a response following a natural disturbance.

8.4.6.1 TEST TECHNIQUE

1. Establish a zero drift, unaccelerated, trimmed OGE hover. Do not retrim control forces or move the collective during the test. Record the initial trim conditions.
2. Determine if a long term response results from a natural disturbance. With the controls either fixed or free, note the open loop aircraft response. If no aircraft response is observed, use an artificial excitation.
3. Excite the aircraft long term response using an aft longitudinal control input to establish a 5 to 10° off trim pitch condition. Return the control exactly to trim. Conduct several tests to find the most representative aircraft responses to include in the report. Do not retrim any control forces and keep the collective constant during the response.
4. If the aircraft is flown hands off during normal operations, obtain control free responses following excitation. The controls are released at trim so the subsequent control motions indicate the effect of attitude changes and gravity force acting on the controls during the response.
5. Record the resulting mode of motion using cockpit data and automatic recording systems. Cockpit displayed attitudes are recorded at selected time increments. The stop watch is started at the completion of the excitation or at a predetermined attitude. The zero time reference point is arbitrary. Use small time intervals to define the shape of irregular responses.

8.4.6.2 DATA REQUIRED

A time history of aircraft attitudes, rates, control positions, heading, airspeed from low airspeed system (if available), FC, Q, N_R, T_o, H_{P_o}, and VAR.

8.4.6.3 TEST CRITERIA

1. Stabilized, unaccelerated flight at trim.
2. All control forces trimmed to zero.
3. Collective fixed.
4. Do not retrim.

8.4.6.4 DATA REQUIREMENTS

1. Stabilize 15 s prior to recording trim data.
2. Record 2 complete cycles or until a test limit is reached.

8.4.6.5 SAFETY CONSIDERATIONS/RISK MANAGEMENT

Be prepared for large attitude excursions. Brief and practice unusual attitude recoveries, using appropriate build up. Be prepared for coupled responses. Operate outside the avoid area of the helicopter's height velocity diagram. Maintain situational and height awareness. Be prepared for power settling.

8.4.7 Control Response

The purpose of these tests is to evaluate the aircraft response to flight control input. The tests are conducted by applying control step inputs of incrementally increasing size against a fixture, and recording a time history of the aircraft response.

8.4.7.1 TEST TECHNIQUE

1. Establish a zero drift, unaccelerated, trimmed hover at the desired altitude. Normally control step inputs are accomplished OGE above the height velocity diagram avoid area.
2. Set the control fixture for the desired displacement and position for the desired input direction. The control input direction and size is verbally and visually (if able) verified between the pilot and copilot/engineer. Start with small displacements and increase incrementally.
3. Voice procedures for cockpit coordination are as follows: “Data on. Stand-by for a (displacement magnitude), (direction of input), (control) input on three. Thousand one, thousand two, thousand three.” The input is executed on three. Example, “Data on, stand-by for a one inch right lateral cyclic input on three. Thousand one, thousand two, thousand three.”
4. Hold the control input fixed. Recover when a steady state rate is obtained or a predetermined flight limit is reached. A common error during these tests is to hold the point longer than required, allowing the aircraft to reach an uncomfortable or even dangerous attitude or rate.

5. When the maneuver is complete, announce the recovery so the control fixture can be quickly and completely removed.
6. Repeat steps 1 through 5 for subsequent data points until the desired control displacement is achieved or a predetermined flight limit is reached.
7. Qualitatively assess the attitude control effectiveness, rate control effectiveness, and the predictability of the control response.

Test limits on bank attitude can prevent achieving steady state roll rates. If the test helicopter can achieve high roll rates but the test program is restricted to low roll attitude excursions, the test pilot must recover prior to achieving steady state roll rate. In this case, a false start technique might be is used. In this technique, the test pilot establishes a bank attitude in the direction opposite to the test input. Make the step input in the desired direction from the banked control position. The use of the false start technique permits a greater roll attitude change and longer time to elapse prior to recovery at a test attitude limit. This technique may permit the helicopter to reach steady state roll rate.

8.4.7.2 DATA REQUIRED

A time history of aircraft attitudes, rates, control positions, heading, airspeed from low airspeed system (if available), FC, Q, N_R, T_o, H_{P_o}, and VAR.

8.4.7.3 TEST CRITERIA

1. Stabilized, unaccelerated flight at trim.
2. All control forces trimmed to zero.
3. Off axis controls fixed.

8.4.7.4 DATA REQUIREMENTS

1. Stabilize 15 s prior to recording data.
2. Record data until steady state rate is achieved or until a test limit is reached.

8.4.7.5 SAFETY CONSIDERATIONS/RISK MANAGEMENT

The considerations for trimmed control positions and long term response generally apply for control response as well. Additionally, all tests have attitude or rate limits which must be observed. You are operating close to the ground, rates or attitudes which may be appropriate for forward flight may not be appropriate for the low airspeed regime. Input size must follow logical build up procedures, usually starting with 1/4 in. Remember you don't have to find the absolute end point, if the trend is rapidly approaching a limit, don't proceed.

8.4.8 Departures and Approaches

The approach to a hover from forward flight uses a descent/deceleration schedule which is repeated over and over again. This schedule can be investigated as a series of fixed trim points. These points are chosen to represent the center and/or extremes of the approach airspeed/altitude envelope. The scope of the test may include repeating the investigation for various types of approach. The following are of interest:

1. Normal approach to a hover.
2. Normal approach to a running landing.
3. Confined area approach to a hover.
4. Precautionary single engine approach.
5. Precision approach to a spot, when thrust to weight precludes a hover.
8. Level flight deceleration (quick stop).

These tests are generally qualitative but if a problem is suspected, time history recordings of the maneuver are used to document the objectionable characteristic.

8.4.8.1 TEST TECHNIQUE

Various approach and departure techniques are investigated. The approaches include steep approach, normal approaches, running landings, and confined area approaches. The departures include: normal take-offs, vertical departures, obstacle clearance take-offs, IGE accelerations to a climb speed, and rolling take-offs.

8.4.8.2 DATA REQUIRED

Qualitative remarks and HQR. A time history is used to document undesirable characteristics. The time history includes aircraft attitudes, rates, control positions, heading, altitude, airspeed from low airspeed system (if available), ground speed (if available), FC, Q, N_R and VAR. Ambient wind and weather conditions are recorded.

8.4.8.3 TEST CRITERIA

Use standard flight techniques for the approaches and departures.

8.5 DATA REDUCTION

8.5.1 General

Data reduction for low airspeed handling qualities tests involves applying the appropriate instrumentation corrections and plotting a number of control positions, aircraft attitudes, rates, and HQRs versus airspeed. Few analytical techniques are needed. Control response is the one exception and is discussed.

8.5.2 Trimmed Control Positions

Instrument corrected control positions and aircraft attitudes are plotted as shown in Figures 8.19 and 8.20. Engineering judgement is applied to determine which aircraft attitudes to depict when the relative wind azimuth is not from a cardinal direction. Additional parameters are shown if a problem is detected and requires discussion.

8.5.3 Critical Azimuth

Critical azimuth data are presented in two ways. First, the trimmed control position data plotted are plotted as a function of relative wind azimuth. An example is provided in Figure 8.21. Additionally, the HQRs and/or VARs obtained during the trimmed control position tests are plotted versus wind azimuth as shown in Figure 8.22. The figure should be further annotated with cross hatched areas depicting what caused the higher numbers. VARs are plotted and annotated similarly. The figure could also be expanded to include areas of restricted field of view, excessive attitude, etc.

8.5.4 Turn on a Spot

Turn on a spot data are reduced and presented similar to trimmed control position data. The data presented are similar to that depicted in Figures 8.21 and 8.22. Minor differences between turn on a spot and trimmed control position data are usually attributable to the cues the pilot has available in conducting these two test techniques.

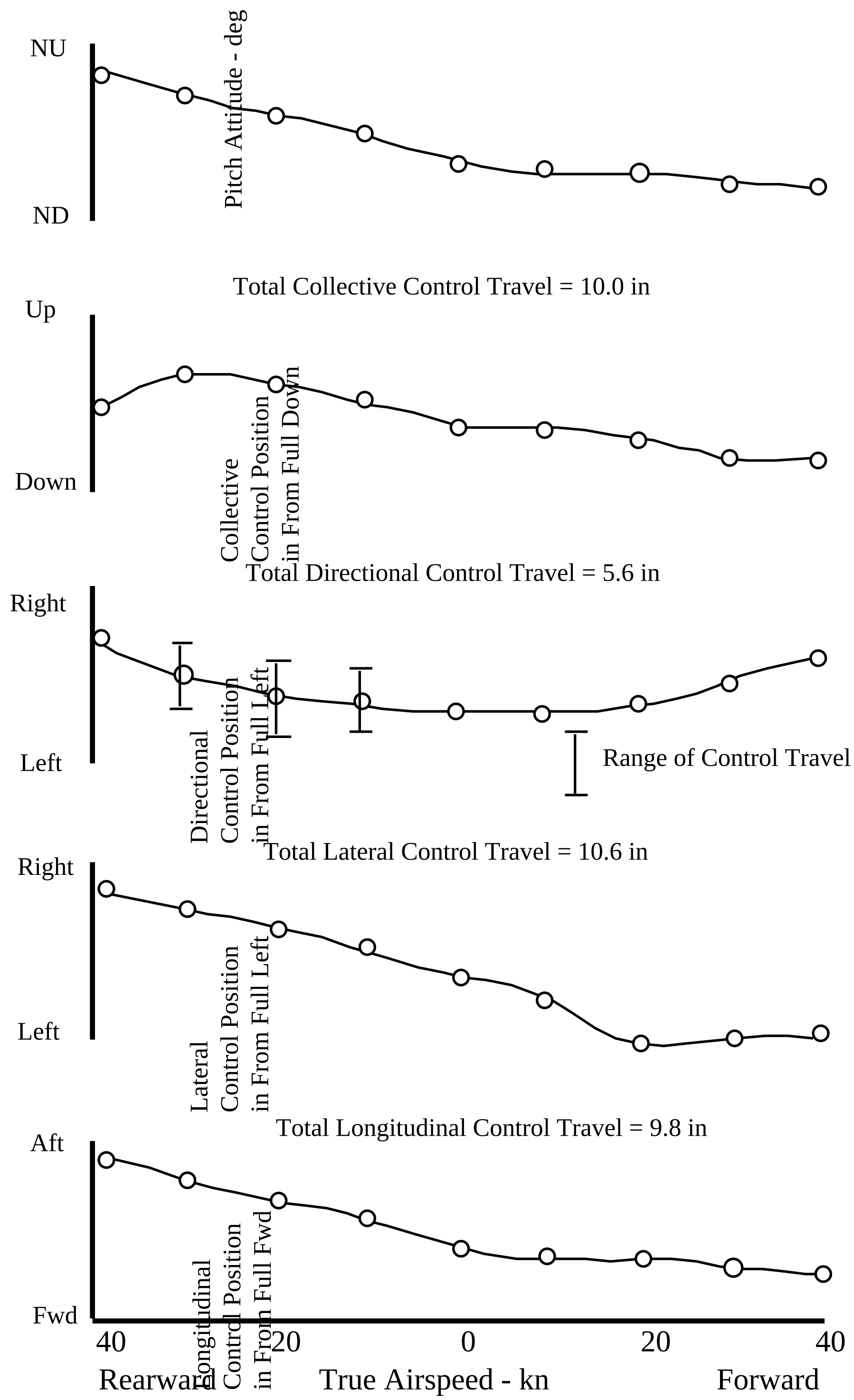


Figure 8.19
Forward and Rearward Trimmed Flight Control Positions

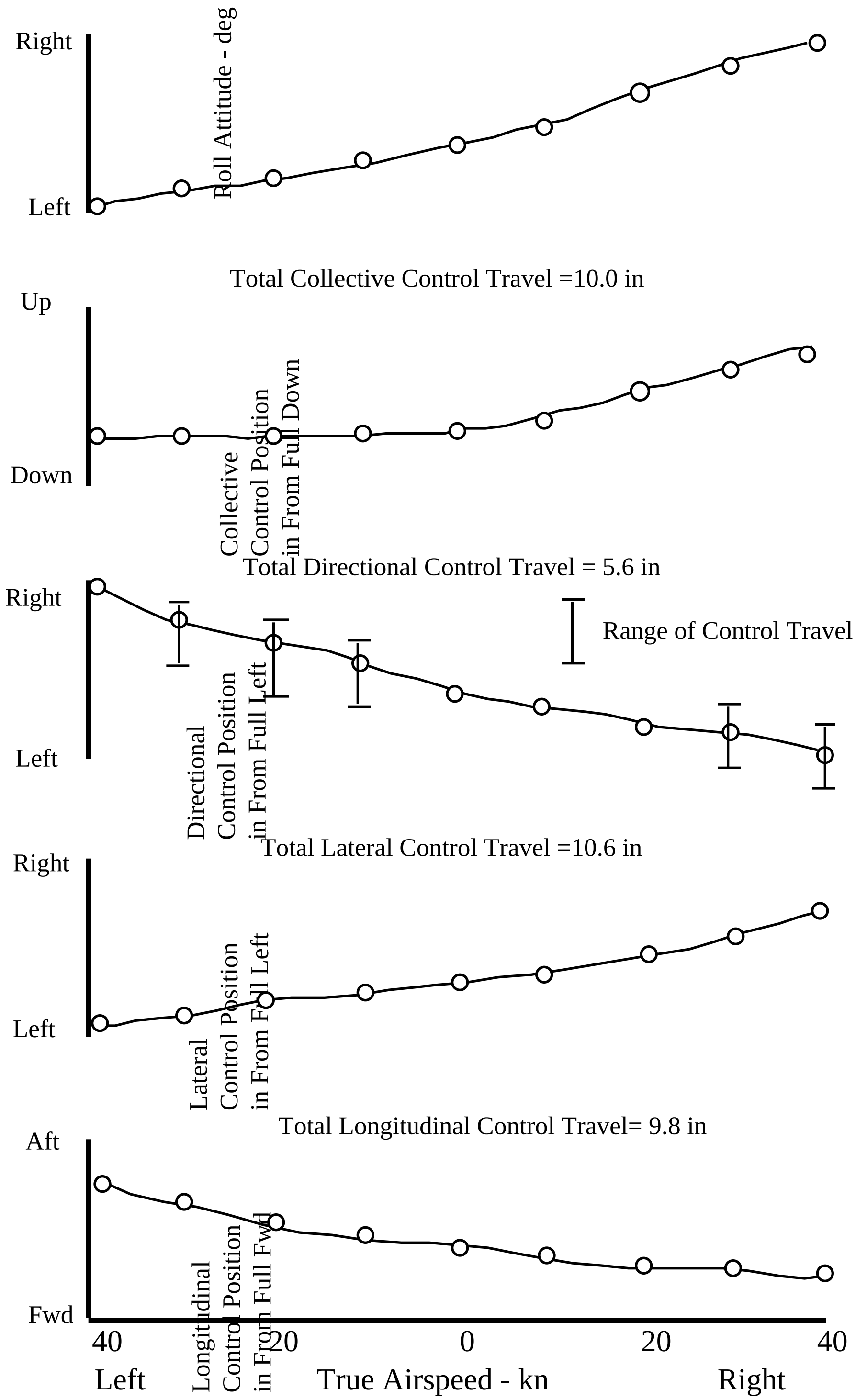


Figure 8.20
Sideward Trimmed Flight Control Positions

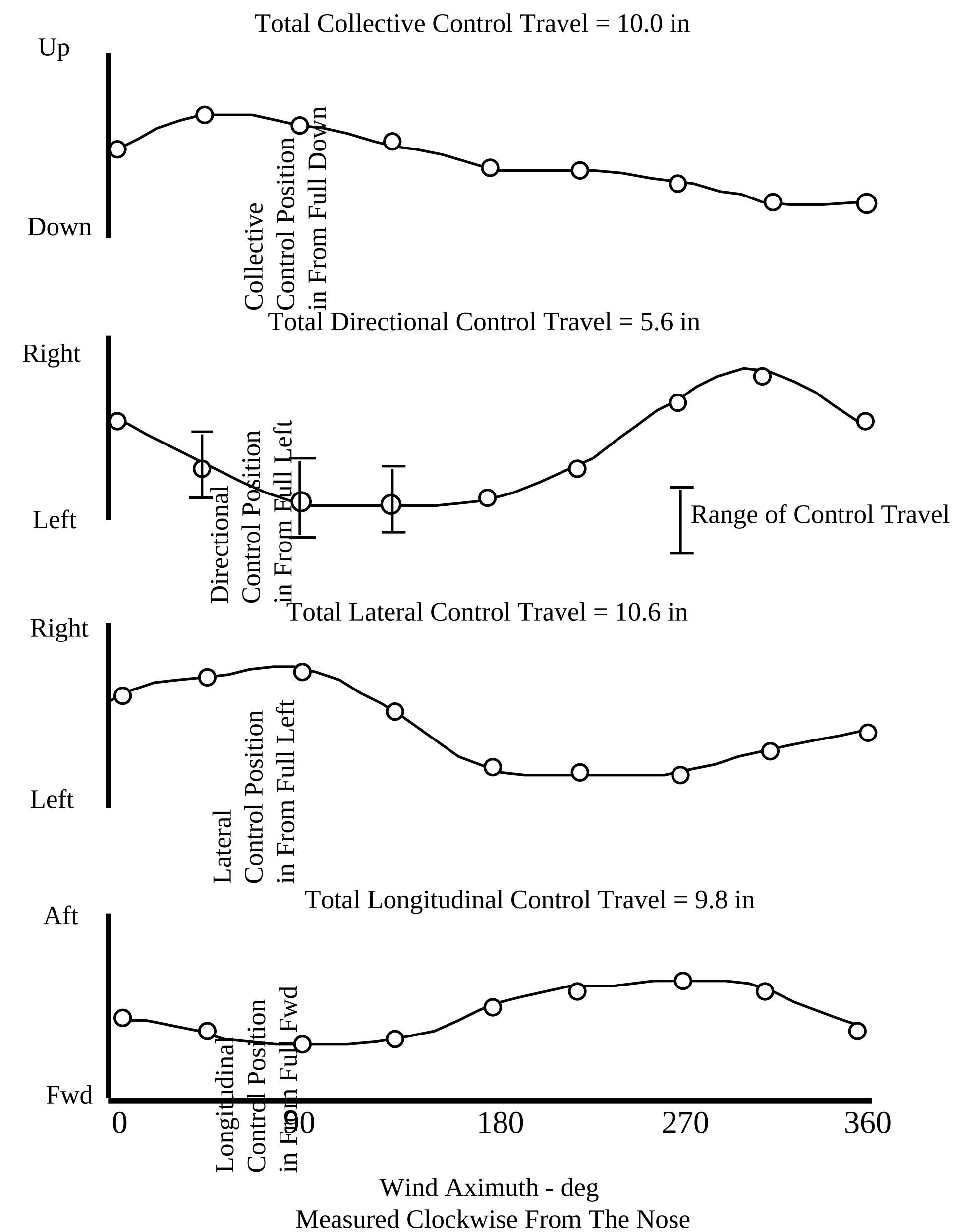


Figure 8.21
Critical Azimuth Determination (20 kn)

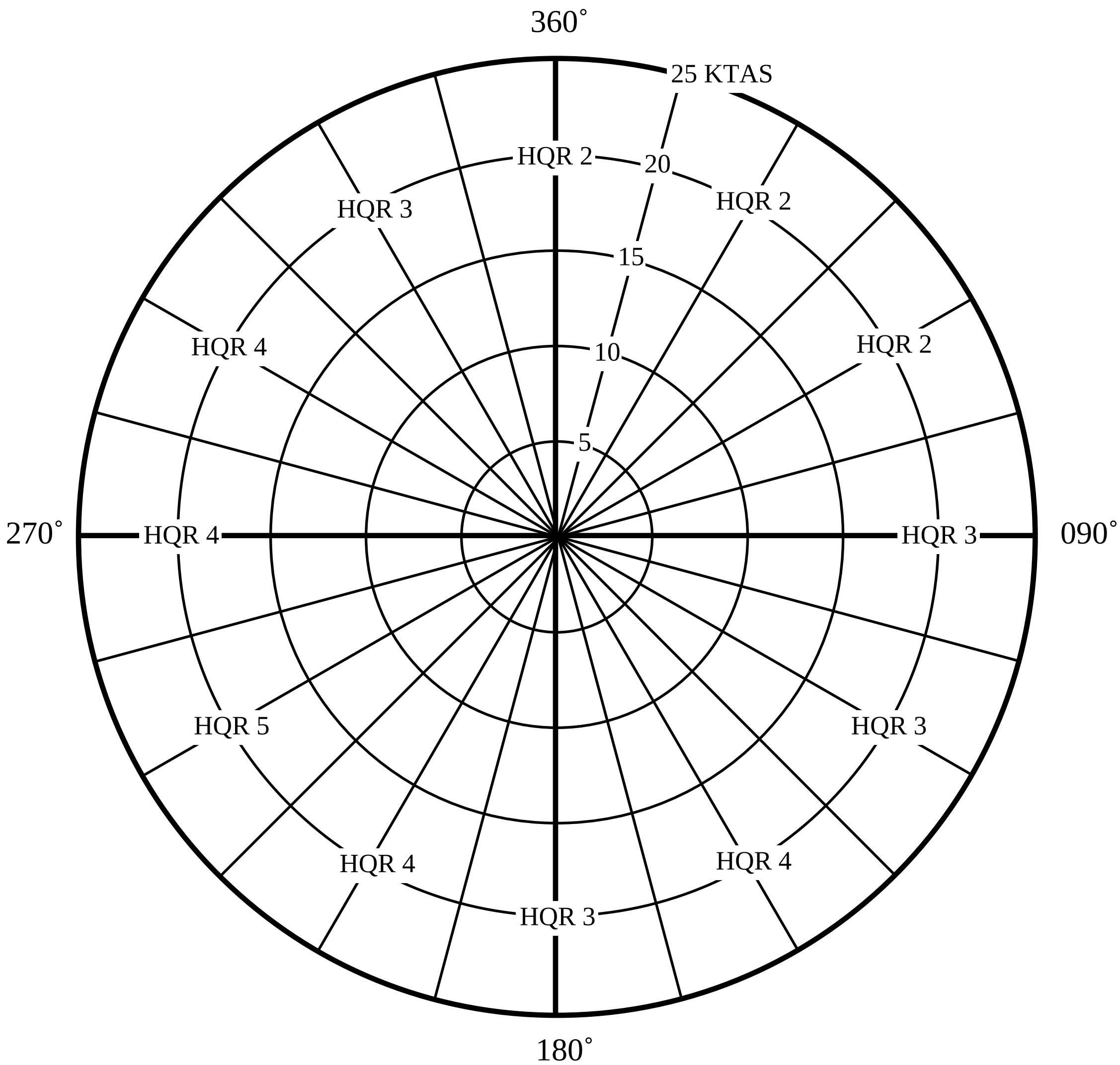


Figure 8.22
HQR Versus Relative Wind Azimuth

8.5.5 Static Stability

Low airspeed static stability is separated generally into longitudinal characteristics and lateral/directional characteristics. Control position and aircraft attitude data are plotted versus airspeeds about trim as shown in Figures 8.23 and 8.24. Note that the trim might not be at zero airspeed depending on the wind conditions at the time of testing.

8.5.6 Long Term Dynamic Stability

Present a representative time history of the aircraft response to a disturbance in the longitudinal axis. Annotate important and descriptive parameters such as τ, Damped frequency (ω_d), Cycles to one-half amplitude ($C_{1/2}$), Cycles to double amplitude (C_d), Time to one-half amplitude ($T_{1/2}$), and Time to double amplitude (T_d). Other parameters of interest are pitch, roll, and yaw rates. Sample data are presented in Figure 8.25.

Tables are used to summarize the engineering characteristics for several test conditions (period $C_{1/2}$, C_d, $T_{1/2}$, T_d, τ, ω_d).

8.5.7 Control Response

Time histories are scaled to distinguish the appropriate responses. Present representative time histories of a 1 in input if available. Proper annotation of the trace is important to assist the reader in visualizing the response. Definitions of control response terms are shown in Figure 8.26. Tables and graphs are used to summarize engineering data for the various responses.

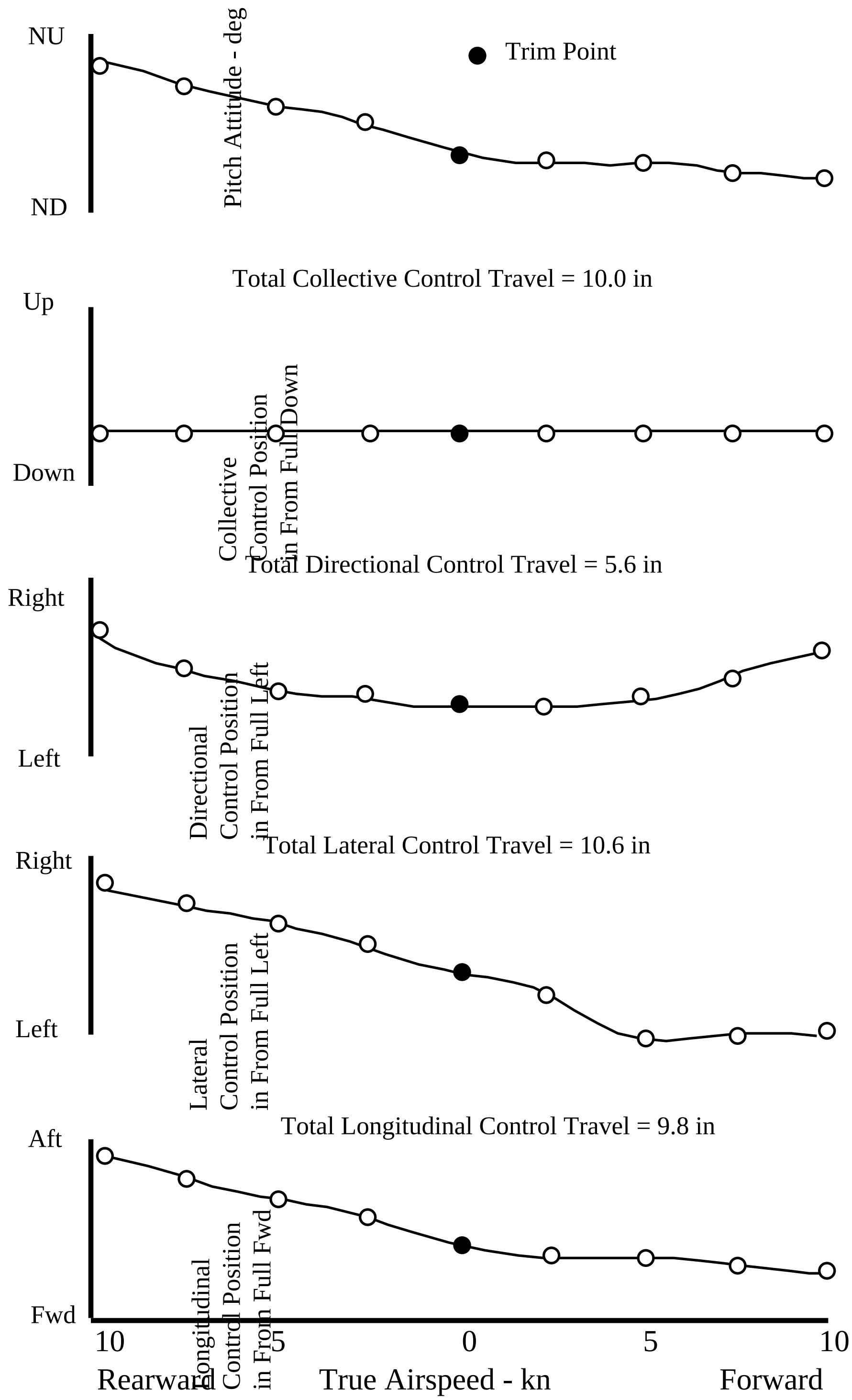


Figure 8.23
Low Airspeed Static Longitudinal Stability

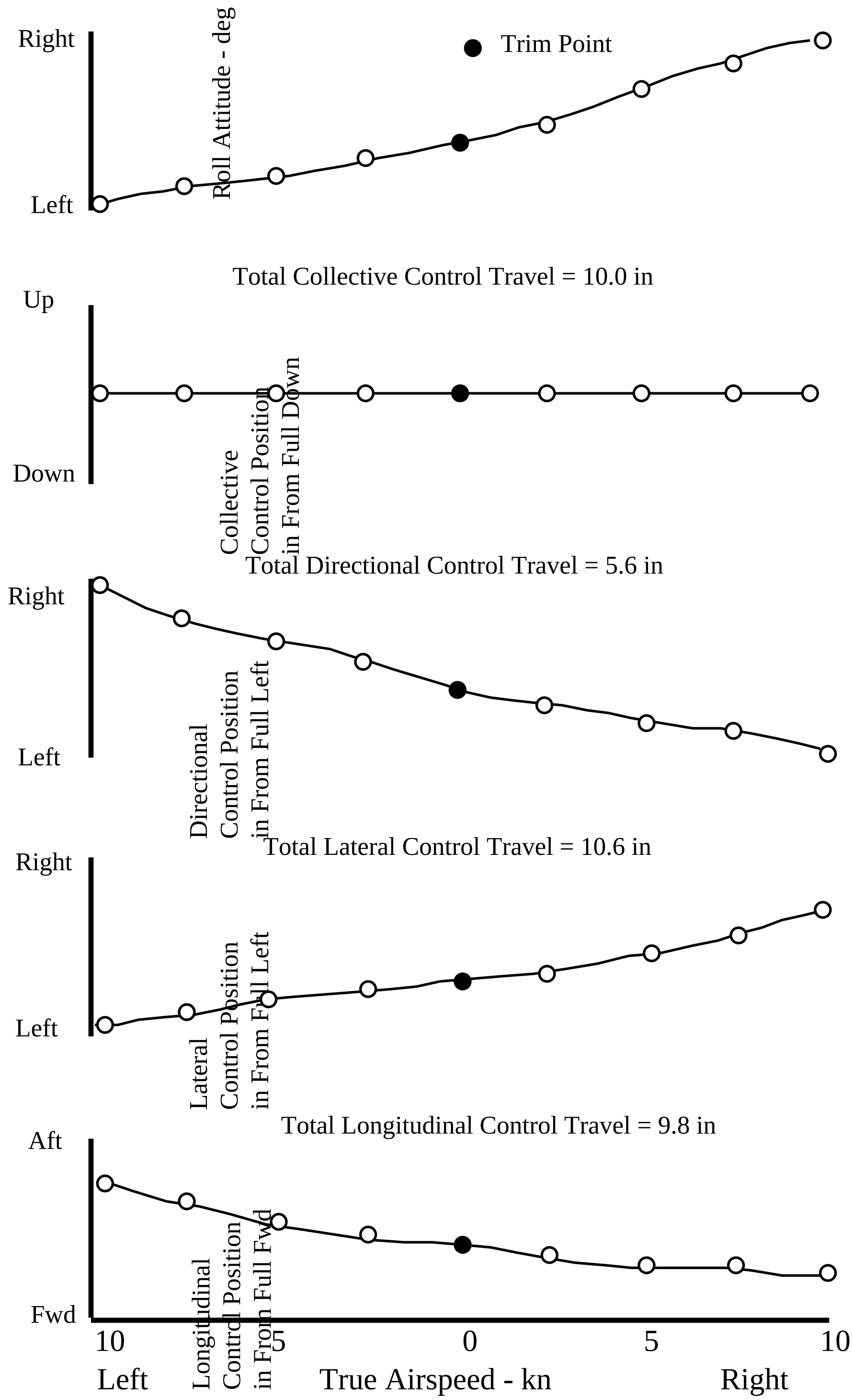


Figure 8.24
Low Airspeed Static Lateral-Directional Stability

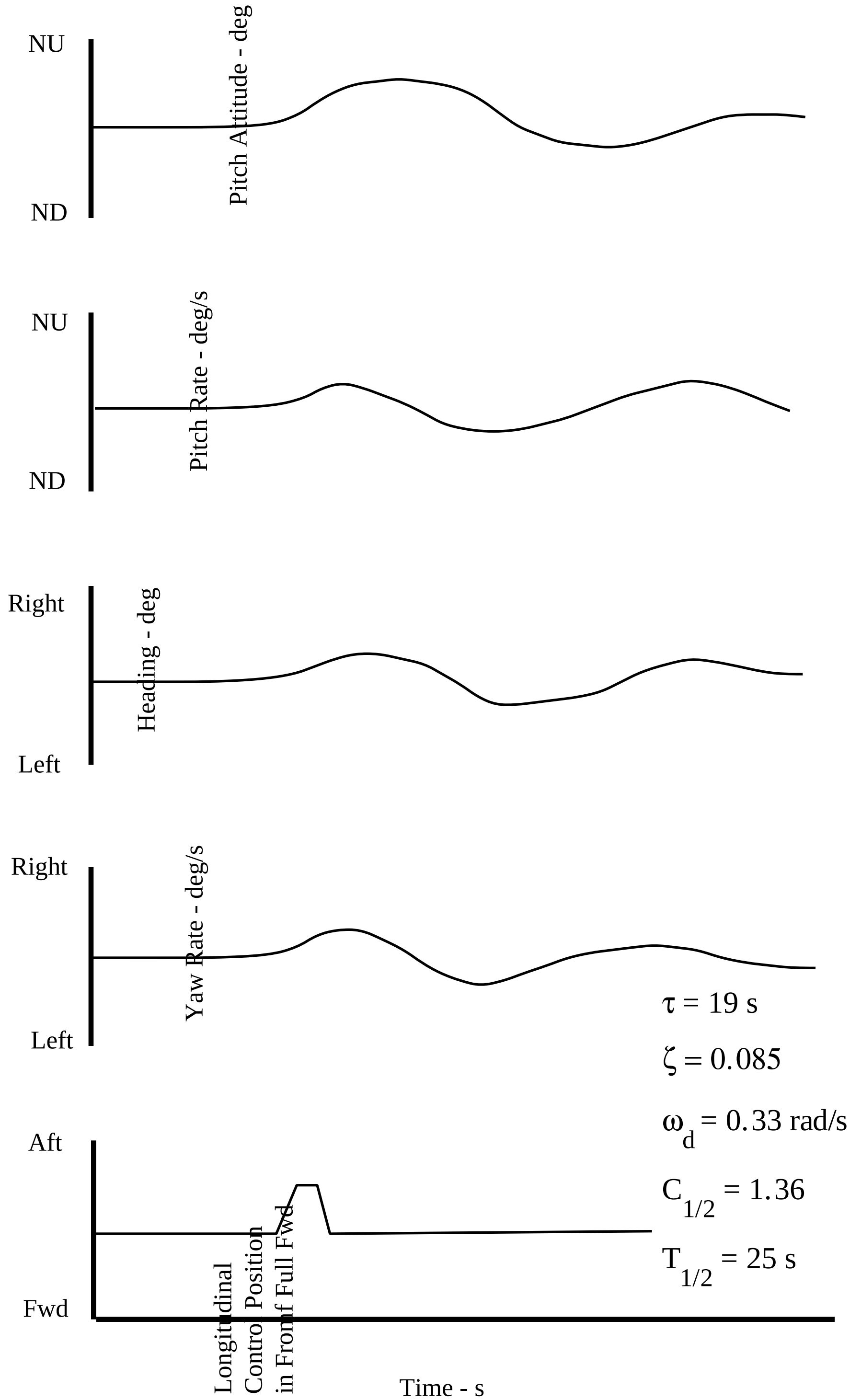


Figure 8.25
Long Term Response

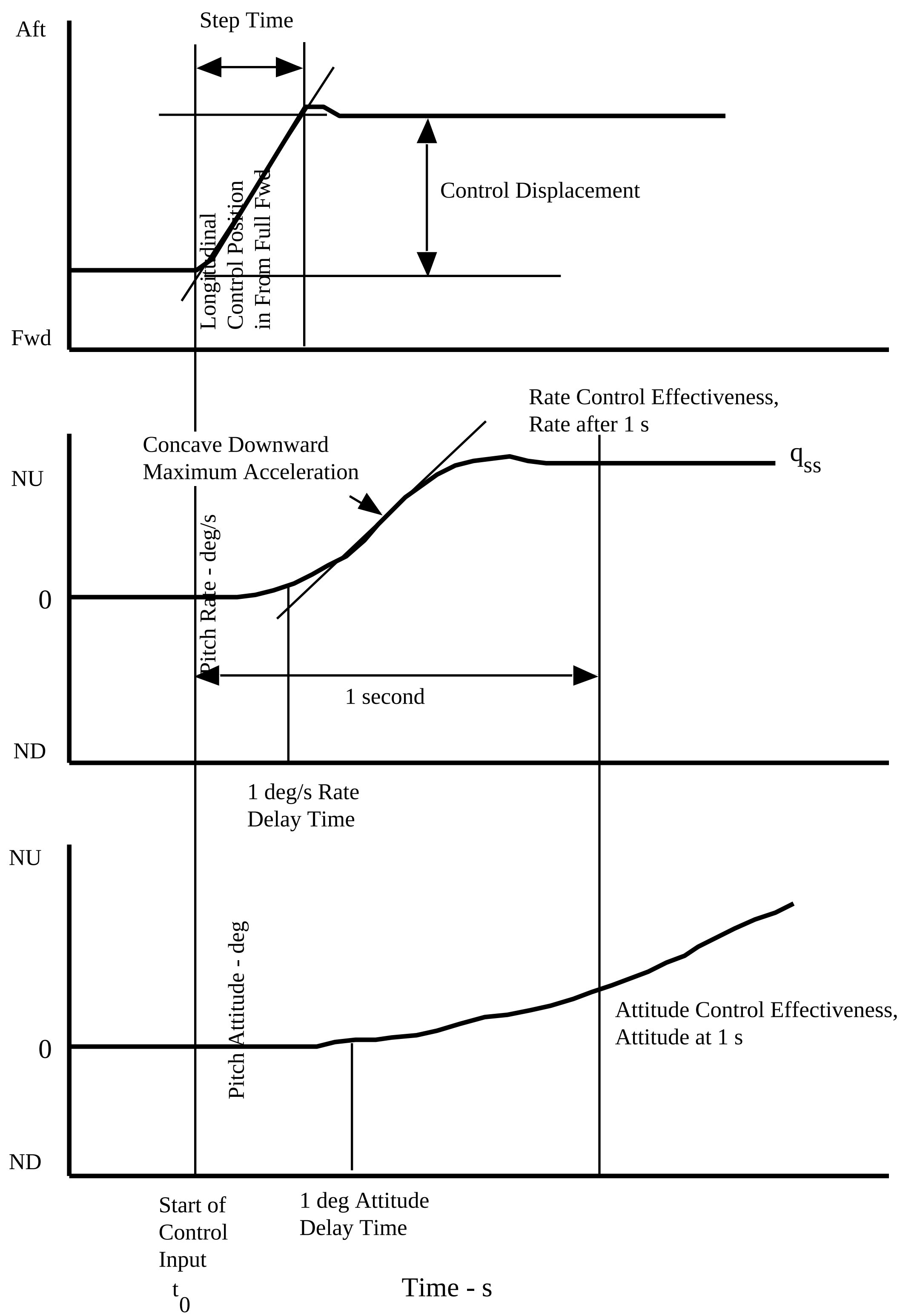


Figure 8.26
Measurement of Step Control Response Characteristics for a First Order System

8.6 DATA ANALYSIS

8.6.1 General

Low airspeed handling qualities are difficult to portray in a pure quantitative manner. The pilot qualitative comments are the most important product of these evaluations. The engineering evaluations are conducted to isolate unsatisfactory or acceptable characteristics. Accurate documentation of the low airspeed flight regime is necessary because the helicopter operates in this environment much of the time.

8.6.2 Trimmed Control Positions

Low airspeed trimmed flight control positions are essentially analogous to hovering (zero ground speed) in a wind of equal velocity from the relative azimuth of the direction of flight. The analysis should include:

1. Were the control margins adequate? Greater than 10% control position or control power margin?
2. Were control movements required in the usual sense (forward cyclic for forward airspeed, right lateral cyclic for right lateral flight)?
3. Were the control position changes with power or airspeed excessive?
4. Could the control forces be trimmed to zero?
5. What were the aircraft attitude changes with airspeed?
6. Was field-of-view adequate?
7. Were the vibrations objectionable?
8. Was there any objectionable coupling between pitch, roll, and yaw?

8.6.3 Critical Azimuth

Critical azimuth determination is an extension of low airspeed trimmed flight control positions and is therefore analogous to hovering in winds of the velocities and azimuths tested. Critical azimuths for a particular aircraft may be based on control margin remaining, pitch, roll or yaw excursions, excessive pitch or roll attitudes, excessively large or frequent control inputs, or high vibration levels. Critical azimuth data analysis should include:

1. What was the limiting factor?
2. For which axis was it most objectionable?

3. Were the control margins adequate?
4. Were the control position changes with variation in azimuth excessive?
5. Could the control forces be trimmed to zero?
6. Was field-of-view adequate?
7. Were the vibrations objectionable?
8. Were pitch or roll attitudes excessive?
9. Are these characteristics suitable for the intended mission?

8.6.4 Turn on a Spot

Turn on a spot data analysis is identical to the evaluation of trimmed control positions and critical azimuth except the pilot must understand different station keeping cues were used in collecting the two sets of information.

8.6.5 Static Stability

Analysis of low airspeed static stability data should consider the following questions:

1. Were the control position gradients positive, neutral, or negative?
2. Did the pilot have force cues to an off trim condition?
3. Are the characteristics suitable for the mission?

8.6.6 Long Term Dynamic Stability

Analysis of the long term response should consider the following questions:

1. How easy or difficult was the long term response excited?
2. Using the methods presented in Chapter 3, discuss the response characteristics (period $C_{1/2}$, C_d, $T_{1/2}$, T_d, τ, ω_d).
3. How does this motion influence the pilot's ability to perform mission tasks?

8.6.7 Control Response

An abundance of quantitative data can be taken from the response time histories. Parameters which define the response are: rate control effectiveness; attitude control effectiveness; steady-state rate, maximum rate, maximum acceleration, initial acceleration, acceleration delay, and time constant. The following considerations are an integral part of control response and effectiveness discussions.

1. Was the steady-state rate adequate?
2. Was the response predictable?
3. Was the response consistent?
4. Were there any over control problems?
5. Was the response suitable for the mission?

8.7 MISSION SUITABILITY

Mission suitability is a very important part of the low airspeed evaluation. The helicopter's unique capabilities in this flight regime provide many potential operational uses if the handling qualities and performance of the vehicle are adequate. Before mission suitability is accessed, a detailed clear understanding of the intended mission must be provided. Additionally, the pilot's experience in conducting similar mission maneuvers is considered. The test pilot must research the mission tasks to establish evaluation maneuvers and determine performance criteria. The mission maneuvers vary greatly with aircraft type, intended use and expected operational environment. Potential maneuvers include everything from sling load operations to rocket firing.

A key element in conducting a mission suitability evaluation is experience. Ensure all operational aspects are considered. Engineering tests conducted in the low airspeed regime provide invaluable insight into the reasons for unfavorable comments relative to operational tasks.

8.8 SPECIFICATION COMPLIANCE

General guidelines for helicopter handling qualities in the hovering and low airspeed flight regime are contained in MIL-H-8501A. Military specifications are only a guide. The following list identifies the paragraph number in MIL-H-8501A and a short description of the requirement.

3.2 Longitudinal characteristics
- 3.2.1 Longitudinal control margin; controls and helicopter shake, vibration, roughness
- 3.2.2 Hovering turns on a spot
- 3.2.3 Longitudinal trimmability, control jump
- 3.2.4 Longitudinal control force gradient
- 3.2.5 Quick stop, rapid acceleration
- 3.2.6 Limit control force
- 3.2.7 Breakout force
- 3.2.8 Control coupling
- 3.2.9 Control response
- 3.2.10 Static longitudinal stability
  - 3.2.10.1 Critical center of gravity
  - 3.2.10.2 Trim change in climbs and descents
- 3.2.11 Dynamic longitudinal stability
  - 3.2.11.1 Dynamic control response
  - 3.2.11.2 Gust response
- 3.2.12 Control response, normal acceleration
- 3.2.13 Control power
- 3.2.14 Control response damping

3.3 Directional and lateral characteristics
- 3.3.1 Directional ground handling
- 3.3.2 Control margins; controls and helicopter shake, vibration, roughness
- 3.3.3 Lateral and directional controls for hovering turns on a spot
- 3.3.4 Asymmetrical lateral center of gravity, lateral control
- 3.3.5 Control power
- 3.3.6 Control margin
- 3.3.7 Control response
- 3.3.8 Directional control for autorotation
- 3.3.10 Trimmability, control jump
- 3.3.11 Control force gradients
- 3.3.12 Limit forces
- 3.3.13 Breakout forces
- 3.3.14 Control coupling

- 3.3.15 Lateral control response, over control
- 3.3.16 Lateral and directional control response
- 3.3.17 Lateral trim with power effects
- 3.3.18 Lateral control power
- 3.3.19 Hovering roll and yaw damping
- 3.4 Vertical characteristics
  - 3.4.1 Vertical positioning
  - 3.4.2 Collective control characteristics
  - 3.4.3 Control coupling
- 3.5 Autorotation, rotor characteristics, and miscellaneous
  - 3.5.4 Takeoffs and landings
    - 3.5.4.1 Wind requirements, vertical takeoff
    - 3.5.4.2 Wind requirements, rolling takeoff
    - 3.5.4.3 Wind requirements, landing
    - 3.5.4.4 Autorotation landing distance
  - 3.5.5 Autorotation delay time
    - 3.5.5.1 Autorotation entry characteristics
  - 3.5.6 Autorotation control forces
  - 3.5.7 Autorotation landing speed

Additional requirements are included in the military specification related to boosted controls, failure modes, automatic stabilization equipment, and vibrations. Some of these paragraphs may apply to specific helicopters equipped with an AFCS or highly augmented systems.

MIL-H-8501A is old and under most circumstances supplemented with additional detailed specifications listed in the procurement documents. These must be researched thoroughly and carefully to ensure contractual compliance and a complete evaluation.

A new handling qualities specification has been under development for several years, but to date not finalized. This new specification may include frequency domain tests to evaluate higher the order aircraft flight control systems expected in future helicopters.

8.9 GLOSSARY

8.9.1 Notations

A_{1s}	Lateral cyclic pitch angle, shaft referenced
a_{1s}	Longitudinal flapping angle, shaft referenced
AFCS	Automatic Flight Control System
AGL	Above ground level
B	Damping constant
B_{1s}	Longitudinal cyclic pitch angle, shaft referenced
b_{1s}	Lateral flapping angle, shaft referenced
$C_{1/2}$	Cycles to one-half amplitude
C_d	Cycles to double amplitude
CG	Center of gravity
deg	Degree
e	Base of natural logarithm
FC	Fuel count
g	Gravity
H	Rotor hub force, $\perp$ to shaft
H_{P_o}	Observed pressure altitude
HQR	Handling Qualities Rating
IGE	In ground effect
in	Inch
I_{xx}	Moment of inertia about x axis, roll moment of inertia
I_{xz}	Product of inertia about x z axes
I_{yy}	Moment of inertia about y axis, pitch moment of inertia
I_{zz}	Moment of inertia about z axis, yaw moment of inertia
kn	Knot
KTAS	Knots true airspeed
L	Net moment about x axis, Roll moment, Lift, Length
$L_{A_{1s}}$	Roll moment due to lateral cyclic pitch angle
$_{PED}$	Roll moment due to pedal control
L_{Equiv}	Equivalent length
L_p	Roll moment due to roll rate

$L_{\theta_{TR}}$	Roll moment due to tail rotor pitch angle
L_r	Roll moment due to yaw rate
L_v	Roll moment due to lateral velocity
M	Net moment about y axis, Pitch moment
m	Mass
$M_{B_{1s}}$	Pitch moment due to longitudinal cyclic pitch angle
$M_{\delta_{LONG}}$	Pitch moment due to longitudinal control
M_q	Pitch moment due to pitch rate
M_{θ_C}	Pitch moment due to collective pitch angle
$M_{\dot{\theta}}$	Pitch moment due to time rate of change of pitch angle
M_u	Pitch moment due to longitudinal velocity
M_w	Pitch moment due to vertical velocity
N	Net moment about z axis, Yaw moment
$N_{A_{1s}}$	Yaw moment due to lateral cyclic pitch angle
$N_{\delta_{LAT}}$	Yaw moment due to lateral control
$N_{\delta_{PED}}$	Yaw moment due to pedal control
N_p	Yaw moment due to roll rate
$N_{\theta_{TR}}$	Yaw moment due to tail rotor pitch angle
N_R	Main rotor speed
N_r	Yaw moment due to yaw rate
N_v	Yaw moment due to lateral velocity
OGE	Out of ground effect
p	Angular velocity about x axis, Roll rate
$\dot{p}$	Angular acceleration about x axis
Q	Engine torque
q	Angular velocity about y axis, Pitch rate
q_{ss}	Steady state pitch rate
$\dot{q}$	Angular acceleration about y axis
r	Angular velocity about z axis, Yaw rate
$\dot{r}$	Angular acceleration about z axis
s	Second
T	Thrust
t	Time

$T_{1/2}$	Time to one-half amplitude
$T_{1/N}$	Time to decay to 1/N of maximum amplitude
T_d	Time to double amplitude
T_o	Observed temperature
TPP	Tip path plane
u	Translational velocity component along x axis
u_0	Initial velocity
u_P	Downward velocity
u_T	Transverse velocity
$\dot{u}$	Linear acceleration along x axis
V	Velocity, Free stream velocity, Relative velocity
v	Translational velocity component along y axis
VAR	Vibration Assessment Rating
$\dot{v}$	Linear acceleration along y axis
w	Translational velocity component along z axis
$\dot{w}$	Linear acceleration along z axis
X	Resultant force in x direction
x	Orthogonal direction along longitudinal axis of the aircraft
$X_{B_{1s}}$	Longitudinal force due to longitudinal cyclic pitch angle
X_δ	Longitudinal force due to control
$X_{\delta_{LONG}}$	Longitudinal force due to longitudinal control
X_q	Longitudinal force due to pitch rate
X_{θ_C}	Longitudinal force due to collective pitch angle
X_u	Longitudinal force due to longitudinal velocity
X_w	Longitudinal force due to vertical velocity
$\dot{x}$	Time rate of change of x
Y	Resultant force in y direction
y	Orthogonal direction along lateral axis of the aircraft
$Y_{A_{1s}}$	Side force due to lateral cyclic pitch angle
$Y_{\delta_{LAT}}$	Side force due to lateral control
$Y_{\delta_{PED}}$	Side force due to pedal control
Y_p	Side force due to roll rate
$Y_{\theta_{TR}}$	Side force due to tail rotor pitch angle

Y_r	Side force due to yaw rate
Y_u	Side force due to longitudinal velocity
Y_v	Side force due to lateral velocity
Z	Resultant force in z direction
z	Orthogonal direction along vertical axis of the aircraft
$Z_{B_{1s}}$	Vertical force due to longitudinal cyclic pitch angle
Z_{δ_C}	Vertical force due to collective control
Z_q	Vertical force due to pitch rate
Z_{θ_C}	Vertical force due to collective pitch angle
Z_u	Vertical force due to longitudinal velocity
Z_w	Vertical force due to vertical velocity

8.9.2 Greek Symbols

α (alpha)	Angle of attack
α_{eff}	Effective angle of attack
δ (delta)	Control
δ_C	Collective control
δ_{LAT}	Lateral control
δ_{LONG}	Longitudinal control
δ_{PED}	Pedal control
ϕ (phi)	Roll angle
φ (psi)	Inflow angle
π (pi)	Mathematical constant
θ (theta)	Pitch angle, shaft angle
θ_C	Collective pitch angle
θ_{max}	Maximum pitch angle
θ_{TR}	Tail rotor pitch angle
$\dot{\theta}$	Time rate of change in θ

$\dot{\theta}_{max}$	Maximum time rate of change in θ
τ (tau)	Time constant
ω_d (omega)	Damped frequency
ζ (zeta)	Damping ratio

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CHAPTER NINE

COUPLED LONGITUDINAL AND LATERAL-DIRECTIONAL STABILITY, CONTROL, AND FLYING QUALITIES

ROTARY WING STABILITY AND CONTROL

CHAPTER NINE

FIGURES

CHAPTER NINE

TABLES

CHAPTER NINE

COUPLED LONGITUDINAL AND LATERAL-DIRECTIONAL STABILITY, CONTROL, AND FLYING QUALITIES

9.1 INTRODUCTION

Previous chapters address helicopter stability, control, and flying qualities topics when coupling between aircraft axes is considered negligible or when the pilot suppresses coupling through manipulation of the flight controls. Helicopters, particularly single rotor vehicles, have highly non symmetric configurations and are subject to trim and response coupling between axes. This chapter discusses stability, control, and flying qualities when longitudinal and lateral-directional coupling is present for hover, low airspeed, and forward flight. Because of the helicopter's asymmetric configuration, coupling is a significant factor in assessing mission related flying qualities in all phases of flight. Understanding, developing meaningful test procedures, and documenting any undesirable coupling effects on flying qualities is essential in evaluating a helicopter's ability to perform its mission.

9.1.1 Definition of Coupling

Providing a comprehensive, all inclusive definition is a difficult undertaking; however, a working definition of coupling is:

"Coupling is the generation of a force along or moment about an aircraft axis resulting from forces, moments, or disturbances associated with another axis."

9.2 PURPOSE OF TEST

The purpose of these tests is to evaluate pilot requirements to compensate for coupling in trimmed steady flight and to suppress coupling in short and long term dynamic situations. As a minimum, coupling evaluations include:

1. Trim control positions for equilibrium in hover, low airspeed, and forward flight.

2. Trim control positions to maintain steady non-rectilinear flight such as forward flight turns.

3. Control positions required to perform long term flight condition changes such as level accelerations, transition to climb, and diving accelerations for ordnance delivery.

4. Short term, off axis aircraft responses to rapid control inputs.

9.3 THEORY

Linear analysis of helicopter characteristics is used to gain insight into uncoupled helicopter characteristics. Linear analysis remains a valid tool to investigate the origins and effects of coupling. The cross-axis derivatives previously neglected, or assumed suppressed by pilot control inputs, provide the basis for understanding coupling.

Chapter 4 gives insight to the various control and flight condition effects on tip path plane (TPP) longitudinal flapping angle (a_{1s}) and lateral flapping angle (b_{1s}). In summary, a_{1s} increases (aft tilt of TPP) are related to negative B_{1s} changes (aft cyclic control movement), airspeed or advance ratio (μ) increases, increased collective pitch (θ_C), negative inflow (λ) changes (upward through rotor), positive roll rates (p), and negative pitch rates (q). The b_{1s} increases (right tilt of TPP) result from positive A_{1s} changes (right cyclic control movement), airspeed (μ) increases with a conned rotor, negative pitch rates, and negative roll rates. The coning angle (a_0) represents a balance between blade thrust forces and blade inertial or centrifugal forces due to rotational velocity. These contributions to TPP position were developed based on longitudinal translation of the helicopter. In lateral translation the same types of TPP tilt occurs in an axis system shifted by 90˚. Table 9.I summarizes control, airspeed, inflow, and aircraft angular rate contributions to TPP tilt for longitudinal and lateral translation flight. A review of Table 9.I reveals several instances where disturbances along or about one axis causes changes with respect to another axis, or coupling.

Table 9.I
Tip Path Plane Position Summary

PARAMETER	FLIGHT CONDITION/TPP TILT INTRODUCED			
	Forward Translation		Right Translation	
	a_{1s} change	b_{1s} change	a_{1s} change	b_{1s} change
Airspeed increase	Aft tilt	Right tilt	Aft tilt	Left tilt
Inflow decrease (Up flow)	Aft tilt, no effect in hover	Right tilt during coning increase, no effect in hover	Aft tilt during coning increase, no effect in hover	Left tilt, no effect in hover
$+B_{1s}$ *	Forward tilt	No effect	Forward tilt	No effect
$+A_{1s}$ *	No effect	Right tilt	No effect	Right tilt
$+\theta_C$	Aft tilt, no effect in hover	Right tilt during coning increase, no effect in hover	Aft tilt during coning increase, no effect in hover	Left tilt, no effect in hover
+q (Nose up)	Forward tilt	Left tilt	Forward tilt	Left tilt
+p (Right)	Aft tilt	Left tilt	Aft tilt	Left Tilt
* Valid for rotor with zero flapping hinge offset and no flapping springs				

Chapter 4 also summarizes the effects of hinge offset on rotor hub moments, and the shift in cyclic control phasing resulting from the difference between the natural flapping frequency and the rotor rotational frequency (forcing function frequency). The shift in cyclic control phasing results in lateral tilt with longitudinal cyclic changes (and vice versa) if the helicopter designer does not reorient the swashplate inputs to blade pitch. As shown in Chapter 4, the cyclic phase shift with flapping hinge offset is dependent on blade flap damping (Lock number/density altitude). The hub moments available through flapping hinge offset provide one method of trimming the helicopter for equilibrium or steady flight in conditions where the rotor thrust axis is not aligned through the helicopter center of gravity (CG). Thrust offset from the CG can result in cross axis response when thrust changes occur.

The configuration of the single rotor helicopter plays a significant part in inter-axis response or coupling. Main rotor torque changes and tail rotor vertical position are substantial contributors. For example, main rotor torque changes resulting from collective and inflow change produce yawing moments. High tail rotor thrust lines produce roll moments with directional control inputs.

9.3.1 Dominant Coupling Derivatives

Coupling effects can be assessed by considering the flight condition (u, v, w, p, q, r) and control (A_{1s}, B_{1s}, θ_C, θ_{TR}) effects on the moment equations. Force contributions along the x, y, and z axes from flight condition and control changes are typically small compared to the gravity terms that exist after attitude changes occur, and are neglected for this discussion. These forces must be considered for a complete equilibrium or performance solution.

Table 9.II lists the dominant coupling derivatives and their major origins.

Table 9.II
Coupling Moment Derivatives

DERIVATIVE	DOMINANT ROTOR PARTIAL	DESCRIPTION
L_u	$\frac{1}{I_{xx}}\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial b_{1s}}{\partial u}$ Where: $\frac{\partial b_{1s}}{\partial u} = \frac{\partial}{\partial u}\left(\frac{4}{3}\mu a_0\right)$ For the tail rotor: $\frac{1}{I_{xx}}\ h_{TR}\ \frac{\partial T_{TR}}{\partial u}$	Roll moment due to longitudinal velocity. Right roll with increased speed expected from main rotor and from high tail rotor for small sideslip angles.
L_w	$\frac{1}{I_{xx}}\ h'_{lat}\frac{\partial T}{\partial w}$	Roll moment due to vertical velocity + or - depending on sign of h'_{lat}. Due to thrust vector displaced laterally from CG with offset hinges, tail rotor height, and lateral CG displacements.
L_q	$\frac{1}{I_{xx}}\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial b_{1s}}{\partial q}$ Where: $\frac{\partial b_{1s}}{\partial q} = \frac{\partial}{\partial q}\left(-\frac{q}{\Omega}\right) = -\frac{1}{\Omega}$	Roll moment due to pitch rate. Left roll expected with nose up pitch rate.
$L_{B_{1s}}$	$\frac{1}{I_{xx}}\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial b_{1s}}{\partial B_{1s}}$	Roll moment due to longitudinal cyclic pitch angle. Due to flapping hinge offset or flapping springs. Right roll with positive longitudinal input when swashplate phasing not incorporated.

Coupling Moment Derivatives (cont'd)

L_{θ_C}	$\frac{1}{I_{xx}} h'_{lat} \frac{\partial T}{\partial \theta_C}$	Roll moment due to collective pitch angle + or - depending on sign of h'_{lat}. Due to thrust vector displaced laterally from CG, tail rotor height, and lateral CG displacements.
$L_{\theta_{TR}}$	$\frac{1}{I_{xx}} h_{TR} \frac{\partial T_{TR}}{\partial \theta_{TR}}$	Roll moment due to tail rotor pitch angle. Right roll with increased pitch (left directional control input) with tail rotor thrust axis above roll axis.
M_v	$\frac{1}{I_{yy}} \left(Th + \frac{ebM_S\Omega^2}{2} \right) \frac{\partial a_{1s}}{\partial v}$ Where: $\frac{\partial a_{1s}}{\partial v} = \frac{4}{3} \frac{a_0}{\Omega r}$ For the tail rotor: $\frac{1}{I_{yy}} l_t \frac{\partial T_{TR}}{\partial v} \sin i_{TR}$	Pitch moment due to lateral velocity. Nose up pitch with increased right lateral speed.
M_p	$\frac{1}{I_{yy}} \left(Th + \frac{ebM_S\Omega^2}{2} \right) \frac{\partial b_{1s}}{\partial p}$ Where: $\frac{\partial b_{1s}}{\partial p} = \frac{\partial}{\partial p} \left(\frac{p}{\Omega} \right) = \frac{1}{\Omega}$ For the tail rotor: $\frac{1}{I_{yy}} l_t \frac{\partial T_{TR}}{\partial p} \sin i_{TR}$	Pitch moment due to roll rate. Nose up moment with right roll rate.
M_{θ_C}	$\frac{1}{I_{yy}} \left(Th + \frac{ebM_S\Omega^2}{2} \right) \frac{\partial a_{1s}}{\partial \theta_C}$ Where: $\frac{\partial a_{1s}}{\partial \theta_C} = \frac{\partial}{\partial \theta_C} \left(\frac{8}{3} \theta_C \mu + 2\lambda\mu \right)$	Pitch moment due to collective pitch angle. Nose up moment with increased collective pitch.
$M_{A_{1s}}$	$\frac{1}{I_{yy}} \left(Th + \frac{ebM_S\Omega^2}{2} \right) \frac{\partial a_{1s}}{\partial A_{1s}}$	Pitch moment due to lateral cyclic pitch angle. Due to flapping hinge offset or flapping springs. Nose up pitch with right lateral inputs when swashplate phasing not incorporated.

Coupling Moment Derivatives (cont'd)

$M_{\theta_{TR}}$	$\frac{1}{I_{yy}} l_t \frac{\partial T_{TR}}{\partial \theta_{TR}} \sin i_{TR}$	Pitch moment due to tail rotor pitch angle. Due to inclination of tail rotor thrust vector out of vertical plane.
N_w	$\frac{1}{I_{zz}} \frac{\partial Q_{MR}}{\partial w}$	Yaw moment due to vertical velocity. Nose left moment with increased (downward) vertical speed, torque decreases as inflow increases.
N_p	$\frac{1}{I_{zz}} l_t \frac{\partial T_{TR}}{\partial p}$ Where: $\frac{\partial T_{TR}}{\partial p} = h_{TR} \frac{\partial T_{TR}}{\partial v}$	Yaw moment due to roll rate. Nose right moment with right roll rate when tail rotor thrust axis above roll axis.
$N_{A_{1s}}$	$\frac{1}{I_{zz}} Th' \frac{\partial b_{1s}}{\partial A_{1s}}$	Yaw moment due to lateral cyclic pitch angle. Adverse/proverse yaw, adverse is nose left yaw moment with right lateral control input.
$N_{B_{1s}}$	$\frac{1}{I_{zz}} \frac{\partial Q_{MR}}{\partial B_{1s}}$ alternatively, $\frac{1}{I_{zz}} \frac{\partial Q_{MR}}{\partial w_{rotor}} \frac{\partial w_{rotor}}{\partial B_{1s}}$	Yaw moment due to longitudinal cyclic pitch angle. Nose right moment with forward longitudinal input.
N_{θ_C}	$\frac{1}{I_{zz}} \frac{\partial Q_{MR}}{\partial \theta_C}$	Yaw moment due to collective pitch angle. Nose right moment with increased collective pitch.

Where:

a_0	- Coning angle
A_{1s}	- Lateral cyclic pitch angle, shaft referenced
a_{1s}	- Longitudinal flapping angle, shaft referenced
b	- Number of blades
B_{1s}	- Longitudinal cyclic pitch angle, shaft referenced
b_{1s}	- Lateral flapping angle, shaft referenced
e	- Flapping hinge offset
h	- Height of hub above CG
h'_{lat}	- Lateral distance between the rotor shaft and the CG
h_{TR}	- Height of the tail rotor above CG
i_{TR}	- Tail rotor inclination out of vertical plane

I_{xx} - Moment of inertia about x axis, roll moment of inertia
I_{yy} - Moment of inertia about y axis, pitch moment of inertia
I_{zz} - Moment of inertia about z axis, yaw moment of inertia
L - Roll moment
λ - Inflow ratio
l_t - Distance from the tail to the CG
M - Pitch moment
μ - Advance ratio
M_S - Blade mass moment
N - Yaw moment
p - Roll rate
q - Pitch rate
θ_C - Collective pitch angle
Q_{MR} - Main rotor torque
θ_{TR} - Tail rotor pitch angle
R - Rotor radius
r - Yaw rate
T - Thrust
T_{TR} - Tail rotor thrust
u - Translational velocity component along x axis
v - Translational velocity component along y axis
Ω - Rotor angular velocity
w - Translational velocity component along z axis
w_{rotor} - TPP translational velocity component along z axis.

9.3.2 Coupling Derivative Origins

In this section a brief description of the potential origins of each of the major coupling derivatives is presented. These descriptions are based on theories presented in Chapter 4. Coupling is considered in terms of main and tail rotor contributions. Some potential interferences are addressed where applicable.

9.3.2.1 ROLL MOMENT DUE TO LONGITUDINAL VELOCITY

Roll moments due to longitudinal velocity changes can be attributed to the TPP coning angle, a_0. As velocity increases with coning present, the forward and aft blades see angle of attack (α) differences. As shown in Figure 9.1, the forward blade at $\psi = 180°$ realizes an increase in α; whereas, the aft blade at $\psi = 0°$ experiences an α reduction. Without flapping hinge offset, this lift dissymmetry produces maximum flapping 90˚ later with the retreating blade flapping up and the advancing flapping down. This flapping generates a right TPP tilt and a right roll moment. With flapping hinge offset, the flapping phase shift is less than 90˚ and the roll moment is increased due to the hub moments.

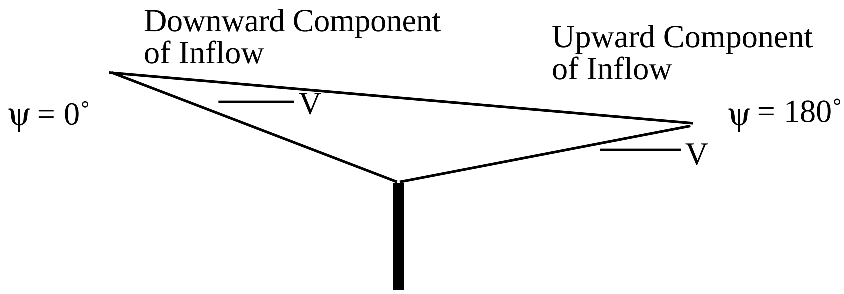


Figure 9.1
Effective Inflow Variation in Translational Flight Due to Coning

Some roll moments may be introduced due to the tail rotor. If the tail rotor is located above the vertical CG, as shown in Figure 9.2, tail rotor thrust changes produce roll moments. If the sideslip angle (β) is small, ΔT_{TR} is to the right, or there is an increase in T_{TR} because of the additional mass flow influenced by the tail rotor. When significant right sideslip angles exist, the ΔT_{TR} is to the left since the inflow (λ_{TR}) is reduced. For left sideslip angles, the momentum and inflow effects are additive giving a ΔT_{TR} to the right.

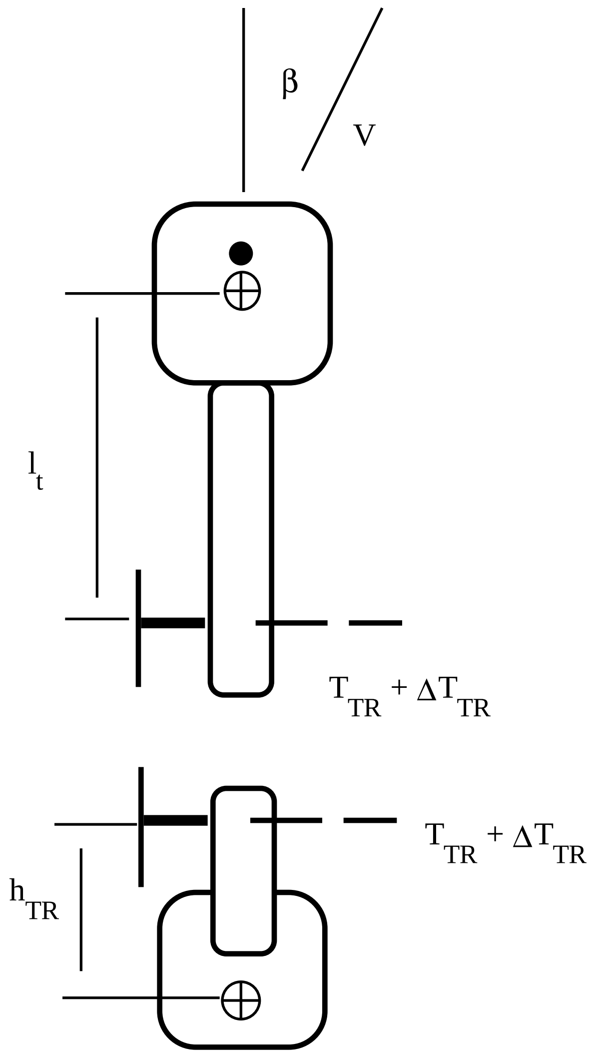


Figure 9.2
Tail Rotor Roll Moments

9.3.2.2 ROLL MOMENT DUE TO VERTICAL VELOCITY

Roll moments due to vertical velocity changes are produced by the main rotor when the thrust vector is not aligned through the aircraft CG. This misalignment increases when the tail rotor thrust axis is raised above the CG or the CG is laterally displaced. Figure 9.3 shows relative TPP inclinations for three conditions of tail rotor vertical position for hover when no other external forces or moments are present. In Figure 9.3 (a) the main rotor thrust (T), T_{TR} and weight (W) all act through the CG and produce no moments. The main rotor thrust balances the weight component (W cosϕ) and the weight component (W sin ϕ) balances the T_{TR}. For Figure 9.3 (b), left TPP tilt is required to balance the tail rotor moment (hT_{TR}). The left TPP tilt also produces a hub moment to the left. The lateral component of T and the T_{TR} are balanced by the weight component (W sinϕ). The moments produced by T_{TR} are balanced by the hub moment and the thrust axis offset (right of CG). In Figure 9.3 (c) the helicopter is level laterally and the T_{TR} is balanced by T sinb_{1s}. The higher tail rotor location produces larger right roll moments which are balanced by the component Tsinb_{1s} and the hub moment. As shown in Figure 9.3, the trend for increased tail rotor height is increased right bank angle and left TPP tilt. If hub moments are not available, the left tilt of the TPP with higher tail rotors is accentuated since greater thrust induced roll moments are required to counter tail rotor roll moments. Although Figure 9.3 is for hover, the concept applies to all flight regimes. Table 9.III summarizes the conditions depicted in Figure 9.3. The greater tail rotor heights cause increased main rotor thrust misalignment with the CG.

Thrust changes occur from vertical velocity changes producing roll moments proportional to the distance from the CG to the main rotor thrust axis. Thrust changes result from rotor inflow variations with vertical speed.

Lateral CG displacement from the main rotor shaft with flapping hinge offset is depicted in Figure 9.4 for hover. Tail rotor thrust is assumed to act through the CG for simplification. As shown, the helicopter requires increased bank angle in the direction of the CG displacement. The rotor thrust produces a left roll moment which is balanced by the hub moment L_H. Vertical velocity changes with the attendant thrust change cause roll moments in the direction of thrust change.

A secondary contribution to the roll moment in forward flight exists since the thrust changes cause coning changes. These coning changes produce lateral TPP tilt of magnitude 4/3 μa_0. The coning changes are transient in nature and subside as the inflow changes during the subsequent helicopter response.

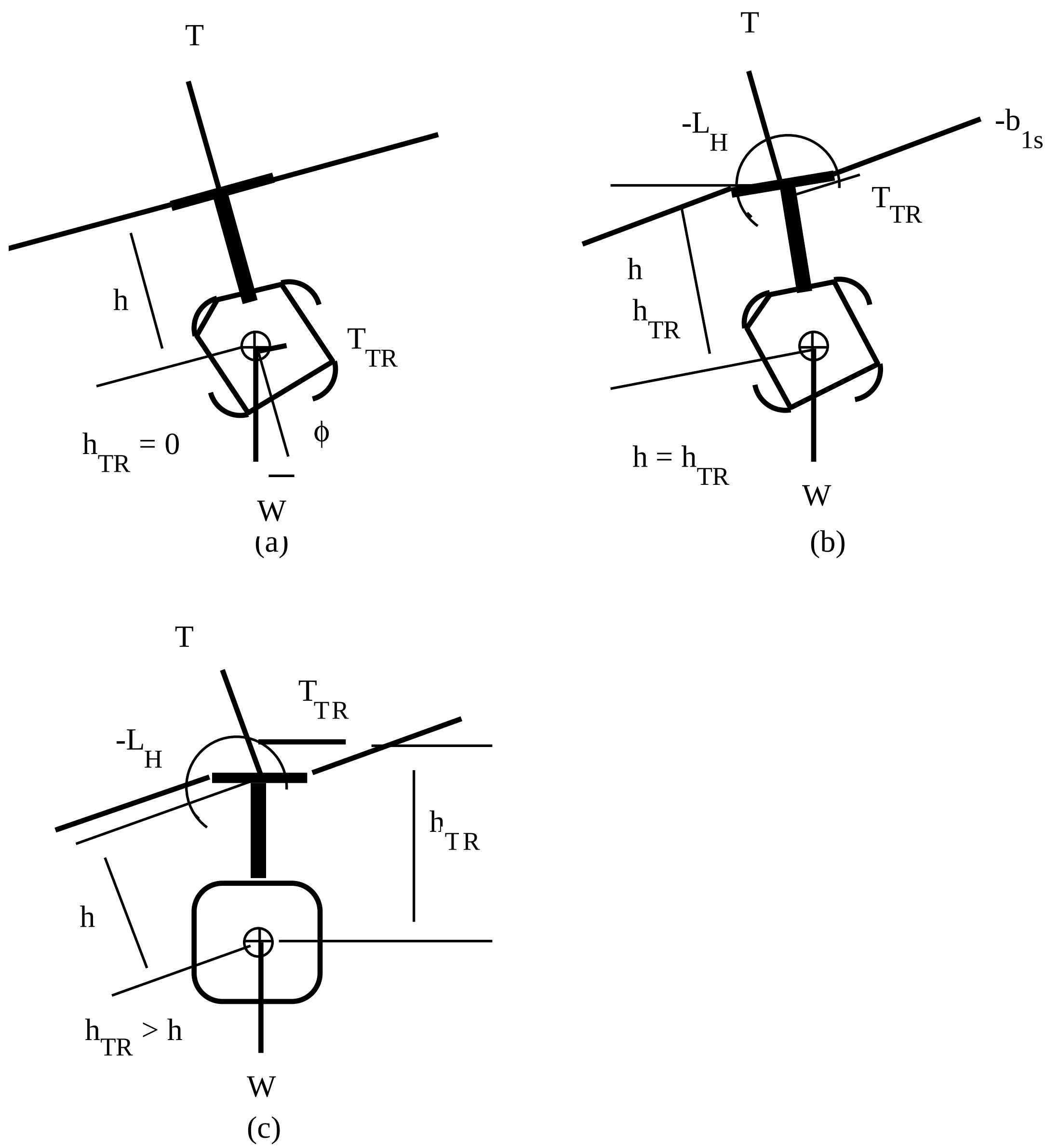
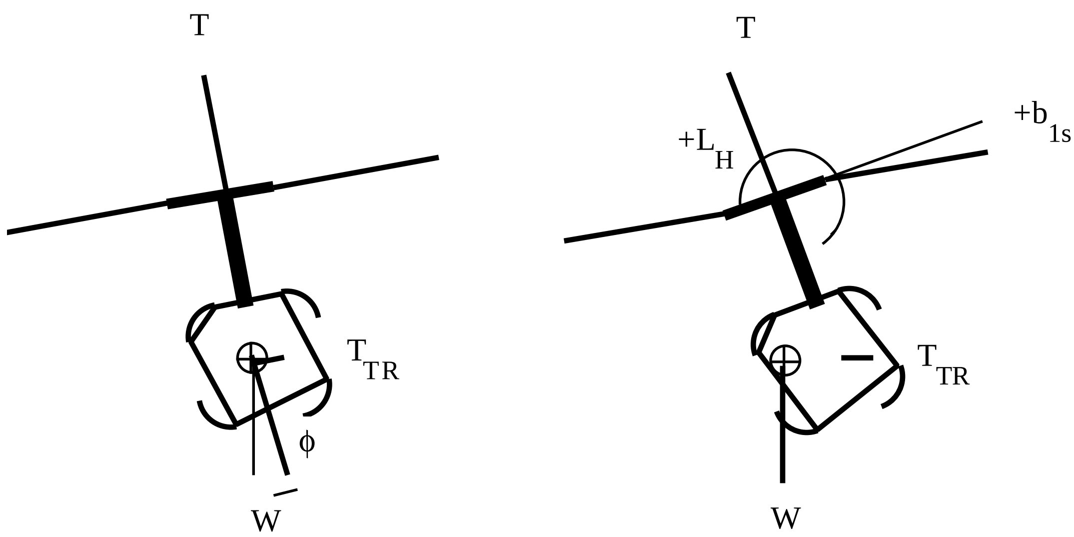


Figure 9.3
Tail Rotor Height Effects on Main Rotor Thrust Alignment



(a) CG Aligned With Mast

(b) Left of Shaft Axis

Figure 9.4
Lateral CG Effects on Main Rotor Thrust Alignment

Table 9.III
Tail Rotor Height Effects on Main Rotor Thrust Alignment Summary

CONDITION	TPP TILT (b_{1s})	BANK ANGLE	REMARKS
(a) $h_{TR} = 0$	0	Left	All forces aligned through CG. No moments required. Weight component ($W \sin\phi$) balances tail rotor thrust (T_{TR})
(b) $h_{TR} = h$	Left	Left, $\lvert\phi_{(b)}\rvert < \lvert\phi_{(a)}\rvert$	$\lvert T \sin b_{1s}\rvert < T_{TR}$; $-L_H$ required to balance moment. $W \sin\phi$ and $T \sin b_{1s}$ balance T_{TR}.
(c) $h_{TR} > h$	Left, $\lvert b_{1s(c)}\rvert > \lvert b_{1s\ (b)}\rvert$		Forces in balance; $\lvert T \sin b_{1s}\rvert = T_{TR}$; $-L_H$ required to balance moments, $\lvert L_{H\ (c)}\rvert > \lvert L_{H\ (b)}\rvert$

9.3.2.3 ROLL MOMENT DUE TO PITCH RATE

Roll moments result from helicopter pitch rates. For an aircraft pitching nose up, the rotor follows the shaft in pitch rate after an initial lag (Δa_{1s}) in angular displacement. Figure 9.5 illustrates the shaft and rotor pitching at the same rate. The forward blade (ψ = 180˚) undergoes an angle of attack reduction, and the aft (ψ = 0˚) blade an angle of attack increase. These angle of attack changes produce lift changes which result in the retreating blade flapping down at ψ = 270˚ (for a rotor with no offset) and the advancing blade up at ψ = 90˚, or a left TPP tilt. The lateral tilt (Δb_{1s}) is $-q/\Omega$ as developed in Chapter 4. The TPP left tilt with nose up rates causes left roll moments.

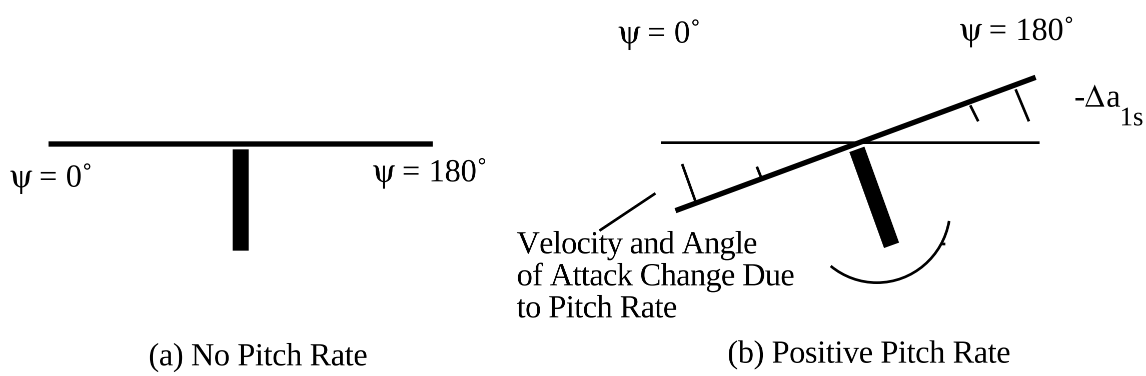


Figure 9.5
Rotor Angle of Attack Changes Due to Pitch Rate

For helicopters having flapping hinge offset, the TPP tilt with pitch rate occurs at a rotor azimuth angle less than 90˚ later; thus, the TPP moves left and forward with nose up pitch. Although the lateral tilt is somewhat reduced with flapping hinge offset, the roll moments are greater since hub moments are present.

9.3.2.4 ROLL MOMENT DUE TO LONGITUDINAL CYCLIC PITCH ANGLE

A rotor without flapping hinge offset normally is rigged so longitudinal cyclic control movements produce the maximum cyclic pitch changes at blade azimuth positions of 90˚ and 270˚. This rigging configuration is a natural fallout of the 90˚ phase shift which occurs for an oscillatory system forced at its natural frequency. When flapping hinge offset or flapping springs are used, the blade natural flapping frequency becomes greater than the blade rotating frequency. The result is a phase shift of less than 90˚. The magnitude of this shift depends on the frequency ratio (ω/ω_n) and blade damping which is affected by Lock number or density altitude. The frequency ratio is dependent on both spring stiffness and rotor speed when flapping springs are used, and only on the offset dimension when flapping hinge offset is present.

During design of the helicopter, efforts are made to phase the cyclic pitch inputs to account for flapping phase shifts less than 90˚. For example, if the sea level phase shift for a rotor with flapping hinge offset is 84˚, the swashplate to pitch change rigging might be such that longitudinal cyclic control movements produce maximum cyclic pitch changes at rotor azimuths of 96˚ and 276˚. The same logic is used for lateral controls. By properly phasing the swashplate, flapping responses for pure longitudinal control inputs are purely longitudinal, and lateral control inputs, purely lateral.

If the helicopter designer did not consider rotor phasing requirements, or if the helicopter is operated off design (density altitude for flapping hinge offset; density altitude and rotor speed for flapping springs), roll moments are generated with longitudinal control movements. In the case of the 84˚ phase angle example, failure to compensate results in a roll moment 10.5% of the pitch moment. The roll acceleration could be higher considering differences in aircraft moments of inertia. Moment of inertia effects are addressed in a subsequent section.

9.3.2.5 ROLL MOMENT DUE TO COLLECTIVE PITCH ANGLE

Roll moments due to collective pitch changes have the same origins as the roll moment due to vertical speed changes. The difference is the mechanics of thrust and coning change due to collective pitch rather than vertical velocity.

9.3.2.6 ROLL MOMENT DUE TO TAIL ROTOR PITCH ANGLE

Roll moments due to tail rotor pitch angle changes occur when the tail rotor thrust axis does not pass through the helicopter CG. Figure 9.6 illustrates this concept for a tail rotor mounted above the CG. If a left directional control input is made to produce a left yawing moment, tail rotor thrust (ΔT_{TR}) is increased to the right. The increased tail rotor thrust results in right roll moments in addition to the desired left yaw moment.

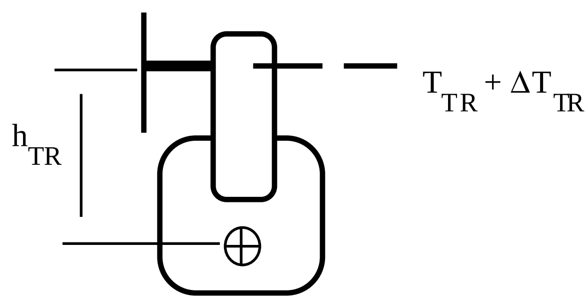


Figure 9.6
Roll Moments Due to Tail Rotor Thrust

9.3.2.7 PITCH MOMENT DUE TO LATERAL VELOCITY

The origins of the pitch moment due to the main rotor are analogous to the roll moment origins. Therefore, the description of coupled pitching moment origins is presented in less detail.

The pitch moment due to lateral velocity results from rotor coning. The rotor flapping equations developed in Chapter 4 can be reoriented for sideward flight. An increase in right sideward flight velocity causes an aft TPP tilt through the coning term similar to the increase in forward velocity producing a right TPP tilt increment. The aft tilt of the TPP with right translational velocities results in nose up pitching moments.

Pitch moments due to lateral velocity also result when the tail rotor is inclined out of the vertical plane. Tail rotor thrust changes caused by lateral velocity have components in the vertical direction thereby producing pitching moments. For a tail rotor providing an upward thrust component, an increase in right lateral velocity results in a decrease in tail rotor thrust and a nose up pitch.

9.3.2.8 PITCH MOMENT DUE TO ROLL RATE

The pitching moment due to roll rate results from increased blade angle of attack changes on the downward blade and reduced angle of attack for the upward blade of a rolling rotor. The maximum flapping caused by the blade angle of attack changes occurs 90˚ later in rotor azimuth when flapping hinge offset or flapping springs are not used, and less than 90˚ later when flapping hinge offset/flapping springs are incorporated. Thus a rotor experiencing a right roll rate experiences aft TPP tilt (Δa_{1s}) of magnitude p/Ω.

Pitch moments due to roll rate result when the tail rotor is inclined out of the vertical plane. When the helicopter is rolling to the right with a high tail rotor, the inflow change at the tail rotor reduces its thrust. For an upward canted tail rotor, the tail rotor thrust has a vertical component and a thrust reduction produces a nose up pitch moment.

9.3.2.9 PITCH MOMENT DUE TO COLLECTIVE PITCH ANGLE

In hover, pitch moments due to collective pitch angle are present when the thrust axis is not aligned with the helicopter CG. Figure 9.7 shows a hovering helicopter with flapping hinge offset and with no external forces or moments involved. If the CG is aligned longitudinally beneath the rotor shaft, the helicopter hovers as shown in Figure 9.7 (a) without longitudinal cyclic input or TPP tilt. As the CG is moved aft, forward cyclic and forward TPP tilt are required for equilibrium. A nose down hub moment ($-M_H$) is balanced by a nose up moment due to thrust. An increase in thrust from an up collective movement results in a nose up pitching moment.

In forward flight, the location of the thrust axis relative to the longitudinal CG location causes pitching moments with collective changes. In addition, the flapping equations developed in Chapter 4 show an aft TPP tilt (Δa_{1s}) associated with collective pitch increases given by $8/3\ \Delta\theta_C\mu$.

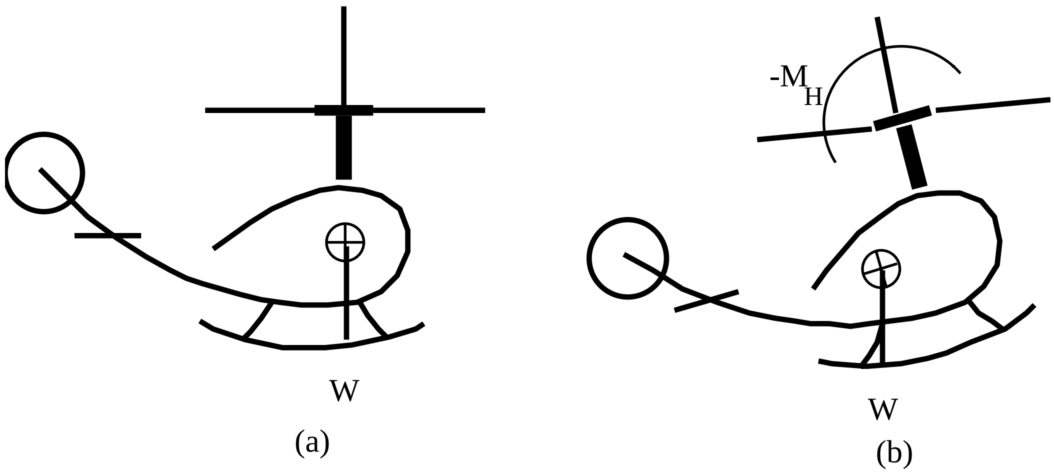


Figure 9.7
Longitudinal CG Effects on Main Rotor Thrust Alignment

9.3.2.10 PITCH MOMENT DUE TO LATERAL CYCLIC PITCH ANGLE

Pitching moments due to lateral control inputs result from non-optimum cyclic control phasing associated with flapping hinge offset or flapping springs and from operation at off design conditions when proper cyclic pitch phasing is a design feature. Off design operation for rotors with flapping hinge offset is a function of density altitude. Off design operation for rotors with flapping springs is affected by both density altitude and rotor speed. If the operating condition is such that the control phasing angle (azimuth angle between maximum cyclic input and maximum flapping) is less than the design phase angle, some nose up pitch accompanies right lateral control inputs.

9.3.2.11 PITCH MOMENT DUE TO VERTICAL VELOCITY

Pitching moments due to vertical velocity changes in forward flight represent the helicopters angle of attack stability. Angle of attack stability is an important contribution to uncoupled longitudinal stability, control, and flying qualities characteristics as discussed in Chapter 6. For this reason, angle of attack stability is not considered a coupling term in forward flight.

In hover, however, angle of attack stability does not have meaning since angle of attack is not defined. Also, in hover, helicopter pilots perceive use of the flight controls as follows:

1. Longitudinal cyclic for control of longitudinal translation and pitching moments.
2. Lateral cyclic for control of lateral translation and roll moments.
3. Directional control for yawing moments/heading.
4. Collective control for vertical translation/position.

Thus, it is desirable that vertical motions not result in pitch, roll or yaw moments. Any pitching moments resulting from vertical velocity changes in hover are observed as coupling.

The origin of pitching moments with vertical velocity is the same as for roll moments with vertical velocity. In the pitch case, if the thrust axis does not pass through the longitudinal CG position, pitching moments are produced with thrust changes. Figure 9.8 illustrates a helicopter having an aft CG where hub moments (flapping hinge offset/flapping springs) are available. The thrust produces a nose up moment which is balanced by the nose down hub moment $-M_H$. Thrust changes resulting from vertical speed changes (Δw) or inflow changes $(\Delta\lambda)$ lead to pitch moments. For the situation depicted in Figure 9.8, a positive Δw (descent) results in a thrust increase and a nose up pitching moment.

9.3.2.12 YAW MOMENT DUE TO VERTICAL VELOCITY

During vertical velocity changes, rotor in plane forces are generated which tend to accelerate or decelerate the rotor, and hence change the rotor torque required for a fixed rotor speed. Figure 9.9 illustrates a rotor blade section element that is initially in equilibrium (a). A vertical velocity change is introduced in Figure 9.9 (b) causing an increase in section lift and forward shift to remain perpendicular to the relative wind. The section drag is essentially unchanged but shifts to remain parallel to the relative wind. In Figure 9.9 (b), the in plane force (dF) is reduced and torque required (r dF) is also reduced. If the engines govern rotor speed well, the torque delivered to the main rotor is reduced and a left yawing moment results. A negative vertical velocities (aircraft climb) produces a right yaw moment. Intuitively, the limiting condition for descent is a stabilized autorotation where the pilot compensates for the left yaw moment by applying right directional control.

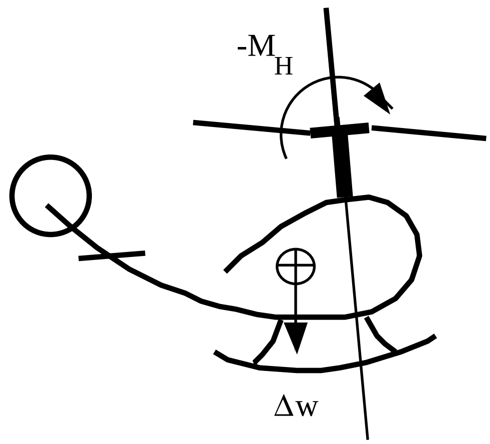


Figure 9.8
Main Rotor Thrust Alignment for Aft CG and Hub Moments

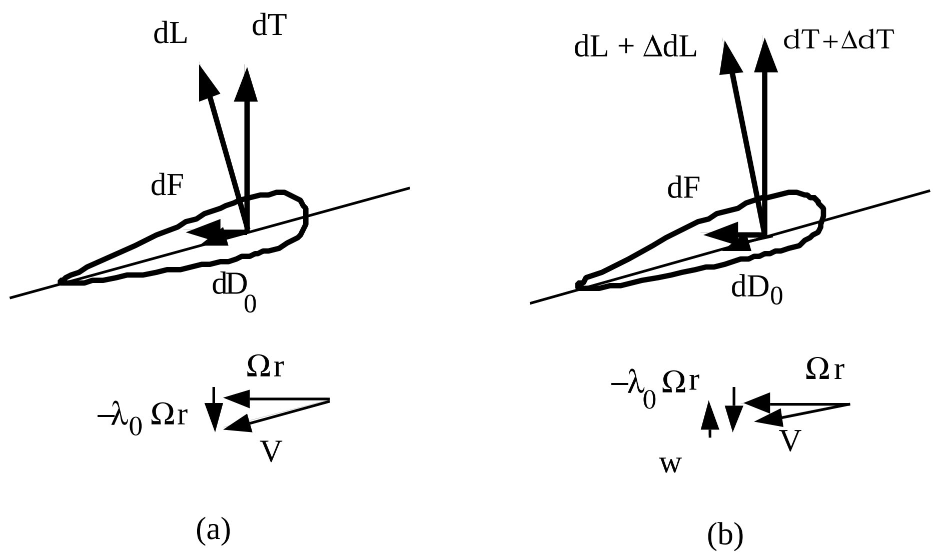


Figure 9.9
Rotor Torque Changes Due to Vertical Velocity

9.3.2.13 YAW MOMENT DUE TO ROLL RATE

Yawing moments are generated by roll rate through the tail rotor and its vertical position relative to the helicopter CG. Figure 9.10 shows a helicopter with a high tail rotor rolling to the right. The roll rate causes an increased inflow velocity to the left of magnitude $h_{TR}p$, generating a left increment in tail rotor thrust. The left tail rotor thrust increment results in a right yaw moment.

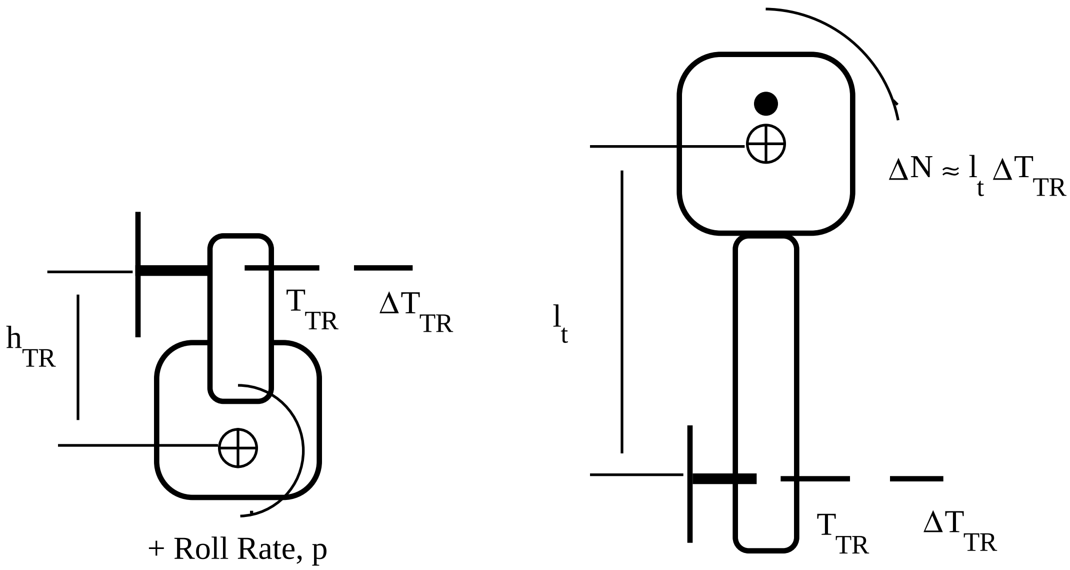


Figure 9.10
Yaw Moments Due to Roll Rate

9.3.2.14 YAW MOMENT DUE TO LATERAL CYCLIC PITCH ANGLE

Lateral control inputs result in yawing moments when the main rotor thrust axis is displaced longitudinally from the CG, as shown in Figure 9.11. The rotor thrust component, $T\sin b_{1s}$, produces a right yawing moment which is trimmed out when the aircraft is in equilibrium. When a right lateral cyclic input is made, there is a change in lateral TPP tilt (Δb_{1s}) proportional to the input producing a proverse yawing moment ($T\,h'\sin\Delta b_{1s}$). If the thrust axis passes behind the CG, lateral cyclic produces adverse yaw.

A secondary source of yawing moments with lateral control might result from rotor wake skewing due to the lateral input. A right lateral input causes an induced rotor flow to the left from the thrust change to the right ($T\sin \Delta b_{1s}$). This skewed flow impinges on the vertical tail and tail rotor causing a reduction in tail rotor thrust and a tail load change to the left. The rotor wake skew leads to a proverse yaw with lateral control input.

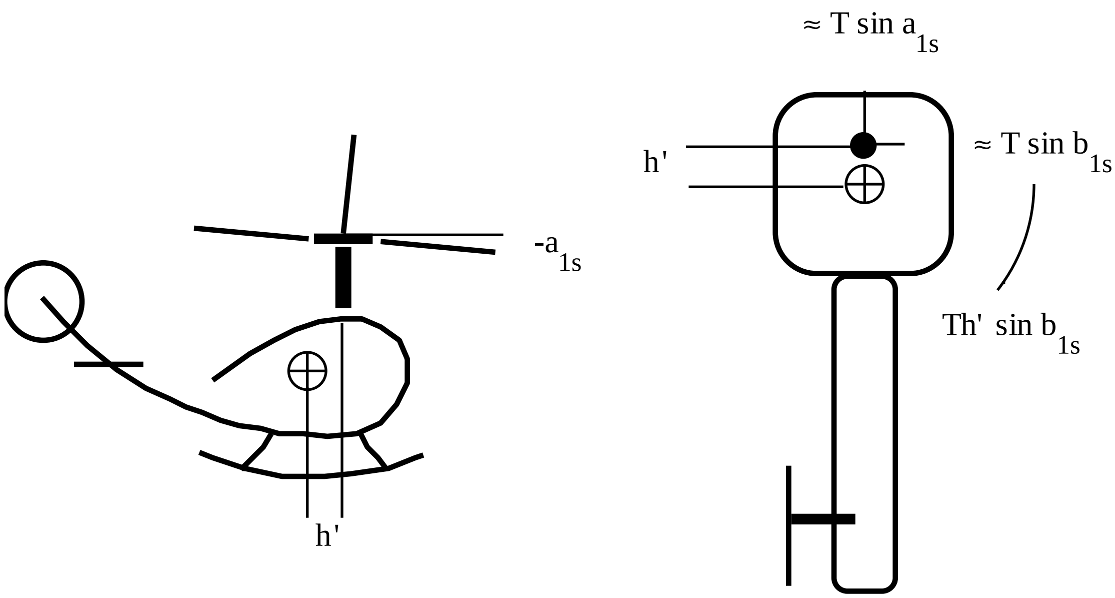


Figure 9.11
Yaw Moments Due to Lateral Cyclic

9.3.2.15 YAW MOMENT DUE TO LONGITUDINAL CYCLIC PITCH ANGLE

Longitudinal control inputs can produce yawing moments through the same mechanism as yawing moments generated by vertical velocity changes. These longitudinal control induced yawing moments are expected only in forward flight. Figure 9.12 illustrates this concept. In Figure 9.12, the pilot makes an aft control input from an initial condition and the TPP tilts aft. The aft tilt introduces a component of flow (Δw_{rotor}) normal to the TPP, upwash in this situation. The upwash increases the blade section angle of attack and reduces the in plane blade section force (dF), as previously illustrated in Figure 9.9. The reduction in dF lessens the main rotor torque required and consequently generates a left yawing moment when the engine governs rotor speed correctly. Aft longitudinal control inputs in forward flight cause torque reductions and hence left yawing moments. Conversely, a forward input produces a right yaw. For a given Δa_{1s}, the yawing moment increases with airspeed since the Δw_{rotor} component is increased proportionally to the airspeed.

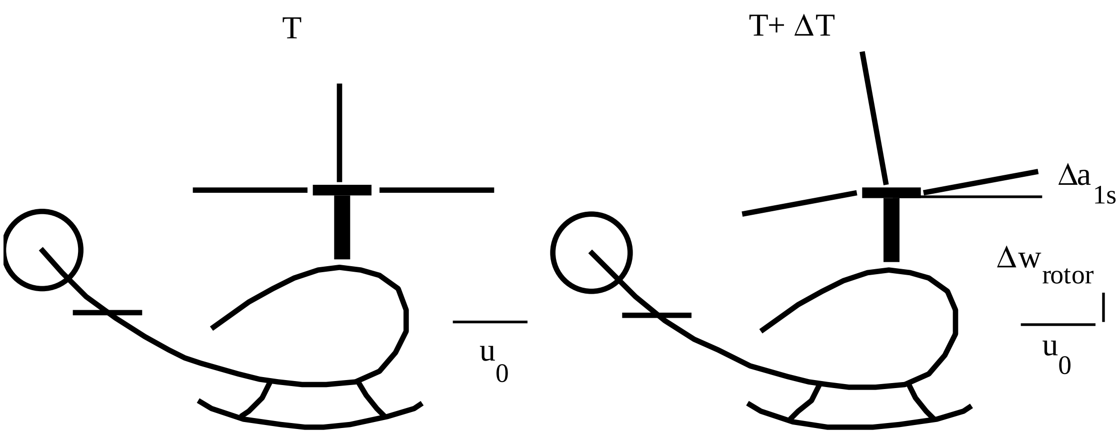


Figure 9.12
Inflow Changes Due to Longitudinal Cyclic

9.3.2.16 YAW MOMENT DUE TO COLLECTIVE PITCH ANGLE

Collective pitch changes in powered flight produce changes in main rotor torque. Figure 9.13 illustrates this concept and shows forces and flow velocities on a rotor blade element which experienced a collective pitch increase from Figure 9.13 (a) to 9.13 (b). In Figure 9.13 (b), the larger collective pitch increased the section lift, thrust, and induced velocity; however, the section drag may be essentially unchanged. The section lift is tilted further aft in the plane of rotation thereby increasing dF. The in plane torque (r dF) requires greater engine torque to maintain the governed rotor speed. The larger engine torque is reflected as an increased torque to the main rotor or an increased right yawing moment.

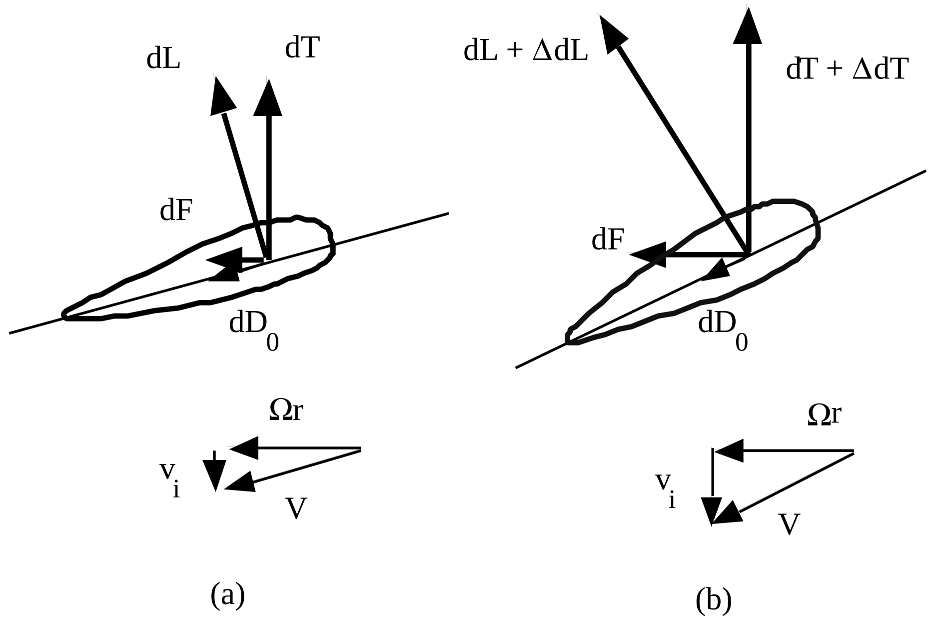


Figure 9.13
Rotor Torque Change Due to Collective Pitch Angle

9.3.2.17 YAW MOMENT DUE TO LATERAL VELOCITY

Yawing moments due to lateral velocity changes in forward flight represent the helicopter's directional stability. Directional stability is an important contribution to uncoupled lateral-directional stability, control, and flying qualities characteristics. Directional stability is not considered as a coupling term in forward flight.

In hover, the basic perception of the lateral control is the management of lateral translation and roll moments. The directional controls are used to manage heading changes. In the simplest form, yawing moments produced in sideward flight can be considered as coupling since they are not desired and the pilot is interested in controlling lateral velocity/bank angle with lateral cyclic control.

The origin of yawing moments with lateral velocity is the same in hover and forward flight. A right lateral relative velocity increases the tail rotor inflow producing a decrease in tail rotor thrust. The reduction in tail rotor thrust results in a left yawing moment.

9.3.3 Coupling Perception

Section 9.3.2 considered the moments resulting from translational velocity (u, v, w), rotational rates (p, q), and control inputs (A_{1s}, B_{1s}, θ_{TR}, θ_C). The moments generated and the aircraft angular acceleration depend on the time development of the flight parameters and control inputs. As an example, consider a helicopter with roll moments produced by longitudinal cyclic, pitch rate, and longitudinal velocity. A forward control input instantaneously produces nose down pitch acceleration and roll acceleration (conceivably right roll). The pilot senses an immediate cross axis response. As the pitch rate develops, an additional roll moment is generated. In addition, the developing right roll rate produces roll damping which causes the roll acceleration to diminish or the roll rate to stabilize. The roll moment with pitch rate reduces the stabilizing effect, and the pilot might perceive a continuing of the roll acceleration. As airspeed increases, some time later a further right roll moment is generated and a follow-on right roll acceleration results. Obviously other moments and responses occur due to the effect of other stability derivative contributions. Accelerations might be generated immediately or some time later as other flight parameters change. If for example, the pilot makes a forward longitudinal control input without control coupling ($L_{B_{1s}}$) present, a single axis (pitch) response is initially perceived, but some time later, after the pitch rate developed, a roll to the right is manifested. Roll control coupling might at first be expected, but further analysis indicates a lag or delay in the roll tendency. This delay cues the evaluator that the coupling present is not control related.

For systems with short term response approximated by first order responses, as discussed in Chapter 8, further confusion could exist. Consider a first order response with a very short time constant. The uncoupled response to a longitudinal control input is the generation of a steady pitch rate almost immediately after the control input. The presence of roll due to pitch rate coupling causes the total roll acceleration to be sensed shortly after the longitudinal input. In this situation, the evaluator might be hard pressed to sort out whether the roll response was caused by the control input or the pitch rate.

Coupling effects may be masked by the feeding of coupling effects from axis to axis. To illustrate this concept, consider a helicopter that exhibits first order pitch and roll response characteristics. Assume the helicopter has no cross axis control coupling, and moderate response time constants. The pilot makes a forward longitudinal input and a pitch rate develops. This pitch rate causes flapping in roll and some time later right roll rates are introduced. The right roll rates produce nose up longitudinal flapping which reduces the pitch rate and also reduces the right flapping and roll moment. The system exhibits coupling, but the continued cross axis effects tend to some extent to reduce the degree of coupling felt.

9.3.4 Moments of Inertia

The flying qualities evaluator is typically not interested in, or sensitive to, the magnitudes of the moments, but rather to the angular accelerations and rates produced by these moments. In the simplest form, a moment produces an angular acceleration with the magnitude directly proportional to the size of the moment and inversely proportional to helicopter moment of inertia (I_{xx} or I_{yy} or I_{zz}). Although not true in all cases, the conventional helicopter layout leads to relative moments of inertia such that the pitch (I_{yy}) and yaw (I_{zz}) moments of inertia are about the same magnitude, while the roll inertia (I_{xx}) may be approximately 20% of the pitch inertia. Two examples are used to show the effects of moment of inertia on the coupling observed.

For the first example, consider the roll moment with longitudinal cyclic. It was argued for a control phase shift of 84˚, 10.5% of the pitching moment generated by a longitudinal input is experienced in roll if the designer neglected phase shift requirements. If moments of inertia were that such I_{xx} equaled 20% I_{yy}, then the roll acceleration is 52.5% of the pitch acceleration, or the total acceleration response occurs approximately 30˚ off the desired axis.

As the second example, consider a helicopter that is first subjected to a pitch rate and then to the same magnitude roll rate. In this case, the roll moment due to pitch rate and the pitch moment due to roll rate is the same size. When relative moments of inertia are considered (I_{xx} equal approximately 20% I_{yy}), the roll acceleration due to the pitch rate would be five times the pitch acceleration due to roll rate. The evaluator might be sensitive to the roll coupling present, but unconcerned or even unaware of the pitch coupling.

Moments of inertia differences between axes may influence significantly the pilot's perception of cross axis aerodynamic or control coupling.

9.3.5 Control Versus Aerodynamic Coupling

9.3.5.1 LONGITUDINAL AND LATERAL CYCLIC INPUT PHASING

Cyclic control coupling may exist when flapping hinge offset or flapping springs are incorporated in the design. The origin of this coupling is the phase shift between the rotor azimuth for control input and the azimuth for maximum flapping. The phase shift is less than 90˚ because the rotating or forcing frequency is less than the natural flapping frequency. The designer can compensate for the control coupling by mechanically introducing the correct azimuth position for cyclic blade pitch inputs. This mechanical azimuth correction is a function of Lock number or density altitude for offset flapping hinges, and a function of both Lock number and rotor speed for flapping springs. Thus, correcting for cyclic control coupling is possible; however, the correction is valid for a single design condition (density altitude or density altitude/rotor speed).

When mechanical cyclic control phasing is used, compensation for rotor coupling due to angular rates and translational velocities is not achieved. To illustrate, a rotor with flapping hinge offset is subjected to a nose up pitch rate. This rotor in a short time pitches at the same rate as the rotor shaft, but lags behind the shaft in pitch angular displacement by the angle (Δa_{1s}). The flapping caused by pitch rate results in left tilt and some aft tilt of the rotor. The left TPP tilt results in the normal roll coupling, and the aft tilt produces an additional pitch rate damping contribution. The designer's use of mechanical phasing for cyclic inputs does not affect the flapping phasing resulting from other disturbances such as angular rates (p, q) and velocities (u, v, w).

9.3.5.2 MECHANICAL CONTROL MIXING

Various helicopter configurations incorporate combinations of longitudinal shaft tilt, lateral shaft tilt, and tail inclination out of the vertical plane. Forward longitudinal shaft tilt is used to provide an optimum cruise attitude and reduce longitudinal cyclic requirements at higher airspeeds. Left lateral shaft tilt can optimize the interplay between cyclic inputs, hub moments, and tail rotor vertical position to attain a laterally level hover. Tail rotor inclination from vertical provides a vertical upwards component of tail rotor thrust to improve performance and may expand aft CG limits.

The design lateral and longitudinal CG positions in conjunction with any shaft tilt may lead to design conditions where the thrust is not aligned through the CG. For this condition thrust change with collective inputs result in undesirable pitch and roll.

The canted tail rotor produces pitch moments when tail rotor thrust is changed. Tail rotor thrust changes also cause roll moments if the tail rotor thrust axis is not vertically aligned through the CG.

To eliminate the inter-axis coupling manifested by these designs, mixing of the cockpit control inputs is performed prior to their application at the flight control servos. Table 9.IV shows possible mechanical control mixing to eliminate inherent inter-axis control induced coupling.

Table 9.IV
Mechanical Control Mixing

FROM	TO	PURPOSE OF MIXING
Collective	Tail rotor pitch	Counter yaw moment due to main rotor torque change
Collective	Lateral cyclic	Counter roll moment due to changes in tail rotor thrust acting above/below the vertical CG. Counter roll moment due to main rotor thrust changes and thrust not acting through lateral CG.
Collective	Longitudinal cyclic	Counter pitch moment due to tail rotor thrust changes for a canted tail rotor (vertical component of T_{TR}). Counter pitch moment due to main rotor thrust changes and thrust not acting through longitudinal CG.
Directional	Longitudinal cyclic	Counter pitch moment due to tail rotor thrust change for a canted tail rotor.
Directional	Lateral cyclic	Counter roll moment due to changes in tail rotor thrust acting above/below roll axis.

Coupling generated by helicopter flight condition variables (u, v, w, p, q, r) for configurations with longitudinal/lateral main rotor shaft tilt and/or canted tail rotors generally are not compensated by the mechanical mixing unless properly scaled flight condition signals are provided to appropriate automatic flight control system servos.

9.4 TEST METHODS AND TECHNIQUES

In general, there are no clearly defined methods for evaluating helicopter response coupling. Existing specifications for helicopters and VSTOL (Vertical/short takeoff and landing) aircraft (MIL-H-8501A and MIL-F-83300) are very general regarding coupling.

MIL-H-8501A states:

"For all operating conditions, longitudinal, lateral, directional, or vertical control motions shall not produce adverse response of the helicopter due to mechanical coupling in the control system".

MIL-F-83300 states:

"Control inputs or aircraft motions about a given aircraft axis shall not induce objectionable control forces or aircraft motions about any other axis."

"The application of any cockpit control input necessary to meet any pitch, roll or yaw performance requirement of this specification shall not result in any objectionable aircraft attitudes or angular rates about the axes not under consideration."

Limits on what is "adverse" or "objectionable" are not quantified. Any evaluation of coupling effects on mission suitability/acceptability is subjective. Coupling assessments are normally not standard elements in helicopter flying qualities programs, but result because of some peculiarity or difficulty encountered during conduct of other standard tests or during performance of mission related maneuvers.

New helicopters are evaluated to support development and acceptance testing of operational flight trainers (OFT). Data are used to compare OFT and aircraft characteristics in efforts to ensure OFT fidelity. Naturally, documentation of coupling, even if not "adverse" or "objectionable", is required to provide acceptable flying qualities fidelity.

9.4.1 Trimmed Control Positions

Trimmed control position information is obtained during equilibrium flight. Data are obtained during any situation desired (level flight, climb, descent, and dive) as long as the aircraft is in rectilinear flight at a steady speed. Data are obtained over an airspeed range at selected airspeed intervals, at constant airspeed for various power settings, or at a given airspeed and power setting for various sideslip angles. For these equilibrium conditions, all angular rates (p, q, r) are zero; therefore, the coupling quantities (M_p, L_q, N_p) are neglected. Other coupling terms are omitted if specific items are constrained during the test. For example, if control position data are obtained during constant airspeed climbs and descents, u is zero and therefore L_u is not considered.

Steady state testing involve equilibrium or steady, non equilibrium conditions. An example of steady non equilibrium flight is the steady turn discussed under maneuvering stability testing in Chapter 6. In a steady turn, all flight parameters except heading (and possibly altitude) are stable and the only acceleration involved is in the vertical axis. In a steady turn, trim control position data are meaningful in terms of the control positions required to trim the other axis to equilibrium. For this reason, coupling in steady turning flight is addressed under trim control position testing.

During any equilibrium or steady state test, coupling parameters are changing as each data point is obtained over the range of test variables. To illustrate, trimmed control position data for level flight are obtained as collective is adjusted to maintain power for level flight, and the other controls are manipulated for trimmed wings level, ball centered flight. Any control position coupling, as well as speed coupling (M_v, L_u), are inherent in the data. If the main rotor cyclic control phasing is correct, such that $L_{B_{1s}}$ and $M_{A_{1s}}$ are not involved, the coupling arising from aerodynamics and geometry such as $L_{\theta_{TR}}$, M_{θ_C}, $N_{A_{1s}}$ and N_{θ_C} may exist. It may be impossible to sort out any one effect.

Control coupling resulting from incorrect cyclic control phasing ($L_{B_{1s}}$ and $M_{A_{1s}}$) are perceived during control response testing and are omitted from equilibrium tests for simplicity. This does not mean cyclic control coupling does not exist in equilibrium data.

9.4.1.1 HOVER AND LOW AIRSPEED

Hover and low airspeed trimmed control position (TCP) data are normally obtained at constant altitude during fore/aft translations and sideward translations, vertical translation, and during translation with the relative wind from various azimuths (critical azimuth). Table 9.V summarizes the main and tail rotor coupling which exist and the effect of each type. The individual stability derivative effect reflects the expected result if no other aerodynamic or control terms change. Obviously, during data acquisition all or most of the stability derivatives are involved.

Table 9.V
Summary of Low Airspeed Coupling Moment Origins

FLIGHT REGIME	STABILITY DERIVATIVE	EFFECT
Forward/aft translation	L_u	Right roll in forward flight, left roll in aft flight.
	L_{θ_C}	Depends on main rotor thrust offset from lateral CG. Normally with symmetrical lateral CG, rotor thrust acts to right of the lateral CG for high tail rotors; therefore, left roll with up collective.
	$L_{\theta_{TR}}$	For tail rotor above the roll axis, increased left directional control produces right roll.
	M_{θ_C}	Depends on main rotor thrust offset from longitudinal CG. For thrust acting forward of CG, up collective produces nose up pitch. For thrust acting aft of CG, up collective produces nose down pitch.
	N_u	Left yaw in forward and aft flight.
	N_{θ_C}	Right yaw with up collective, left yaw with down collective
Sideward translation	L_{θ_C}, $L_{\theta_{TR}}$, M_{θ_C}, N_{θ_C}	Same effects as for forward/aft translation.
	M_v	Nose up pitch with right translation, nose down pitch with left translation.
	N_v	Right yaw in right sideward flight, left yaw in left sideward flight.
Vertical flight	L_{θ_C}, $L_{\theta_{TR}}$, M_{θ_C}, N_{θ_C}	Same effects as for forward/aft translation.
	L_w	Depends on main rotor thrust offset from lateral CG. Normally with symmetrical lateral CG, rotor thrust acts to right of CG for a high tail rotor; therefore, left roll in descent and right roll in climb.
	M_w	Depends on main rotor thrust offset from longitudinal CG. For thrust acting forward of CG, descent produces nose up pitch. For thrust acting aft of CG, descent produces nose down pitch.
	N_w	Right yaw in climb, left yaw in descent.

Summary of Low Airspeed Coupling Moment Origins (cont'd)

Critical azimuth	L_{θ_C}, $L_{\theta_{TR}}$, M_{θ_C}, N_{θ_C}	Same effects as for forward/aft translation.
	L_u, M_v	Same effects as for forward/aft and sideward translation since critical azimuth involves both u an v velocities.
	N_u, N_v	Similar effects as for forward/aft translation. N_u effect expected to dominate with relative winds near 0˚ and 180˚. N_v effects dominate with relative winds near 90˚ and 270˚.

9.4.1.1.1 Test Technique

Hover and low airspeed test techniques for TCP are addressed in Chapter 8. No additional procedures are required to assess coupling. If significant pilot effort is required to obtain the data or if the data have non-linearities/reversals, then additional quantitative testing is appropriate to investigate the mission suitability characteristics. These tests include gentle and rapid translation and aircraft repositioning tasks for the cardinal directions as well as realistic quartering translations/repositions. Turns on a spot using the techniques of Chapter 8 for several wind conditions are considered for heading changes up to 360˚.

Steady vertical climbs are performed for several torque values between hover and maximum power. Vertical descent to a realistic descent rate is appropriate. If significant trim changes are required to maintain vertical flight then gradual and rapid climb/descent transitions are used to assess mission suitability. Climb/descent reversals (transition to climb from descent) represent the most aggravated situation.

Specific climb/descent procedures for equilibrium data acquisition are as follows:

1. Establish a zero drift trimmed hover. Record initial conditions and establish the difference between hover torque and maximum torque.
2. Divide the excess torque available into 4 or 5 intervals.
3. Once out of ground effect (OGE) hover torque is established, transition to a low in ground effect (IGE) hover before torque application. Starting at low altitude allows stabilization at reduced altitudes for improved drift cues.
4. Smoothly, but briskly, increase torque to the desired interval.

5. Manipulate the flight controls as required to maintain heading and climb without drift.

6. When the vertical accelerometer (g meter) reads 1.0, or when the vertical speed indicator is stabilized, record data.

7. Assign an Handling Qualities Rating (HQR) and Vibration Assessment Rating (VAR) to each data point.

8. Reposition for the next data point at the next torque interval.

9. Repeat process until maximum torque is achieved.

10. Before starting descent tests determine torque required for the desired descent rate.

11. For descent, use the same procedure, but establish initial hover high enough to obtain data, and provide sufficient clearance for recovery.

9.4.1.1.2 Data Required

Cockpit: Run number, control positions, heading, airspeed from low airspeed system (if available), ground speed (if available), vertical speed, fuel counts (FC), Q, N_R, θ, ϕ, HQR and VAR for each equilibrium data point. Use automatic data recording devices for all quantitative transition tasks.

Ground station: Run numbers, wind direction and speed, T_o, and H_{P_o}.

9.4.1.1.3 Test Criteria

Equilibrium Data.

1. No relative motion between helicopter and pace vehicle for horizontal flight data.
2. Stabilized, unaccelerated flight.
3. No vertical climb/descent for horizontal flight data. No drift for vertical flight data.
4. Minimum control movement.
5. All control forces trimmed to zero.

Transition Tasks.

1. Start from steady initial flight condition.
2. Move controls as required to perform transition task.

9.4.1.1.4 Data Requirements

1. Stabilize 15 s prior to recording.
2. Record 10 s data for equilibrium points, time history for transition tasks.

9.4.1.2 FORWARD FLIGHT

Forward flight TCP data are obtained over the airspeed ranges of interest for level flight, turning flight, climbs at constant torque (up to maximum torque), descents at constant torque (to autorotation at constant N_R), and diving flight at constant torque. Climb and descent data are taken over an airspeed range at constant torque/N_R or at constant airspeed for various torque settings. Normally, the data are obtained in balanced (ball centered) flight, but in some instances data are taken for zero sideslip conditions. Test data obtained during equilibrium flight have all the effects of translational velocity (u, v, w) and control coupling inherent in the results, and represent the total trim changes involved.

9.4.1.2.1 Test Technique

Forward flight quantitative data test techniques are addressed in Chapters 6 and 7. If during level flight tests large excursions or abrupt reversals in data trends exist, constant altitude accelerations and decelerations (including quick stop) are performed to assess the effects on mission related tasks. Transitions to maximum climb and descent/autorotation from level flight, as well as flight path angle reversal (enter climb from descending flight, enter descent from climbing flight), are also appropriate. These transition maneuvers provide information on the transient trim changes involved and the pilot's ability to cope with the trim changes.

9.4.1.2.2 Data Required

Control positions, V_o, H_{P_o}, T_o, V_v (climb and descent) θ, ϕ, β, Q, N_R, FC, HQR, and VAR.

Transient maneuvers require the use of automatic data recording equipment. During transient maneuvers the parameters are recorded for analysis.

9.4.1.2.3 Test Criteria

1. Stabilized, ball centered flight.
2. Wings level for level flight or constant bank angle for turning flight.
3. No vertical velocity (level flight).
4. Constant vertical velocity (climb and descent).
5. All control forces trimmed to zero.

9.4.1.2.4 Data Requirements

1. Stabilize 15 s prior to recording data.
2. Record 10 s data.
3. H_{P_o} ± 5 ft (level flight).
4. V_v level flight, ±10 fpm; climb and descent, stabilized ± 25 fpm.
5. N_R ± 1%.
6. V_o ± 1 kn.
7. $\phi \pm 2^\circ$.

For transient maneuvers.

1. Stabilize as soon as possible prior to starting maneuver.
2. Record time history of trim and conduct of maneuver from start to finish including the trim point.

9.4.1.3 STEADY TURNS

Steady turns at constant collective position are normally conducted as part of the assessment of maneuvering stability as discussed in Chapter 6. These tests are primarily designed to assess longitudinal flying qualities. Test criteria for conduct of steady turns is balanced (ball centered) flight during data acquisition for each test point. A direct fallout of this test is the observation of the overall inter-axis coupling involved as reflected in the lateral and directional control requirements to establish ball centered flight. Coupling expected to occur during constant collective steady turns are:

1. Right lateral control displacements in both turn directions to counter roll with nose up pitch rate (left roll moment).

2. Lateral control displacements to counter roll moments due to thrust changes when the thrust is not laterally aligned through the CG. The direction of required control displacement depends on thrust axis orientation (left or right of CG). This effect is similar for left and right turns.
3. Lateral control displacements to counter roll moments due to lateral velocity (dihedral effect) if changes in sideslip angle are required for balanced flight.
4. Lateral control displacements to counter roll moments due to tail rotor thrust changes associated with directional control inputs (yaw damping and directional stability).
5. Directional control inputs to counter directional stability and yaw damping effects. The directional control requirements may be different for left and right turns.
6. Directional control inputs to counter rotor torque changes.

Steady turns are conducted at constant altitude and constant airspeed. These terms represent operational conditions reflective of Search and Rescue (SAR) and Antisubmarine Warfare (ASW) search patterns, and holding patterns during landing zone reconnaissance. The constant altitude steady turns give similar insight to the lateral and directional control requirements as determined during constant collective turns. The constant altitude turns are not maneuvering stability tests since the collective position is being varied, but they do give insight to additional coupling obtained with collective movement. Increased collective required to maintain altitude during turns results in increased aft rotor tilt. This aft tilt may result in a shallowing of the longitudinal control position/force gradient with load factor, even to the condition where the gradient becomes neutral or negative.

9.4.1.3.1 Test Technique

Techniques for conducting constant collective steady turns are described in Chapter 6. The technique for constant altitude turns is described in Chapter 7. Lateral, directional, and collective control forces are trimmed to zero at the pilot's discretion.

9.4.1.3.2 Data Required

All control positions, longitudinal control force, g, V_o, bank angle, pitch attitude, turn direction, sideslip angle (if available), HQR, VAR, V_v, FC, H_{P_o}, T_o, Q, and N_R.

9.4.1.3.3 Test Criteria

1. Ball centered, balanced flight.
2. Collective fixed/varied depending on test.
3. Constant bank angle/pitch attitude.
4. Constant airspeed.
5. Steady load factor.

9.4.1.3.4 Data Requirement

1. Stabilize 15 s prior to taking data.
2. Record 10 s stabilized data.
3. Trim V_o ± 1 kn.
4. Ball centered, ± 1/4 ball.
5. H_{P_o} ± 5 ft (level flight).
6. N_R ± 1%.
7. V_v level flight, ±10 fpm; climb and descent, stabilized ± 25 fpm.

9.4.2 Steady Pull Ups/Push Overs

Steady symmetrical pull ups and push overs at constant collective position are normally conducted as part of the assessment of maneuvering stability. Maneuvering stability testing is discussed in Sections 6.4. Test criteria for the conduct of pull ups and push overs requires ball centered, balanced flight during the test. Inter-axis coupling may exist during pull ups and push overs and is reflected in the lateral and direction control requirements to counter trim changes and maintain balanced flight. Expected coupling:

1. Right lateral control to counter left roll moments with nose up pitch rate (pull up), left lateral control with nose down pitch rate (push over).
2. Lateral control displacements to counter roll moments due to thrust changes when the thrust is not laterally aligned through the CG. The direction of required control displacement depends on thrust axis orientation (left or right CG). This effect is opposite for pull ups and push overs.
3. Lateral control displacements to counter roll moments due to lateral velocity if changes in sideslip angle are introduced by main rotor torque changes.
4. Lateral control displacement to counter roll moments due to tail rotor thrust changes associated with directional control inputs.

5. Directional control inputs to counter main rotor torque changes. Directional control requirements may be in opposite directions for pull ups and push overs. Right directional control inputs are expected in pull ups. When rapid longitudinal inputs are used during pull ups and push overs, roll coupling associated with non optimum cyclic input phasing may exist. If this coupling is present, it is observed as an immediate roll tendency.

Collective inputs may be used to change load factor as in a bob up or bob down maneuver. These maneuvers are operationally representative in terms of low altitude terrain following, vertical repositioning in formation flight/aerial refueling, and recovery from low power diving flight. Collective maneuvering gives insight to the collective ability to produce load factor as well as coupling involved with collective movement. The coupling perceived depends on the final aircraft response the pilot desires. For example, the pilot may desire to hold heading and pitch attitude during formation flight, whereas during a diving pullout or terrain following he might be interested in coordinated (ball centered) flight and some use of pitch attitude change in his control of flight path angles. In either case, expected coupling include:

1. Nose up pitching moments with up collective input due to aft TPP tilt.
2. Nose up or down pitching moments with up collective when the thrust axis is not longitudinally aligned with the CG. The thrust axis could pass on either side of the CG depending on aircraft loading and external forces and moments (tail loads). These pitching moment result from changes in thrust magnitude.
3. Heading/sideslip changes due to yaw moments from collective inputs (torque changes).
4. Roll moments from lateral flapping associated with coning (thrust changes). These moments are to the left for collective increases.
5. Roll moments from aircraft dihedral effect when sideslip changes occur.
6. Roll moments from tail rotor thrust changes associated with directional control inputs used to control heading/sideslip.
7. Roll moments from thrust changes when the thrust axis is not laterally aligned with the CG.

9.4.2.1 TEST TECHNIQUE

Steady symmetrical pull up and push over test procedures for longitudinal control inputs are provided in Section 6.4. No adjustments to these techniques are required to investigate coupling, unless significant coupling is present. The pull ups and push overs can be performed with the other controls fixed to assess the degree of coupling.

The collective bob up or bob down technique is as follows:

1. Stabilize in ball-centered, coordinated flight at the desired trim airspeed.
2. Reduce all control forces to zero.
3. Rapidly increase or decrease collective pitch. If a control fixture is used, establish the fixture increment and place the fixture appropriately. Start with small increments and use a build up procedure.
4. Manipulate the other controls to maintain the desired flight condition (pitch attitude, or heading, or bank angle, etc.).
5. Repeat the procedure until the predetermined limit (load factor, torque, control margin, etc) is reached.

9.4.2.2 DATA REQUIRED

Longitudinal, lateral, and directional control positions, longitudinal control force, g, V_o, bank angle, pitch attitude, HQR, VAR, size of longitudinal or collective input, FC, H_{P_o}, T_o, Q, and N_R for coordinated cyclic control pull up/push over flight maneuvers. For uncoordinated cyclic only pull up/push over tests (lateral and directional controls fixed) and for collective bob up and bob down maneuvers, time histories of the data parameters as well as collective position, pitch rate, roll rate, yaw rate, heading and sideslip angle.

9.4.2.3 TEST CRITERIA

1. Ball centered, balanced flight for coordinated maneuvers.
2. Constant bank angle for coordinated maneuvers.
3. Lateral cyclic and directional control fixed for uncoordinated maneuvers.
4. Collective fixed for cyclic pull ups/push overs.
5. Constant airspeed.
6. Load factor data are valid only as the aircraft passes through the trimmed pitch attitude during cyclic pull ups/push overs.

9.4.2.4 DATA REQUIREMENTS

1. Record steady trim data when on airspeed, on attitude.
2. Trim V_o ± 2 kn.
3. Ball centered for coordinated maneuvers, ± 1/4 ball.
4. H_{P_o} ± 1000 ft of the target test attitude.
5. N_R ± 1%.

9.4.3 Steady Heading Sideslips

Steady heading sideslip tests are performed to assess static lateral-directional flying qualities. The only meaningful insight to coupling obtained from steady heading sideslip tests is of pitching moments resulting from lateral velocity changes. Main rotor TPP longitudinal tilt is expected when the lateral velocity is changed. The longitudinal tilt is a function of coning and the magnitude of the lateral velocity change. During steady heading sideslip tests an aft TPP tilt is expected in right sideslips and forward tilt in left sideslips. These moments, if unchecked, produce pitch attitude and airspeed excursions. During conduct of steady heading sideslips, forward cyclic control inputs in right sideslips, and aft cyclic inputs in left sideslips are normally required to maintain airspeed.

Test techniques, data required, test criteria, data requirements, and Safety Considerations/Risk Management for steady heading sideslip testing are given in Section 7.4.

9.4.4 Control Response

Control response tests are normally conducted for all four controls (longitudinal, lateral, directional, and vertical) in each direction. The purpose of control response testing is to investigate the primary axis helicopter response to abrupt (step) control inputs. In the simplest form, the responses achieved give insight to the aircraft damping and the pilots ability to establish angular rates and attitude changes. Control response tests are performed using single axis control inputs, with the control positions for the other axes fixed. The control response data are obtained during the first several seconds immediately following the control step.

Control response testing provides an excellent avenue to investigate short term coupling. This short term coupling plays a significant role in the pilot's perception of his ability to rapidly and accurately perform flight path changes. Since the response is short term in nature, the slower developing responses such as airspeed changes with longitudinal control steps and lateral velocity changes with lateral control steps are neglected.

For the purpose of discussing coupling, it is assumed the pilot uses the cockpit controls as follows:

1. Longitudinal cyclic to control pitch attitude, pitch rate, and airspeed.
2. Lateral cyclic to control bank angle, roll rate, and lateral velocity.
3. Directional control for heading and yaw rate.
4. Vertical/Collective control for altitude or vertical speed.

With this simplified control approach, off axis responses which occur are considered as coupling. Generally the coupling which occurs is not desirable. For example, a left directional control input is made in forward flight on a helicopter with a high tail rotor. This directional input produces a left yaw acceleration (desired uncoupled response), an initial right roll which tries to turn the aircraft right (undesired coupled response), and with time a right sideslip causing left roll and left turn (desired coupled response).

Hover and forward flight control response and the resultant coupling may or may not be similar. Consequently control response testing is usually performed in hover and at several representative forward flight airspeeds.

9.4.4.1 LONGITUDINAL CONTROL RESPONSE

Potential coupling expected during longitudinal control response testing is summarized in Table 9.VI. There is no simple response. The coupling evaluator should be aware of the timing of the perceived response, its magnitude, and how other quantities are changing. The timing of the perceived response is an indicator of whether the coupling is control input related, rate related, or a secondary effect. When an acceleration is observed immediately following a control input, it is probably control related. If the response is delayed, it is probably rate related or a secondary effect such as the dihedral characteristic.

The largest cross axis responses observed for longitudinal control inputs are normally experienced in roll. The apparent coupling response is larger as the ratio of pitch moment of inertia (I_{yy}) to roll moment of inertia (I_{xx}) increases. The yaw coupling is typically suppressed due to the smaller moments involved and the higher yaw moment of inertia (I_{zz}).

Table 9.VI
Longitudinal Control Response Coupling

FLIGHT REGIME	COUPLING DERIVATIVE	EFFECT
Hover	$L_{B_{1s}}$, Roll moment due to longitudinal cyclic pitch angle	Coupling due to improper cyclic control phasing associated with use of flapping hinge offset/flapping springs. If control phasing is greater than required (90˚ with flapping hinge offset), immediate right roll occurs with forward control input, left roll with aft input.
	L_q, Roll moment due to pitch rate	Coupling due to blade angle of attack changes at 0˚ and 180˚ azimuths. Nose up pitch rate produces a left roll and nose down a right roll. Pilot perceives a delay in roll response. The delay is reduced with increased pitch damping.
Forward Flight	$L_{B_{1s}}$, Roll moment due to longitudinal cyclic pitch angle	Cyclic control phasing effect same as in hover. An additional effect in forward flight due to rotor thrust changes if the thrust axis is not laterally aligned with CG. Rotor thrust increase with aft control input generates right roll if the thrust axis is left of the CG, left roll if right of the CG. Rotor thrust change effect may add to or subtract from the cyclic phasing effect. Rotor thrust effect may increase with airspeed. Another effect in forward flight is due to coning change with thrust change. Coning increase with aft control input causes left roll and adds to the cyclic phasing effect. Coning effect increases with airspeed.
	L_q, Roll moment due to pitch rate	Same effect as in hover.
	$N_{B_{1s}}$, Yaw moment due to longitudinal cyclic pitch angle	Effect due to torque changes with longitudinal control input. Aft control inputs decrease torque and result in left yaw, forward inputs increase torque and produce right yaw. Secondary (delayed) roll occurs as sideslip angle changes. Aft control input causes right sideslip and left roll, forward input causes left sideslip and right roll for a helicopter with positive dihedral effect.

9.4.4.1.1 Test Technique

Longitudinal control response test techniques are described in Section 8.4 for hover and Section 6.4 for forward flight.

9.4.4.1.2 Data Required

Hover.

A time history of aircraft attitudes, rates, control positions, heading, airspeed from low airspeed system (if available), FC, Q, N_R, T_o, H_{P_o}, and VAR.

Forward Flight.

V_o, H_{P_o}, θ, FC, T_o, Q, N_R, cockpit control positions, control input size and direction.

Automatic recording systems are required for data collection. A time history of the longitudinal control position, attitudes, rates, and accelerations are essential data elements.

9.4.4.1.3 Test Criteria

1. Stabilized, unaccelerated flight at trim.
2. All control forces trimmed to zero.
3. Off axis controls fixed.

9.4.4.1.4 Data Requirements

1. Stabilize 15 s prior to recording data.
2. Trim $V_o \pm 1$ kn.
3. Record data until steady state rate is achieved or until a test limit is reached.

9.4.4.2 LATERAL CONTROL RESPONSE

Potential coupling expected during lateral control response testing is summarized in Table 9.VII. As in the longitudinal case, there is no simple, expected response. The coupling evaluator should be aware of the timing of the perceived response, its magnitude and how other quantities are changing. The timing of the perceived response is indicative of whether the coupling is control input related, rate related, or a secondary effect. When an acceleration is observed immediately following a control input it is probably control related. If the response is delayed, it is probably rate related or a secondary effect.

The cross axis responses observed in pitch and yaw with lateral control inputs are normally significantly less than the roll coupling observed for other control inputs. Pitch and yaw moments of inertia (I_{yy} and Izz) tend to be substantially larger than the roll moment of inertia (I_{xx}). These greater moments of inertia reduce the cross axis accelerations resulting from lateral control inputs and roll rates.

9.4.4.2.1 Test Technique

Lateral control response test techniques are described in Section 8.4 for hover and Section 7.4 for forward flight.

9.4.4.2.2 Data Required

Hover.

Control positions, all angular accelerations, pitch rate, roll rate, yaw rate, pitch attitude, bank angle, heading, FC, T_o, Q, N_R, H_{P_o}, control input size and direction.

Forward Flight.

All hover data plus load factor, airspeed, sideslip and V_v. All data except FC and T_o are measured in time history form. Measure and record the Automatic Flight Control System (AFCS) control movements.

Certain data requirements listed are not expected to be affected during lateral control response testing. These data are measured regardless to verify that coupling does not exist.

9.4.4.2.3 Test Criteria

1. Airspeed, altitude stabilized.
2. Ball centered in forward flight.
3. Longitudinal, directional, and collective controls fixed.

Table 9.VII

Lateral Control Response Coupling

FLIGHT REGIME	COUPLING DERIVATIVE	EFFECT
Hover	$M_{A_{1s}}$, Pitch moment due to lateral cyclic pitch angle	Coupling due to improper cyclic control phasing associated with use of flapping hinge offset/flapping springs. If control phasing is greater than required (90˚ with flapping hinge offset), immediate nose up pitch occurs with right control input, nose down with left input. Magnitude of coupling generally less than roll due to longitudinal cyclic.
	M_p, Pitch moment due to roll rate	Coupling due to blade angle of attack changes at $\psi = 90^\circ$ and 270˚ azimuths. Right roll rate produces a nose up pitch. Pilot perceives a delay in pitch response. The delay is reduced with increased pitch damping. Magnitude of coupling generally is less than roll due to pitch rate.
	$N_{A_{1s}}$, Yaw moment due to lateral cyclic pitch angle	Coupling occurs if the CG is displaced longitudinally from the rotor shaft. Right lateral inputs produce right yaw when the shaft is forward of the CG, and left yaw aft of the CG. Coupling effect may be small.
	N_p, Yaw moment due to roll rate	Coupling occurs when the tail rotor thrust axis is displaced vertically from the CG. Right yaw is expected with right roll rate when the tail rotor is high. Pilot perceives a delay in yaw response. The delay is reduced with increased roll damping.
Forward flight	$M_{A_{1s}}$, Pitch moment due to lateral cyclic pitch angle	Effect same as in hover.
	M_p, Pitch moment due to roll rate	Effect same as in hover.
	$N_{A_{1s}}$, Yaw moment due to lateral cyclic pitch angle	Effect same as in hover. Follow on response is suppressed by directional stability.
	N_p, Yaw moment due to roll rate	Effect is similar to hover, but is modified depending on the trim pitch attitude and the location of the roll axis. Initial effect slightly greater in forward flight than hover for some tail rotor to CG geometry. Follow on response is suppressed by directional stability.

9.4.4.2.4 Data Requirements

1. Stabilize 15 s prior to taking data.
2. Trim V_o ± 1 kn.
3. H_{P_o} ± 1000 ft of the target test altitude.

9.4.4.3 DIRECTIONAL CONTROL RESPONSE

Potential coupling expected during directional control response testing is summarized in Table 9.VIII. In general, coupling in roll due to directional control changes is tail rotor height dependent. If the tail rotor is high, initial right roll with left directional control is expected, followed by a reduction in roll response as yaw rate develops. In forward flight the final bank angle change may be in the opposite direction (in the direction of directional control) when dihedral effect is positive. The coupling evaluator should be aware of the timing of the perceived response, its magnitude and how other quantities (yaw rate and sideslip) are changing. The timing of the perceived response gives insight to whether the coupling is control input related or a secondary effect. When an acceleration is observed immediately following a control input, it is probably control related. If the response is delayed, it is probably rate related or a secondary dihedral effect.

9.4.4.3.1 Test Technique

Directional control response test techniques are described in Section 8.4 for hover, and Section 7.4 for forward flight.

9.4.4.3.2 Data Required

Hover.

Control positions, all angular accelerations, pitch rate, roll rate, yaw rate, pitch attitude, bank angle, heading, FC, T_o, Q, N_R, H_{P_o}, control input size and direction.

Forward Flight.

All hover data plus, load factor, airspeed, sideslip, and V_v.

All data except FC and T_o are measured in time history form. Measure and record the AFCS control movements.

Certain data requirements listed are not expected to be affected during directional control response testing. These data are measured regardless to verify that coupling does not exist.

9.4.4.3.3 Test Criteria

1. Airspeed, altitude stabilized.
2. Ball centered in forward flight.
3. Longitudinal, lateral and collective controls fixed.

9.4.4.3.4 Data Requirements

1. Stabilize 15 s prior to taking data.
2. Trim $V_o \pm 1$ kn.
3. $H_{P_o} \pm 1000$ ft of the target test altitude.

Table 9. VIII
Directional Control Response Coupling

FLIGHT REGIME	COUPLING DERIVATIVE	EFFECT
Hover	$L_{\theta_{TR}}$, Roll moment due to tail rotor pitch angle	Coupling occurs when the tail rotor thrust axis is displaced vertically from the CG. Left directional control inputs produce right roll acceleration, right directional control inputs produce left roll accelerations when the tail rotor is high.
	L_r, Roll moment due to yaw rate	Coupling occurs when the tail rotor thrust axis is displaced vertically from the CG. Left yaw rate produces left roll, right yaw rate produces right roll when the tail rotor is high. Effect opposes $L_{\theta_{TR}}$ and the pilot perceives a reduction in roll with directional control input as time increases.
Forward Flight	$L_{\theta_{TR}}$, Roll moment due to tail rotor pitch angle	Effect same as in hover, modified dependent on trim pitch attitude and the location of the roll axis. Secondary (delayed) roll effects occurs as sideslip angle changes. With positive dihedral effect, the helicopter eventually rolls in the direction of the directional control input.
	L_r, Roll moment due to yaw rate	Effect similar to hover but may be completely masked by sideslip changes and dihedral effect.

9.4.4.4 VERTICAL CONTROL RESPONSE

Potential coupling expected during vertical control response testing is summarized in Table 9.IX. Coupling with collective inputs may be expected in all axes for hover and forward flight. The coupling evaluator should be aware of the timing of the perceived response, its magnitude and how other quantities (pitch rate, roll rate, yaw rate, sideslip) are changing. The timing of the perceived response is an indication of whether the coupling is control input related, rate related, or a secondary effect. When an acceleration is observed immediately following a control input, it is probably control related. If the response is delayed it is probably rate or velocity related, or a secondary effect such as the dihedral effect. The coupling effect may be significant in all axes due to the size of the coupling moments produced in pitch and yaw, and the relatively low roll moments of inertia (I_{xx}).

9.4.4.4.1 Test Technique

Vertical control response test techniques are described in Section 8.4. for hover. Longitudinal control response test techniques given in Section 6.4 are used during forward flight vertical control response testing, with the exception the collective control is not held fixed.

9.4.4.4.2 Data Required

Hover.

Control positions, all angular accelerations, pitch rate, roll rate, yaw rate, pitch attitude, bank angle, heading, load factor, V_v, FC, V_o, Q, N_R, H_{P_o}, control input and size.

Forward Flight.

All hover data plus airspeed and sideslip.

9.4.4.4.3 Test Criteria

1. Airspeed, altitude stabilized.
2. Ball centered in forward flight.
3. Longitudinal, lateral and directional controls fixed.

Table 9.IX

Vertical Control Response Coupling

FLIGHT REGIME	COUPLING DERIVATIVE	EFFECT
Hover	$L_{\theta C}$, Roll moment due to collective pitch angle	Coupling occurs if the thrust axis is not aligned laterally with the CG. Rotor thrust increase with up collective generates right roll if the thrust axis is left of the CG, and left roll if the thrust axis is right of the CG. The thrust axis is normally right of the CG in hover with high tail rotor configurations and no lateral CG offset.
	L_W, Roll moment due to vertical velocity	Coupling has the same origin as $L_{\theta C}$. Climb produces left roll if the thrust axis is left of the CG, right roll if the thrust axis is right of the CG. Effect opposes $L_{\theta C}$ and the pilot perceives a reduction in roll response as rate of climb or descent develops.
	$M_{\theta C}$, Pitch moment due to collective pitch angle	Coupling occurs if the thrust axis is not aligned longitudinally with the CG. Rotor thrust increase with up collective generates nose up pitch if the thrust axis is forward of the CG; and nose down pitch if the thrust axis is aft of the CG. The thrust axis is normally forward of the CG, when flapping hinge offset/flapping springs are used.
	M_W, Pitch moment due to vertical velocity	Coupling has the same origin as $M_{\theta C}$. Climb produces nose down pitch if the thrust axis is forward of the CG, nose up pitch if the thrust axis is aft of the CG. Effect opposes $M_{\theta C}$ and the pilot perceives a reduction in pitch response as rate of climb or descent develops.
	$N_{\theta C}$, Yaw moment due to collective pitch angle	Coupling results from torque changes with collective movement. Increased collective produces nose right yaw, decreased collective produces nose left yaw.
	N_W, Yaw moment due to vertical velocity	Coupling results from inflow (vertical velocity) effects on torque required. Climb results in a torque increase and right yaw, descent results in a torque decrease and left yaw. During vertical control response testing the pilot may not perceive effect. A slight secondary contribution is the change of mass flow to the tail rotor. Increases in tail rotor thrust for the same pitch setting produces follow on or delayed left yaw with both climb and descent.

Vertical Control Response Coupling (cont'd)

Forward Flight [1]	L_{θ_C}, Roll moment due to collective pitch angle	The thrust axis lateral alignment effect is the same as in hover. An additional effect in forward flight is due to coning change with thrust change. Coning increase with up collective causes left roll and add/subtract from the thrust alignment effect.
	L_W, Roll moment due to vertical velocity	The thrust axis alignment effect is the same as in hover. An additional transient effect in forward flight is due to the coning change with thrust change. Descent increases coning and causes left roll, climb decreases coning, causing right roll. Opposes L_{θ_C} and the pilot perceives a reduction in roll response as rate of climb or descent develops.
	M_{θ_C}, Pitch moment due to collective pitch angle	Thrust axis longitudinal effect is the same as in hover. An additional effect is due to aft TPP tilt with up collective; as well as thrust increases with collective, aft tilt, and inflow. The effect is always nose up with up collective, nose down with down collective. If nose up pitch is observed in hover with up collective, nose up pitch is expected to be larger in forward flight. If nose down pitch is observed with up collective in hover, nose down pitch is expected to be smaller in forward flight.
	N_{θ_C}, Yaw moment due to collective pitch angle	Effect same as in hover.
	N_W, Yaw moment due to vertical velocity	No effect is observed in collective control response testing since response time is short and w does not develop.

Note: (1) Follow on responses in roll (dihedral effect) and pitch with lateral velocity may be observed.

9.4.4.4.4 Data Requirements

1. Stabilize 15 s prior to taking data.
2. Trim $V_o \pm 1$ kn.
3. $H_{P_o} \pm 1000$ ft of the target test altitude.

9.4.5 Longitudinal Long Term Coupling

The longitudinal long term mode is typically a low frequency oscillatory motion or an aperiodic divergent motion for helicopters. The oscillatory mode may be convergent or divergent. For either the oscillatory or divergent modes, the motion consists of an interchange of airspeed and altitude at modest pitch rates. Classically the long term mode is considered to be uncoupled in roll and yaw.

Coupling may appear during investigation of long term characteristics if it is not actively suppressed by pilot inputs. As airspeed variations occur about the nominal trim airspeed, roll moments are expected. These roll moment (L_u) contributions, acting through coning and airspeed changes, generate right roll tendencies during higher than trim airspeed conditions, and left roll tendencies for slower airspeeds. Additionally, when airspeeds are slower than trim (aircraft pitching down) or faster than trim (aircraft pitching up), some roll with pitch rate is expected. The roll with airspeed and roll with pitch rate work to some degree to cancel each other. Figure 9.14 illustrates the airspeed, attitude, and pitch relationships for an undamped long term oscillation. This figure shows that the two roll contributions are opposed. For very low frequency (long period) oscillations with low pitch rates, the roll with airspeed term dominates. For higher frequency oscillations, the roll with pitch rate contributions may have a stronger influence. Figures 9.15 and 9.16 illustrate the phasing of bank angle with airspeed and altitude. Figure 9.15 illustrates roll with airspeed only coupling, and Figure 9.16 is for roll with pitch rate only coupling. Comparing figures 9.15 and 9.16, the phasing of the bank angle excursions is dependent on the dominant form of coupling encountered. The actual phasing of the bank angle changes during long term longitudinal motions depends on the relative magnitudes of each contribution.

1. Trim Airspeed, No Roll Due to Airspeed
   Maximum Nose Down Attitude, No Pitch Rate

2. Maximum Airspeed, Maximum Right Roll Due to Airspeed
   Maximum Nose Up Pitch Rate, Maximum Left Roll Due to Pitch Rate

3. Trim Airspeed, No Roll Due to Airspeed
   Maximum Nose Up Attitude, No Pitch Rate

4. Minimum Airspeed, Maximum Left Roll Due to Airspeed
   Maximum Nose Down Pitch Rate, Maximum Right Roll Due to Pitch Rate

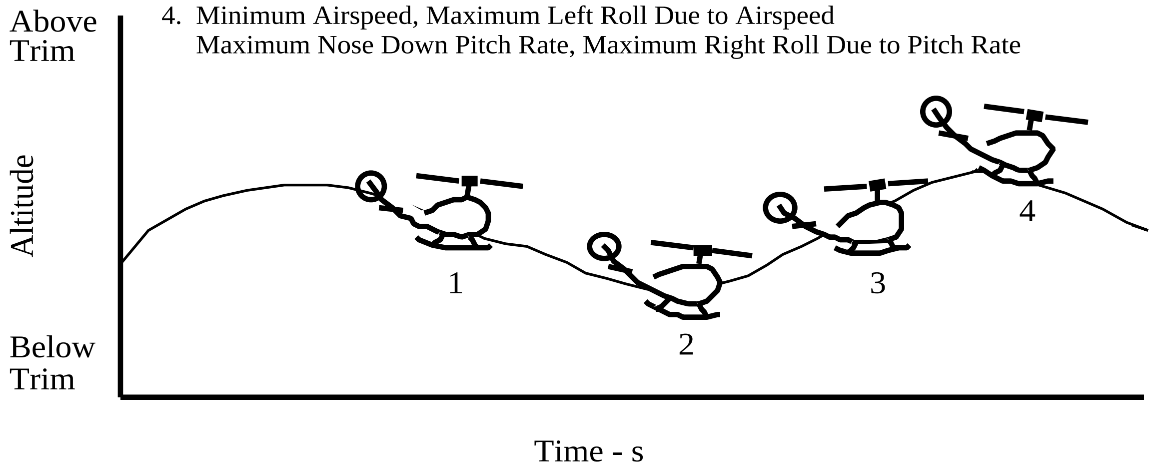


Figure 9.14
Airspeed, Altitude, and Pitch Relationships
During Long Term Oscillation

1. Highest Altitude, Slowest Airspeed, Maximum Right Roll
2. Trim Altitude, Trim Airspeed, Trim Bank Angle
3. Lowest Altitude, Fastest Airspeed, Maximum Left Roll
4. Trim Altitude, Trim Airspeed, Trim Bank Angle

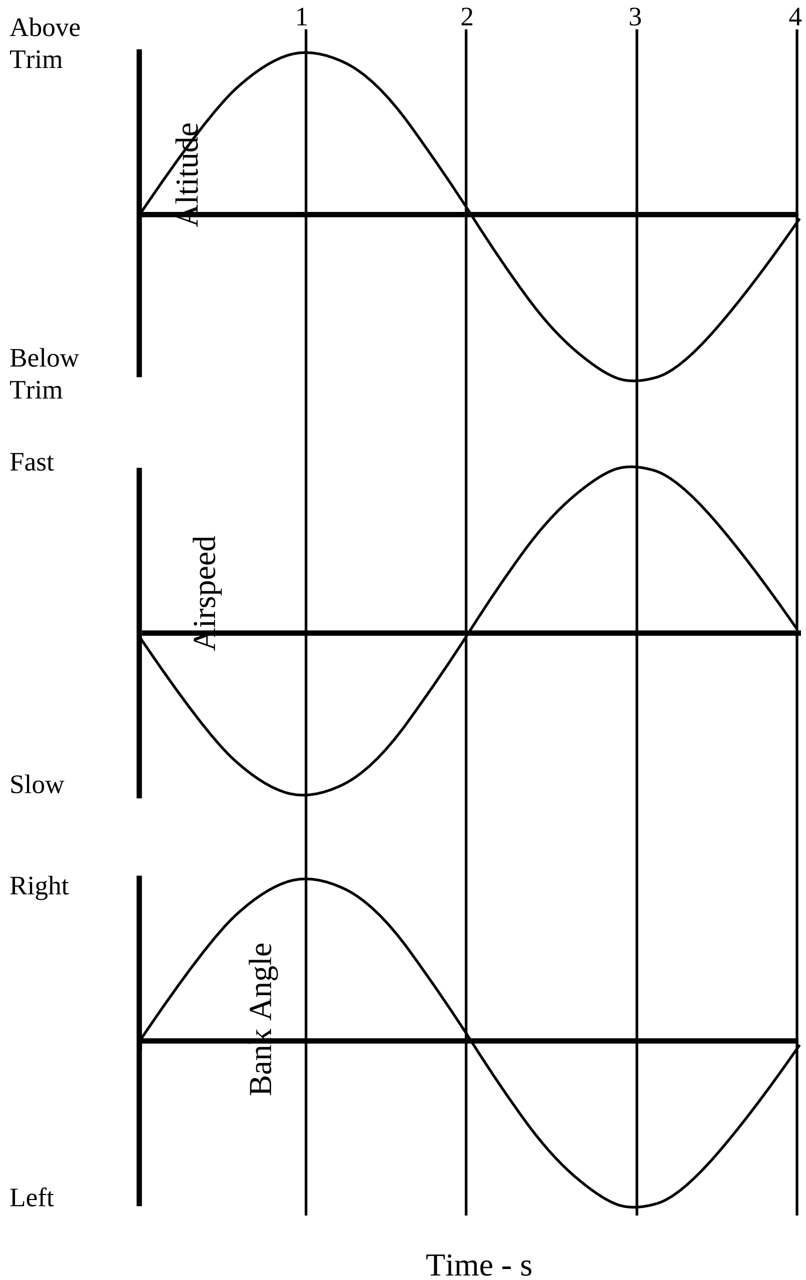


Figure 9.15
Bank Angle Due to Airspeed During Long Term Oscillation

1. Highest Altitude, Slowest Airspeed, Maximum Left Roll
2. Trim Altitude, Trim Airspeed, Maximum Nose Up Pich Rate, and Trim Bank Angle
3. Lowest Altitude, Fastest Airspeed, Maximum Right Roll
4. Trim Altitude, Trim Airspeed, Maximum Nose Down Pitch Rate, and Trim Bank Angle

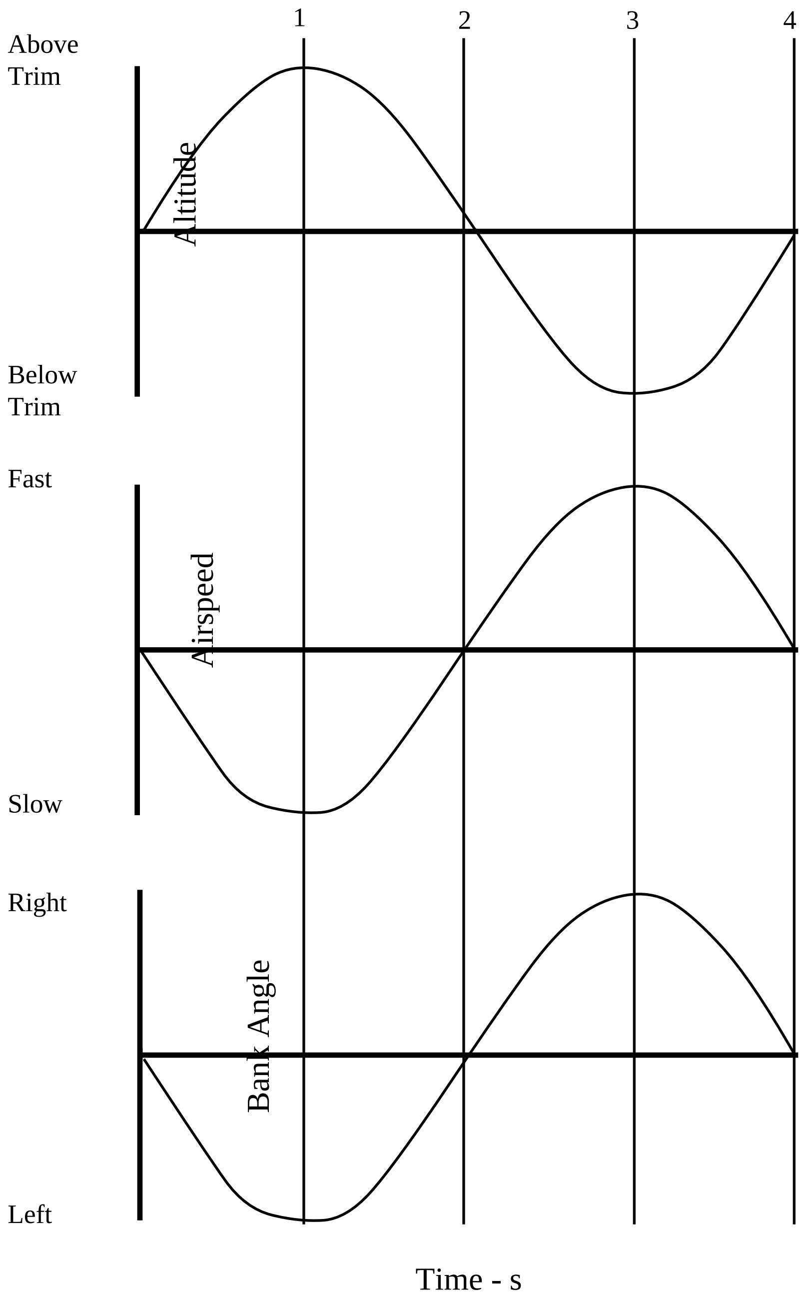


Figure 9.16
Bank Angle Due to Pitch Rate During Long Term Oscillation

If bank angle excursions are allowed to develop during testing of long term oscillations, then sideslip angle changes can be expected through introduction of spiral mode effects. The spiral mode may be convergent or divergent.

During investigation of longitudinal long term oscillatory modes of motion, the evaluator may observe rolling and yawing motions superimposed on the classical pitch motion with the controls fixed or free.

When the longitudinal long term motion is aperiodic the aircraft is statically unstable (normally with weak pitch rates) and the dynamics involve what appear to be spiral mode characteristics. Statically unstable helicopters exhibit rolling motions as airspeed changes occur. When airspeed increases during the divergence, right roll will be observed as the helicopter pitches down and accelerates. When the airspeed decreases, the roll tendency is to the left. The pilot may measure the controls required to counter roll tendencies, or may investigate the motions with controls free. Obviously, if the controls are free, there may be an interaction between the longitudinal and spiral modes.

9.4.5.1 EXCITATION METHODS

1. No conscious excitation. Controls fixed and no pilot inputs. Imperfect trim conditions or atmospheric disturbance may excite an aperiodic divergent mode or a lightly damped oscillatory mode. This is an unsatisfactory excitation technique if no response is obtained, but usually indicates a desirable aircraft characteristic.
2. Use the longitudinal control to accelerate or decelerate to an airspeed faster or slower than trim. Then smoothly return the control to the original trim position and record the aircraft motion. An off trim airspeed variation of 5 to 15 kn is normally used to excite the motion.
3. The method of excitation can determine the type of response documented. Natural disturbances which result in a long term response are desirable but these responses are usually contaminated by another disturbance before the motion is completed. This makes quantifying the mode of motion difficult.

9.4.5.2 TEST TECHNIQUE

1. Stabilize at the desired trim airspeed (these tests can be conducted at trimmed conditions other than level flight) and reduce all control forces to zero. Do not retrim control forces or move the collective during the test.

2. Record trim conditions. Assure the ability of returning to trim control positions and of holding them after exciting the response.

3. Determine if a long term response results from a natural disturbance. With the controls either fixed or free, note the open loop aircraft response. If no aircraft response is observed, an artificial excitation is used.

4. Excite the aircraft using one or all of the artificial methods discussed above. Try several different methods to find the most representative aircraft responses to include in the report. Do not retrim any control forces and keep the collective constant during the response.

5. If the aircraft is flown with controls free (hands off), obtain responses following the excitation. The controls are released at trim so the subsequent control motions indicate the effect of attitude changes and gravity force acting during the response.

6. Record the resulting mode of motion using cockpit data and automatic recording systems. Cockpit displayed airspeed, altitude, and attitudes are recorded at selected increments of time. Start the stop watch at the completion of the excitation or at a predetermined airspeed. The zero time reference point is arbitrary. Small enough time intervals are used to define the shape of irregular responses.

7. The resulting mode of motion may not be a classical single axis response. Use lateral cyclic and directional controls as necessary to maintain a single axis response for the initial excitation. If you find the lateral-directional inputs are significant, record a time history of the aircraft response with all controls free.

8. For hand held data, a plot of airspeed at regular intervals (5 to 10 s) adequately defines the response. An alternate method is to record peak (maximum and minimum) airspeeds, and determine the period by timing the response as the helicopter passes through the trim airspeed.

9.4.5.3 DATA REQUIRED

V_o, H_{P_o}, V_v, θ, ϕ, heading, HQR, VAR, FC, T_o, Q, N_R, and cockpit control positions.

Following the excitation, record airspeed, altitude, and attitudes as a function of time. Use automatic recording systems to simplify this task.

9.4.5.4 TEST CRITERIA

1. Ball centered, balanced flight.
2. Controls fixed (at trim) and/or free.
3. Collective fixed.

9.4.5.5 DATA REQUIREMENTS

1. Stabilize 15 s prior to taking data.
2. Trim $V_o \pm 1$ kn.
3. $H_{P_o} \pm 1000$ ft of the target test altitude.

9.4.6 Spiral Motion Coupling

The helicopter spiral mode is typically a slowly developing long term aperiodic motion which may be stable, neutral or unstable. The spiral mode characteristics are defined by the time history of the bank angle after the helicopter is disturbed in bank angle. Tests are conducted normally at constant airspeed by using the longitudinal control as required to maintain airspeed. If airspeed is not maintained then other effects may be superimposed. When the bank angle is disturbed, naturally or artificially, sideslip is generated, usually in the direction of bank. One of the effects of the sideslip change is the generation of a directional stability yawing moment (nose right for right wing down). Because of the bank angle, the yaw moment produces a nose down attitude change tending to accelerate the helicopter longitudinally. When this acceleration is unchecked by the pilot, additional roll moments are produced as well as pitch moments. The pitch moments may be stabilizing or destabilizing, as evidenced by the longitudinal control or force gradient measured during static longitudinal stability tests. With a longitudinally statically unstable aircraft, airspeed increases persist and a further right roll results when starting from an initial right bank, and a reduction in left bank angle when starting from initial left roll. Disturbances of the long term divergence mode produce a stabilizing effect for spiral stability tests initiated from left bank angles, and destabilizing from initial right bank angles.

For helicopters having positive longitudinal static stability characteristics nose up pitching moments with airspeed increases excite the long term oscillatory longitudinal motion and a coupling between the spiral and longitudinal modes is evident.

During spiral stability testing, the pilot should be aware of coupling. He may wish to maintain airspeed with longitudinal cyclic and measure the control required to hold airspeed. Since spiral and long term longitudinal motions are nuisance modes which may require periodic suppression during conduct of some tasks, the pilot investigates the constrained motion (hold airspeed during spiral tests) as well as the free motion. By allowing the interplay between modes, the evaluator can assess expected aircraft motions and the degree to which pilot attention is required if distracted by other mission related tasks.

9.4.6.1 EXCITATION METHODS

1. No conscious excitation. Controls fixed and no pilot inputs. Imperfect trim conditions or atmospheric disturbance may excite an aperiodic divergent mode or a lightly damped oscillatory mode. This is an unsatisfactory excitation technique if no response is obtained, but usually indicates a desirable aircraft characteristic.
2. Use the lateral or directional control to establish a 5˚ to 10˚ bank angle. Then smoothly return the control to the original trim position and record the aircraft motion.

9.4.6.2 TEST TECHNIQUE

1. Trim the helicopter in wings level, steady heading, coordinated flight at the desired test conditions.
2. Use directional controls to develop a sideslip angle. Hold sideslip until the helicopter rolls to the desired bank angle.
3. For tail rotor helicopters, the rate of directional control application should not create a roll moment due to change in tail rotor thrust. Lateral control is fixed throughout the evaluation at the initial trim position.
4. Use a range of bank angles from 5˚ to approximately 20˚ unless significantly larger bank angles are routinely experienced in accomplishing the mission of the helicopter.

5. After the bank angle is established, smoothly return the directional controls to the trim position and observe the helicopter response.

6. If the effective dihedral of the helicopter is not sufficient to achieve the desired bank angle, the alternate method is to fix the directional controls at trim, use lateral cyclic to establish the desired bank angle, smoothly return the lateral cyclic to trim, and observe the helicopter response. Since the spiral mode is usually weak, the response is easily contaminated if the controls are not returned exactly to trim.

7. Record a time history of the response or record the bank angle at selected time intervals to generate a plot of bank angle as a function of time.

8. Repeat steps 1 through 7 as required to evaluate increasingly larger bank angles both left and right.

9.4.6.3 DATA REQUIRED

Control positions, initial and final lateral or directional control positions, V_o, V_v, ϕ, β, θ, Q, N_R, FC, H_{P_o}, and T_o.

Hand recorded or automatic time history of bank angle versus time.

9.4.6.4 TEST CRITERIA

1. Wings level, ball centered, unaccelerated flight at trim.
2. Controls fixed except for the excitation control.
3. Excitation control returned exactly to trim.
4. No retrimming.

9.4.6.5 DATA REQUIREMENTS

1. Stabilize 15 s prior to returning excitation control to trim.
2. $V_o \pm 1$ kn.
3. $H_{P_o} \pm 1000$ ft of the target test altitude.
4. $N_R \pm 0.5\%$.

9.4.7 Lateral-Directional Oscillation Coupling

The helicopter lateral-directional oscillation (LDO) mode is typically a relatively short term, positively damped, oscillatory motion with a period of 3 to 5 s. Depending on the relative sizes of stability derivatives involved, the motion may be flat with little or no roll involved or a combination of yawing, sideslipping, and rolling. In the classical LDO, pitch attitude and airspeed excursions are minimal. Since the conventional helicopter has a non symmetric configuration, coupling of the LDO into pitching motions can be expected. Potential contributions to pitch motions are the pitch moment due to lateral velocity (M_v) or sideslip, and the pitch moment due to roll rate (M_p). When the sideslip angle is to the right a nose up moment is provided from the rotor coning. Left sideslips cause a nose down moment. Roll moments (accelerations) and subsequent roll rates result from the dihedral effect as sideslip changes, and from roll moment due to yaw rate (L_r). The initial development of pitch response due to lateral velocity and roll rate are through the short term longitudinal mode of motion. When the LDO is heavily damped, minimal pitch excursions are observed. For lightly damped LDOs, pitching motions at the LDO frequency can become apparent. These pitch motions with lightly damped LDOs aggravate tracking precision to a greater degree than poor LDO damping. If the LDO is low frequency, the pitch attitude excursions may generate commensurate airspeed changes.

Long term pitching motions are possible after complete subsidence of the LDO, whether heavily or lightly damped. Just as for uncoupled dynamics, the short term motion through minor airspeed or attitude changes provides the energy required to initiate long term dynamics. These residual longitudinal dynamics and spiral characteristics can be coupled, leading to increased pilot effort to suppress undesired motions.

9.4.7.1 EXCITATION METHODS

1. No conscious pilot inputs. Imperfect trim conditions or atmospheric disturbance may excite an aperiodic divergent mode or lightly damped oscillatory mode. If no significant response is obtained, it could indicate a desirable helicopter characteristic particularly for flight in turbulence.

2. Pilot excitation.

a. Release from steady-heading sideslip. This is generally the preferred technique because it closely resembles conditions encountered in flight. The technique requires that the cyclic and directional controls are simultaneously returned to the trim conditions. Control system mechanical characteristics often complicate this requirement. These difficulties can be overcome with use of control fixtures.

b. Release from pedal driven LDO. After trimming the helicopter at the desired conditions, drive the helicopter in yaw with a sinusoidal pedal input. Identify the damped frequency of the yaw oscillation. After exciting the yaw oscillation, stop the sinusoidal input with the directional control at the trim position and observe (record) the helicopter response.

c. Lateral cyclic or collective pulse. Secondary test techniques used to create a transient sideslip angle include a lateral cyclic pulse or a collective pulse input. The lateral cyclic pulse technique is applicable where the adverse/proverse yaw characteristics are appreciable and AFCS provides no heading hold or coordinated turn feature. The collective pulse technique is applicable for tail rotor configured helicopter.

9.4.7.2 TEST TECHNIQUE

1. Stabilize at the desired trim airspeed and reduce all control forces to zero. Subsequently, do not retrim control forces. Assure you can return and maintain the controls at the trim position after exciting the response.

2. Record trim condition.

3. First determine if a LDO results from a natural disturbance. With the controls either fixed or free, note the open loop helicopter response. If no helicopter response is observed, use an artificial excitation.

4. Excite the helicopter response using one of the artificial methods.

5. If the helicopter is operationally flown hands off, obtain controls free responses following excitation. Controls are released at trim so the subsequent response indicates the effect of control motions.

6. Record the resulting motion using cockpit data and automatic recording system.

7. The effects of various degraded AFCS modes should be considered and investigated if warranted.

9.4.7.3 DATA REQUIRED

β, ϕ, θ, V_o, H_{P_o}, V_v, FC, T_o, Q, N_R, and cockpit control positions.

Time history of the helicopter response to excitation. Attitudes (θ, ϕ, and β), rates (q, p, and r), and control positions. Estimate the ϕ/β ratio, period, and damping.

9.4.7.4 TEST CRITERIA

1. Ball centered, balanced flight at trim.
2. Longitudinal, lateral, and pedal controls fixed (at trim) and/or free.
3. Collective fixed.

9.4.7.5 DATA REQUIREMENTS

1. Trim $V_o \pm 2$ kn.
2. $H_{P_o} \pm 1000$ ft of the target test altitude.
3. $N_R \pm 0.5\%$.

9.4.8 Cyclic Only Turns

Cyclic only turns are used to give insight to several lateral-directional handling qualities characteristics. Once the turn is established (steady turn), cyclic opposite the turn or cyclic into the turn may be required. Opposite cyclic is an indicator of an unstable spiral mode if the lateral control gradient in steady heading sideslip is in the proper sense, right lateral control in right sideslip. Cyclic into the turn indicates just the opposite. During turn entry, either smooth gradual or abrupt lateral control inputs are used. For the smooth entry, insight is gained to the helicopter's overall ability to maintain coordinated flight. The abrupt input provides information about the control response and the degree to which coupling effects are manifested. The main coupling expected is the yaw moment due to lateral cyclic pitch angle ($N_{A_{1s}}$). The sign of $N_{A_{1s}}$ is dependent on longitudinal CG position relative to the rotor shaft. When the CG is aft of the shaft, right lateral control inputs are expected to produce proverse right yaw moments. An additional contribution to proverse yaw results from rotor wake skew (rotor side wash to the left with right control) and its effect on tail rotor thrust and fin loads. Adverse yaw is expected with forward CG locations. During abrupt lateral control inputs with associated yaw moments, the pilot observes simultaneous roll and yaw accelerations. When yaw is in the adverse sense a false start (yaw away from roll) or hesitation in the development of yaw rate is experienced. With proverse yaw immediate roll and yaw acceleration in the same direction are evident. Yaw with lateral cyclic is considered a coupling term. At first observation it appears proverse yaw is desirable. In reality, too much coupling in either direction is undesirable. With significant adverse yaw an excessive delay in the generation of yaw rate or turn rate may exist. In extreme cases too much proverse yaw can result in the pilot driving the LDO mode unstable during high gain closed loop control of bank angle.

During abrupt lateral control induced bank angle changes, information from the degree of uncoordinated (ball not centered) flight which occurs reflects qualitatively the adequacy of the aircraft directional stability.

Pilot attention while conducting lateral control only turns is directed to initial yawing accelerations, initial ball position, transient side force cues, follow-on ball movement and lateral control requirements in the stabilized turn. If serious lack of coordination exists, pilot ability to provide coordination through directional control inputs is assessed. When significant adverse or proverse yaw tendencies are observed, tight pilot in the loop tasks such as precise bank angle captures, abrupt roll outs from established turns, and roll reversals are evaluated for mission suitability.

9.4.8.1 TEST TECHNIQUE

1. Stabilize at the desired trim airspeed and reduce all control forces to zero. Maintain longitudinal, directional and collective controls fixed during initial tests.
2. Record trim conditions.
3. Starting with slower, more gradual lateral control inputs, establish left and right turns to predetermined bank angles.
4. Increase initial control application rate up to step inputs of predetermined sizes, with follow-on lateral inputs to establish desired bank angle. A control fixture is used to size inputs. Review methods in Section 7.4 for use of control fixtures.
5. If significant yaw excursions are observed during initial tests, attempt to coordinate with directional control.

6. Perform tests of bank angle capture, roll reversals, etc., if warranted.
7. Record time histories of representative aircraft responses, as desired for documentation purposes.

9.4.8.2 DATA REQUIRED

Control positions θ, ϕ, β, p, q, r, V_o, H_{P_o}, FC, To, Q, N_R, size and direction of inputs.

Use automatic recording systems.

9.4.8.3 TEST CRITERIA

1. Ball centered, balanced flight at trim.
2. Longitudinal, directional and collective controls fixed. Directional control movement as required when evaluating coordination efforts.

9.4.8.4 DATA REQUIREMENTS

1. Stabilize 15 s prior to taking data.
2. Trim V_o ± 2 kn.
3. H_{P_o} ± 1000 ft from the target test altitude.
4. Ball centered, ± 1/4 ball, for trim.

9.4.9 Pedal Only Turns

Pedal only turns are used to provide information about several lateral-directional handling qualities characteristics. Once the turn is established (steady turn), pedal opposite the turn or pedal into the turn may be required. Opposite pedal indicates an unstable spiral mode if the directional control gradient in steady heading sideslips indicates positive directional stability. It is highly unlikely the directional control gradient is in the unstable direction. Pedal into the turn indicates a stable spiral mode. During turn entry and bank angle capture either smooth gradual or abrupt directional control inputs are used. For the smooth entry, insight is obtained about the pilot's ability to induce and stop roll by use of the directional controls. The abrupt input provides information about the yaw control response and the degree to which coupling occurs. Coupling of roll with directional control inputs is expected when the tail rotor thrust axis is not aligned with the vertical CG. For helicopters with high tail rotors, a left directional control input causes immediate left yaw and right roll accelerations. The roll acceleration in time may produce a bank angle change attempting to turn the aircraft in a direction opposite to the control input.

When the helicopter has positive dihedral effect, a left roll moment is generated as right sideslip increases and this left roll moment eventually arrests the roll due to directional control, and the aircraft rolls in the direction of turn. The initial roll away from the desired direction of turn is interpreted as coupling or at least a distinct hesitation in the production of roll rates, and attitude changes in the intended direction.

In extreme cases with inadequate dihedral effect, the helicopter may initially roll away from the heading change and maintain a left yawed and right banked condition with little or no ground track change, or may even continue to roll away from the intended direction of turn.

During abrupt directional control induced turns, the immediate response gives insight to the roll with directional control involved and the follow on roll after sideslip develops (insight about the level of dihedral effect).

Pilot attention while conducting directional control only turns is directed to initial roll accelerations, rates, and attitude changes, follow on roll tendencies, anticipation required to establish the desired bank angle using directional control only, and the final directional control required (pedal into or opposite the turn) for the steady turning condition. Recovery to zero bank flight from steady turns using directional control only can also be used to assess coupling and dihedral effect.

9.4.9.1 TEST TECHNIQUE

1. Stabilize at the desired trim airspeed and reduce all control forces to zero. Maintain longitudinal, lateral and collective controls fixed.
2. Record trim conditions.
3. Starting with slower, more gradual directional control inputs, establish left and right turns to predetermined bank angles.
4. Increase initial control application rate up to step inputs of predetermined sizes. A control fixture is used to size inputs. Review methods of Section 7.4 for use of control fixtures.

5. Perform tests to look at initial turn capability, bank angle capture, turn reversals, etc.

6. Record time histories of representative aircraft responses as desired for documentation purposes.

9.4.9.2 DATA REQUIRED

Control positions θ, ϕ, β, p, q, r, V_o, H_{P_o}, FC, T_o, Q, N_R, size and direction of inputs.

Use automatic recording systems.

9.4.9.3 TEST CRITERIA

1. Ball centered, balanced flight at trim.
2. Longitudinal, lateral and collective controls fixed.

9.4.9.4 DATA REQUIREMENTS

1. Stabilize 15 s prior to taking data.
2. Trim $V_o \pm 2$ kn.
3. $H_{P_o} \pm 1000$ ft of the target test altitude.
4. Ball centered, ± 1/4 ball, for trim.

9.5 DATA REDUCTION

Data reduction requirements for these tests are limited to the application of instrument corrections to recorded data. Airspeed and altitude data are corrected for both instrument and position errors in forward flight. Automatic data recording systems normally include total instrumentation system corrections in the process used to change a flight tape (or telemetry) raw data into engineering units. Most quantitative information required for plotted data are read from time histories of selected data parameters. In some instances, time histories are used in dynamic tests to show the overall response and the interrelation/timing of parameter changes. Time histories are appropriately annotated to show the phenomena leading to the final assessment/conclusions drawn from aircraft responses.

9.5.1 Trimmed Control Positions

9.5.1.1 HOVER AND LOW AIRSPEED

Plot all control positions, pitch attitude, banks angles, torque (optimal), versus relative true wind velocity (longitudinal, lateral and vertical translation) or wind direction (critical azimuth) for all stabilized data points.

Provide time histories of all control positions, velocity if available (Doppler, or low airspeed system), aircraft attitudes and angular rates, load factor torque (optional) to show unsuitable characteristics obtained during translation/repositioning dynamic tests. Time histories of suitable characteristics are optional.

Plots and time histories include tabulations of weight, CG, pressure altitude, ambient temperature, height above ground level, AFCS operating mode, and N_R.

9.5.1.2 FORWARD FLIGHT

Plot all control positions, pitch attitude, bank angle, sideslip, and torque (optional) versus calibrated airspeed for all stabilized data points.

Provide time histories of all control positions, airspeed, altitude, aircraft attitudes and angular rates, load factor, sideslip, torque, and N_R to show unsuitable characteristics obtained during dynamic transition (climb, descent, acceleration, deceleration, etc.) tests. Time histories of suitable characteristics are optional.

Plots and time histories include tabulations of weight, CG, pressure altitude (not required if shown on time history) ambient temperature, AFCS operating mode and N_R (not required if shown on time history).

9.5.1.3 STEADY TURNS

Plot all control positions, pitch attitude, bank angle, sideslip, torque, rate of descent (constant collective turns), versus load factor. Distinguish between left and right turns. Plot calibrated airspeed to show quality of airspeed control.

Tabulate weight, CG, pressure altitude, airspeed (if not plotted), ambient temperature, AFCS operating mode and N_R.

9.5.2 Steady Pull Ups/Push Overs

Plot all control positions, pitch rate, sideslip, torque versus load factor for all ball centered pull up and push over data points obtained with longitudinal cyclic inputs.

Provide time histories of all control positions, pitch attitude, bank angle, heading, all angular rates, load factor, torque, and sideslip for:

1. Cyclic pull ups/push overs where excessive amount of pilot effort was required to maintain balanced flight.
2. Cyclic pull ups/push overs which were intentionally not coordinated with other controls if undesirable characteristics exist.
3. Collective pull ups/push overs which required excessive amount of pilot effort to obtain required response (coordinated flight, or constant heading, or constant pitch attitude, etc.).

Plots and time histories include tabulations of weight, CG, pressure altitude, ambient temperature, AFCS operating mode and N_R.

9.5.3 Steady Heading Sideslips

Plot all control positions, pitch attitude, bank angle, rate of climb or descent versus sideslip angle.

Plots include tabulations of weight, CG, pressure altitude, AFCS operating mode, torque and N_R.

9.5.4 Control Response

Provide time histories of all control positions, all angular accelerations, all angular rates, pitch attitude, bank angle, heading, load factor, airspeed, altitude, torque, AFCS positions (optional), sideslip, and N_R. Time history parameters are selectively eliminated if certain coupling responses are not observed.

Tabulations of weight, CG, ambient temperature, and AFCS operating mode are included. Chapters 6, 7 and 8 address additional required annotations on time histories.

9.5.5 Longitudinal Long Term Coupling

Provide time histories of all control positions, pitch attitude, bank angle, heading, all angular rates (optional if rates are low), airspeed, altitude, load factor (optional), and sideslip. Time histories involving pilot control manipulation to maintain coordinated flight and time histories of aircraft response with controls fixed and free are appropriate.

Tabulations of weight, CG, ambient temperature, AFCS operating mode, torque, and N_R are included.

9.5.6 Spiral Motion Coupling

Provide time histories of all control positions, pitch attitude, bank angle, heading, all angular rates (optional if rates are low), airspeed, altitude, load factor (optional), and sideslip. Time histories involving pilot longitudinal control manipulation to maintain airspeed, and time histories of aircraft response with controls fixed and free are appropriate.

Tabulations of weight, CG, ambient temperature, AFCS operating mode, torque, and N_R are included.

9.5.7 Lateral-Directional Oscillation Coupling

Provide time histories of all control positions, pitch attitude, bank angle, heading, all angular rates (pitch rate optional), and sideslip. Time histories involving pilot longitudinal control manipulation to maintain pitch attitude, and time histories of aircraft response with controls fixed and free are appropriate.

Tabulations of weight, CG, ambient temperature, AFCS operating mode, torque, and N_R are included.

9.5.8 Cyclic Only Turns

Time histories of the more abrupt lateral control inputs for turn entries, bank angle captures, and roll reversals are presented to show undesirable aircraft characteristics. Provide time histories of all control positions, pitch attitude, bank angle, heading, all angular rates (pitch rate optional), load factor (optional), and sideslip.

Tabulations of weight, CG, airspeed, altitude, ambient temperature, AFCS operating mode, torque, and N_R are included.

9.5.9 Pedal Only Turns

Time histories of the more abrupt directional control inputs for turn entries, bank angle captures, turn roll outs, heading reversals, etc. for representative responses are presented to show undesirable aircraft characteristics. Provide time histories of all control positions, pitch attitude, bank angle, heading, all angular rates (pitch rate optional), load factor (optional), and sideslip.

Tabulations of weight, CG, airspeed, altitude, ambient temperature, AFCS operating mode, torque, and N_R are included.

9.6 DATA ANALYSIS

9.6.1 Trimmed Control Positions

9.6.1.1 HOVER AND LOW AIRSPEED

Low airspeed trimmed flight control positions are essentially analogous to hovering (zero ground speed) in a wind of equal velocity. The analysis includes:

1. Were there any limiting factors?

2. For which axis was the limiting factor most objectionable?
3. Were the control margins adequate? Greater than 10% control position or control power margin?
4. Were control movements required in the usual sense (forward cyclic for forward airspeed, right lateral cyclic for right lateral flight)?
5. Were the control position changes with power, airspeed, or azimuth excessive?
6. Could the control forces be trimmed to zero?
7. What were the aircraft attitude changes with airspeed, rate of climb? Were they objectionable?
8. Was field-of-view adequate?
9. Were the vibrations objectionable?
10. Was there any objectionable coupling between pitch, roll, and yaw?
11. Could the pilot keep up with trim changes/coupling during dynamic situations?
12. Are these characteristics suitable for the intended mission?

9.6.1.2 FORWARD FLIGHT

The following considerations are generally included in the analysis of trimmed control position in forward flight data:

1. Were there adequate control margins?
2. Was forward cyclic required with increasing airspeed?
3. Are the control position changes with changes in airspeed and power excessive?
4. Were there significant non linearities or abrupt reversals in data trends?
5. Could forces be trimmed to zero?
6. What were pitch attitude changes with airspeed?
7. Was field-of-view adequate?
8. Were there objectionable vibrations?
9. What was the coupling between the pitch, yaw and roll axes?
10. Could small precise collective changes be made?
11. Could the pilot keep up with trim changes and coupling during dynamic situations?

9.6.1.3 STEADY TURNS

The following considerations are generally included in the analysis of steady turn test data:

1. What was the basic longitudinal stability?
2. What was the effect of power addition in constant altitude turns on longitudinal control position gradients?
3. What longitudinal force/position cues were available at increased load factors?
4. What trim changes/coupling were involved in other axes?
5. Could the trim changes be easily compensated?
6. What pitch attitudes were required to maintain airspeed?
7. Was field-of-view adequate?
8. Were vibrations objectionable?

9.6.2 Steady Pull Ups/Push Overs

The following considerations are generally included in the analysis of steady pull ups/push overs:

1. What was the basic longitudinal stability?
2. What longitudinal force and position cues were available at increased load factors during cyclic induced pull up/push overs?
3. What trim changes and coupling were involved in other axes during cyclic and collective induced pull ups/push overs?
4. Could the trim changes be easily compensated?
5. Were vibrations objectionable?

9.6.3 Steady Heading Sideslips

The following are normally included in the analysis of steady heading sideslips:

1. What were the lateral and directional control gradients? Were they adequate?
2. What was the bank angle gradient? Did it provide sufficient cues to prevent cross control trimming.

3. What pitch attitudes and longitudinal control inputs were required to maintain airspeed?
4. Were control and attitude gradients linear and aircraft cross control changes predictable?
5. Was the field-of-view adequate?
6. Were the vibrations objectionable?

9.6.4 Control Response

An abundance of quantitative data are taken from the response time histories. Parameters which define the response are: rate control effectiveness; attitude control effectiveness; steady-state rate, maximum rate, maximum acceleration, initial acceleration, acceleration delay, and time constant. The following considerations are an integral part of control response and effectiveness discussions.

1. Was the steady-state rate adequate?
2. Was the response predictable?
3. Was the response consistent?
4. Were there any over control problems?
5. What coupling was observed between axes?
6. Was the coupling immediate or did it appear delayed?
7. Was the coupling desirable or undesirable?
8. What pilot efforts were required to compensate for undesired coupling?
9. What vibrations were introduced?
10. Was the response suitable for the mission?

Figures 9.17 and 9.18 illustrate the expected trends when an off axis response is due to the control input (Figure 9.17) or due to subsequent rate or secondary effects (Figure 9.18). As shown, the pitch accelerations (maximum slope of angular rate) occur at nearly the same time shortly after the input is made if the off axis response is control generated. When the off axis coupling is related to the angular rate a distinct lag is involved in the development of the off axis acceleration, and the development of off axis angular rate is delayed.

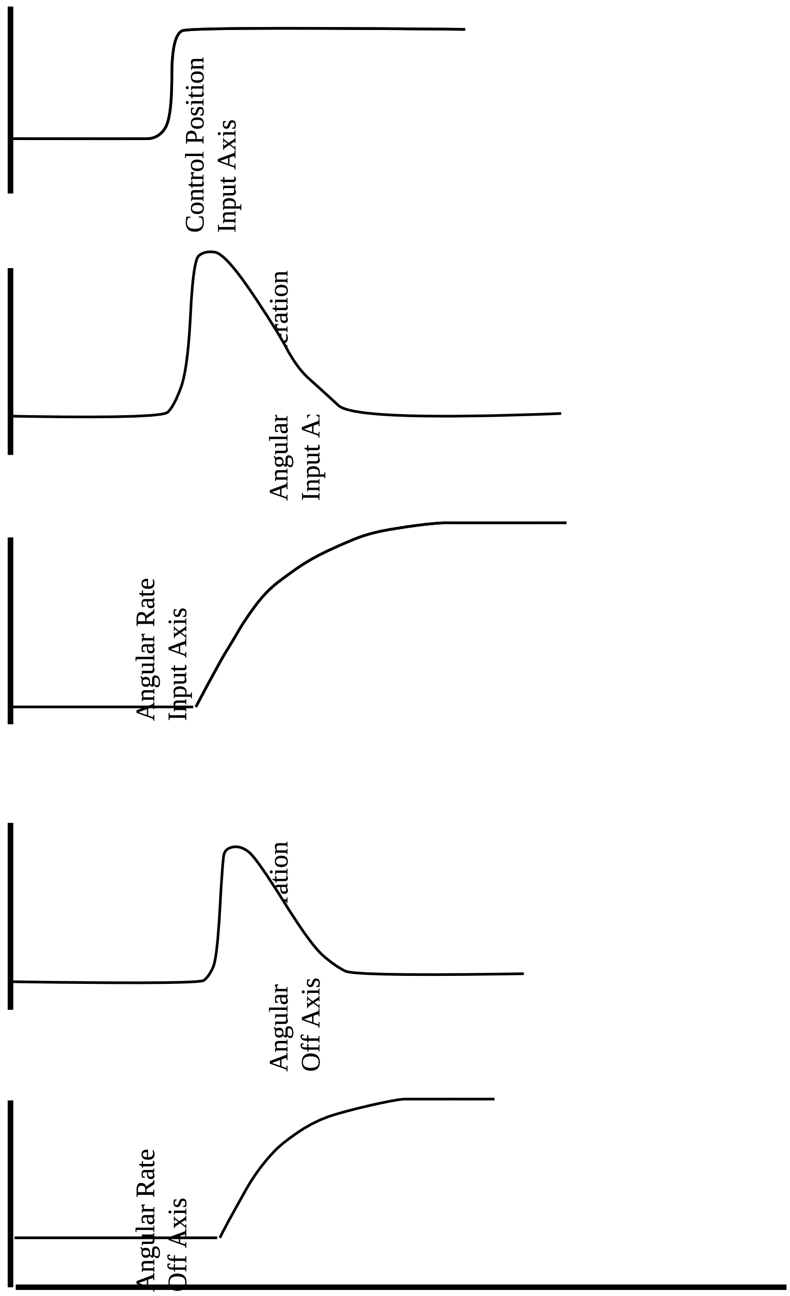


Figure 9.17
Coupling Due to Control Input

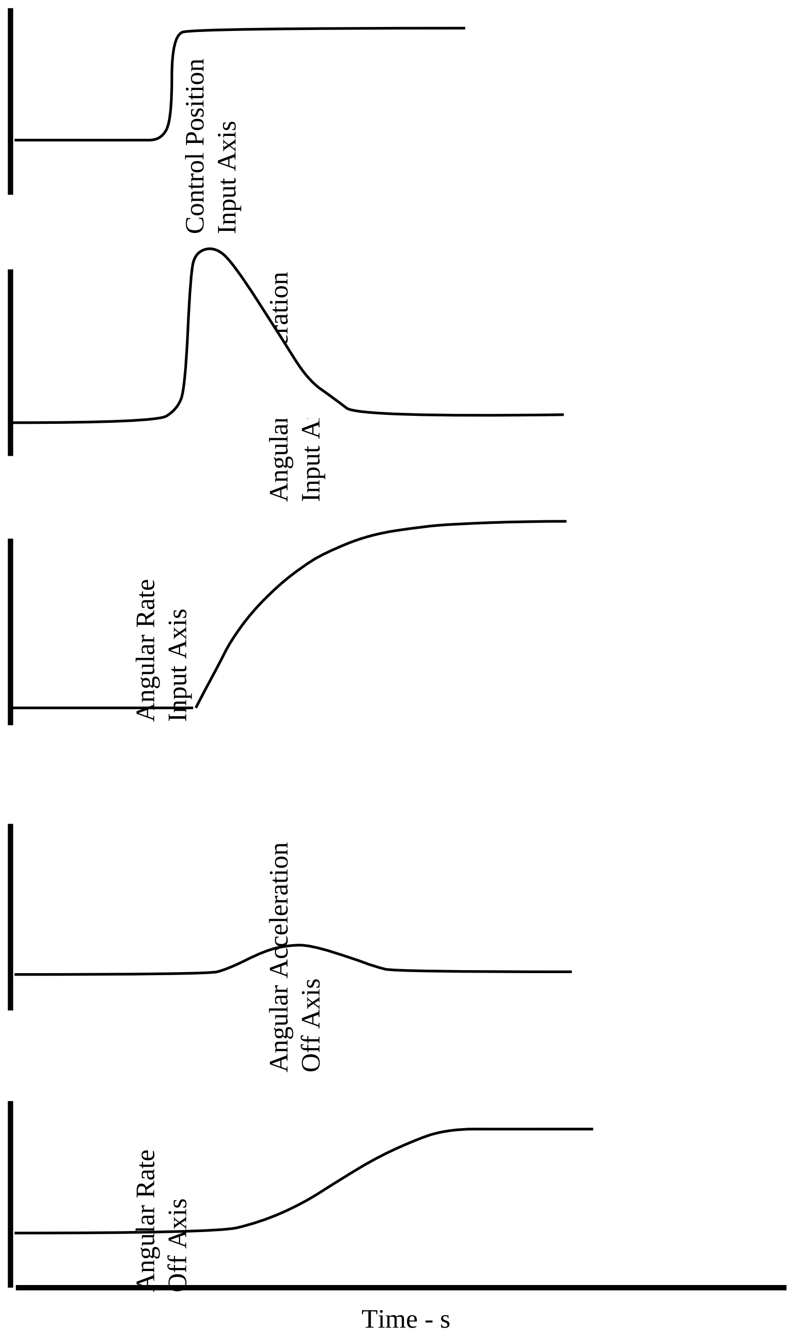


Figure 9.18
Coupling Due to Angular Rate

9.6.5 Longitudinal Long Term Coupling

The longitudinal long term motion is a nuisance mode, particularly when lightly damped or when statically or dynamically unstable. The following considerations are generally included in the analysis of the longitudinal long term motion:

1. How easy or difficult was it to excite the long term response?
2. In engineering terms, how do you describe the uncoupled aircraft mode of motion? Use the period, damping ratio, time to half amplitude or time to double amplitude to tell the reader what you saw.
3. What coupling or off axis trim changes exist?
4. What were controls fixed and free responses?
5. Are off axis motions more or less significant than the single axis motion?
6. How do the coupled and uncoupled motions influence the pilot's ability to perform mission tasks? The significance of the response is the level to which it degrades or helps the pilot perform a specific task. An evaluation is made of the pilot effort required to suppress or correct the airspeed, attitude, and altitude variations about trim.

9.6.6 Spiral Motion Coupling

The spiral mode of motion is a nuisance mode, particularly when neutrally stable or unstable. The following considerations are generally included in the analysis of the spiral mode:

1. To what extent do natural disturbances or other motions excite the spiral mode?
2. In engineering terms, how do you describe the uncoupled aircraft mode of motion? Use the time to half amplitude or time to double amplitude to tell the reader what you saw.
3. What coupling or off axis trim changes occur?
4. What were the controls fixed and free responses?
5. Were airspeed changes significant or dangerous?
6. Are off axis motions more or less significant than the basic spiral motions?
7. How does the motion influence the pilot's ability to perform mission tasks?

The significance of the response is the level to which it degrades or helps the pilot perform a specific task. An evaluation is made of the pilot effort required to suppress or correct the airspeed, attitude, and altitude variations from trim.

9.6.7 Lateral-Directional Oscillation Coupling

The following considerations are generally included in the analysis of lateral-directional oscillation (LDO) characteristics:

1. How easily is the LDO excited during other mission tasks?
2. What is the basic character of the LDO in terms of frequency, damping ratio, cycles or time to half amplitude, yawing motion or rolling motion or combination of both?
3. Does pitch coupling exist?
4. What is pilot effort required to suppress the LDO and cross axis coupling?
5. Does the LDO degenerate into spiral or longitudinal long term motions?
6. Can the LDO mode be driven unstable with tight pilot in the loop bank angle control tasks?
7. How does the motion influence the pilot's ability to perform mission tasks?

9.6.8 Cyclic Only Turns

The following considerations are generally included in the analysis of lateral cyclic only turns:

1. Did the aircraft initially yaw opposite to the control input or display excessive hesitation in developing turn in the direction of roll?
2. To what degree was transient uncoordinated (ball out) motion observed?
3. Could steady bank angles be obtained precisely and quickly ?
4. What direction of lateral control was required in the established turn?
5. What was the level of coordination (ball position) in the established turn?
6. What were the roll out responses?
7. Was the LDO excited? To what extent?
8. Could the turn be coordinated easily if desired?

9.6.9 Pedal Only Turns

The following considerations are generally included in the analysis of directional control only turns:

1. Did the aircraft initially roll opposite to the control input or display excessive hesitation in developing roll in the direction of yaw?
2. Could steady bank angles be obtained with a reasonable degree of directional control input leading?
3. What direction of directional control was required in the established turn?
4. What was the level of coordination (ball position) in the established turn?
5. Could roll out from the established turn be accomplished with directional control only?
6. Was the LDO excited? To what extent?
7. Did the LDO excitation significantly affect desired performance?

9.7 MISSION SUITABILITY

Mission suitability is the ultimate reason for conducting any handling qualities evaluation. The helicopter's unique capabilities in the low airspeed regime and its forward flight capabilities provide many potential operational uses if the handling qualities and performance of the vehicle are adequate. Before mission suitability is accessed, a detailed clear understanding of the intended mission is developed. Additionally, the pilot's experience in conducting similar mission maneuvers is considered. The test pilot researches the mission tasks to establish evaluation maneuvers and determine performance criteria. The mission maneuvers vary greatly with aircraft type, use, and operational environment. Potential maneuvers include everything from sling load operations to rocket firing.

Chapters 6, 7, and 8 address test procedures to evaluate handling qualities suitability when the inter-axis coupling is neglected or easily overcome. The test pilot now assesses mission effectiveness of the vehicle when all the considerations of the previous chapters are included, with the realization that the aircraft is asymmetrically configured and naturally exhibits coupled responses.

In addition to the engineering tests, the test pilot investigates how the helicopter is used and closely duplicates these mission tasks. Knowledge of coupling origins and measurement of coupling phenomena allows the test team to design operational scenarios which reflect the effects of undesirable coupling characteristics. Inter-axis coupling can aggravate long term dynamics and require significant compensation in short term maneuvering. The mission of the aircraft determines the maneuvers flown. Cargo helicopters require close attention to straight and level cruise tasks (long term dynamic stability) as well as maneuvering in confined areas. Attack aircraft may need special consideration for tracking tasks and maneuverability. Each test program requires a unique set of mission maneuvers.

The most important aspect of mission suitability testing is to keep the operator in mind. How the test vehicle is used in the field, and who is operating the aircraft are important factors to consider.

9.8 SPECIFICATION COMPLIANCE

Requirements for helicopter handling qualities are contained in MIL-H-8501A. Coupling requirements stated in MIL-H-8501A are very brief and are contained in paragraph 3.5.16. Paragraph 3.5.11.1 of MIL-H-8501A allows for mechanical control mixing to the extent "… no adverse limitations in control power shall exist with any possible combination of control inputs...".

9.9 GLOSSARY

9.9.1 Notations

a_0	Coning angle
A_{1s}	Lateral cyclic pitch angle, shaft referenced
a_{1s}	Longitudinal flapping angle, shaft referenced
AFCS	Automatic Flight Control System
ASW	Anti-submarine warfare
b	Number of blades
B_{1s}	Longitudinal cyclic pitch angle, shaft referenced
b_{1s}	Lateral flapping angle, shaft referenced
dD_0	Blade element profile drag

dF	Blade element torque force
dL	Blade element lift
dT	Blade element thrust
e	Flapping hinge offset
F	Force
FC	Fuel count
g	Gravity
h	Height of hub above CG
h'	Longitudinal distance between the rotor shaft and the CG
h'_{lat}	Lateral distance between the rotor shaft and the CG
H_{P_o}	Observed pressure altitude
HQR	Handling Qualities Rating
h_{TR}	Height of the tail rotor above the CG
IGE	In ground effect
i_{TR}	Tail rotor inclination out of vertical plane
I_{xx}	Moment of inertia about x axis, roll moment of inertia
I_{yy}	Moment of inertia about y axis, pitch moment of inertia
I_{zz}	Moment of inertia about z axis, yaw moment of inertia
L	Net moment about x axis, Roll moment
$L_{B_{1s}}$	Roll moment due to longitudinal cyclic pitch angle
LDO	Lateral-directional oscillation
L_H	Roll moment due to rotor hub forces
L_q	Roll moment due to pitch rate
L_{θ_C}	Roll moment due to collective pitch angle
$L_{\theta_{TR}}$	Roll moment due to tail rotor pitch angle
L_r	Roll moment due to yaw rate
l_t	Distance from the tail to the CG
L_u	Roll moment due to longitudinal velocity
L_w	Roll moment due to vertical velocity
M	Net moment about y axis, Pitch moment
$M_{A_{1s}}$	Pitch moment due to lateral cyclic pitch angle
M_H	Pitch moment due to rotor hub force

M_p	Pitch moment due to roll rate
M_{θ_C}	Pitch moment due to collective pitch angle
$M_{\theta_{TR}}$	Pitch moment due to tail rotor pitch angle
M_S	Blade mass moment
M_v	Pitch moment due to lateral velocity
M_w	Pitch moment due to vertical velocity
N	Net moment about z axis, Yaw moment
$N_{A_{1s}}$	Yaw moment due to lateral cyclic pitch angle
$N_{B_{1s}}$	Yaw moment due to longitudinal cyclic pitch angle
N_p	Yaw moment due to roll rate
N_{θ_C}	Yaw moment due to collective pitch angle
N_R	Main rotor speed
N_u	Yaw moment due to longitudinal velocity
N_v	Yaw moment due to lateral velocity
N_w	Yaw moment due to vertical velocity
OFT	Operational flight trainer
OGE	Out of ground effect
p	Angular velocity about x axis, Roll rate
$\dot{p}$	Angular acceleration about x axis
Q	Engine torque
q	Angular velocity about y axis, Pitch rate
Q_{MR}	Main rotor torque
$\dot{q}$	Angular acceleration about y axis, Pitch acceleration
R	Rotor radius
r	Angular velocity about z axis, Radius along blade, Yaw rate
$\dot{r}$	Angular acceleration about z axis, Yaw acceleration
SAR	Search and rescue
T	Thrust
t	Time
TCP	Trim control position
T_o	Observed temperature
TPP	Tip path plane
T_{TR}	Tail rotor thrust

u	Translational velocity component along x axis
u_0	Initial velocity
$\dot{u}$	Linear acceleration along x axis
V	Velocity, Free stream velocity, Relative velocity
v	Translational velocity component along y axis
VAR	Vibration Assessment Rating
v_i	Induced velocity at hover
V_o	Observed airspeed
VSTOL	Vertical/short takeoff and landing
V_v	Vertical velocity
$\dot{v}$	Linear acceleration along y axis
W	Weight
w	Translational velocity component along z axis
w_{rotor}	TPP translational velocity along z axis
$\dot{w}$	Linear acceleration along z axis
x	Orthogonal direction along longitudinal axis of the aircraft; Distance along x axis
y	Orthogonal direction along lateral axis of the aircraft
z	Orthogonal direction along vertical axis of the aircraft

9.9.2 Greek Symbols

α (alpha)	Angle of attack
β (beta)	Flapping angle, Sideslip angle
ϕ (phi)	Roll angle
λ (lambda)	Inflow ratio
λ_{TR}	Tail rotor inflow ratio
μ (mu)	Advance ratio
θ (theta)	Pitch angle
θ_C	Collective pitch angle

θ_{TR}	Tail rotor pitch angle
ω (omega)	Forcing frequency
ω_n	Natural frequency
ψ (psi)	Blade azimuth angle, Phase angle, Yaw angle
Ω (Omega)	Rotor angular velocity

9.10 REFERENCES

1. Johnson, W., *Helicopter Theory*, Princeton University Pres. 1980.

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3. Military Specification, “Helicopter Flying and Ground Handling Qualities; General Requirements For” MIL-H-8501A, with Amendment 1, 3 April, 1962.

4. Prouty, Raymond W., *Helicopter Performance, Stability, and Control*, PWS Publishers, Boston, MA, 1986.

5. U. S. Naval Test Pilot School Flight Test Manual, *Rotary Wing Performance*, USNTPS-FTM-No. 106 USNTPS, Patuxent River, MD, 31 December 1987.

CHAPTER TEN

SUDDEN ENGINE FAILURES, AUTOROTATIVE FLIGHT, AND AUTOROTATIVE LANDINGS

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EQUATIONS

		PAGE
$\frac{d\Omega}{dt} = \dot{\Omega} = -\frac{1}{I_R} Q$	*eq 10.1*	10.2
$dT = 4\rho\pi \; rdr \; v_i^2$	*eq 10.2*	10.3
$C_Q = \frac{Q}{\rho A_D \Omega^2 R^3}$	*eq 10.3*	10.4
$\frac{Q}{Q_0} = \frac{\Omega^2}{\Omega_0^2}$	*eq 10.4*	10.4
$t = \frac{I_R \Omega_0}{Q_0}\left(\frac{\Omega_0}{\Omega} - 1\right)$	*eq 10.5*	10.5
$\frac{1}{I_{zz}}\left(\frac{\partial N}{\partial \theta_{TR}}\right)\theta_{TR} = \frac{1}{I_{zz}}\left(\frac{\partial N}{\partial \delta_{PED}}\right)\delta_{PED} = \dot{r}_0$	*eq 10.6*	10.9
$T_{TR} \, l_t = Q_{MR}$	*eq 10.7*	10.9
$T_{TR}\left(\cos i_{TR}\right) l_t = Q_{MR}$	*eq 10.8*	10.9
$N_{\delta_{PED}} \approx - \frac{1}{I_{zz}}\left(\frac{\partial T_{TR}}{\partial \delta_{PED}}\right) l_t$	*eq 10.9*	10.11

$$- \frac{Q_{MR}}{I_{zz}} = \frac{N}{I_{zz}} = \dot{r} - N_r r$$ *eq 10.10* 10.11

$$\dot{r}_0 = - \frac{Q_{MR}}{I_{zz}}$$ *eq 10.11* 10.15

$$\delta_{PED} = - \frac{Q_{MR}}{N_{\delta_{PED}}}$$ *eq 10.12* 10.15

$$L_H = \left(\frac{ebM_S\Omega^2}{2} \right) b_{1s}$$ *eq 10.13* 10.17

$$M_H = \left(\frac{ebM_S\Omega^2}{2} \right) a_{1s}$$ *eq 10.14* 10.18

$$\frac{\partial a_{1s}}{\partial B_{1s}} = - \frac{\left(1 + \frac{3}{2} \mu^2 \right)}{\left(1 - \frac{\mu^2}{2} \right)}$$ *eq 10.15* 10.35

$$M_{CG} = \left(Th + \frac{ebM_S\Omega^2}{2} \right) a_{1s}$$ *eq 10.16* 10.36

$$L_{CG} = \left(Th + \frac{ebM_S\Omega^2}{2} \right) b_{1s}$$ *eq 10.17* 10.42

$$\dot{h}_{TD} = \sqrt{2gh_{AGL}}$$ *eq 10.18* 10.54

$$\frac{d\Omega}{dt} = - \frac{1}{I_R}\left(\frac{\Omega^2}{\Omega_0^2}\right) Q_0$$

eq 10.19 10.55

$$\frac{T}{T_0} = \frac{\Omega^2}{\Omega_0^2}$$

eq 10.20 10.56

$$\ddot{z} = \ddot{h} = \frac{W - T}{m} = g\left(1 - \frac{T}{W}\right)$$

eq 10.21 10.56

$$\dot{h} = gt^2\left(\frac{Q_0}{I_R\Omega_0}\right)\left(1 + \frac{tQ_0}{I_R\Omega_0}\right)^{-1}$$

eq 10.22 10.57

$$\Delta h = \frac{gt^2}{2}\left[1 - \frac{2I_R\Omega_0}{Q_0 t} + \frac{2\left(I_R\Omega_0\right)^2}{\left(Q_0 t\right)^2} \ln\left(1 + \frac{Q_0 t}{I_R\Omega_0}\right)\right]$$

eq 10.23 10.57

CHAPTER TEN

SUDDEN ENGINE FAILURES, AUTOROTATIVE FLIGHT, AND AUTOROTATIVE LANDINGS

10.1 INTRODUCTION

This chapter discusses helicopter stability, control, and flying qualities during engine failure, autorotation entry, steady autorotative flight, and landing following a successful autorotation. Although the emphasis is on flying qualities, knowledge of helicopter performance requirements and power deficits is essential to understanding the autorotation evaluation starting with engine failure and ending with a satisfactory landing. Flying qualities and performance characteristics are both required to analyze and test autorotation capability.

10.2 PURPOSE

The purpose of these tests is to evaluate the handling qualities of the helicopter from engine failure to completion of the landing. Tests are conducted to evaluate the following characteristics:

1. Engine failure and autorotation entry.
2. Steady autorotative flight.
3. Autorotative landings.

Tests investigate the effects of flight condition (hover, low airspeed, and forward flight), power setting (climb, level, and descent), and altitude. Further testing may be conducted to assess the effects of parameters such as gross weight (GW) and center of gravity (CG).

10.3 THEORY

10.3.1 Engine Failure

Sudden engine failures for single rotor helicopters in powered flight result in large unbalanced moments on both the rotor and airframe. In time these moments produce rotor speed changes and attitude excursions resulting in aircraft deviations from the original flight condition. The pilot eventually responds to these excursions with control inputs to bring the helicopter back under control, establish the steady autorotative glide, and perform the landing maneuver.

10.3.1.1 ROTOR SPEED DECAY

At the time of power loss there is a decelerating torque on the drive train proportional to the engine power that was being delivered to the main rotor, tail rotor, accessories, and to overcome mechanical losses in the drive train. The decelerating torque produces a drive train deceleration given by:

$$\frac{d\Omega}{dt} = \dot{\Omega} = -\frac{1}{I_R} Q \qquad \textit{eq 10.1}$$

Where:

I_R	- Rotor rotational moment of inertia
Q	- Torque
t	- Time
Ω	- Rotor angular velocity
$\dot{\Omega}$	- Rotor angular acceleration.

If flight conditions (airspeed, rate of climb/descent) are constant over the time interval of interest, the time for rotor decay to some prescribed percentage of the initial value can be predicted. Consider the main rotor for a hovering helicopter. Figure 10.1 shows the geometry of a rotor blade section at two different rotational speeds for the same collective pitch angle.

The thrust (T) produced for each blade section is proportional to the square of the rotational velocity (Ω_0 or Ω). Simple momentum theory relates thrust and induced velocity by:

$$dT = 4\rho\pi \; rdr \; v_i^2 \qquad \textit{eq 10.2}$$

Where:

π	- Mathematical constant
ρ	- Density
r	- Radius along blade
T	- Thrust
v_i	- Induced velocity.

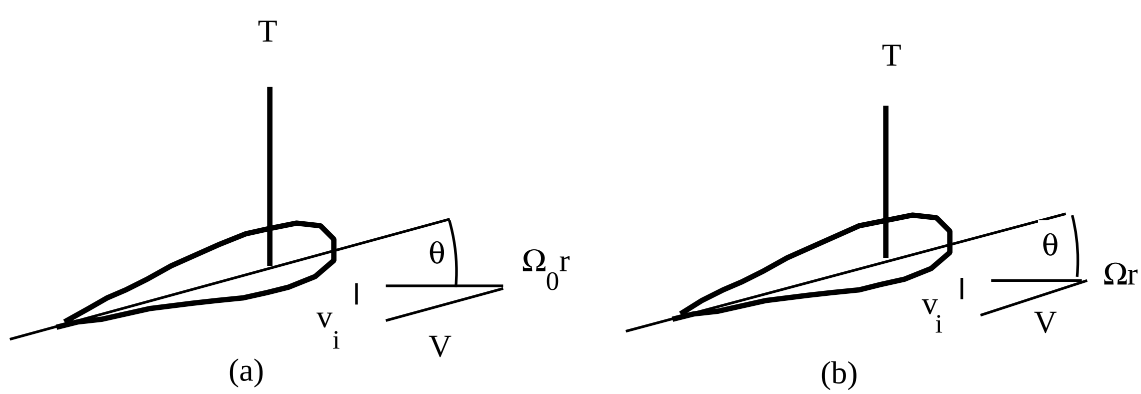


Figure 10.1
Rotor Blade Section Velocities

Since the thrust is proportional to the square of the rotational velocity (Figure 10.1) and the square of the induced velocity, thrust changes caused by different rotor speeds do not affect the ratio of v_i to Ωr or the blade angle of attack. The blade angle of attack dictates the lift coefficient and the power or torque coefficient (C_P or C_Q). The torque coefficient (Reference 10.3) is defined as:

$$C_Q = \frac{Q}{\rho A_D \Omega^2 R^3}$$

eq 10.3

Where:

C_Q	- Torque coefficient
Q	- Torque
ρ	- Density
A_D	- Rotor disc area
Ω	- Rotor angular velocity
R	- Rotor radius.

Since the torque coefficient is constant for constant blade pitch angle, then:

$$\frac{Q}{Q_0} = \frac{\Omega^2}{\Omega_0^2}$$

eq 10.4

Where:

Q	- Torque
Q_0	- Initial torque
Ω	- Rotor angular velocity
Ω_0	- Initial rotor angular velocity.

Substitution of Equation 10.4 into Equation 10.1 for main rotor torque and integrating the differential quantities gives the rotor decay time as:

$$t = \frac{I_R \Omega_0}{Q_0}\left(\frac{\Omega_0}{\Omega} - 1\right)$$

eq 10.5

Where:

I_R	- Rotor rotational moment of inertia
Q_0	- Initial torque
t	- Time
Ω	- Rotor angular velocity
Ω_0	- Initial rotor angular velocity.

Eq 10.5 states the time required for the rotor speed to decay to some value is directly proportional to the rotor inertia and initial speed, and inversely proportional to initial torque. Figures 10.2, 10.3 and 10.4 are graphical representations of Equation 10.5.

Plots similar to Figures 10.2, 10.3, or 10.4 for the test helicopter are produced with manufacturer supplied moment of inertia data. Test results may be used to update the decay plot computed for the test helicopter.

10.3.1.2 THRUST DECAY

After power loss, with no control inputs or significant change in flight condition in the time interval of interest, rotor thrust decays as the square of the rotor speed. As a first cut, a 10% decay in rotor speed at constant collective produces a 21% reduction in thrust.

10.3.1.3 YAW RESPONSE

A large contribution to yaw moment equilibrium is provided by tail rotor thrust. An instantaneous power loss produces an initial yaw acceleration to the left for U.S. designs because of the unbalanced yaw moment.

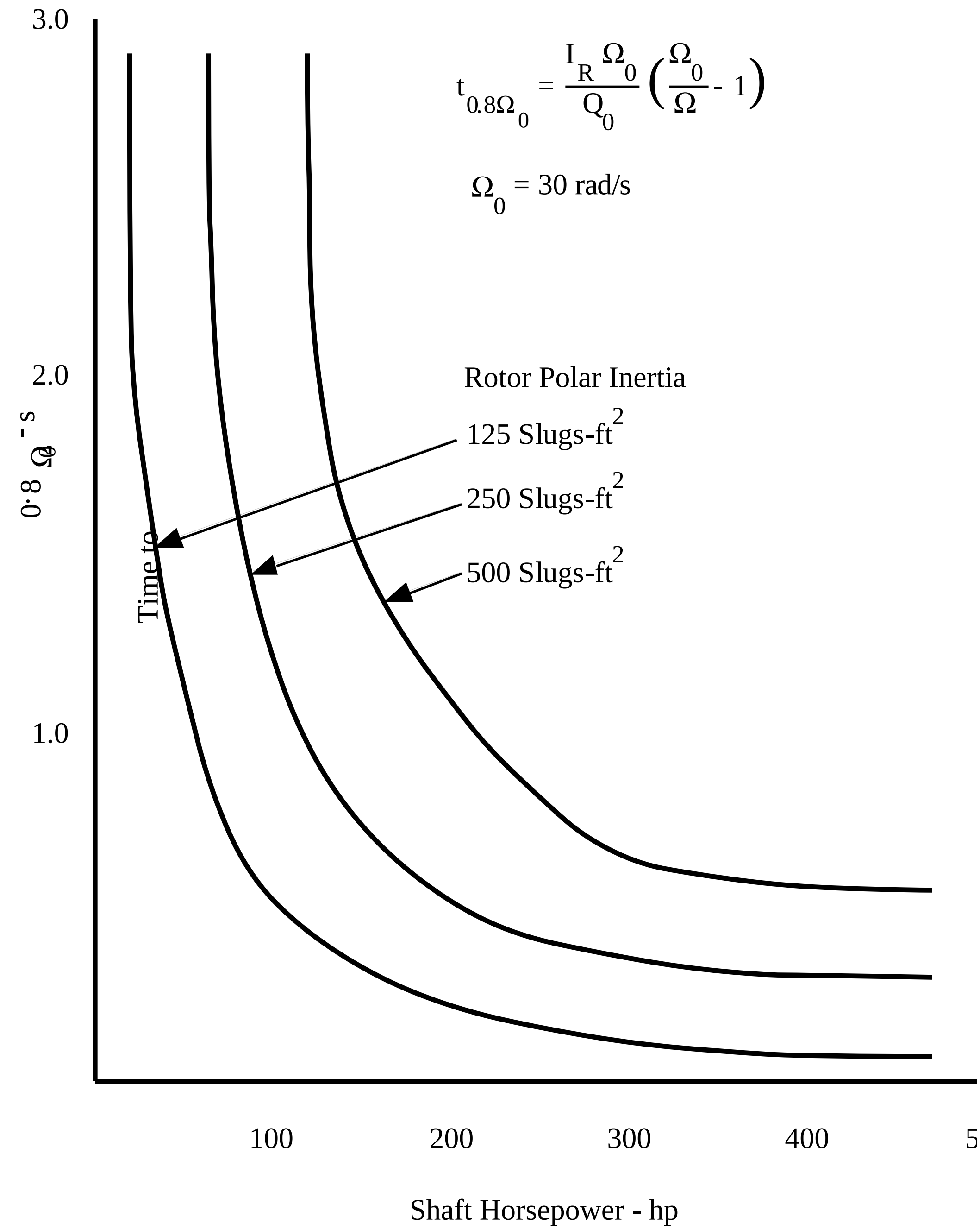


Figure 10.2
Rotor Speed Decay for Typical Light Helicopters

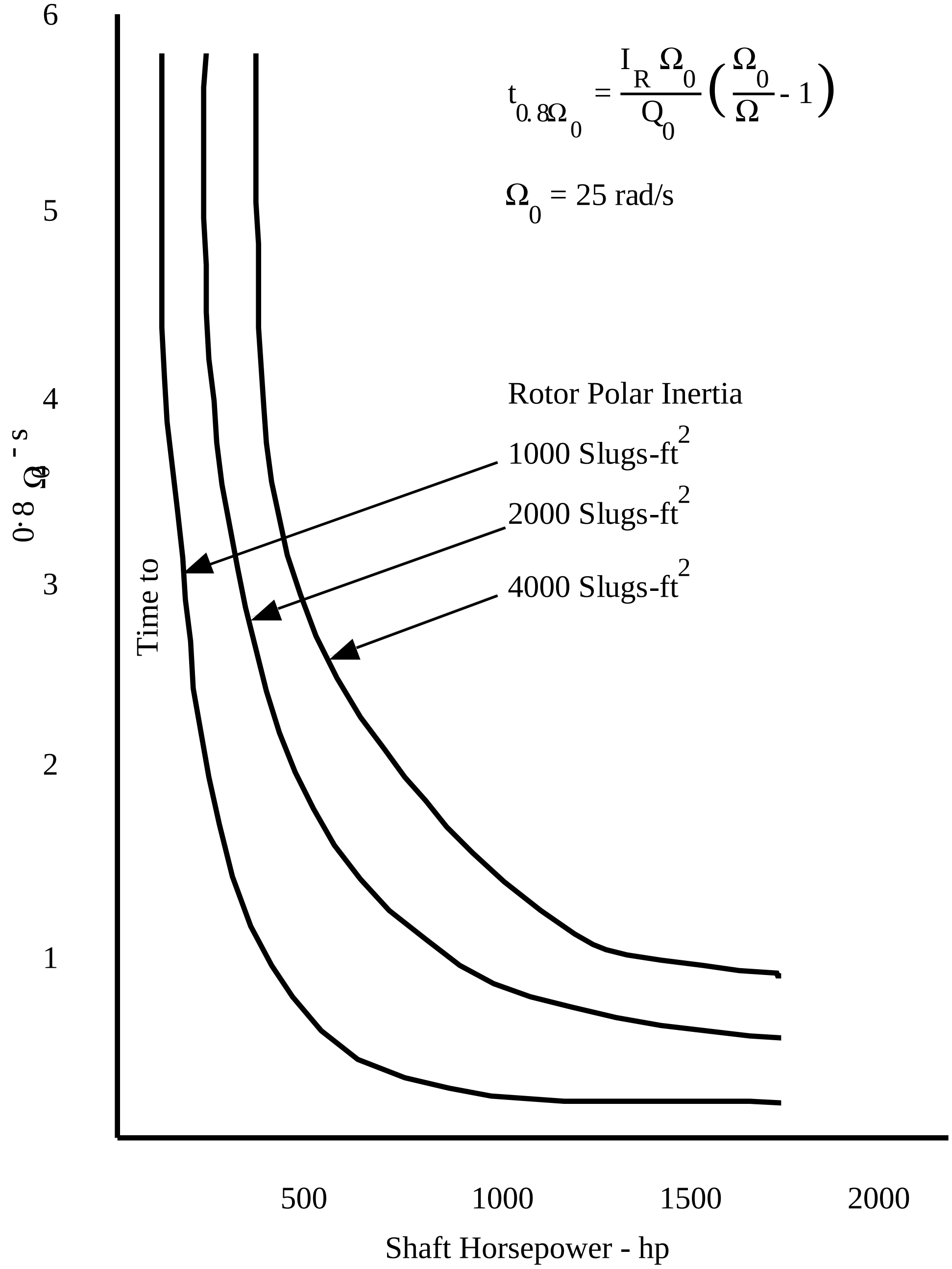


Figure 10.3
Rotor Speed Decay for Typical Medium Helicopters

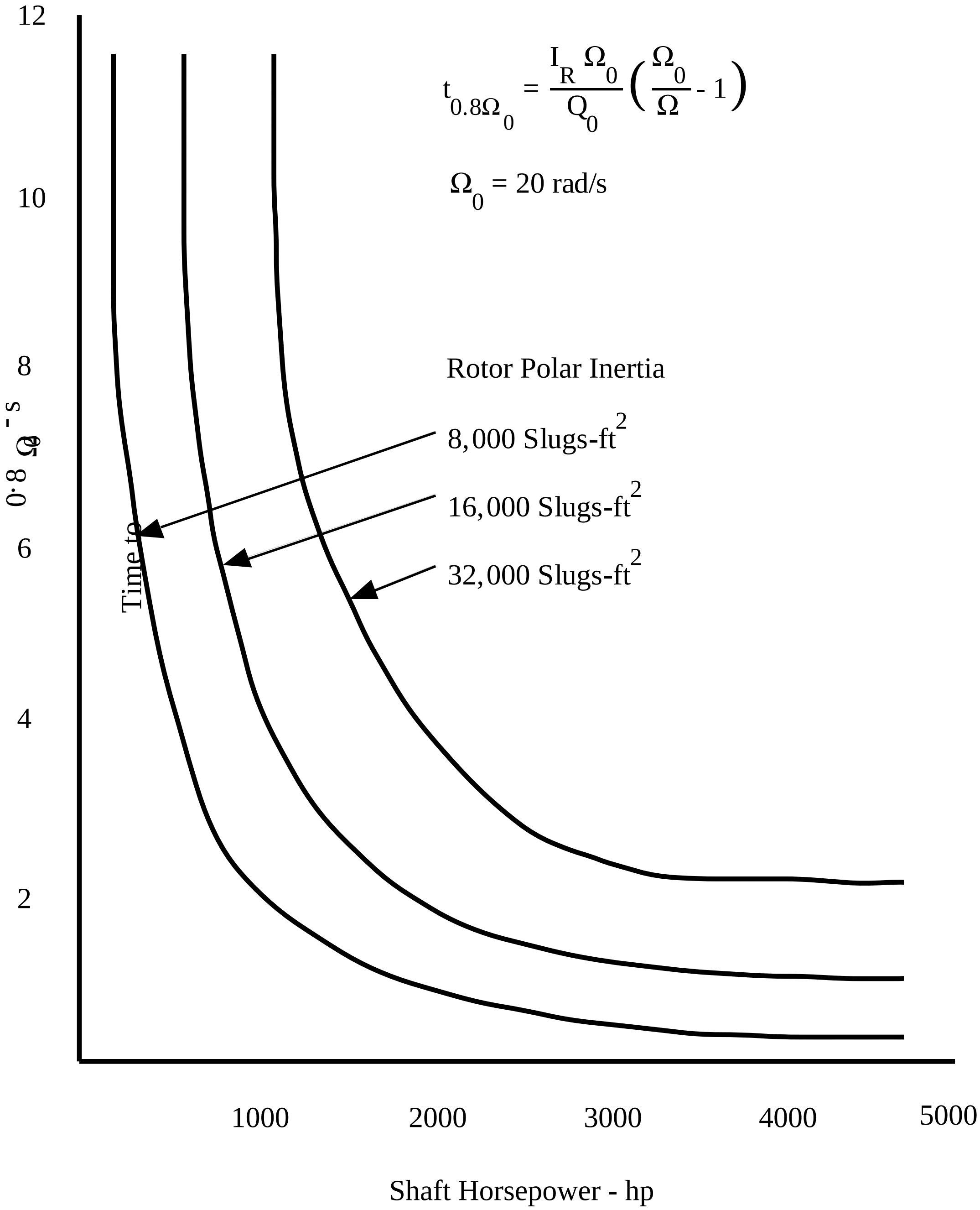


Figure 10.4
Rotor Speed Decay for Typical Heavy Helicopters

10.3.1.3.1 Hover

Data available from hover directional control response tests and hover performance tests are helpful in predicting initial yaw response and pedal requirements to counter yaw moments following power failure.

In hover, directional control response is essentially first order and largely a single axis response. Directional control response time histories are used to determine control sensitivity and yaw rate damping. The yaw moment of inertia may be available from the airframe manufacturer or can be estimated as follows:

$$\frac{1}{I_{zz}}\left(\frac{\partial N}{\partial \theta_{TR}}\right)\theta_{TR} = \frac{1}{I_{zz}}\left(\frac{\partial N}{\partial \delta_{PED}}\right)\delta_{PED} = \dot{r}_0 \qquad \textit{eq 10.6}$$

Where:

δ_{PED}	- Pedal control
I_{zz}	- Moment of inertia about z axis, yaw moment of inertia
N	- Net moment about z axis, Yaw moment
θ_{TR}	- Tail rotor pitch angle
$\dot{r}_0$	- Initial angular acceleration about z axis.

The ratio of $\dot{r}_0$ to δ_{PED} yields the yaw moment due to pedal control (pedal control sensitivity, $N_{\delta_{PED}}$) and is shown in Figure 10.5.

When main rotor torque data are available from hover performance testing, the tail rotor thrust required to balance main rotor torque can be estimated from:

$$T_{TR}\, l_t = Q_{MR} \qquad \textit{eq 10.7}$$

or:

$$T_{TR}\left(\cos i_{TR}\right) l_t = Q_{MR} \qquad \textit{eq 10.8}$$

Where:

i_{TR}	- Tail rotor inclination out of vertical plane
l_t	- Distance from the tail to the CG
Q_{MR}	- Main rotor torque
T_{TR}	- Tail rotor thrust.

Sample plots of tail rotor parameters available from hover performance tests are depicted in Figures 10.6 and 10.7 for generalized data.

The data of Figures 10.6 and 10.7 may be converted to engineering units as shown in Figures 10.8 and 10.9.

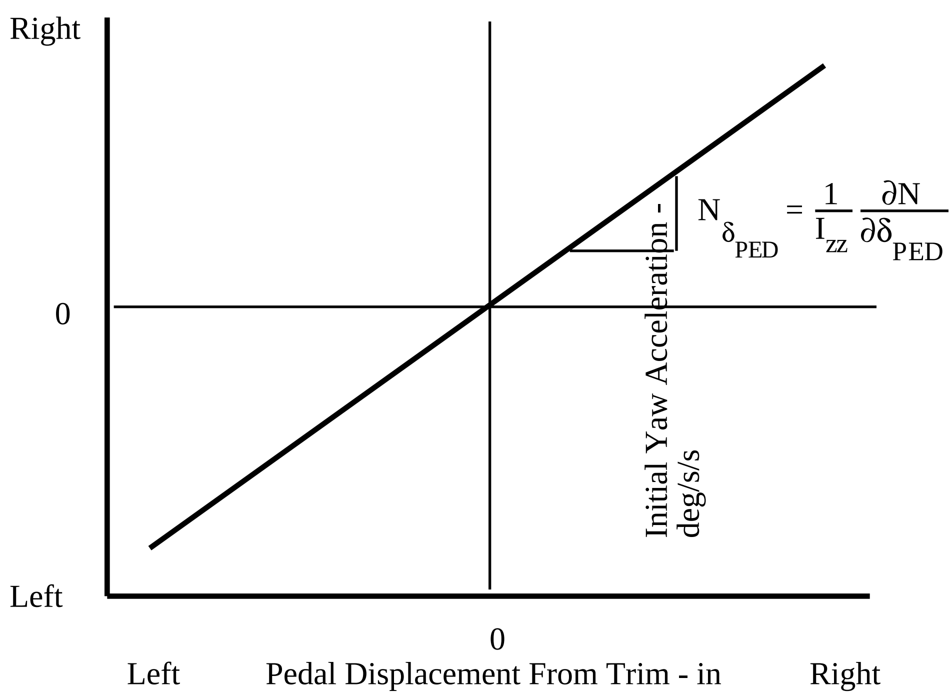


Figure 10.5
Determination of Directional Control Sensitivity

The data of Figures 10.8 and 10.9 in terms of stability derivatives are:

$$N_{\delta_{PED}} \approx -\frac{1}{I_{zz}}\left(\frac{\partial T_{TR}}{\partial \delta_{PED}}\right) l_t \qquad \textit{eq 10.9}$$

Where:

δ_{PED}	- Pedal control
I_{zz}	- Moment of inertia about z axis, yaw moment of inertia
l_t	- Distance from the tail to the CG
$N_{\delta_{PED}}$	- Yaw moment due to pedal control
T_{TR}	- Tail rotor thrust.

The directional control sensitivity is obtained from Figure 10.5 and the partial $\partial T_{TR}/\partial \delta_{PED}$ from Figure 10.6, yielding I_{zz}.

For a given hover condition (GW and N_R), the main rotor torque is obtained from Figure 10.9 or other hover plots. In the event of an instantaneous power loss, the yaw response is obtained from:

$$-\frac{Q_{MR}}{I_{zz}} = \frac{N}{I_{zz}} = \dot{r} - N_r r \qquad \textit{eq 10.10}$$

Where:

I_{zz}	- Moment of inertia about z axis, yaw moment of inertia
N	- Net moment about z axis
N_r	- Yaw moment due to yaw rate
Q_{MR}	- Main rotor torque
r	- Angular velocity about z axis
$\dot{r}$	- Angular acceleration about z axis.

The yaw damping derivative can be determined from directional control response tests. Equation 10.10 can be integrated to determine yaw rate and heading changes expected with time, following an abrupt power loss with controls fixed.

During the test planning stages prior to conducting simulated engine failure tests, hover performance and control response data are analyzed to gain insight to expected response in the first moments following power loss. Moment of inertia values (I_{ZZ}) may be updated if required, to adjust yaw stability derivatives when different aircraft loadings are used for control response and engine failure tests.

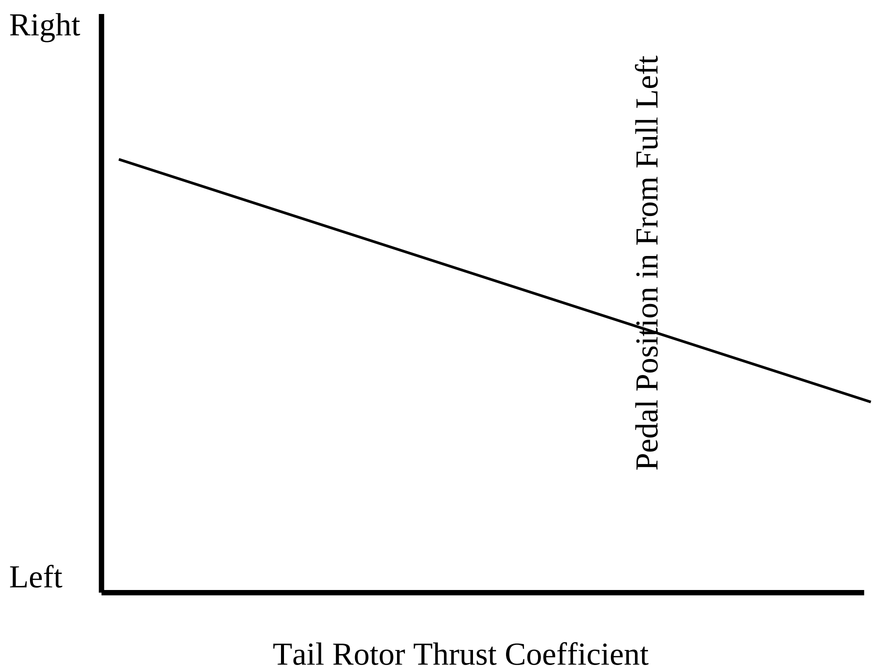


Figure 10.6
Pedal Position Versus Tail Rotor Thrust Coefficient

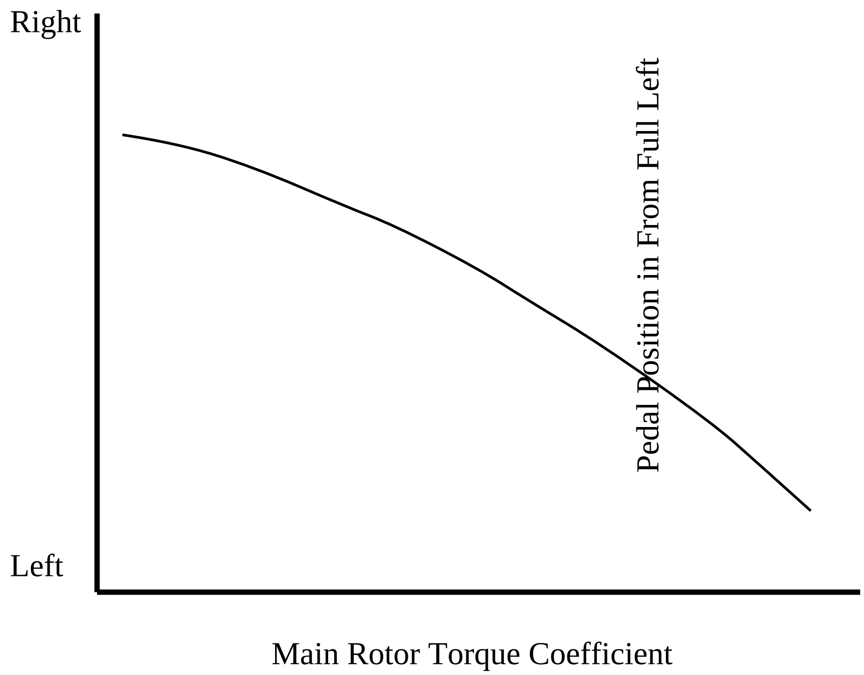


Figure 10.7
Pedal Position Versus Main Rotor Torque Coefficient

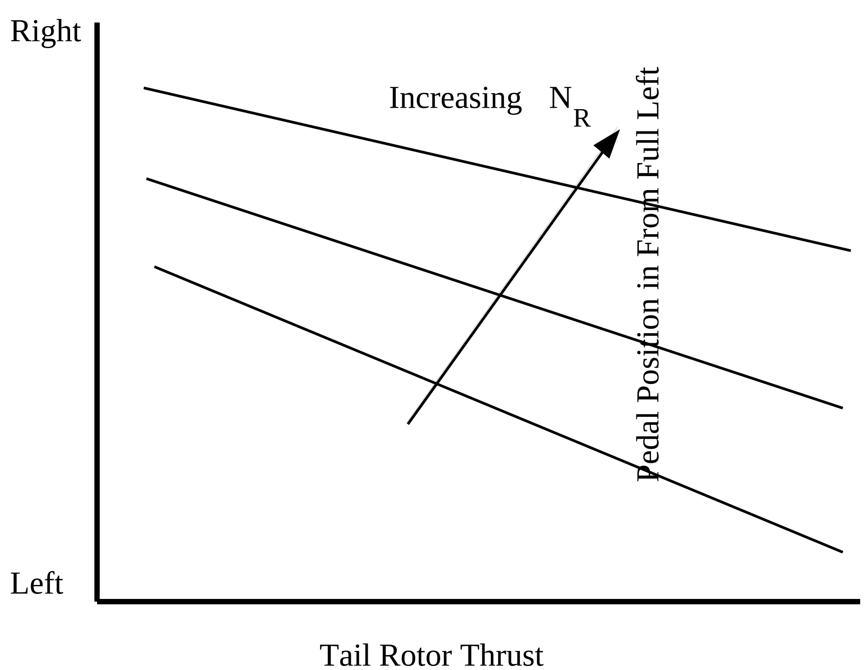


Figure 10.8
Pedal Position Versus Tail Rotor Thrust

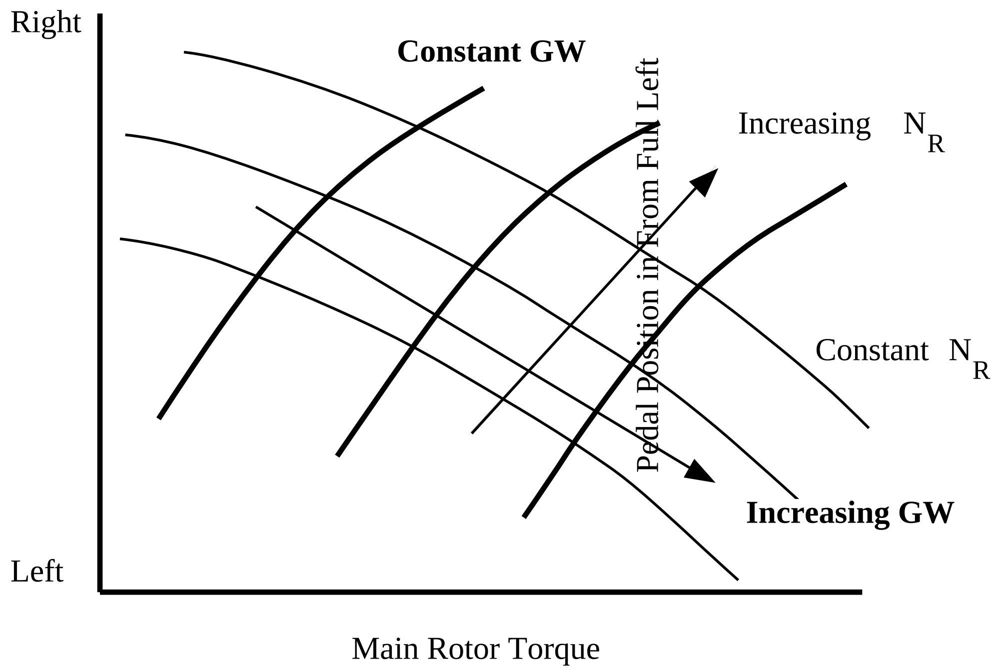


Figure 10.9
Pedal Position Versus Main Rotor Torque

10.3.1.3.2 Level Forward Flight

The yaw response in forward flight is somewhat different than hover largely due to the directional stability contribution. Torque contributions to initial yaw acceleration following power loss are similar to hover. Power required and torque varies with airspeed and GW, as shown in Figure 10.10. Therefore, the initial yaw acceleration with engine failure becomes milder as airspeed increases from hover, to the airspeed for minimum power, then becomes progressively stronger as airspeed continues to increase. The effect of increased weight or altitude is to accentuate the initial yaw acceleration, particularly at the slower airspeeds.

Following power loss without corrective action, the unbalanced left yaw moment produces a right sideslip. As the yaw rate develops, right yaw moments due to yaw damping reduce the yaw acceleration. In hover, if sufficient time is available before control input, the yaw approaches a quasi steady rate, modified by tail rotor thrust reduction as drive train speed decays. In forward flight, the unchecked response is a lateral-directional oscillation centered about a quasi steady right sideslip. Data obtained from directional control response tests in forward flight provide insight into the overall aircraft response expected from a power loss. Forward flight data plotted as shown in Figure 10.5 provide pedal sensitivity. The product $N_{\delta_{PED}}\delta_{PED}$ is the initial yaw acceleration ($\dot{r}_0$) from Equation 10.6. The initial yaw acceleration can be related to main rotor torque and pedal input as follows:

$$\dot{r}_0 = -\frac{Q_{MR}}{I_{zz}} \qquad \textit{eq 10.11}$$

$$\delta_{PED} = -\frac{Q_{MR}}{N_{\delta_{PED}}} \qquad \textit{eq 10.12}$$

Where:

δ_{PED}	- Pedal control
I_{zz}	- Moment of inertia about z axis, yaw moment of inertia
Q_{MR}	- Main rotor torque
$\dot{r}_0$	- Initial angular acceleration about z axis.
$N_{\delta_{PED}}$	- Yaw moment due to pedal control.

The size of a left pedal input required to simulate yaw response following a sudden engine failure is predicted by Equation 10.12. If this pedal input size is not excessively large or prohibited, a limited simulation of engine failure characteristics can be obtained from directional control response tests. Obvious limitations of this simulation are due to thrust changes with rotor speed decay and roll moments generated by tail rotor thrust changes.

Roll responses, particularly those associated with sideslip, are expected. Use of pedal inputs to simulate aircraft response to power loss for a single rotor helicopter provide a reasonable approximation to roll characteristics following engine failure without pilot corrective action.

Yaw response of the tandem helicopter usually is less severe than for the single rotor helicopter. The counter-rotating rotors produce corresponding opposed rotor torques. If the torques are balanced, no significant yaw occurs for a sudden power loss. If one rotor is delivering significantly more torque than the other, the sudden torque loss to both rotors results in a net yaw moment.

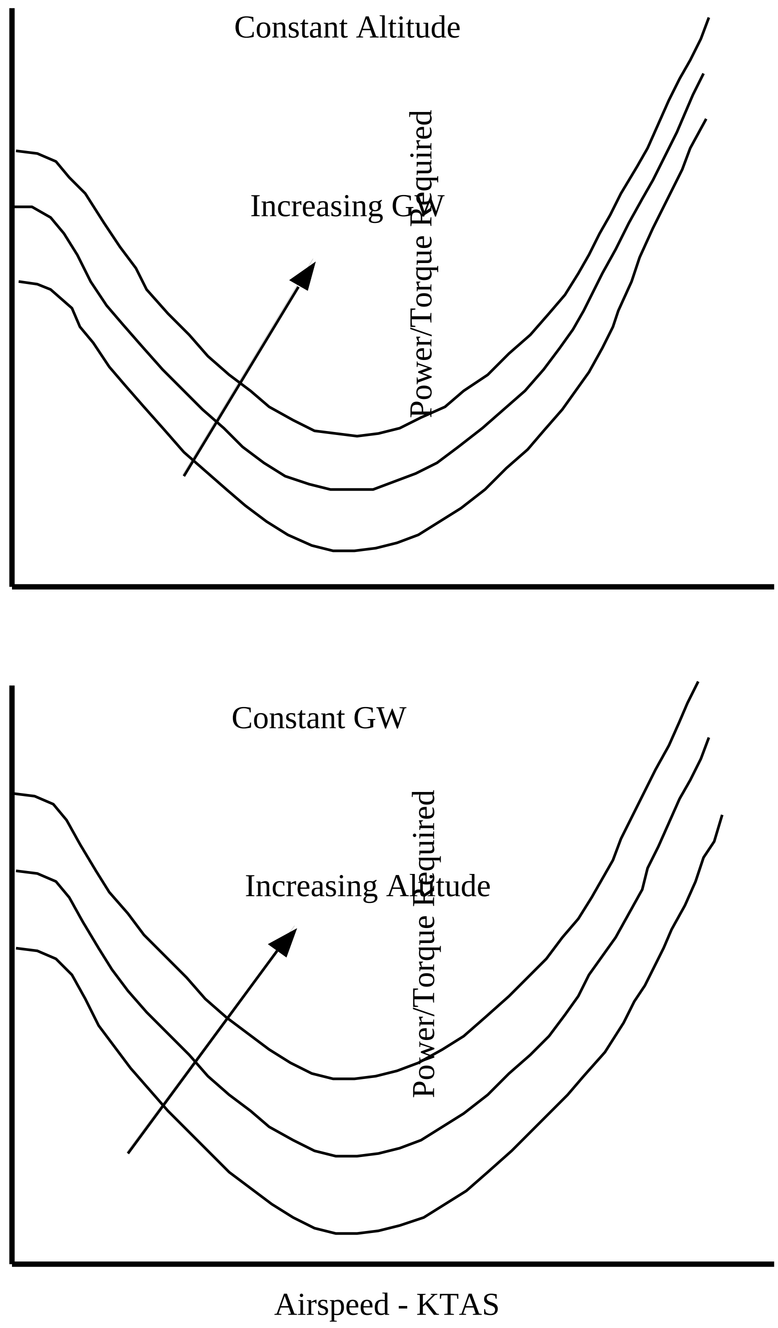


Figure 10.10
Level Flight Power/Torque Required

10.3.1.4 ROLL RESPONSE

10.3.1.4.1 Hover

The airframe response to sudden power loss in hover is predominantly a yaw response. Some roll response due to thrust decay with rotor speed when the main rotor thrust axis is not aligned laterally with the aircraft CG. If the helicopter has flapping hinge offset, changes in roll hub moment result from rotor speed decay. The roll hub moment equation as developed in Chapter 4 is:

$$L_H = \left(\frac{ebM_S\Omega^2}{2}\right) b_{1s} \qquad \textit{eq 10.13}$$

Where:

L_H	- Roll moment due to rotor hub forces
e	- Flapping hinge offset
b	- Number of blades
M_S	- Blade mass moment
Ω	- Rotor angular velocity
b_{1s}	- Lateral flapping angle, shaft referenced.

The initial hub moment before power loss depends on lateral tip path plane (TPP) tilt (b_{1s}). For a given b_{1s}, the hub moments decrease with rotor speed producing a net moment opposite to TPP tilt. For helicopters with the tail rotor located vertically above the CG and without lateral CG asymmetry, thrust reduction and hub moment reduction with rotor speed decay produce right roll moments. Another possible source of roll moments is the reduction in tail rotor thrust with drive train speed decay and the resulting left yaw rates. The roll moment due to tail rotor thrust change depends on tail rotor vertical position relative to the CG. The roll produced by main rotor and tail rotor thrust changes are not expected to be significant. Proper buildup procedures during simulated engine failure tests determine the importance of roll accelerations.

10.3.1.4.2 Level Forward Flight

In forward flight, appreciable roll moments may exist following sudden power loss as the sideslip angle increases. The dihedral effect produces roll with increased sideslip angle. There may be some roll due to yaw rate. With positive effective dihedral, right sideslips produce left roll moments and left yaw rates produce left roll moments. Positive dihedral effect and roll due to yaw rates are expected from the main rotor and from high tail rotors. When the helicopter exhibits positive dihedral effect, roll in the direction of yaw is observed following power loss. The relative strength of dihedral effect is observed during pedal only turns and lateral-directional oscillation testing (amount of roll manifested during lateral-directional oscillations).

10.3.1.5 PITCH RESPONSE

10.3.1.5.1 Hover

Pitch excursions in hover following sudden power loss result from thrust changes as rotor speed decays when the thrust axis is not aligned longitudinally with the helicopter CG. With flapping hinge offset, changes in hub pitch moments result from rotor speed decay. The pitch hub moment equation as developed in Chapter 4 is:

$$M_H = \left(\frac{ebM_S\Omega^2}{2}\right) a_{1s} \qquad \textit{eq 10.14}$$

Where:

a_{1s}	- Longitudinal flapping angle, shaft referenced
b	- Number of blades
e	- Flapping hinge offset
M_H	- Pitch moment due to rotor hub force
M_S	- Blade mass moment
Ω	- Rotor angular velocity.

The initial hub moment before power loss depends on longitudinal TPP tilt (a_{1s}). For a given a_{1s}, the hub moments are reduced with rotor speed. The loss in rotor speed generates a net moment away from the direction of TPP tilt. For helicopters with offset flapping hinges, displacement of the thrust vector longitudinally from the CG causes the thrust moments and hub moments to be in opposite directions. As thrust and hub moments decrease with rotor speed decay, their changes are opposing. For example, when the CG is aft of the shaft, TPP tilt is forward and the thrust axis passes ahead of the CG. Rotor speed loss reduces the thrust producing a nose down moment and also reduces the hub moment producing a nose up moment. The onset of any pitch accelerations lag the yaw accelerations since they depend on rotor speed changes and require a finite amount of time to develop. As in roll, in the first few moments following power loss, significant pitch moments are not anticipated.

10.3.1.5.2 Level Forward Flight

Thrust decay and hub moment contributions in forward flight have the same origins as in hover. Additional considerations are:

1. Fuselage aerodynamics (tail loads) and their effects on thrust axis longitudinal orientation with respect to the CG.
2. Blowback effects.
3. Sideslip generated pitch moments.
4. Trim lateral TPP position.

Fuselage aerodynamics contribute to the trimmed a_{1S}. If significant tail down loads exist in trimmed forward flight, additional forward a_{1S} is required to balance tail loads. Tail down loads work to favor the hub moment changes with rotor speed decay for aft CG positions, and to favor the thrust effect for forward CG position. Figures 10.11 and 10.12 depict these concepts.

For the aft CG case (Figure 10.11), increase in tail down load requires forward TPP tilt for equilibrium with the thrust vector through the CG. For the forward CG (Figure 10.12), the increase in tail down load increases the forward tilt for equilibrium with the TPP aligned with the shaft when the tail load is present. Some differences in aircraft pitch attitude (and trim tail loads) exist between the helicopters represented in Figures 10.11 and 10.12, but analysis of these differences is not required to illustrate the basic trends. In Figure 10.11 (aft CG), the thrust and the hub moments both decrease with rotor speed decay. When the tail load is present, loss of rotor speed only affects the hub moment contribution. Reduction of the nose down hub moment results in a net nose up moment. For the forward CG, with the down tail load, reduction of thrust causes a nose up moment and no hub moment change occurs. In both cases, the down tail load at trim causes a net nose up moment as rotor speed decays.

Rotor blowback is a function of collective pitch, inflow, and advance ratio. With controls fixed, θ_C is constant. The inflow contribution is less than the collective pitch contribution for positive thrust. For positive thrust, advance ratio changes produce blowback phenomenon in the usual sense. As rotor speed decays, the advance ratio increases proportional to rotor speed loss producing aft TPP tilt. The aft tilt results in a nose up moment and a secondary effect on thrust as shown in Figure 10.13. The rotor blowback increases the upwash component through the rotor and increases blade angle of attack thereby reducing to some extent the rotor thrust loss resulting from rotor speed decay.

As right sideslip increases after power loss, the TPP coning and the right lateral velocity generate blade lift changes between the advancing blade ($\psi = 90^\circ$) and retreating blade ($\psi = 270^\circ$) which leads to flapping approximately 90˚ later or longitudinal flapping. The flapping for the coned TPP produces aft tilt and nose up moments.

The trimmed, lateral TPP position before power loss is dictated essentially by the requirement to balance side forces and roll moments. If the TPP is initially trimmed left, sideslip changes increase upwash through the rotor, thereby delaying rotor thrust loss and the pitch and roll moments associated with thrust changes.

The total pitch moment involved with sudden power loss in forward flight is usually nose up. Moderate nose up pitch is probably desirable if operating at airspeeds above optimum autorotative airspeeds. Pitch moments developed after power loss result from rotor speed and sideslip changes and are observed as delayed responses occurring after torque reduction.

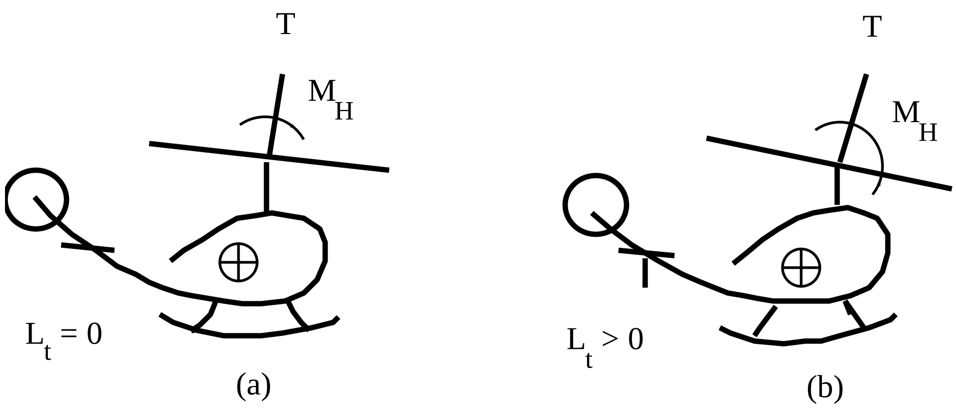


Figure 10.11
Rotor Pitch Moment Changes at Aft CG

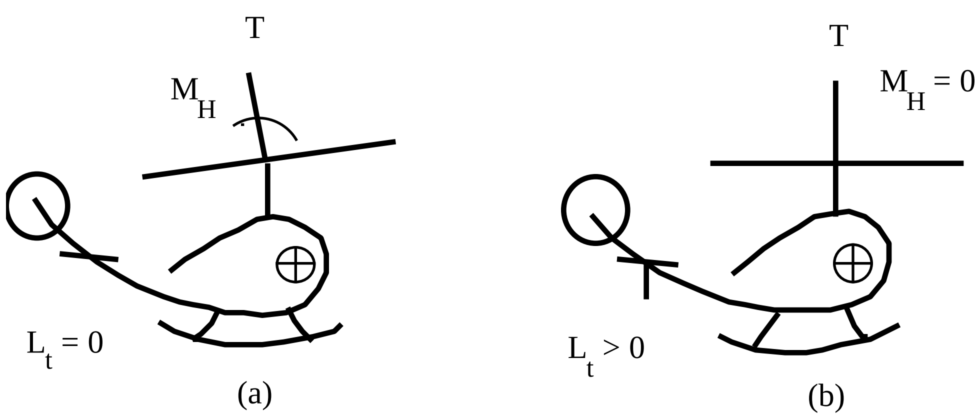
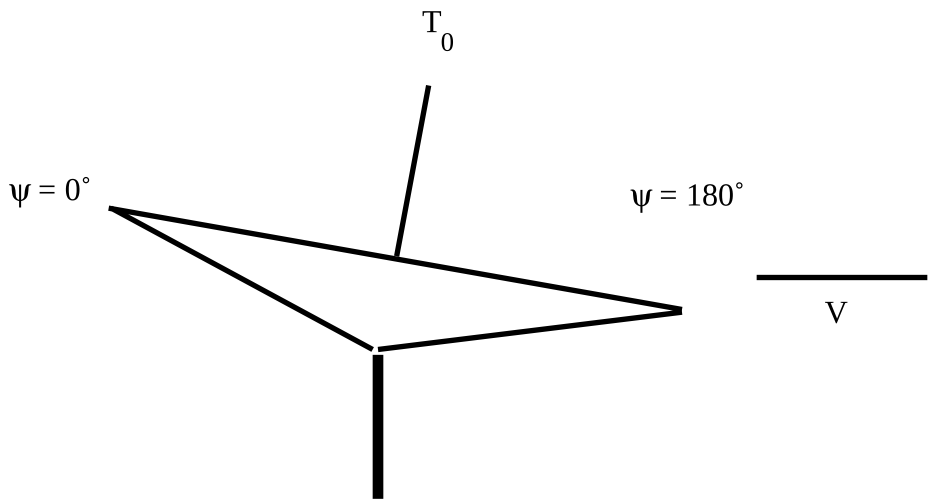
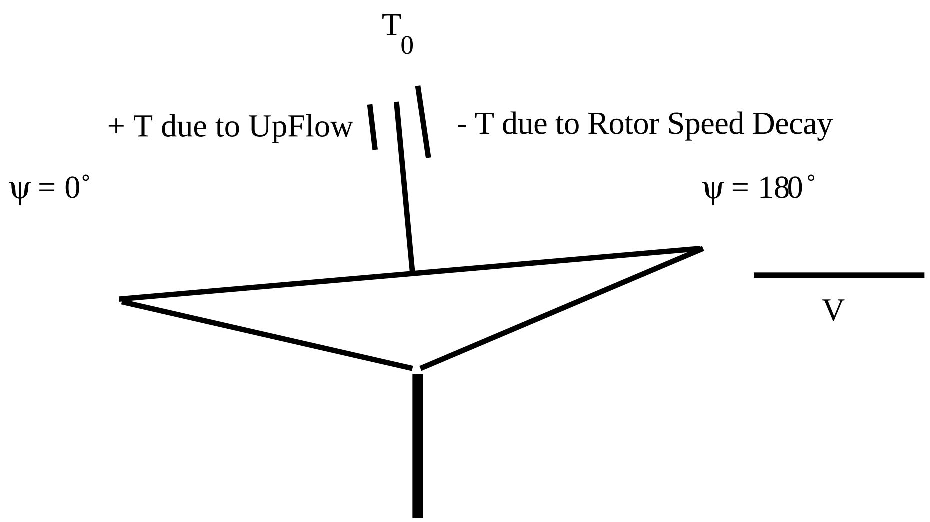


Figure 10.12
Rotor Pitch Moment Changes at Forward CG



(a) TPP Position and Thrust Before Rotor Speed Decay



(b) TPP Position and Thrust After Rotor Speed Decay and μ Increase

Figure 10.13
Aft TPP Tilt with Increased Advance Ratio

Tandem rotor helicopters may experience nose up or nose down pitch following power loss. If the blowback effect due to increased μ with reduced rotor speed dominates, both rotors tilt aft giving some relief in thrust loss and nose up moments from the aft tilt. However, with significant thrust loss, the forward rotor may undergo a larger thrust loss than the aft rotor. The aft rotor in powered flight operates in the wake of the forward rotor and operates usually at higher collective pitch to counter the downwash of the forward rotor. When the forward rotor loses thrust with rotor speed decay, the downwash on the aft rotor is reduced and the aft rotor thrust loss may not be as severe as the forward rotor. The difference in thrust changes causes nose down moments.

10.3.1.6 CLIMB VERSUS DESCENT

The initial responses to sudden power loss (rotor speed decay and yaw acceleration) depend on the magnitude of torque being delivered. Later responses result from sideslips and rotor speed decay. In low power descending flight, less torque is lost during engine failure and the overall aircraft response is milder and takes more time to develop. Additionally, the helicopter is close to the stabilized autorotation condition and less transition is required.

During high power climbs just the opposite occurs, rotor decay and yaw accelerations are greater. In a high power climb the flight path angle must be reversed before steady autorotative flight is achieved.

During evaluation of aircraft characteristics following engine failure, a build-up technique starting with low power is used. This build up approach reduces initial and follow on responses, and allows more controlled entry to autorotation during initial testing. Surprises are minimized as tests progress to high power settings.

10.3.1.7 DESIRED PITCH AND ROLL RESPONSES

MIL-H-8501A places limits on the pitch, roll, and yaw (heading) excursions allowed before pilot intervention following sudden power loss. Within the constraints of allowable excursions, it is desirable to have pitch and roll excursions which:

1. Minimize rotor speed loss.
2. Assist the helicopter toward the required steady flight condition.

These two desirable characteristics are not necessarily complimentary.

If minimizing rotor speed loss is the prime concern, nose up pitch and left roll seem desirable in forward flight. Nose up pitch results from aft TPP displacement caused by increase advance ratio as rotor speed decreases. This aft tilt generates a component of rotor upwash and some rotor accelerating moments. Additional aft tilt is provided by coning and sideslip. Left roll due to left TPP tilt after generation of right sideslip also gives incremental rotor accelerating moments. Figure 10.14 shows the upwash contributions from tilt and sideslip, and Figure 10.15 illustrates the reduced in plane forces.

In Figure 10.15 the upwash change has realigned the relative wind and tilted the lift (dL) forward, thereby reducing the retarding force (dF) and moment (rdF).

The pitch moment produced tends to slow the helicopter. The reduction in airspeed is favorable when operating at high forward speeds since the aircraft is decelerating toward the optimum steady autorotative airspeed. If the helicopter is operating below the optimum autorotative airspeed, aft pitch may not be desirable.

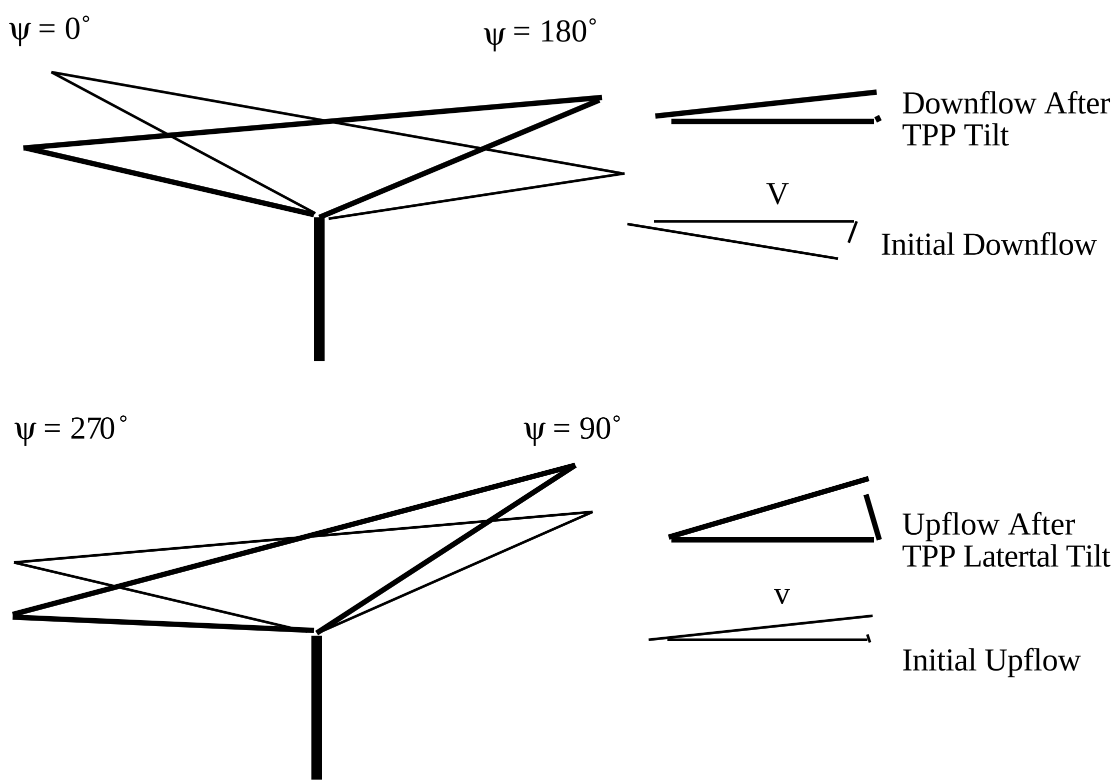


Figure 10.14
Rotor Upwash Due to Aft and Left Tilt

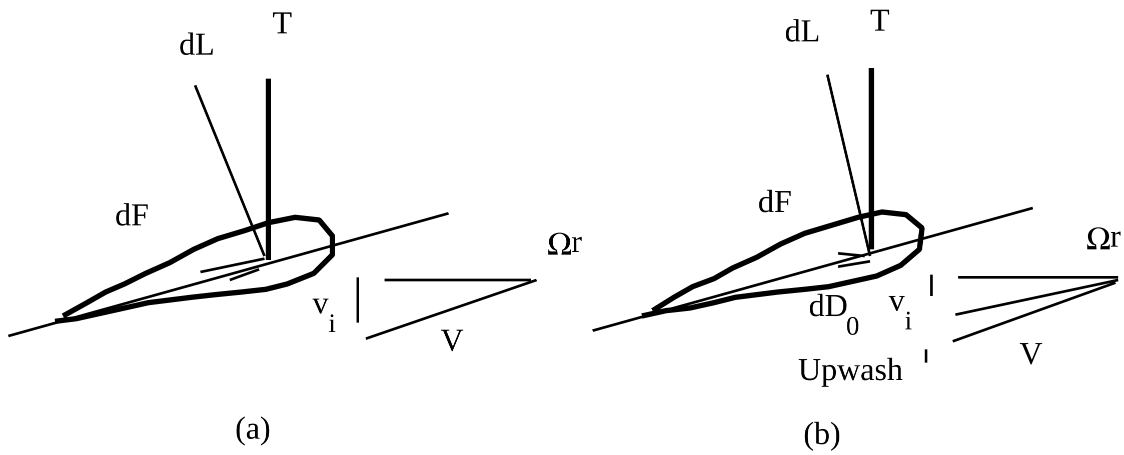


Figure 10.15
Accelerating in Plane Rotor Forces

10.3.1.8 MULTI-ENGINE OPERATION

On multiengine helicopters it is possible to experience partial or total power loss. The effects of partial power loss depend on the initial operating condition. Two cases worthy of consideration are:

1. Limited partial power loss which is less than or equal to the additional power available from the remaining engine(s).
2. Extreme partial power loss which is greater than the additional power available from the remaining engine(s).

In the first situation assuming good rotor speed governing, with an instantaneous power loss, the engine control system senses the rotor speed decay and responds to maintain rotor speed. During the time for the engine(s) to accelerate, rotor torque imbalance and transient rotor speed loss is experienced. Figure 10.16 depicts engine and rotor parameter variations for the limited partial power loss on a two engine helicopter.

Transient yaw, roll, and pitch responses previously discussed are observed, but with the onset of additional torque from the remaining engine(s), the transient responses diminish and the aircraft returns to a quasi steady condition. In the steady condition, sideslip, bank angle, and airspeed disturbances may have occurred exciting long term dynamic responses.

When the power loss is greater than can be absorbed by the remaining engine(s), the engine controls sense rotor speed loss and respond with maximum power available. At maximum power, insufficient power is available to restore full rotor speed. In this situation, there are transient rotor speed losses and a final or steady reduction in rotor speed. Characteristic transient conditions following extreme power loss are shown in Figure 10.17 for a two engine helicopter.

Transient yaw, roll, and pitch responses are observed which are greater than those for the limited partial power loss. As the remaining engine(s) absorb the torque deficits, the transient response diminishes. In the steady condition, trim changes/aircraft responses are proportional to the net torque deficit and the steady loss of rotor speed.

Use of pedal control response data to simulate the anticipated aircraft response following a sudden power loss may not be as useful in evaluating airframe response to partial power loss since additional forces and moments are present as the other engine(s) assume the load.

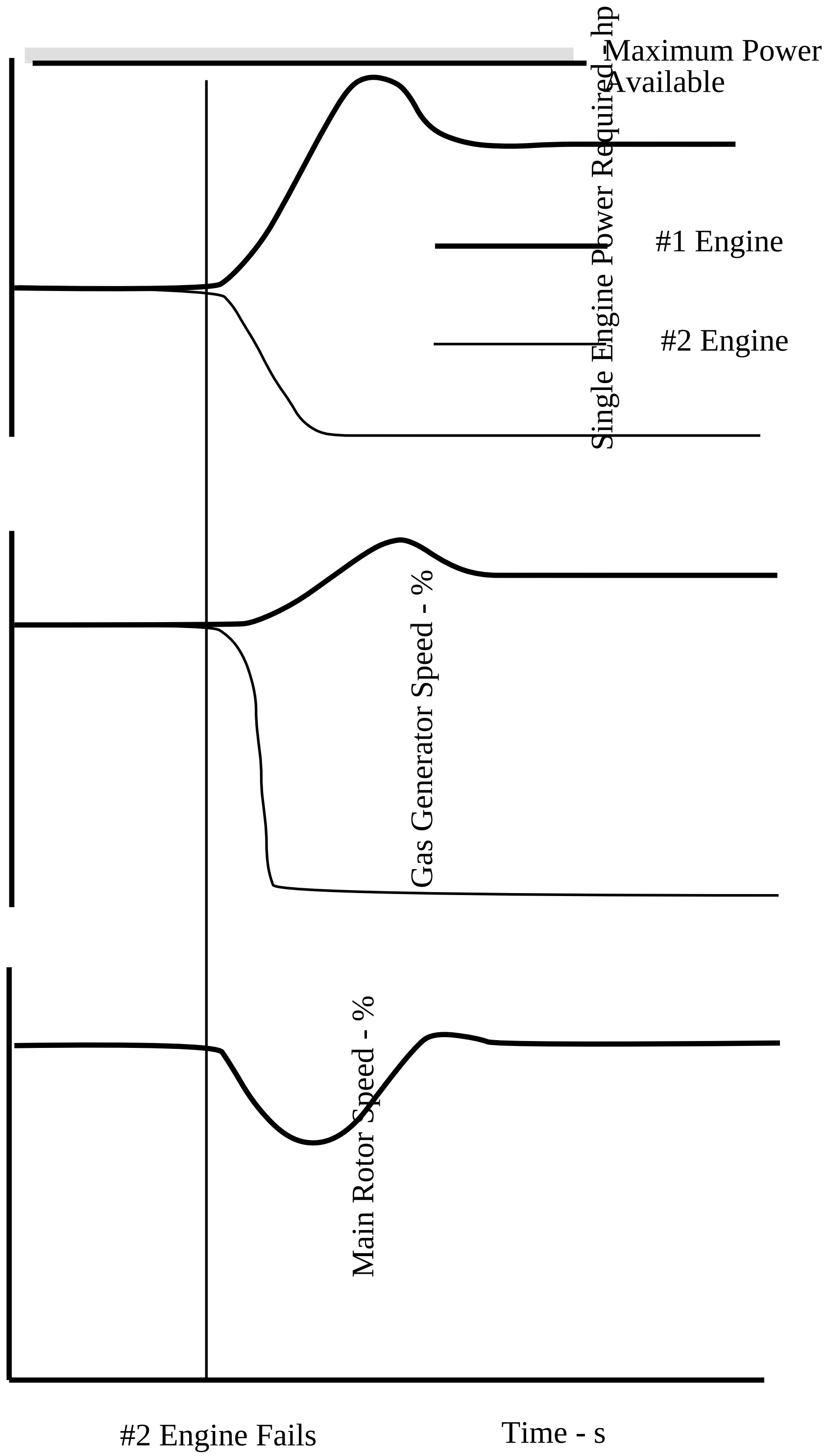


Figure 10.16
Power Train Response to Limited Partial Power Loss

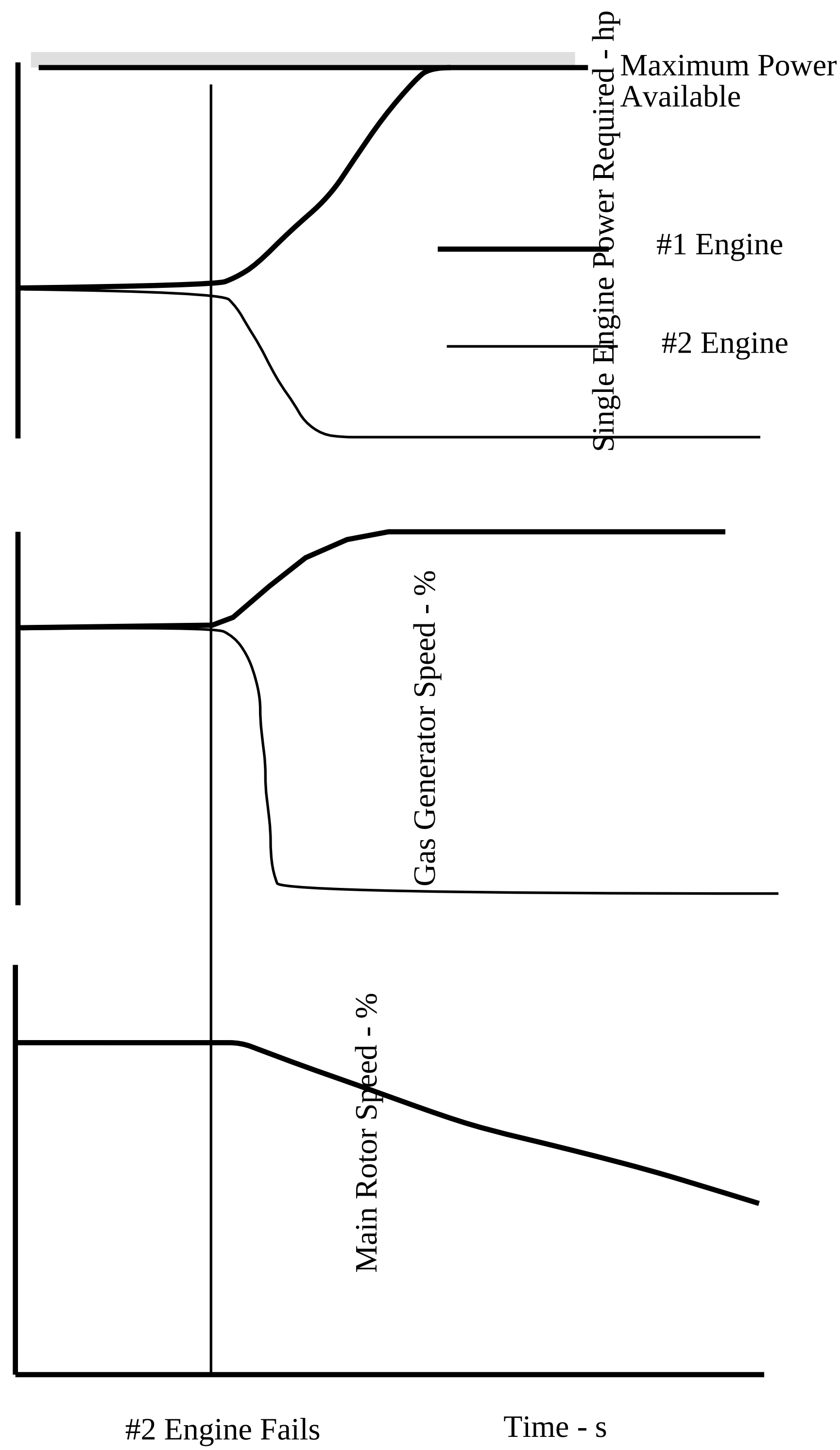


Figure 10.17
Power Train Response to Extreme Partial Power Loss

10.3.1.9 ENGINE FAILURE WARNING

10.3.1.9.1 Natural Warning

Natural warning of an engine failure can be provided by engine noise, rotor speed noise, acceleration, or attitude variations. Engine noise may be a poor cue in a helicopter equipped with turbo-shaft engines due to masking of engine noise by transmission and other noises. If engine noise reduction does not provide adequate warning of engine failure, acceleration/attitude change or the sound of rotor speed decay may provide a warning. Attitude change (except yaw in a hovering single rotor helicopter) may not be a good warning for two reasons. First, Automatic Flight Control Systems (AFCS) tend to limit attitude excursions. Second, the attitude changes due to engine failure are often masked by turbulent atmospheric conditions or, are similar in nature to AFCS or hydraulic hardovers. The natural warning must be unambiguous. If the pilot is required to determine the exact nature of the problem, the warning effectiveness is lost.

Low power descent or decelerating maneuvers make unambiguous natural warning unlikely. The pilot of an aircraft in a low power condition may not know he has a power failure until after he increases collective to terminate the low power maneuver. Additionally, some transient rotor droop is not unusual. The pilot may neglect the natural aural warning due to rotor speed decay until a severe loss in rotor speed is experienced.

10.3.1.9.2 Artificial Warning

Artificial warning devices are installed in most aircraft to improve engine failure warning characteristics but many of them have common deficiencies. These deficiencies include a capability to be deactivated, false activation during ground and flight operations, and warning which is related to a low rotor speed and not a power failure. More sophisticated systems provide different audible tone warnings for engine power malfunctions and low rotor speed indications. A warning device which produces false indications is ineffective.

10.3.2 Steady Autorotation

Following a power failure, attaining and maintaining steady state or quasi steady autorotative flight is a primary goal. The recovery of autorotative rotor speed and descent airspeed, is considered under the category of steady autorotative flight. The pilot attempts to acquire the optimum possible conditions for an autorotative flare and landing during autorotative flight.

Autorotative performance theory provides an understanding of rotor response to variations in collective pitch, cyclic pitch, and inflow velocity as well as the effects of weight, density altitude, and maneuvering on steady state autorotative characteristics.

10.3.2.1 ROTOR EQUILIBRIUM

Rotor equilibrium is obtained when net drive train torque and torque required are balanced. Figure 10.18 presents some simplified diagrams to illustrate a few of the many possible operating states which are applicable to blade elements of an autorotating rotor.

In Figure 10.18 (b), the pro and anti-autorotation torque vectors are equal and the rotor angular velocity is constant. In Figure 10.18 (c) the rotor speed is increased and the inflow velocity constant. This produces an increased blade element speed, increased drag, and decreased angle of attack. The reduced angle of attack causes the blade element lift vector (dL) to tilt toward the trailing edge. Since the anti-torque vector is greater than the pro-torque vector, the rotor decelerates. In Figure 10.18 (d), the blade angle of attack is decreased and the inflow velocity is increased by a reduction of thrust and induced velocity. The result of these changes is rotational acceleration of the rotor. After the rotor in Figure 10.18 (d) stabilizes, the steady state condition resembles Figure 10.18 (e).

The blade elements along a twisted, autorotating rotor blade are all operating under approximately the same inflow conditions (at low forward speeds). The outer elements are operating at the highest tangential velocities and lowest angle of attack. The result of the above conditions is anti-rotational blade element forces over the outer portion of a rotor while the inner portion produces the predominance of pro-autorotational blade element forces.

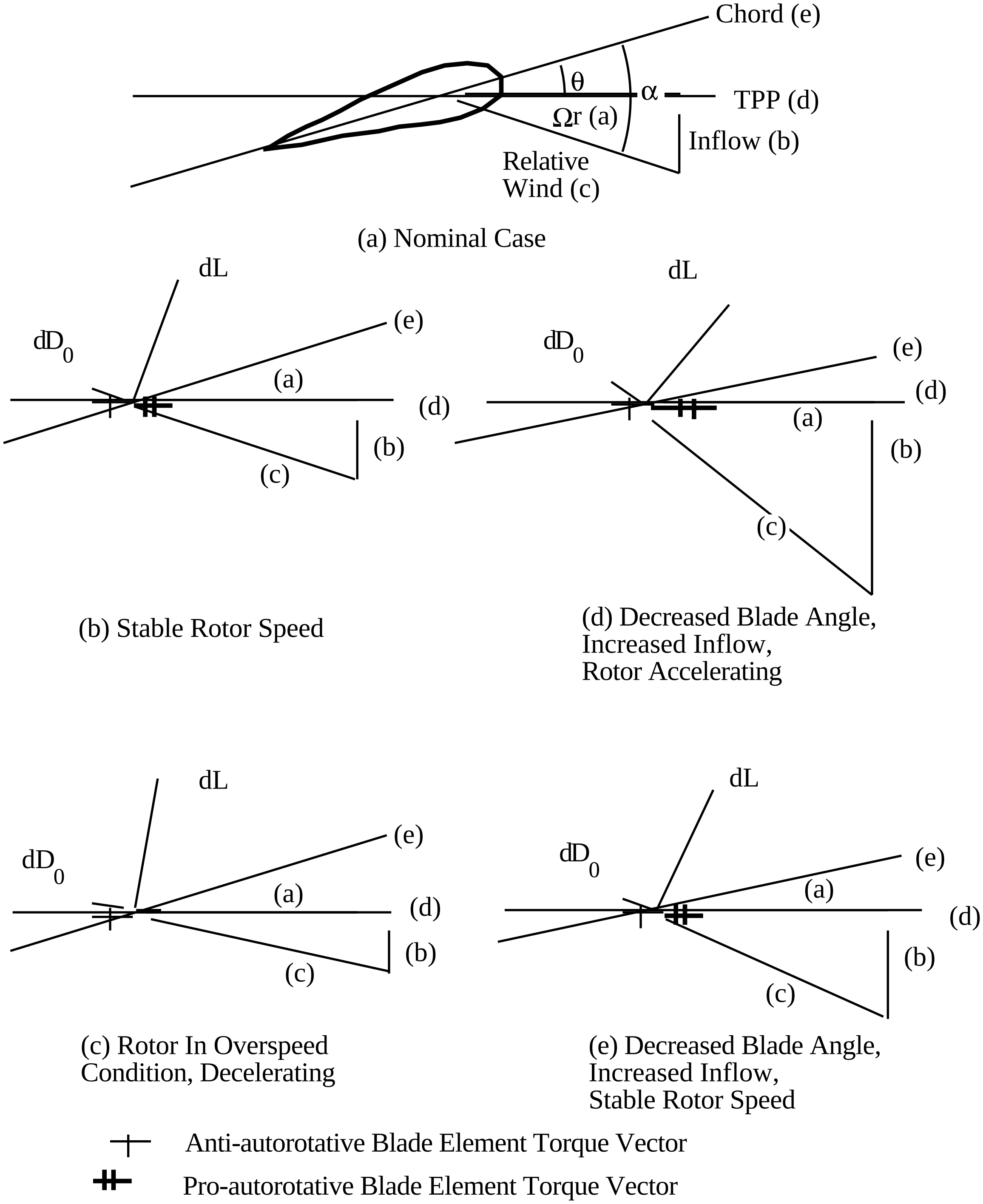


Figure 10.18
Rotor Blade Section Autorotative Equilibrium

10.3.2.2 RATE OF DESCENT

The power to drive the rotor during autorotation comes from the exchange of potential energy (altitude) for kinetic, rotational energy. The power requirements of the rotor in autorotative flight are quite similar to those in powered flight, the rotor produces thrust (induced power) to support the aircraft vertically, and propel it at some forward flight speed (parasite power). There is also the requirement to turn the rotor (profile power).

Since engine power is not available, the autorotative rate of descent is proportional to the deficit in power, or the power required for level flight. Rate of descent is therefore dependent on the items which affect power required: airspeed, density altitude, gross weight, rotor speed, and external configuration. Examination of a power required plot (Figure 10.10) shows that high rates of descent are expected at lower forward airspeeds, rates of descent decrease up to the airspeed for minimum power required, and rates of descent again increase for speeds above the minimum power required airspeed.

Higher rotor speeds usually increase rate of descent since rotor profile power is increased. Increased weight and density altitude appear to increase rate of descent; however, a trade-off must be considered. If it is assumed that the pilot uses collective to maintain a given rotor speed (say 100%) at all weights/altitudes, some relief in rate of descent may be observed. Figure 10.19 illustrates the relief expected.

Performance theory for a blade section in autorotative equilibrium (resultant force (dR) perpendicular to Ωr) is dependent only on the ratio of the two dimensional drag (dD_0) to the two dimensional lift (dL). The dL/dD_0 ratio is dictated only by the two dimensional angle of attack which in Figure 10.19 is θ_C plus inflow angle. The inflow angle is defined by the ratio of inflow velocity to Ωr. All blade sections in Figure 10.19 are in autorotative equilibrium.

Blade elements in Figure 10.19 (a) and (b) are at the same collective position, but the helicopter in (b) is at a heavier GW or higher density altitude. Because of the higher GW/altitude, rate of descent (and inflow) is higher causing the rotor to operate at a greater rotational speed.

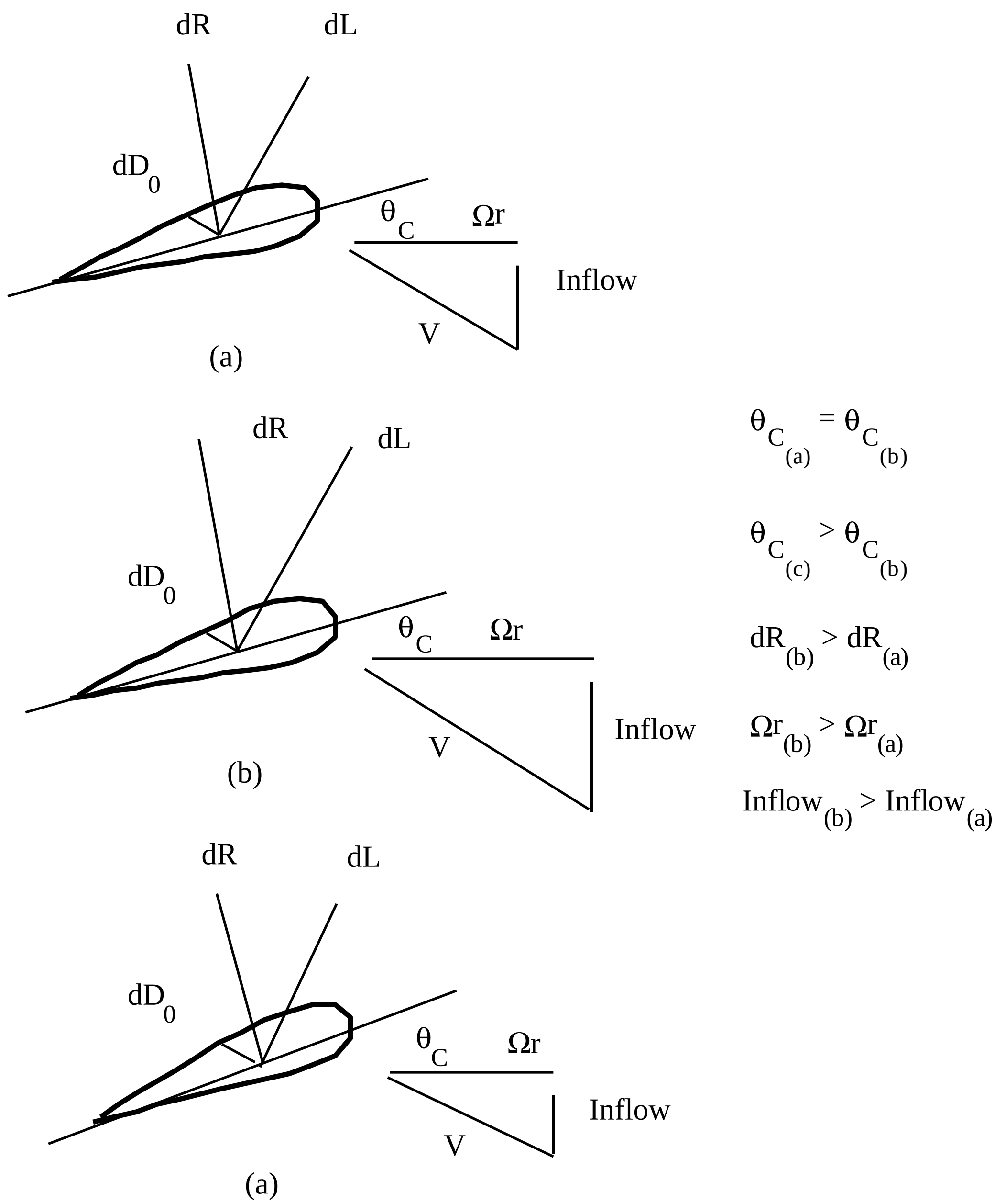


Figure 10.19
Effects of Weight/Density Altitude on Rate of Descent at Constant Rotor Speed

The pilot reacts to the high rotor speed in (b) by increasing collective. The rotor speed is reduced, but the inflow velocity (due largely to rate of descent) must be reduced to achieve the equilibrium dL/dD_0 ratio. Some fine adjustments to this analysis are required to account for variations in induced velocity required.

There is a trade off in weight and rate of descent as the collective is adjusted to maintain rotor speed. In some instances, no appreciable differences in rates of descent are observed with gross weight changes when the descent rotor speed is held constant.

10.3.2.3 MINIMUM COLLECTIVE PITCH

The minimum collective pitch angle which provides satisfactory steady state rotor speed is governed by airspeed, gross weight, and density altitude. An increase in airspeed increases the rotor profile power requirement which decreases the steady state value of rotor speed. An increase in altitude tends to decrease the profile power required resulting in an increase in rotor speed for a given collective. An increase in inflow velocity associated with both high altitude and high gross weight increases blade element angle of attack and the pro-autorotative torque force. The same trends are also evident during evaluations of rotor speed recovery characteristics. Rotor speed recovery at light GW and low density altitude may be slow, while high GW and altitude may cause rapid rotor speed increase or overspeed. The minimum collective for a given aircraft provides satisfactory steady state minimum rotor speed and provides adequate rotor speed recovery characteristics throughout the entire allowable envelope of the aircraft.

10.3.2.4 OPTIMUM AUTOROTATIVE AIRSPEEDS

Two airspeeds of interest for optimizing autorotation are airspeed for minimum rate of descent and maximum glide distance. Since rate of descent in autorotation is proportional to level flight power required, investigation of flight characteristics for minimum rate of descent airspeeds begin near the airspeed for minimum power required.

Performance analysis shows the best glide ratio is obtained at the airspeed where the ratio of descent rate to airspeed is minimized. Since rate of descent is proportional to the deficit in power required, a tangent to the level flight power required curve drawn through the origin as illustrated in Figure 10.20, provides a starting point for investigating flight characteristics at best glide conditions.

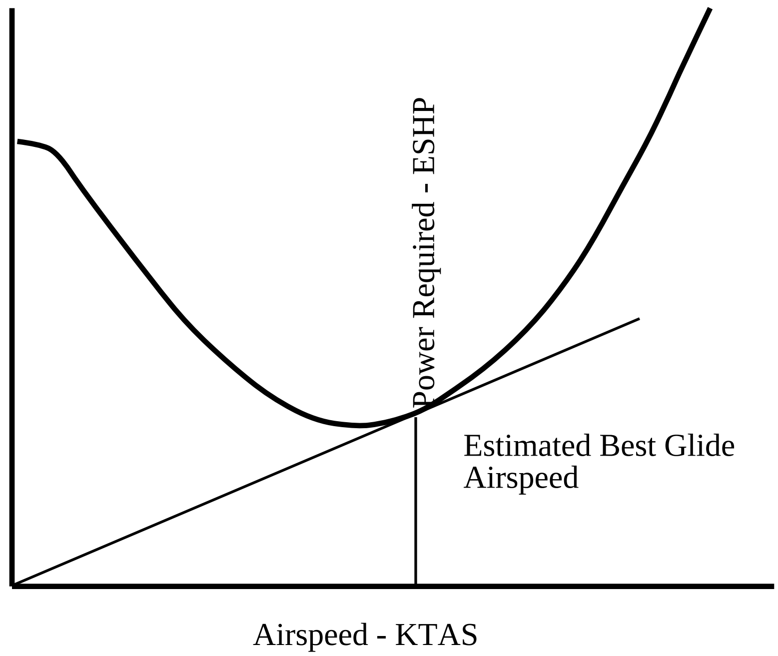


Figure 10.20
Estimation of Airspeed for Best Glide Ratio

10.3.2.5 TRIMMED CONTROL POSITIONS, STATIC, AND DYNAMIC STABILITY

The theory and test techniques for trimmed control positions, static, and dynamic stability presented in Chapters 6, 7, and 8 are equally applicable to the autorotative flight regime.

10.3.2.6 CYCLIC AND COLLECTIVE MANEUVERING

During steady autorotation, a vertical thrust component is required to balance aircraft weight. Rotor longitudinal and lateral TPP position is controlled using cyclic pitch. The variation of longitudinal TPP tilt with cyclic is given by:

$$\frac{\partial a_{1s}}{\partial B_{1s}} = - \frac{\left(1 + \frac{3}{2}\mu^2\right)}{\left(1 - \frac{\mu^2}{2}\right)}$$

eq 10.15

Where:

a_{1s} - Longitudinal flapping angle, shaft referenced

B_{1s} - Longitudinal cyclic pitch angle, shaft referenced

μ - Advance ratio.

With flapping hinge offset, the pitch moments produced by cyclic are:

$$M_{CG} = \left(Th + \frac{ebM_S\Omega^2}{2} \right) a_{1s} \qquad \textit{eq 10.16}$$

Where:

a_{1s} - Longitudinal flapping angle, shaft referenced

b - Number of blades

e - Flapping hinge offset

h - Height of hub above CG

M_{CG} - Pitch moment about CG

M_S - Blade mass moment

T - Thrust

Ω - Rotor angular velocity.

Equations 10.15 and 10.16 indicate the cyclic ability to change TPP position and produce moments in autorotation are the same as in level flight. Tilt of the TPP using cyclic in forward flight produces thrust changes due to inflow (upwash or downwash from free stream) changes as illustrated in Figure 10.14. Generation of thrust changes and pitch moments with cyclic in autorotation and the ability to maneuver are similar to forward level flight.

Inflow changes introduced during forward flight through cyclic inputs result in rotor acceleration for aft control inputs and decelerations for forward inputs because of the blade section angle of attack differences with inflow (Figure 10.18). Collective inputs are used, as required, to control rotor speed changes which occur during cyclic maneuvers.

Collective only maneuvers produce thrust changes (possibly coupling with pitch moments) and rotor speed changes. Collective maneuvers are limited generally to maintaining rotor speed within an acceptable range.

10.3.2.7 ROTOR SPEED CONTROL

Steady state autorotative rotor speed is controlled largely by collective application. Rotor accelerating/decelerating moments depend on collective position and inflow. Rotor acceleration/deceleration is dictated by the size of the in plane moment and the rotor's rotational moment of inertia. Low inertia rotors accelerate rapidly and high inertia rotors accelerate slowly for a given in plane moment.

When rotor speed recovery is extremely rapid the pilot may experience a rotor speed control problem. The pilot may be required to manage rotor speed in deference to other emergency procedures. This problem may be aggravated at high density altitude, high GW, and following a severe loss of rotor speed which precipitates a high rate of descent. This excess rate of descent may tend to induce rotor overspeed requiring additional pilot manipulation of the collective. Precise, small, continuous collective inputs may be required to manage rotor speed for low inertia rotors.

When rotor speed recovery is sluggish, the pilot may have difficulty determining how much collective input is required to effect a final rotor speed change. If severe rotor speed loss is encountered, electrical systems, AFCS, and conceivably hydraulic systems may be lost. If the pilot is distracted with controlling attitude or reengaging systems (such as AFCS), an overspeed may occur. The high inertia rotor system is basically a lightly damped first order system which may require large pilot inputs to achieve desired rotor acceleration, followed by periodic pilot inputs to modify the acceleration and some collective leading to establish the final rotor speed.

10.3.3 Transition To Autorotation Following Power Loss

There is a finite period of time between a power failure and pilot response referred to as pilot reaction time. The magnitude of pilot reaction time varies with circumstances, as does the maximum allowable delay time which can be tolerated. The acceptability of either pilot reaction time or the available delay time cannot be determined independently, they must be considered collectively. The predictability and complexity of the recovery control inputs is an important aspect in both reaction time and delay time and can be a limiting factor. An elementary evaluation of pilot response to a total power failure is accomplished readily when the elements of the above factors are considered separately with continual awareness of their interdependency.

10.3.3.1 AVAILABLE DELAY AND PILOT REACTION TIMES

A total power failure with controls fixed produces a rotor speed decay resulting in a loss of thrust as well as angular accelerations about all axes and linear acceleration along all axes. When control inputs are made by the pilot to enter an autorotation, the result is a reversal in rotor speed decay and possibly a decrease in normal acceleration. The length of time the controls can remain fixed during a transition is equal to the time required for divergence of one or more characteristics as follows:

1. Minimum tolerable rotor speed.
2. Attitude or angular rate or combination of attitude and angular rate which does not afford 100% probability for a successful recovery.
3. Conditions that do not allow the pilot to move the flight controls as rapidly as required without producing acceleration or blade flapping exceeding structural limits.

When operations are conducted in close proximity to the ground an additional consideration for defining divergence is in order. It is possible to make a safe transition into an autorotative flight condition which does not provide the combinations of rotor speed, airspeed, and rate of descent which allows a sufficient margin for a safe landing. In such cases a modification to the definition of divergence is required to insure the minimum performance margin required for the degree of landing safety desired. For example, the minimum rotor speed for an entry at 100 ft Above Ground Level (AGL) and 10 kn may be higher than the minimum rotor speed for the same entry at 1000 ft AGL. Problems related to retreating blade stall, structural interference, etc. are analyzed in the same fashion as discussed above for ground clearance. Divergence with respect to structural integrity should be predicted by the manufacturer, and realistic margins added for safety.

Pilot reaction time varies from pilot to pilot; but as a rule, the variations are minimal. The reaction times of interest are those following pilot sensation of aural, visual, vestibular, and proprioceptor cues resulting from engine failure. The response time of a pilot, is a function of a certain number of defined delays or lags. The elements of pilot response time are receptor delays, transmission delays, central process delays, muscle latency, and activation time.

Studies indicate that when all factors including initial aircraft responses (motion, aural warning, etc.) are considered, the minimum probable pilot reaction time to a sudden unexpected power loss is about one second.

Receptor delays are a function of the operator's vigilance, the type of stimulation, and the intensity and quality of the sensation. Reaction times are shortest for auditory cues, with proprioceptory cues next, and visual cues producing the longest reaction times. Transmission delays involve signal passage to/from the brain. The central process delays involve detection, recognition, discrimination, identification, thinking, planning, and the decision making processes. When there is more than one stimulus, each of which requires a particular, correct response, there is usually a significant increase in the central process delay time. The muscle latency and activation time refers to the time required to obtain muscle action once the transmission of the command from the brain is complete.

The helicopter's mission may dictate the requirement for an extremely effective engine failure warning characteristic and/or an extremely long available delay time. Pilots are often required to concentrate on flight instruments, fire control systems, or other special and demanding mission tasks. The relatively simple tasks of visual flight rules (VFR) formation flying and terrain avoidance requires pilot attention outside the cockpit and also tends to reduce pilot awareness of power failure cues. When a pilot is involved in a special mission task, a finite period of time is required to switch from the mission task to the control task required for transition into autorotative flight.

Many helicopters have altitude and attitude hold features. Some have a collective control hold system. These systems promote hands off flight. When engine failure is recognized during hands off flight, a finite period of time is required to move hands and feet back to the flight controls.

10.3.3.2 CONTROLS FIXED NATURAL RESPONSE

To some extent, the natural helicopter response to power loss is in the favorable direction for establishing steady autorotation. Loss of thrust associated with rotor speed decay provides an acceleration downward which eventually produces rate of descent which is required for steady autorotation. Additionally, in forward flight, pitch up tendencies and right sideslips (left TPP tilt), with resulting inflow changes, decrease rotor speed decay.

10.3.3.3 RESPONSE TO COLLECTIVE INPUT

The collective is usually the primary control for autorotative entry. A few noteworthy exceptions include sudden power losses at low wheel heights and sudden power losses at high tip speed ratios. These exceptions involve the use of longitudinal cyclic and are related to limited areas of operation discussed later.

10.3.3.3.1 Hover

When the collective is lowered in hovering flight there is an immediate decrease in normal acceleration as the direct result of decreased thrust. The decreased collective pitch significantly decreases the power required to turn the rotor, hence a decrease in the power deficit. The decreased power deficit diminishes rotor speed decay and allows the rotor speed to stabilize and increase as the flow up through the rotor establishes a predominance of pro-autorotative blade element torque. When the collective is decreased, after a hovering engine failure, the rate of collective motion, in part, determines the rotor speed lost during transition. The faster the motion to full down, the higher the minimum rotor speed and in some cases the higher the recovered steady state value. Slow decreases in collective pitch provide a smoother entry and reduce vertical velocity obtained during the first few seconds. The large loss in rotor speed, associated with slow rates of collective reduction can be regained and may not be significant unless there is insufficient altitude remaining.

Downward collective movement combined with reduced rotor speed decreases cyclic control effectiveness. Equation 10.16 shows pitch moments generated by TPP movement are reduced as both rotor speed and thrust are reduced. Likewise the ability to produce roll moments is decreased.

10.3.3.3.2 Forward Flight

In forward flight, the rate and magnitude of collective movement becomes more significant since control effectiveness is reduced. Rotor speed decay may not be arrested and blade flapping limits may be encountered. When the collective is lowered in forward flight the TPP tilts forward or flaps down at $\psi = 180^\circ$. After the TPP flaps forward because of collective reduction, it tilts further forward in response to its inherent angle of attack instability. The total effect of lowering the collective is a nose down pitch moment, increased flow down through the rotor, a subsequent reduction in normal acceleration, and continual loss of rotor speed. In addition, the forward flapping of the TPP can present a structural clearance problem in terms of droop stop clearance or mast strikes for semirigid rotors.

10.3.3.4 RESPONSE TO LONGITUDINAL CYCLIC INPUT

10.3.3.4.1 Hover

Significant longitudinal control inputs following power loss are required usually to accelerate to a more favorable airspeed and rate of descent combination prior to initiation of the landing flare. Cyclic effectiveness to produce pitch moments is reduced because of rotor speed decay and thrust losses.

10.3.3.4.2 Forward Flight

Forward longitudinal cyclic control input in forward flight cause the TPP to flap forward driving the rotor angle of attack more negative. This forward tilt of the TPP increases the flow down through the rotor resulting in a decrease in rotor pro-autorotative blade element torque. The down flow increases the forward flapping due to the rotor's inherent angle of attack instability. The net effect of forward cyclic is a nose down pitch moment, increased flow down through the rotor, a reduction in normal acceleration, continued loss of rotor speed, and an increase in rate of descent. The forward flapping with increased down flow significantly increases at higher airspeeds. If a pilot over controls the aircraft, significant time can elapse between full down collective and attaining satisfactory autorotative rotor speed. The low rotor speed and large potential dive angle combine to exaggerate the rapid increase in rate of descent.

The flapping response to control inputs and down flow changes has the potential to produce droop stop or mast strikes. The TPP to fuselage clearance margins for forward control inputs is reduced as the CG moves aft.

At higher forward airspeeds, an initial aft cyclic input may be required following sudden power loss. Aft control inputs provide a rotor up flow and pro-autorotative rotor torque moments, increase load factor, improve cyclic control effectiveness, and reduce concern for droop stop pounding/mast strikes.

10.3.3.5 CONTROL REVERSAL

Equations developed in Chapter 4 for rotor contributions to roll and pitch moments with flapping hinge offset are:

$$M_{CG} = \left(Th + \frac{ebM_S\Omega^2}{2} \right) a_{1s} \qquad \textit{eq 10.16}$$

$$L_{CG} = \left(Th + \frac{ebM_S\Omega^2}{2} \right) b_{1s} \qquad \textit{eq 10.17}$$

Where:

a_{1s}	- Longitudinal flapping angle, shaft referenced
b	- Number of blades
b_{1s}	- Lateral flapping angle, shaft referenced
e	- Flapping hinge offset
h	- Height of hub above CG
L_{CG}	- Roll moment about CG
M_{CG}	- Pitch moment about CG
M_S	- Blade mass moment
T	- Thrust
Ω	- Rotor angular velocity.

These equations indicate that moments generated by a_{1s} and b_{1s} are dependent on thrust and rotor speed. When thrust and/or rotor speed are significantly reduced, there is a reduction in the moment generated by a given TPP tilt. If the rotor thrust approaches zero (zero load factor), moments are produced by only the hub moment term ($ebM_S\ \Omega^2/2$). If negative thrust is encountered (load factor less than zero) the thrust and hub moment terms oppose each other until at some negative load factor, no moments are produced. Further reductions in load factor (for a given Ω) causes the sign of the moments to reverse and the helicopter experiences control reversal. Control reversal implies that forward control inputs produce nose up moments and right control input yield left roll moments. Figure 10.21 illustrates this concept. Figure 10.21 (a) shows the helicopter in 1.0 g flight.

Figure 10.21 (b) through (e) show reduction of thrust until at (e) a large negative thrust exists. For simplification, longitudinal TPP position and rotor speed are maintained constant. As thrust decreases the total rotor moment available is reduced (but still has the proper sign) until at (e) the thrust moment and hub moment are opposite, and equal and no moments are therefore available. In (e) the thrust moment is greater than the hub moment. If the pilot moves the longitudinal control at (e) TPP movement is in the proper sense (aft tilt for aft input), but the thrust moment dominates. In case (e), an aft control input tilts the TPP aft, producing a nose up hub moment and a nose down thrust moment, which is stronger than the hub moment. The result is a net nose down pitch moment for an aft control input, or a reversal in the control sense (control reversal).

If the helicopter is not equipped with flapping hinge offset or flapping springs, control reversal can be expected as soon as the thrust (or load factor) goes negative. Similarly, the same conditions results in lateral control reversal.

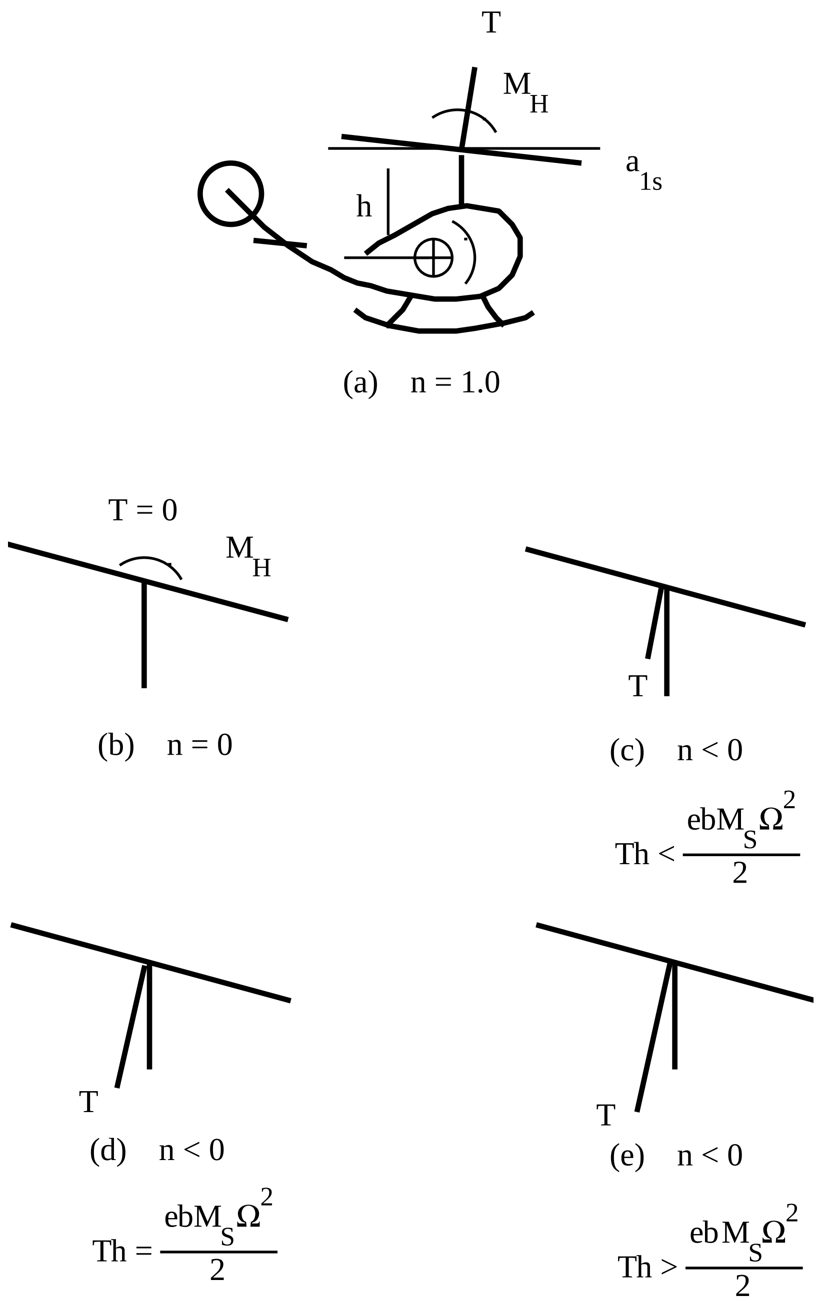


Figure 10.21
Longitudinal Cyclic Control Reversal

If rotor speed decays significantly, the ability to produce hub moments is reduced and control reversal occurs at smaller values of negative thrust or negative load factor.

Cautious use of down collective and forward cyclic control inputs during forward flight power loss tests is imperative (particularly at higher airspeeds), to prevent aggravated loss of rotor speed, excessive forward flapping, excessive nose down pitch attitudes/descent rates and control reversal.

10.3.3.6 RESPONSE TO LATERAL CYCLIC INPUT

It is extremely difficult to predict the lateral cyclic inputs required during entry into autorotative flight. The most probable input is to the right, as this tends to counter the predominate roll away from the right sideslip (experienced during power loss in a single rotor helicopter). One interesting aspect of lateral inputs is the effect these inputs have on aircraft response when the collective motion lags the lateral control displacement. Lateral inputs used to keep the aircraft wings level as a sideslip angle develops result in approximately a steady heading sideslip. When the response of the aircraft to rapid down collective or forward control motion is compared for steady heading balanced flight and steady heading sideslip flight, the sideslip entry may exhibit a more violent characteristic. When down collective or forward control inputs are made rapidly there is a reduction (or reversal) in lateral control effectiveness. In addition, the TPP tilts left because of the reduced coning (low thrust). The relatively small lateral aircraft moment of inertia may contribute to the severity of the situation allowing comparatively large roll angular acceleration to develop.

10.3.3.7 RESPONSE TO DIRECTIONAL CONTROL INPUT

Directional control inputs are very important during recovery from sudden power loss. The application of right pedal reduces the tail rotor thrust. This reduction in thrust is accompanied by an immediate reduction in tail rotor torque. The reduction in tail rotor torque effectively reduces the total power train torque deficiency. For extreme right sideslips the tail rotor itself may be approaching an autorotative state and provide some initial torque to the system as the right pedal is applied.

A second major effect of corrective directional control inputs is to reduce sideslip angles both before and after the collective is lowered. The rapid or timely application of the proper control greatly reduces the possibility of divergence in yaw and roll.

One potential adverse effect of right pedal inputs is the incremental left roll moment produced for a helicopter configured with a high tail rotor.

10.3.3.8 RESPONSE IN REARWARD FLIGHT

An engine failure during rearward flight can result in a very difficult situation from a lateral and directional controllability aspect. Such an engine failure in a single rotor helicopter usually results in a rapid yaw to the left, due first to the decrease in main rotor torque, then later due to directional stability. If a substantial yaw rate develops before there is a pilot response, right yaw control may be insufficient to prevent a rapid 180˚ turn to the left. A rapid turn to the left during rearward flight also results in an almost instantaneous introduction of a lateral wind component at the left side of the rotor. A rotor which exhibits high positive dihedral effect produces a rapid roll away from the lateral velocity. The roll due to sideslip can be reduced by anticipation of the problem and roll into the direction of yaw.

10.3.3.9 CONTROL INPUTS AT LOW LOAD FACTORS

Control inputs at low load factors may be introduced by the pilot or by the AFCS. At low load factors the cyclic control effectiveness is reduced. For single rotor helicopters this loss in effectiveness is felt in pitch and roll, and for the tandem configurations, in roll and yaw. When the pilot or AFCS makes inputs at reduced load factor with little or no aircraft response, the tendency is to make larger inputs. If the rotor thrust is then restored rapidly, the effects of these control inputs can be excessive. Cautious use of flight controls in low g flight is mandatory.

10.3.4 Autorotative Landing

A successful autorotative landing is only possible after an entry into some form of autorotative, descending flight. The definition of a successful landing is generally established by the procuring agency, while the test activity is left to determine what limits are imposed on operations to assure such landings. The success of an autorotative flare and landing are strongly dependent upon energy transfers, rotor performance characteristics, and landing gear design. The nature of the parameters being tested denotes a performance type test is in order; but, the emergency nature of the entire task and the dynamic aspects of maneuvers, dictates that stability and control tests must be conducted. A clever approach is to conduct a stability and control type test while paying strict attention to performance.

There are a number of very important performance characteristics which must be monitored and understood. These characteristics are related to the power required for level flight, the kinetic energy of the rotor, the kinetic energy of the aircraft, the rate of descent, and the potential energy of the aircraft at the instant the landing flare is commenced. The ability to decrease the horizontal and vertical velocities is related to the rotor energy and the thrust produced by the rotor during energy transfers. The energy not absorbed and used by the rotor during the landing phase is absorbed by the landing gear.

Touch down conditions required in a successful landing reflect the landing surface characteristics expected. Different airspeed/rate of descent combinations are allowable when landing on a smooth prepared surface rather than a plowed field.

Landings are considered from several possible flight conditions as follows:

1. Steady autorotative descent.
2. Low above ground level (AGL) altitude hovers.
3. High AGL altitude hovers.
4. Intermediate AGL altitude flight at low airspeed.
5. Low AGL altitude flight with low to high airspeeds.

In situations 2 through 5, it is probable that no steady autorotative flight condition exists prior to landing; rather, the pilot must use the helicopter and rotor energy available to him to execute the successful landing. Situations 2 through 5 give rise to the concept of the Height-Velocity Diagram (H-V diagram) sometimes referred to as the Dead Man's Curve.

10.3.4.1 LANDING FROM STEADY AUTOROTATIVE DESCENT

Landing from steady autorotative descent usually involves a flare phase to retard rate of descent and airspeed. The flare is executed usually with cyclic and some possible collective adjustments to control rotor speed and descent rate. Rotor speed usually increases during the cyclic flare with rotor speed increase dependent on the rate of flare and the airspeed at flare commencement.

10.3.4.1.1 Rotor Speed Increase During Flare

The origin of rotor speed increases during an autorotative flare is illustrated in Figure 10.22. In Figure 10.22, the aircraft is descending at essentially constant conditions when the pilot makes an aft cyclic input.

The cyclic input tilts the TPP aft thereby increasing the rotor angle of attack and thrust and providing nose up pitch moments. After the helicopter pitches to some predetermined attitude, the pilot adjusts cyclic pitch to maintain this attitude, while ground speed and rate of descent decrease due to the increased thrust. In the flared attitude, the rotor senses increased angle of attack (or upwash) equivalent to an increased rate of descent. Increased rate of descent provides pro-autorotative rotor moments or rotor acceleration torque.

If the flare rate is high, ballooning (climbing) may result, and down collective may be required to control ballooning. High flare rates or high pitch attitudes may also lead to rotor overspeeding which must be controlled with collective. Optimization of flare technique may be required to keep all parameters within satisfactory bounds.

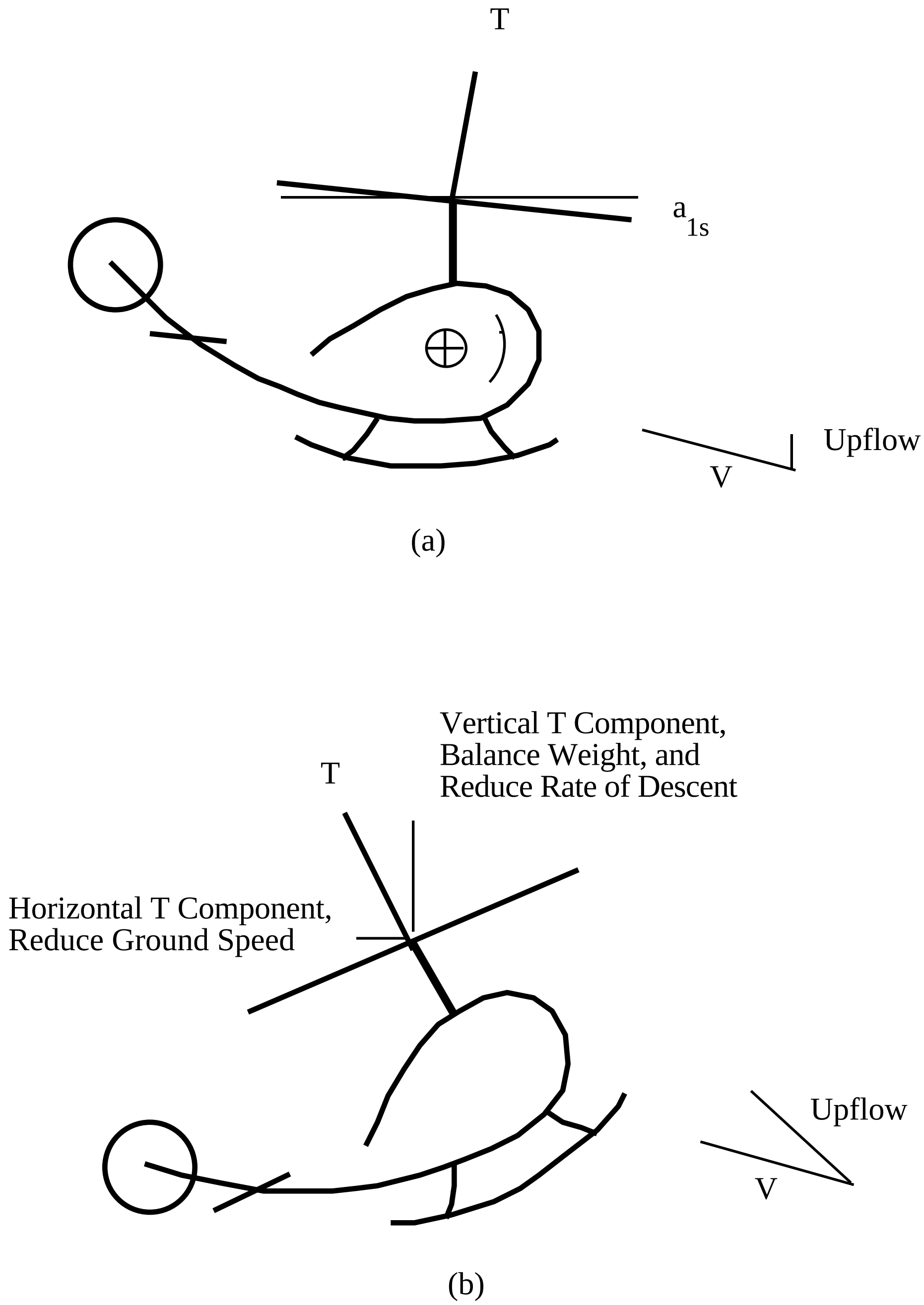


Figure 10.22
Rotor Acceleration During Cyclic Flare

10.3.4.1.2 Landing

Aircraft attitude is adjusted between the flare phase and touch down to obtain the required/desired touchdown attitude. The required touchdown attitude may be nose up (tail wheel first landing for aircraft so configured) to near level for three point landing or for landing skid configured helicopters. Landing attitudes involve structural considerations (tail wheel/tail boom loads), ability to control pitch rate after a nose high touchdown, and uncontrolled TPP flapping (structural or ground interference) associated with high rates of descent and low rotor speeds at touchdown.

Adjustments to aircraft attitude in preparation for landing are essentially cyclic maneuvers. When the flare is terminated rotor speed may be high (desirable), but the accelerating moments achieved during the flare are lost or reduced and rotor speed decays. The available energy stored in the rotor is dissipated using collective pitch inputs to cushion the landing and obtain acceptable touchdown descent rates. The use of increased collective naturally increases the rotor speed decay rate.

10.3.4.1.3 Full Flare Landing

The full flare autorotative landing is the most difficult to perform. The term full flare refers to a landing where airspeed and ground speed are zero on touchdown. The conditions are achieved by conducting a continuous decelerating flare from some higher airspeed. The degree of difficulty actually experienced during this maneuver is dependent upon many obvious factors such as atmosphere, rate of descent, aircraft size, field of view, etc. The pilot may be required to modify the control technique because of aircraft structural limits. Some considerations are fear of tail rotor to ground contact, main rotor to structure interference, landing gear failure or upset due to lateral drift. The ability to land tail wheel first generally eases the demand for pilot proficiency, while any consideration which requires a near level landing tends to increase the degree of difficulty.

The full flare landing has greater practical application than any other type of landing. This landing is used to obtain the best results on touchdown over the vast majority of surfaces available; water, rough terrain, marsh land, and wooded areas. The goal is to touchdown at a zero ground speed to prevent upset.

10.3.4.1.4 Modified Full Flare Landing

A full flare landing is not always required to achieve the desired goal of a zero ground speed touchdown. Winds of 5 to 10 kn are representative of the minimum surface winds on any given day. In addition, ground speeds of up to about 5 kn (sum of lateral and longitudinal components) provide touchdown safety comparable to that attained with zero ground speed. A modified full flare landing is then defined as a full flare type landing where the touchdown ground speed does not exceed 5 kn and the touchdown calibrated airspeed does not exceed 10 kn.

10.3.4.1.5 Vertical Landing

A vertical landing is required usually when power is lost in a hover. Like the full flare landing it alludes to zero wind and zero ground speed conditions. A landing of this type is accomplished by the use of collective only, to cushion landing impact. This type landing also has a modified category which refers to ground contact when either the values of horizontal drift (ground speed) or calibrated airspeeds exceed about 5 kn. The overall success of vertical landing lies in pilot skill and the structural strength of the landing gear.

10.3.4.1.6 Run-On Landing

The run-on landing is one where there is either no requirement or capability to attain a near zero ground speed. Touchdown is achieved at calibrated airspeeds greater than 10 kn and/or ground speeds in excess of 5 kn. There are three major variations to this landing; cyclic only flare, collective only flare and coordinated cyclic-collective flare. The coordinated cyclic-collective maneuver is the most universally used in autorotative landing techniques and may result in either a tail first or a flat attitude touchdown. The cyclic only maneuver requires the tail wheel or tail skag to touchdown first and sustain some of the aircraft weight while the main mounts fall through as thrust is lost on the main rotor. The collective only run-on landing is required when a flare is either not possible or not practical (low altitude). The cyclic is used to attain and maintain the desired landing attitude but the collective is the only control used to reduce sink rate and select the landing point.

Run-on landings, as a rule, require the least pilot skill to accomplish a satisfactory touchdown attitude and impact velocity. The success of the landing is dependent upon the landing surface, landing gear configuration and pilot skill in controlling the aircraft through the ground run out. Two of the problems expected during a water landing are tucking and loss of directional control. A soft surface such as mud or soft sod may produce enough sliding friction to separate the landing gear from the aircraft. If the run-on ground speed is sufficiently high and initial touchdown is on one main mount or end of a skid, a dynamic pitch-roll-yaw bounce may ensue. This type of gyration may turn an otherwise routine landing into one with damage.

Helicopters which may be configured with different landing gear depending on the type of terrain or landing surface in use should be tested with each gear configuration and landing surface combination. An example is a helicopter configured with either skid gear for use on land or float gear for use on land and water. For a helicopter with different external configurations, not only must the autorotative landings be evaluated, the transition into autorotative flight and the steady state autorotation must be evaluated for each configuration.

The effect of gear type on the run -on landing must be evaluated. Additionally use of wheel brakes, collective lowering for braking, and control margins during the run-on landing must be evaluated. Pedal control and margins are especially important during the run-on landing. The helicopter must be aligned in the direction of run-out during the landing to reduce the change of roll over.

10.3.4.1.7 Blade Stall

The potential for blade stall during the autorotative flare and landing phases is a concern. Stall is related to blade section angle of attack which is a combination of collective input, rotor speed, airspeed, flapping, and inflow (upwash) contributions. High upwash values in the flare, aft TPP tilt (downward flapping on retreating blade) and reduced total velocity on the retreating blade during forward flight flares are all conducive to stall production. High collective positions, low rotor speeds and high descent rates can induce stall in the landing. Stall results in thrust losses, reduced cyclic effectiveness to produce moments, increased downward acceleration and aggravated rotor speed decay associated with increased blade drag coefficients.

10.3.4.2 HEIGHT-VELOCITY DIAGRAM

The H-V diagram serves to define those conditions for which a successful autorotative landing is not possible. Operation inside the H-V diagram boundaries should be avoided or minimized based on operational requirements. Figure 10.23 represents a generic H-V diagram. Points A and B represent conditions 2 and 3, respectively, of section 10.3.4.1, and ranges A-C-B and D-E-F, conditions 4 and 5, respectively. Table 10.I defines the boundaries of Figure 10.23.

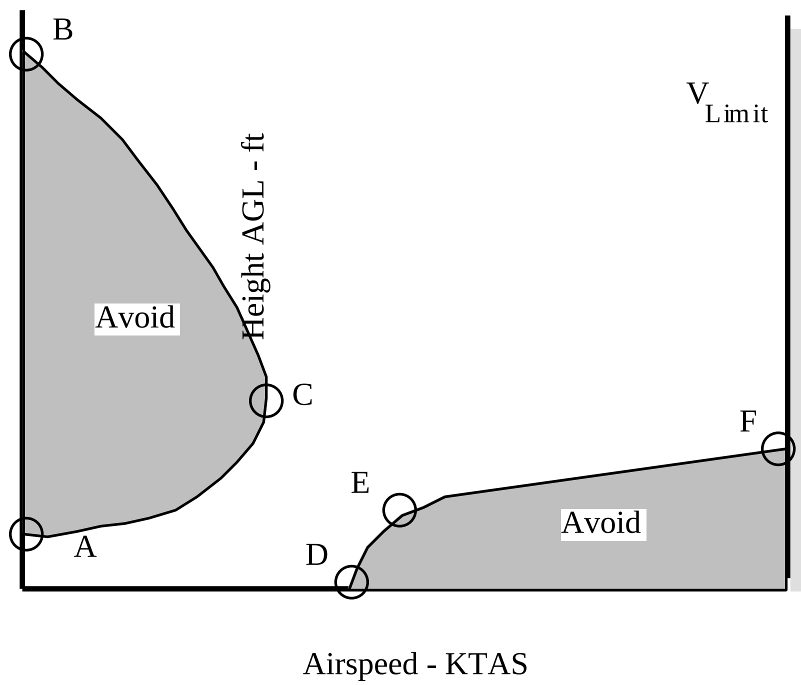


Figure 10.23
Generic Height-Velocity Diagram

Table 10.I
Height-Velocity Diagram Descriptions

POINT	DEFINING PHRASE
A	Maximum height for low hover (ft AGL)
B	Minimum height for high hover (ft AGL)
C	Knee, low speed regime
D	Maximum allowable touchdown speed (Ground Speed)
E	Knee, high speed regime (KIAS)
F	Minimum height at V_H or V_{max} (ft AGL)
A-B-C	Low speed regime
B-C	Upper boundary, low speed regime
A-C	Lower boundary, low speed regime
D-E-F	High speed, low height regime
D-E	Initial increase in high speed, low height regime

The contractor usually furnishes the first H-V diagram for a new aircraft and is required usually to perform at least a minimum demonstration of its validity. The demonstration is conducted at mission and maximum mission weights, typically using a 2 s delay. Specification MIL-D-23222A (AS) (Reference 10.5) describes required demonstration points. In addition to a demonstration of landings following simulated engine failures, the contractor may be required to demonstrate high forward speed, low forward speed, and high sink rate landings from practice entries. The landing surface for landing is generally dictated by contractual agreement.

10.3.4.2.1 Maximum Height for Low Hover

The least complex autorotative flare and landing is executed from a low altitude hover using collective pitch only. If the thrust of the rotor is ignored (thrust equal zero, free fall) and the aircraft allowed to fall from an initial static condition through a distance of h ft the aircraft would strike the ground at a predictable velocity given by:

$$\dot{h}_{TD} = \sqrt{2gh_{AGL}} \qquad \text{eq 10.18}$$

Where:

g	- Gravity
h_{AGL}	- Height above ground level
$\dot{h}_{TD}$	- Vertical velocity at touchdown.

By specifying the limit sink speed ($\dot{h}_{TD}$), the free fall limit height is calculated. For example, if the design limit sink speed were 15 ft/s (900 fpm), the limit height is approximately 3.5 ft. This estimation of the maximum height for a vertical landing is extremely conservative since it does not account for rotor thrust being produced.

Rotor thrust effects are examined by considering the collective fixed autorotation as rotor speed decays. Lowering the collective following power loss may not always be desirable since thrust reduction with collective input may lead to excessive rates of descent at touchdown, even though collective reduction lowers the rotor speed decay rate. The fixed collective setting is maintained until limit sink speed ($\dot{h}_{TD}$) is obtained. The pilot then increases collective rapidly to achieve thrust equal weight (zero vertical acceleration) and continues to smoothly increase collective (maintaining thrust equal weight) until touchdown is achieved just prior to reaching the collective up limit or rotor stall. For this model, rotor speed bleeds off according to:

$$\frac{d\Omega}{dt} = -\frac{1}{I_R}\left(\frac{\Omega^2}{\Omega_0^2}\right)Q_0$$

eq 10.19

Where:

I_R	- Rotor rotational moment of inertia
Q_0	- Initial torque
t	- Time
Ω	- Rotor angular velocity
Ω_0	- Initial rotor angular velocity.

Because rotor speed decays, the thrust is given by:

$$\frac{T}{T_0} = \frac{\Omega^2}{\Omega_0^2} \qquad \textit{eq 10.20}$$

Where:

T	- Thrust
T_0	- Initial thrust
Ω	- Rotor angular velocity
Ω_0	- Initial rotor angular velocity.

The concept of Equation 10.20 was developed in section 10.3.1.1. The vertical acceleration is obtained from:

$$\ddot{z} = \ddot{h} = \frac{W - T}{m} = g\left(1 - \frac{T}{W}\right) \qquad \textit{eq 10.21}$$

Where:

g	- Gravity
$\ddot{h}$	- Vertical acceleration
m	- Mass
T	- Thrust
W	- Weight
$\ddot{z}$	- Acceleration along z axis.

Since the vertical acceleration is dependent on the thrust to weight ratio (where $W = T_0$) Equations 10.20 and 10.21 are combined and integrated with time to obtain vertical speed. If the vertical speed obtained is the limit value $\dot{h}_{TD}$, then the altitude lost in accelerating to $\dot{h}_{TD}$ is determined by a second integration. The descent rate and altitude lost expressions are:

$$\dot{h} = gt^2\left(\frac{Q_0}{I_R\Omega_0}\right)\left(1 + \frac{tQ_0}{I_R\Omega_0}\right)^{-1} \qquad \textit{eq 10.22}$$

$$\Delta h = \frac{gt^2}{2}\left[1 - \frac{2I_R\Omega_0}{Q_0 t} + \frac{2\left(I_R\Omega_0\right)^2}{\left(Q_0 t\right)^2}\ln\left(1 + \frac{Q_0 t}{I_R\Omega_0}\right)\right] \qquad \textit{eq 10.23}$$

Where:

$\dot{h}$	- Vertical velocity
g	- Gravity
t	- Time
Q_0	- Initial torque
I_R	- Moment of inertia, rotor
Ω_0	- Initial rotor angular velocity.

The loss in altitude (Δh) during this phase of the landing is improved with increased rotor speed and rotor inertia and degraded by the initial torque (or power) required.

Rapid collective application during the second phase (to establish thrust equal weight) with follow on smooth collective application to maintain thrust aggravates rotor speed decay. Any additional height available provided by follow on use of collective depends on the degree to which rotor speed decays in accelerating vertically to $\dot{h}_{TD}$ and the rotor stall boundaries. For lower inertia rotors, higher hover power requirements, and high blade lift loadings, there may be no thrust (or useable energy) available from the rotor to provide additional height capability.

10.3.4.2.2 Slow Speed Low Boundary

As speed increases from the low hover point there is some decrease in power required. The forward entry from the lower boundary probably provides improved rotor decay characteristics due to the lower torque required but little or no cyclic flare is generally possible and the landing consists of a collect flare at near entry speed. The forward speed landing may give the pilot better field of view than in hover, but the forward speed and low rotor speed may become critical on the roll out. Very little down collective, if any, is used between power loss and touchdown when on, or below, the lower boundary.

10.3.4.2.3 Minimum Height for High Hover and Slow Speed High Boundary

At the high hover condition, sufficient altitude (potential energy) is available to convert to airspeed (kinetic energy) thereby reducing power required and providing energy to the rotor lost during rotor speed decay. Studies indicate the minimum height for high hover is related to the airspeed defined for the low speed knee (section 10.3.4.2.5). The minimum height for high hover with a 2 s delay generally is in the range of 500 to 800 ft.

Entry at the minimum height for high hover generally requires down collective to minimize rotor speed loss, a significant dive angle to obtain forward speed, and a rapid flare prior to touchdown. Pilot judgement for airspeed and flare, becomes crucial to the successful accomplishment of this maneuver. As height decreases and airspeed increases, the requirement to dive decreases. In fact, as the entry altitude approaches that of the low speed knee, a significant dive is not possible and there is, in some cases, a significant change in pilot technique for control of the collective. The collective may not reach full down because of the vertical acceleration and ground proximity.

10.3.4.2.4 Slow Speed Knee

The slow speed knee represents an area of the curve with all the bad features of the upper and lower boundaries. The landing from this area probably results in a touchdown at a speed not much less than the entry speed. The flare must come soon after a rate of descent is established. The rate of descent is controlled (established) by the pilot by either a slight dive, a small decrease in collective or a combination of the two. The time between power failure and landing may be only a few seconds.

Analytical studies, backed up with actual aircraft data, indicate the airspeed for the low speed knee is related to the airspeed for minimum power required in level flight and the helicopter gross weight. The same study predicts the minimum high hover altitude as a function of the airspeed for the slow speed knee, but data indicate that high hover altitude results are optimistic, i.e. the altitudes predicted are too low. Typically, the low speed knee with a 2 s delay time occurs in the range of 40 to 60 kn and 120 to 150 ft.

10.3.4.2.5 Maximum Touchdown Speed

The maximum touchdown speed in the high speed region is limited by landing gear design, etc. and is certainly tied to the landing surface characteristics. The touchdown is covered under autorotative landings. The ability to control direction of motion on the ground and stop the aircraft is important when landing in unprepared areas.

10.3.4.2.6 High Speed Region

The initial H-V curve rise in the high speed region is required to allow for some flare capability. The flare is needed here to slow to the maximum touchdown speed. The landing attitude (level or tail wheel first) somewhat governs the shape of this curve as does the flare capability of the aircraft. As speed increases toward limit speed, there is a tendency for some aircraft to lose a significant amount of altitude during rotor decay (controls fixed). In addition, an extended level autorotative deceleration (or pull up) from near V_H may be a difficult piloting task. This is especially true of entries at speeds in excess of about 90 kn. Decelerations or pull ups are considered from a ground clearance standpoint during pitch attitude changes. Practice is required usually to develop pilot proficiency for the successful accomplishment of a near V_H entry. The engine characteristics of some aircraft offset some of the value gained from practicing this entry, in that the static power during push over (after cyclic only flare when needles are not split) may be enough to reduce rotor speed decay. However, in a real situation rotor speed decay is extreme and drastically changes the technique required to land. The constant altitude wind up turn or climbing turn may be a good way to decelerate in some high speed situations, as it allows rapid change in rotor angle of attack (increased upwash) to recover rotor speed loss. This turning maneuver may be required when obstacles do not allow a straight ahead deceleration.

10.4 TEST METHODS AND TECHNIQUES

Helicopter flying qualities following sudden power loss through completion of a successful landing reflect the performance capabilities of the aircraft and the ease with which the pilot can attain and maintain optimum performance. Tests generally included in autorotative flying qualities evaluations are designed to investigate aircraft responses to engine failure, warnings provided, aircraft response to pilot inputs during the transition to autorotation, steady autorotative flight characteristics, and the pilot's ability to flare and land safely.

Before conducting any autorotative testing, perform a data search of contractor and government test results for the type aircraft to be tested, and other similar aircraft. This search includes assessment of:

1. Trimmed control position data in hover, low speed flight, vertical climbs/descents, and forward flight climbs/descents.
2. Control positions to counter trim changes associated with transition maneuvers for the above conditions.
3. Control response (and coupling) data for all axes in hover and forward flight.
4. Rotor acceleration/deceleration tendencies as indicated by changes in torque required during longitudinal control response tests and maneuvering stability tests (constant collective turns).

Consider using build up simulation maneuvers. Examples are:

1. Directional control response type tests to simulate torque changes and aircraft response following sudden power loss.
2. Conduct of autorotative flares, transitions from flare to landing attitude, and limited collective increase to a simulated runway at a safe altitude AGL. This simulation provides insight to rotor acceleration in the flare, rotor decay during landing attitude transition, and rotor decay associated with collective increases, as well as a rough indication of altitude lost from flare initiation to start of the collective increase.

Autorotation testing is accomplished usually in the following order:

1. Steady state autorotation performance (rate of descent with airspeed) and flying qualities (trimmed control positions, autorotation turns, etc.) evaluations.
2. Flare and landing characteristics.
3. Simulated engine failures characteristics.
4. H-V diagram investigation.

Steady state autorotation test techniques are addressed in reference 10.9 and Chapters 6 and 7, and are not repeated in this section. Flare and landing characteristics are conducted before throttle chops and H-V diagram tests in the event unforeseen problems exist after the throttle chop requiring an actual autorotation. H-V diagram tests are conducted last to benefit from knowledge gained during flare, landing, and throttle chop tests.

Subjective evaluation of the autorotative flying qualities is accomplished with a complete understanding of mission requirements. Handling Qualities Ratings (HQRs) are assigned with respect to pilot workload and attaining adequate and desired performance for the test conditions.

10.4.1 Flare and Landing

The purpose of these tests is to determine the optimum flare attitude and rate required to maximize rotor speed and maintain ground speed and descent rate within acceptable limits, to determine cyclic control requirements to establish the desired landing attitude, and to determine collective application requirements to effect a satisfactory autorotative landing.

The effect of different landing gear configurations on the touchdown and landing run as well as use of braking, either wheel brakes or collective lowering for braking, along with control margins during the flare and landing are evaluated.

10.4.1.1 TEST TECHNIQUE

1. At a safe altitude, practice power recoveries from autorotation at minimum rate of descent airspeed and several increasing flare attitudes. Investigate faster and slower airspeeds at each flare attitude.
2. At a safe altitude, practice simulated flares and landings from full autorotation at minimum rate of descent airspeed and several increasing flare attitudes and flare rates. Vary rotor speed for each entry. Investigate faster and slower airspeeds at each flare condition. Pay close attention to rotor speed changes, airspeed, altitude lost during flare and simulated landing, aircraft responses/trim changes which provide useful cues or which cause concern, and note the field of view.
3. Establish stabilized autorotation at the optimum conditions (airspeed, rotor speed) in preparation for actual autorotative flare/landing.
4. At a predetermined height based on contractor tests and/or altitude simulations, flare at the optimum rate to the optimum attitude.
5. As ground speed decreases to a comfortable level, establish a landing attitude and complete the landing.
6. Repeat tests systematically varying approach airspeed, flare characteristics, approach rotor speed, flare altitude, attitude, and touchdown ground speed. During these systematic variations, change only one parameter at a time and use small incremental changes initially to establish the trend produced by each variation.
7. Once the optimum flare and landing requirements are determined, investigate the effect of braking on the landing run. Vary the use of wheel brakes and collective lowering for braking.
8. Evaluate each available landing gear and landing surface combination for the helicopter’s mission.

10.4.1.2 DATA REQUIRED

Cockpit. Run number; approach airspeed; flare attitude; flare altitude, flare rate (low, medium, high); approach, maximum and touchdown rotor speeds; touchdown ground speed (estimate if not available); fuel count (FC); roll out distance; tower ambient conditions.

Ground personnel if used. Run number, wind speed and direction, temperature, pressure.

Ground station (telemetry). Control positions, attitudes, rates, load factor, airspeed, altitude (radar altitude if available), rotor speed. Use of ground station personnel observing data real time is recommended to provide information on exact pitch rate, pitch attitude, control margins, and rotor speeds during build up to the test limits.

10.4.1.3 TEST CRITERIA

1. Stabilized autorotation (airspeed, rotor speed) prior to flare initiation.
2. Forces initially trimmed to zero (optional).

10.4.1.4 DATA REQUIREMENTS

1. Stabilize as soon as possible before reaching flare altitude.
2. Start automatic recording of data 10 s prior to initiating flare. Record data until aircraft is on the ground.
3. Approach airspeed ± 1 kn.
4. Approach rotor speed ± 1%.
5. Flare altitude ± 5 ft.
6. Wind < 5 kn, ± 20° of the landing heading.

10.4.1.5 SAFETY CONSIDERATIONS/RISK MANAGEMENT

This test involves operation in an extreme situation where ground contact conditions are controlled. Execute a power recovery at any time the situation does not feel right. In the event a power recovery is required, engine response may be sluggish. When the engine engages the rotor, abrupt torque and heading changes may occur. Tail boom/tail wheel first landings at high pitch attitudes may lead to asymmetric main gear/skid touchdown and a subsequent pitch, roll, yaw bounce and/or nose tuck and/or lateral drift. After touchdown, heading control is reduced with low rotor speed. Ability to rapidly reposition TPP angle is reduced with low drive train speeds. Adequate fire fighting, crash rescue, and emergency medical treatment facilities should be available based on risk level. Follow careful build up procedures to investigate the use of braking during the landing run to avoid unexpected helicopter attitude changes on the deck.

10.4.2 Response to Engine Failure and Pilot Inputs

The purpose of this test is to evaluate aircraft response to sudden power loss, effectiveness of warning cues, available time for the pilot to take corrective action, initial pilot actions required, and follow up pilot actions required to transition to a steady or quasi steady autorotative condition. Tests are initiated at a safe altitude, usually 2000 to 3000 ft AGL. Desired end points are those associated with high power flight and representative power loss situations, i.e. total power loss in a single engine aircraft, or loss of one or more engines simultaneously in multiengine helicopters. Test techniques discussed in this section are designed to provide for a logical build up to end point conditions. Power loss is simulated by abrupt step input of the engine power control from the flight condition to the ground idle condition.

Engine deceleration (and torque loss) for this simulation is governed by the fuel control scheduling characteristics and may not be representative of actual lost power conditions (instantaneous failure of engine power turbine drive shaft or fuel starvation). Actual engine shut downs usually are not performed since they limit the power recovery option.

Abrupt power control movement involves coordinated efforts between flight crew members and ground station personnel similar to control response testing. These tests involve control response type inputs using the engine power control and voice procedures to establish cockpit coordination.

A systematic build up is necessary to eliminate surprises. Figure 10.24 illustrates a possible built up sequence. Point 1 is the starting point and is selected since steady autorotations and autorotative flares/landings were conducted starting at this airspeed.

Once Point 2 is achieved two paths are available, 2 to 3A or 2 to 3B. Either path is acceptable. If Path 2 to 3B is chosen, side paths (airspeed variations at constant torque) ending at the level flight curve are considered. An alternate path similar to Path 2 to 3A on the back side of the power curve is also realistic. Operation on the back side of the power curve provides information used later during H-V diagram investigations, which are conducted at lower entry altitudes.

At each stopping point of Figure 10.24 sub-investigations of other parameters such as pilot delay time, trimmed rotor speed, type of initial pilot input required (collective or cyclic) are pursued.

Systematic build up is a necessity. Regardless of the path chosen on Figure 10.24, it is imperative that only one variable is changed during progression to each stopping point during the test evolution.

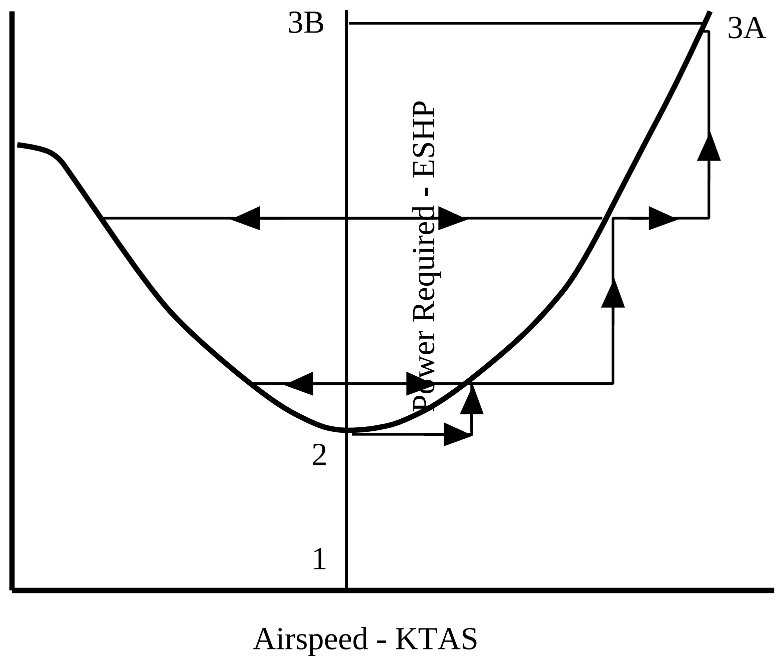


Figure 10.24
Simulated Power Failure Test Sequence

10.4.2.1 TEST TECHNIQUE

1. Determine the delay time for minimum allowable rotor speed at which positive corrective action (down collective) is required. Initially use small values of delay time or rotor speed change.

2. Establish stabilized flight at the desired test airspeed, torque setting, and rotor speed. The aircraft may be descending, level, or climbing.

3. As test altitude is approached begin voice procedures for cockpit coordination as follows: “Data on. Standby for a throttle chop on three. Thousand one, thousand two, thousand three”. The sudden power reduction is executed on three.
4. Maintain controls fixed for a predetermined delay time or rotor speed decay.
5. Make predetermined primary corrective control input (cyclic or collective) at the prescribed delay time or minimum rotor speed.
6. Make secondary inputs as required to fly the aircraft.
7. Transition to steady state autorotation.
8. Perform power recovery.
9. Assign HQR.
10. Repeat process for next test condition or variable change.

10.4.2.2 DATA REQUIRED

Cockpit. Entry flight conditions (airspeed, rotor speed), delay time, rotor speed at control input, minimum rotor speed observed, FC, altitude lost, cues, warnings, ambient conditions and comments.

Automatic recording devices (aircraft or ground station) are used to obtain time histories of aircraft attitudes, rates, control positions, heading, airspeed, altitude, load factor, rotor speed, throttle position, torque, and power turbine speed.

10.4.2.3 TEST CRITERIA

1. Stabilized conditions (airspeed, rotor speed, rate of descent/climb) prior to throttle chop.
2. Forces initially trimmed to zero.

10.4.2.4 DATA REQUIREMENTS

1. Stabilize as soon as possible before reaching test altitude if descending or climbing.
2. Start automatic recording of data 10 s prior to sudden power reduction. Record data until power recovery is completed.
3. Airspeed ± 1 kn.

4. Altitude ± 200 ft.
5. Rotor speed ± 1 %.
6. Torque ± 1%.

10.4.2.5 SAFETY CONSIDERATIONS/RISK MANAGEMENT

Simulated engine failure testing presents the possibility of an inadvertent or uncommanded engine shut down. Before conducting tests check throttle control linkages and throttle stops for proper rigging. Perform throttle chops on the deck before conducting airborne tests. Extreme aircraft responses are possible. Practice recoveries from unusual attitudes at low power and during power applications. Be cautious of control inputs and aware of reduced control effectiveness in low g flight. Brief other potential emergencies such as electrical failure, hydraulic failure, and flight control system (AFCS) failure. Brief aircraft limits and bail out procedures. Wear parachutes. Maintain spatial awareness/orientation at all times. Consider flight termination and analysis of results if any unexpected conditions are encountered. A chase aircraft may be helpful in providing visual traffic clearance.

10.4.3 Height-Velocity Diagram

The purpose of these tests is to determine or validate manufacturer provided H-V diagram. Tests involve extension of engine failure and flare/landing characteristics to the near ground flight regime. End points involve definition of the airspeed and height envelope within which the fleet aviator can reasonably recover and land following a sudden power loss. Tests involve operation near the ground at low and high airspeeds and procedures are designed to provide for a logical build up to the end points. Sudden power loss is simulated usually by abrupt movement of the engine power control from the flight condition to the ground idle condition.

Engine deceleration for this simulation is governed by fuel control scheduling characteristics and may not be representative of actual power loss conditions. Actual engine shut downs are not performed usually since margins for error are reduced or eliminated.

Abrupt power control movements, the potential for violent aircraft reactions, close to ground operation and the limited time available to perform the correct actions necessitate close coordination of flight crew members and ground station personnel. Voice procedures to coordinate activities are mandatory.

A systematic build up is necessary to provide safety margins and eliminate surprises. Figure 10.25 illustrates the possible build up sequences employed during investigation of the low speed area of the H-V Diagram.

In Figure 10.25, either point 1 or 4 is an acceptable starting point. Point 1 represents a very low hover, or even possibly an on the ground point with thrust greater than zero but less than the aircraft weight (light on the landing gear). Point 4 represents a safe condition as determined by engine failure tests at altitude.

Path 1 to 2 develops the maximum height for a safe vertical hovering autorotation. Path 2 to 3 involves the landing technique of a collective increase near touchdown and run-on landing with some ground speed.

Path 4 to 5 represents a build up of decreasing airspeed and increasing power required until point 5 is reached. Intervals along this path are reduced as point 5 is approached. Once point 5 is obtained, either path 5 to 7 or 5 to 7A is acceptable. Paths 5 to 7/7A represent conditions where the initial pilot reactions are down collective to reduce rotor decay and forward cyclic to increase airspeed to arrive at an acceptable rotor speed and flight path combination for flare/landing.

Figure 10.26 proposes potential sequences to investigate the high speed area of the H-V diagram. Point 1, 7 or 8 are all acceptable starting conditions. Point 5 is the maximum allowable landing ground speed.

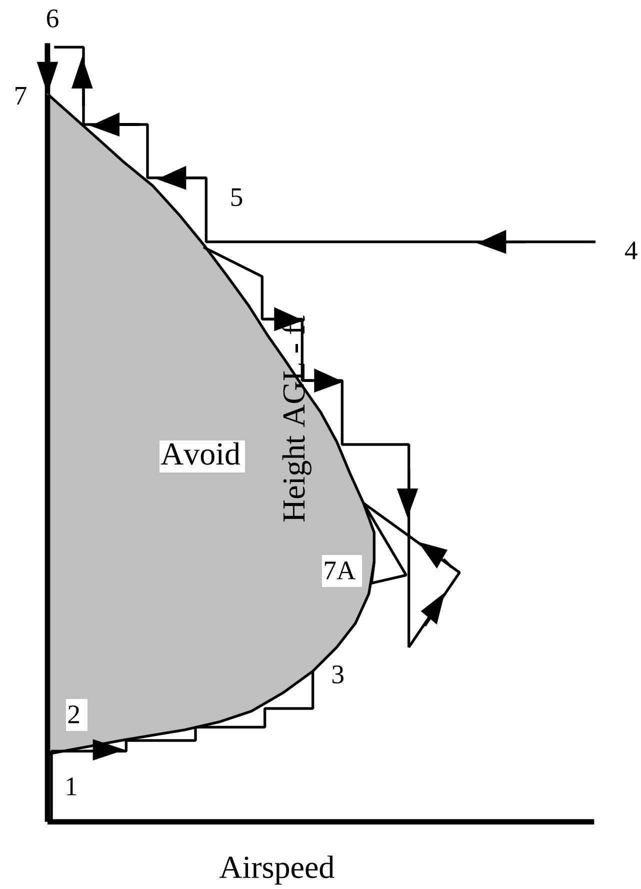


Figure 10.25
Height-Velocity Diagram Low Speed Area Test Sequence

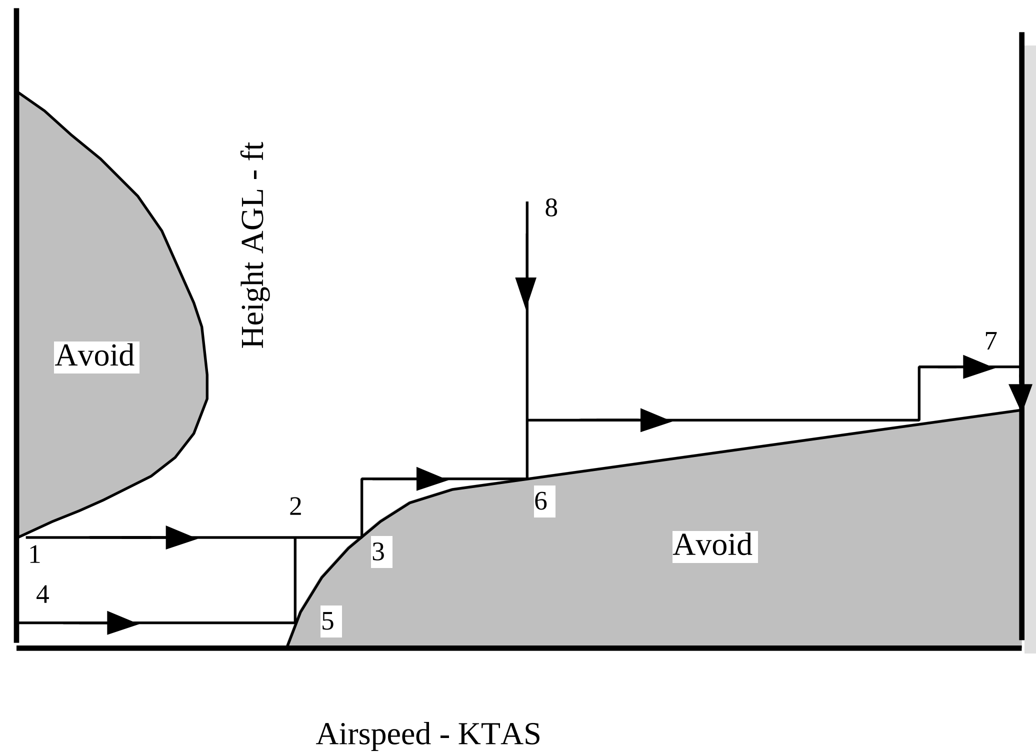


Figure 10.26
Height Velocity Diagram High Speed Area Test Sequence

Point 1 was determined as safe from low speed area investigations. Path 1 to 3 represents intervals of increased speed with reduced power required. Landing procedures on path 1 to 2 involve collective flares for touchdowns and a run-on landing. From 2 to 3 some cyclic flare is required to arrive at a ground speed less than or equal to the maximum allowable (point 5).

Path 4 to 5 represents a very low altitude test sequence involving collective cushioning and run-on landing. Path 4 to 5 also allows investigation of ground roll out characteristics as affected by touchdown ground speed.

If sequence 1 to 3 is chosen initially, a build up from 3 to 5 is used to obtain point 5.

Path 3 to 7 or 5 to 7 allow for build up at a previously defined safe altitude. These sequences involve some cyclic flaring at low speeds and greater cyclic flare at higher speeds to the extent that zoom climbs or decelerating turns are considered.

Starting point 8 represents a predetermined safe airspeed and attitude combination obtained either from engine failure tests at altitude or H-V diagram investigation for the low speed area. If point 8 is chosen as a starting point a build up is used to approach point 5 from point 6.

At each stopping point investigation of effects such as delay time and initial rotor speed are performed.

Careful review of flare/landing and engine failure data are performed prior to starting H-V diagram testing.

10.4.3.1 TEST TECHNIQUE

1. Determine the delay time or minimum rotor speed allowable at which appropriate positive corrective action (down collective, cyclic flare, or push over) is taken. Initially use small values of delay time or rotor speed change.
2. Establish stabilized conditions at the desired test speed (airspeed or ground speed), altitude, and rotor speed. Record stabilized conditions.
3. Initiate voice procedures for cockpit coordination as follows: "Data on. Standby for a throttle chop on three. Thousand one, thousand two, thousand three". The throttle chop is executed on three.
4. Maintain controls fixed for a predetermined delay time or rotor speed decay.
5. Make predetermined primary corrective control input at prescribed delay time or minimum rotor speed.
6. Make secondary inputs as required to fly the aircraft and transition to flare/landing.
7. Assign an HQR to the task of completing a successful landing.
8. Repeat process for next the test condition or variable change.

10.4.3.2 DATA REQUIRED

Cockpit. Airspeed/ground speed, altitude, rotor speed, delay time, rotor speed at control input, minimum rotor speed, cues, warnings, ground roll, fuel remaining, ambient conditions, HQR, FC, and comments.

Automatic recording devices (aircraft or ground station are used to obtain time histories of aircraft attitudes, rates, control positions, heading, airspeed, altitude (radar), load factor, rotor speed, throttle position, torque, and power turbine speed.

10.4.3.3 TEST CRITERIA

1. Stabilized flight prior to throttle chop.
2. Forces initially trimmed to zero.
3. Constant altitude, no rate of climb or descent.

10.4.3.4 DATA REQUIREMENTS

1. Stabilize as soon as possible before recording data.
2. Start automatic recording of data 10 s prior to throttle chop. Record data until ground run is complete.
3. Airspeed/ground speed ± 1 kn.
4. Altitude ± 5 ft above 50 ft AGL, ± 10% altitude below 50 ft AGL.
5. Rotor speed ± 1 %.
6. Wind < 3 kn.
7. Torque as required for level flight.

10.4.3.5 SAFETY CONSIDERATIONS/RISK MANAGEMENT

Simulated engine failure testing always presents the possibility of an inadvertent or uncommanded engine shut down. Before conducting these tests check throttle control linkage rigging and perform ground throttle chops. Be cautious of control inputs and aware of reduced effectiveness of control inputs in low g flight. Tests involve operation in an extreme situation where ground contact conditions must be controlled. Execute a power recovery if a safe autorotative landing is doubtful. In the event a power recovery is required, engine response from ground idle may be sluggish. When the engine engages the rotor, abrupt torque and heading changes may occur. Consider power recovery practice in the low speed environment. At ground contact, tail wheel first landings at high pitch attitudes may result in asymmetric main gear/skid touchdown and a subsequent pitch, roll, yaw bounce and/or nose tuck and/or lateral drift. After touchdown, heading control is reduced with low rotor speed. Ability to rapidly reposition TPP angle is reduced with low rotor speed. Maintain spatial awareness/orientation and conduct operations clear of airport hazards and traffic.

10.5 DATA REDUCTION

Data reduction for steady state autorotative flying qualities involves presentation of selected dependent variables (control positions, attitudes, etc.) as functions of pertinent independent variable (airspeed or main rotor speed). Dynamic characteristics are represented usually in time history form with appropriate annotations. Some graphical data presentations for dynamic characteristics may be warranted. Automatic data recording systems usually include total instrumentation system corrections in the process used to change a flight or telemetry tape raw data into engineering units.

Label all graphs and time histories to show configuration, GW, CG, ambient conditions, AFCS status, and other appropriate constants.

10.5.1 Steady Autorotation

Data reduction requirements and methods for steady autorotation are outlined in Chapters 6 and 7 for longitudinal and lateral-directional tests respectively.

10.5.2 Flare and Landing

Present time histories showing complete flare, landing, and landing run for appropriate parameters and maneuvers. Annotate flare initiation, maximum flare attitude, maximum rotor speed, control inputs to achieve landing attitude, touchdown, and termination of landing run. A sample time history showing the touchdown and roll out phase of an autorotative landing is presented in Figure 10.27.

Consider graphical presentations of parameters showing trends or limiting conditions, such as maximum rotor speed with flare rate for a fixed flare attitude to show overspeed tendencies, or touchdown rotor speed with flare height. Make graphical presentations to illustrate points of interest.

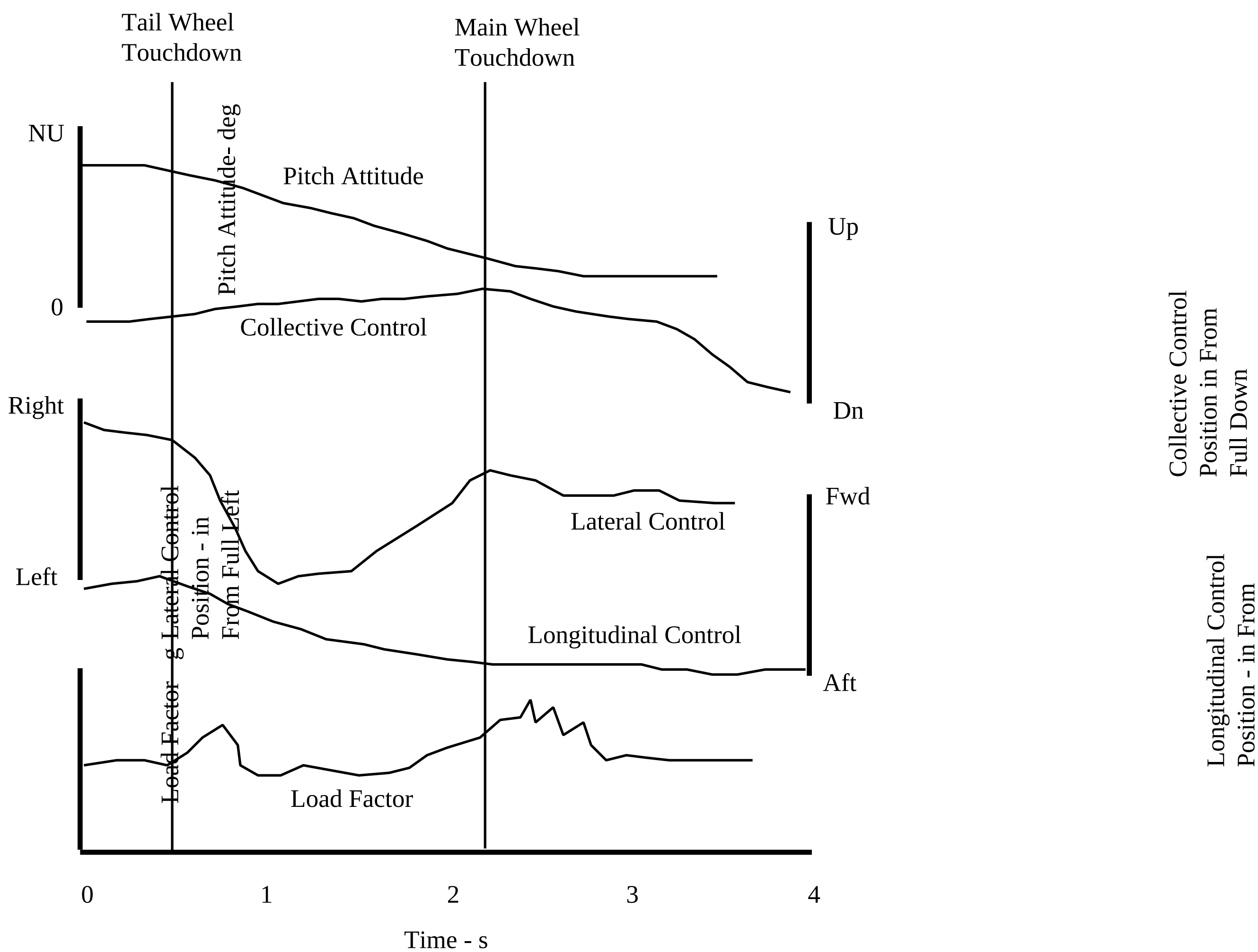


Figure 10.27
Autorotative Touchdown of a Single Rotor Helicopter

10.5.3 Response to Engine Failure and Pilot Inputs

Present time histories from initiation of simulated engine failure to steady autorotative flight. Annotate throttle reduction, pilot reaction (delay time), minimum rotor speed, any approaches to control, attitude or load factor limitations, occurrence of any highly desirable or undesirable characteristics. A sample time history is shown in Figure 10.28.

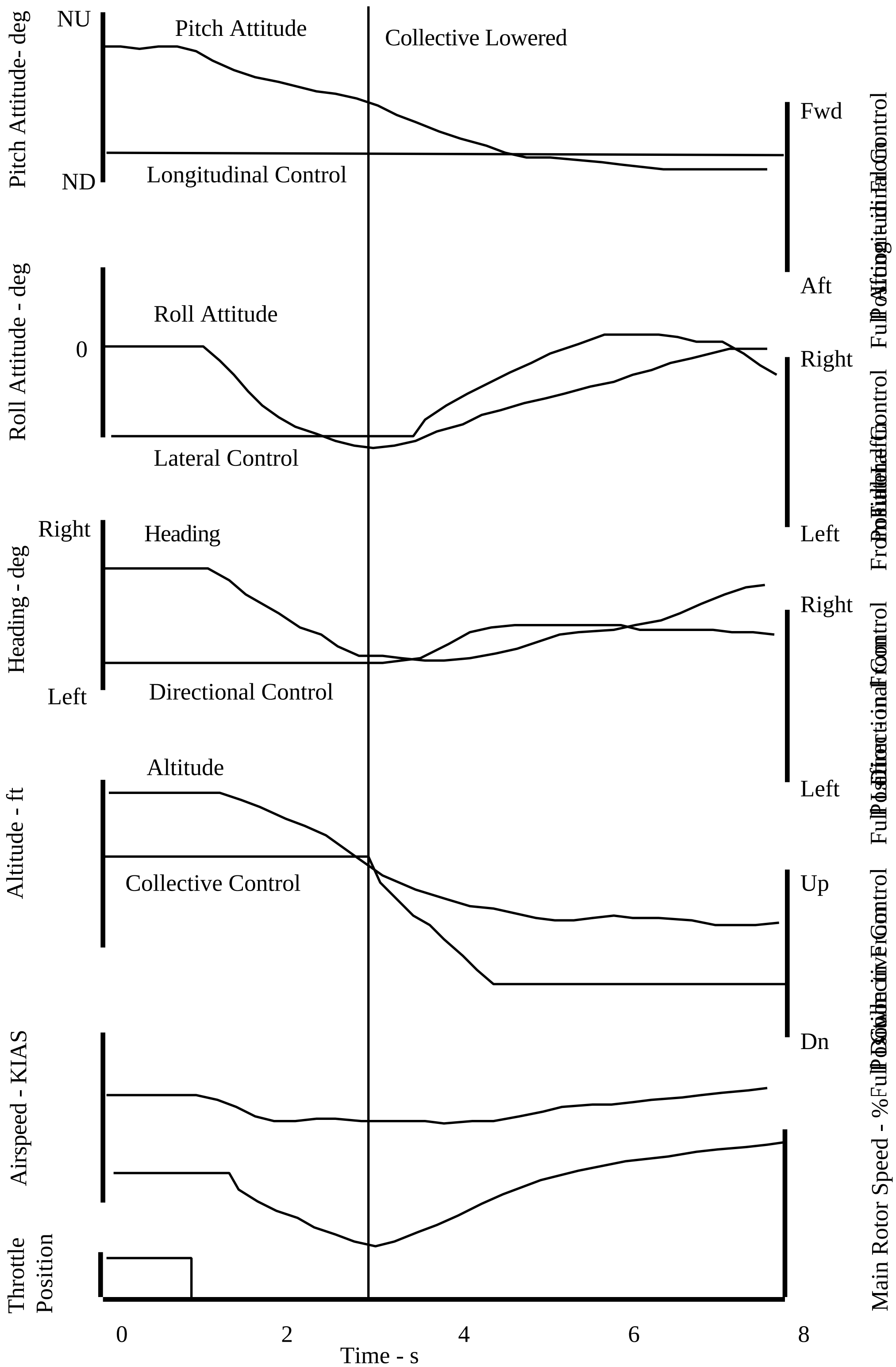


Figure 10.28
Simulated Engine Failure of Single Engine Helicopter

Consider graphical presentations of parameters showing trends or limiting conditions. Samples are shown in Figures 10.29, 10.30, and 10.31. Figures 10.29 and 10.30 show possible presentations of delay time information. Figure 10.31 illustrates total load factor changes without pilot corrective action and those resulting from pilot control input. These plots are shown as examples. Any time a significant and meaningful trend with some parameter appears to exist, use a graphical presentation. Use ingenuity in such graphical presentations and be specific about those parameters that are constants for the graph.

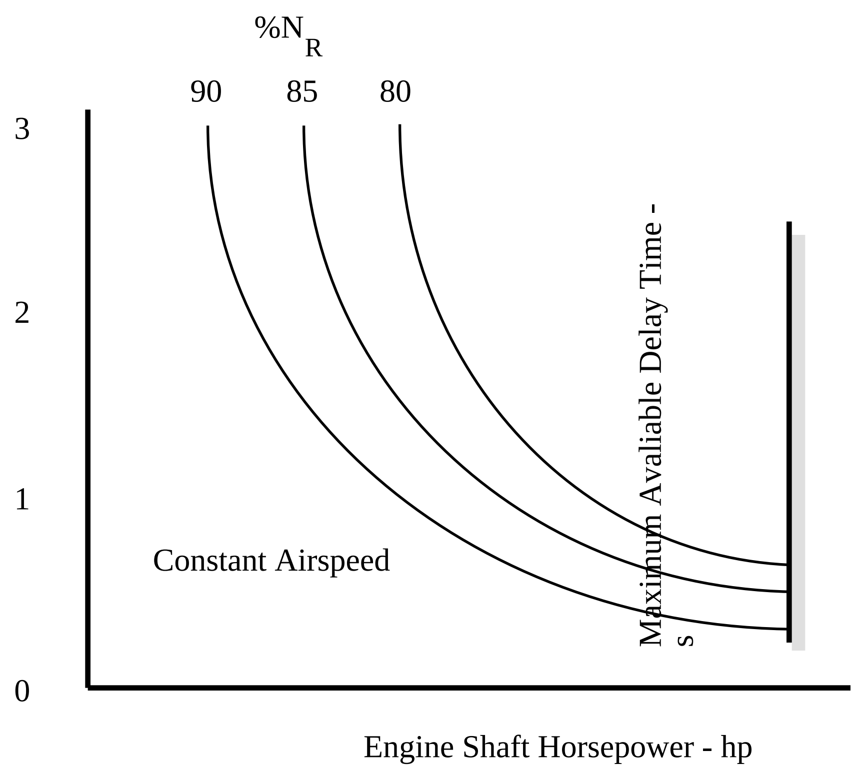


Figure 10.29
Rotor Speed Decay Characteristics

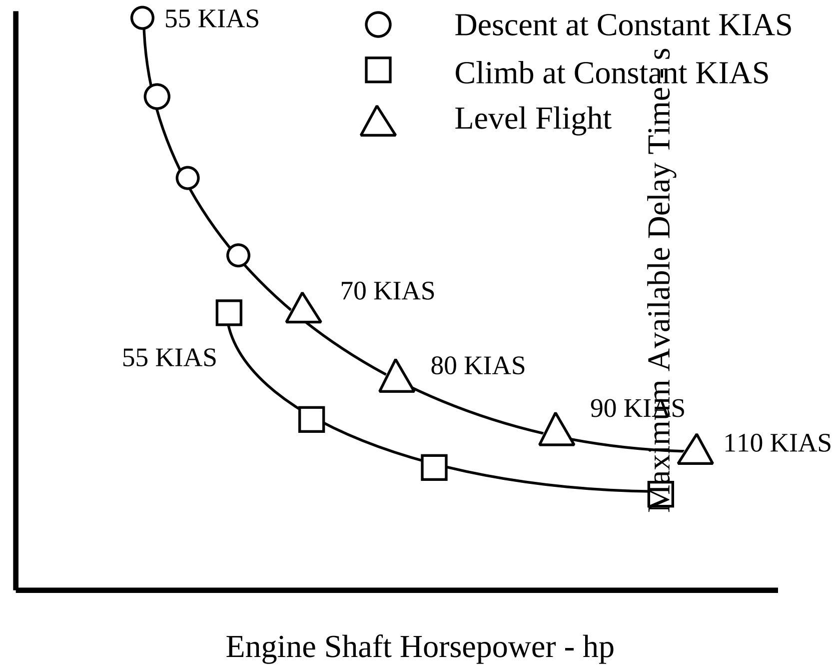


Figure 10.30
Delay Time for Fixed Rotor Speed Change

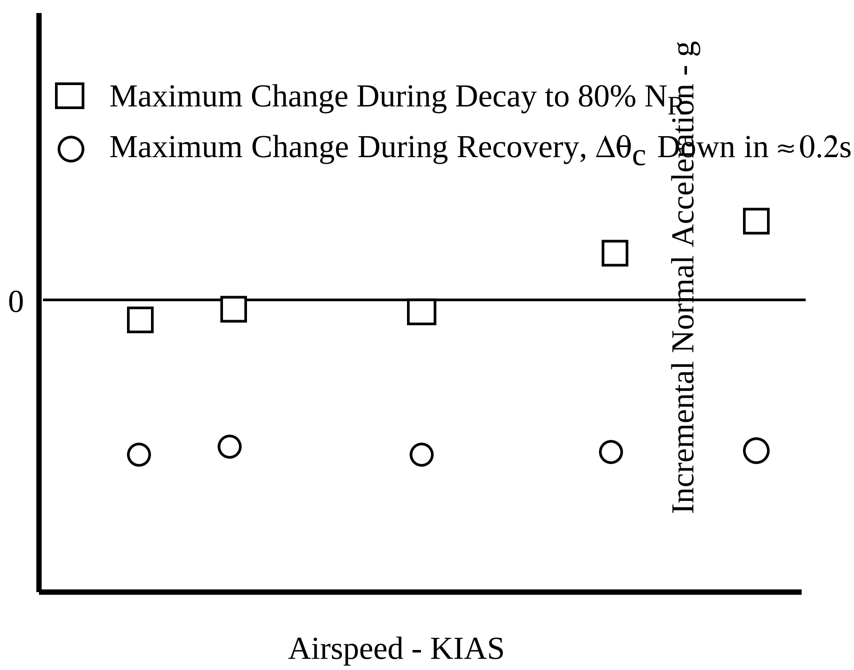


Figure 10.31
Load Factor Changes During Decay and Recovery

10.5.4 Height-Velocity Diagram

Present time histories showing complete maneuvers from simulated engine failure to completion of the landing run. Use appropriate parameters for representative maneuvers. Annotate throttle chop; pilot reaction (delay time); minimum, maximum and touchdown rotor speeds; approaches to control or attitude limitations; touchdown and termination of landing run. A sample time history is shown in Figure 10.32.

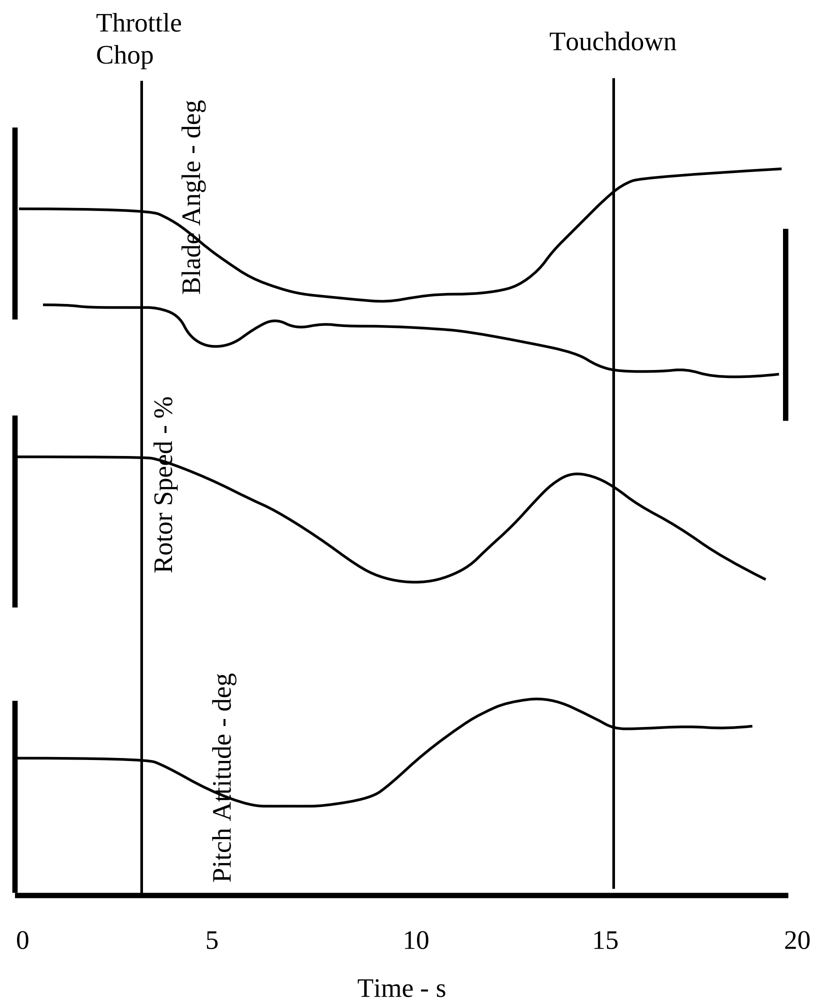


Figure 10.32
Simulated Engine Failure and Landing

Plot the end points obtained during H-V diagram testing (altitude and airspeed/ground speed) as shown in Figure 10.33. Other plots as suggested in sections 10.5.3 and 10.5.4 may be appropriate.

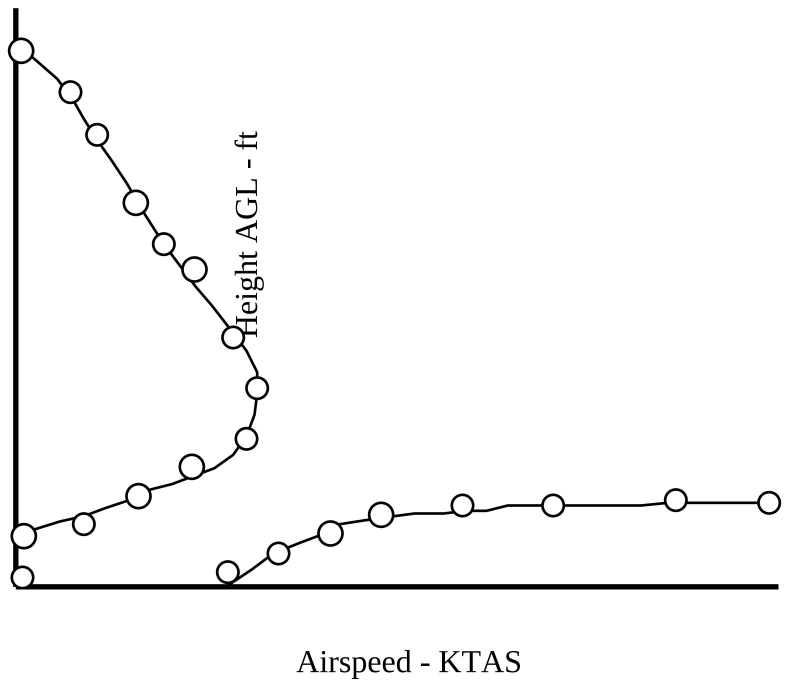


Figure 10.33
Tested Height-Velocity Diagram

10.6 DATA ANALYSIS

10.6.1 Steady Autorotation

Analysis of steady autorotative flying qualities is addressed in Chapters 6 and 7. A summary of possible considerations follows:

1. Were the control margins adequate?
2. What were static stability characteristics in terms of control, force, and attitude gradients with airspeed? Were gradients shallow or significant trim speed bands requiring airspeed monitoring to maintain optimum conditions?
3. What were maneuvering stability characteristics in terms of control and force gradients with load factor? Was rate of descent in turns excessive? How much effort was dedicated to rotor speed control in maneuvering?

4. Did longitudinal long term and spiral motions require close monitoring of airspeed and attitude?

5. Were static lateral-directional characteristics usual? What were rate of descent changes in steady heading sideslip?

6. What is optimum rotor speed for steady state autorotation?

10.6.2 Flare and Landing

The following considerations are generally included in analysis of autorotative flare and landing test data:

1. Were control response characteristics adequate to make precise predictable attitude changes during flare and recovery to landing attitude?
2. How did rotor speed build in the flare?
3. Were there outside cues to rotor speed build up (rotor sound)?
4. Was rotor speed control a problem?
5. How effective was flare in arresting ground speed and rate of descent? Did the aircraft tend to mush or balloon?
6. How critical was flare initiation altitude on safe landing parameters?
7. How much collective/rotor speed was available for landing?
8. What were landing attitudes and touchdown speeds?
9. What were field of view and altitude cues at landing?
10. What were the landing run characteristics and were there problems with asymmetric touchdown, bounce, tuck, TPP control, or heading control? Do these characteristics warrant further tests on other surfaces? What were the landing run distances and were they excessive for the mission?
11. Were there indications of approach to stall as indicated by vibration, control feedback, or loss of TPP control?

10.6.3 Response to Engine Failure and Pilot Inputs

The following considerations are generally included in the analysis of engine failure and pilot response to engine failure test data:

1. What warning was provided and what was the timing and effectiveness of the warning?
2. What were the initial yaw, roll, and pitch responses to engine failure? Were these responses in excess of specification requirements?
3. What time is available delay time for the pilot to take corrective action? What parameter dictates available delay time?
4. How much rotor speed is lost before pilot input and what was the minimum rotor speed experienced?
5. Were aircraft systems (electrical, hydraulic, AFCS) lost and what was the significance of these losses?
6. What is the sequence, rate, direction, and magnitude of pilot inputs required?
7. What was the control response/effectiveness during recovery and were coupled responses a problem?
8. Are structural limits such as load factor or excessive flapping a cause for concern?
9. How long did it take to regain rotor speed and lost systems? Are transients involved as systems come back on line? Is restoration of desired/acceptable rotor speed easily controlled?
10. Were pilot actions required to restore lost systems? Are transients involved?
11. How much altitude was lost between engine failure and (a) recovery of acceptable rotor speed and (b) establishment of stabilized autorotation?

10.6.4 Height-Velocity Diagram

Height velocity diagram investigation is basically a combination of response to engine failure, effects of pilot inputs, and flare/landing characteristics. The combined considerations of sections 10.6.2 and 10.6.3 are addressed during H-V diagram testing.

10.7 MISSION SUITABILITY

The suitability of the test aircraft for the intended mission is the ultimate reason for conducting usual handling qualities testing. However, in the event of a sudden power loss the aircraft is permanently out of the mission and mission suitability reverts to the pilot's ability to save the personnel involved and possibly the airframe.

Engine failure testing from occurrence to landing result in defining those actions which result in a safe landing. To define these actions the test team must be aware of the flight environment, the flight tasks, and the internal and external information available to the operator during conduct of the task.

Engineering and quantitative information should be included in the NATOPS manual for all mission phases to allow meaningful practice in lost power situations.

The most important aspects of autorotative handling qualities is to keep the operator and his required flight and landing area environment in mind.

10.8 SPECIFICATION COMPLIANCE

Guidelines for helicopter handling qualities during engine failure, recovery, steady autorotation and landing are contained in MIL-H-8501A. Military specifications are guides to acceptability and are used in conjunction with mission requirements. Paragraphs 3.2 (Longitudinal characteristics) and 3.3 (Directional and lateral characteristics), and subparagraphs of 3.2 and 3.3, address autorotative flying qualities covered in Chapters 6 and 7 of this manual. These requirements are not repeated here. The following list identifies those paragraphs of MIL-H-8501A which specifically address autorotation requirements.

3.5 Autorotation, rotor characteristics, and miscellaneous requirements.

3.5.4.3 Paved surface landings to at least 35 kn.

3.5.4.4 Stopping distance.

3.5.4.5 Water landings.

3.5.5 Autorotation entry.

3.5.5.1 Attitude changes and delay time following sudden power reduction.

3.5.6 Control forces during transition to autorotation.

3.5.7 Landings at speeds of 15 kn or less.

3.5.9(e)(2) Control margins in steady heading sideslips.

Additional requirements are included in MIL-H-8501 which relate to boosted controls failure modes, and automatic stabilization equipment. These requirements are considered in view of the real potential to lose systems during low rotor speed conditions. MIL-H-8501A is supplemented with additional detailed specifications listed in the procurement documents. These should be researched to ensure contractual compliance and a complete evaluation.

10.9 GLOSSARY

10.9.1 Notations

a_{1s}	Longitudinal flapping angle, shaft referenced
A_D	Rotor disc area
AFCS	Automatic Flight Control System
AGL	Above ground level
b	Number of blades
B_{1s}	Longitudinal cyclic pitch angle, shaft referenced
b_{1s}	Lateral flapping angle, shaft referenced
CG	Center of gravity
C_P	Power coefficient
C_Q	Torque coefficient
dD_0	Blade element profile drag
dF	Blade element torque force

dL	Blade element lift
dR	Blade element resultant aerodynamic force
e	Flapping hinge offset
ESHP	Engine shaft horsepower
F	Force
FC	Fuel count
g	Gravity
GW	Gross weight
h	Height of hub above CG
H-V	Height -Velocity
h_{AGL}	Height above ground level
HQR	Handling Qualities Rating
$\ddot{h}$	Vertical acceleration
$\dot{h}$	Vertical velocity
$\dot{h}_{TD}$	Vertical velocity at touchdown
I_R	Rotor rotational moment of inertia
i_{TR}	Tail rotor inclination out of vertical plane
I_{zz}	Moment of inertia about z axis, yaw moment of inertia
KIAS	Knots indicated airspeed
KTAS	Knots true airspeed
L	Lift
L_{CG}	Roll moment about CG
L_H	Roll moment due to rotor hub forces
L_r	Roll moment due to yaw rate
L_t	Tail lift
l_t	Distance from the tail to the CG
L_v	Roll moment due to lateral velocity
m	Mass
M_{CG}	Moment about CG
M_H	Pitch moment due to rotor hub force
M_S	Blade mass moment
N	Net moment about z axis, Yaw moment
$N_{\delta_{PED}}$	Yaw moment due to pedal control
N_R	Main rotor speed

N_r	Yaw moment due to yaw rate
Q	Torque
Q_0	Initial torque
Q_{MR}	Main rotor torque
θ_{TR}	Tail rotor pitch angle
R	Resultant aerodynamic force, Rotor radius
r	Angular velocity about z axis, Radius along blade
$\dot{r}_0$	Initial angular acceleration about z axis
$\dot{r}$	Angular acceleration about z axis
T	Thrust
t	Time
T_0	Initial thrust
TPP	Tip path plane
T_{TR}	Tail rotor thrust
V	Velocity, free stream velocity
VFR	Visual flight rules
V_H	Maximum level flight airspeed
v_i	Induced velocity
V_{Limit}	Limit velocity
V_{max}	Airframe limited level flight airspeed
W	Weight
$\ddot{z}$	Acceleration along z axis

10.9.2 Greek Symbols

α (alpha)	Angle of attack
δ_{PED} (delta)	Pedal control
λ (lambda)	Inflow ratio
μ (mu)	Advance ratio
π (pi)	Mathematical constant

θ (theta)	Blade pitch angle
θ_C	Collective pitch angle
θ_{TR}	Tail rotor pitch angle
ρ (rho)	Density
Ω (Omega)	Rotor angular velocity
Ω_0	Initial rotor angular velocity
$\dot{\Omega}$	Rotor angular acceleration
ψ (psi)	Blade azimuth angle

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APPENDIX I

GLOSSARY

APPENDIX I

GLOSSARY

NOTATIONS

A	Rate response value
a	Lift curve slope
A_0	Collective pitch angle (not a function of ψ)
a_0	Coning angle
A_{0s}	Collective pitch angle, shaft referenced (not a function of ψ)
A_1	Lateral cyclic pitch angle
a_1	Longitudinal flapping angle
A_{1s}	Lateral cyclic pitch angle, shaft referenced
a_{1s}	Longitudinal flapping angle, shaft referenced
ABC	Advancing Blade Concept
A_D	Rotor disc area
AFCS	Automatic Flight Control System
AGL	Above ground level
APU	Auxiliary power unit
ASW	Anti-submarine warfare
Attitude control effectiveness	Change in aircraft attitude achieved in one second following a unit step control displacement
B	Damping constant
b	Number of blades
B_1	Longitudinal cyclic pitch angle
b_1	Lateral flapping angle
B_{1s}	Longitudinal cyclic pitch angle, shaft referenced
b_{1s}	Lateral flapping angle, shaft referenced
B_{CRIT}	Critical damping

$\dot{B}_{1s}$	Time rate of change of longitudinal cyclic pitch angle, shaft referenced
c	Blade chord
$C_{1,2}$	Constants affected by initial conditions of the motion
$C_{1/2}$	Cycles to one-half amplitude
$C_{1/N}$	Cycles to 1/N amplitude
C_d	Cycles to double amplitude
CF	Centrifugal force
CG	Center of gravity
Control power	Measure of the moment produced per unit of control displacement
Control sensitivity	Initial angular acceleration produced by a unit step control displacement
C_P	Power coefficient
C_Q	Torque coefficient
dD_0	Blade element profile drag
deg	Degree
D_f	Fuselage drag force
dF	Blade element torque force
dL	Blade element lift
dR	Blade element resultant aerodynamic force
DT IIA	Developmental Test IIA
dT	Blade element thrust
e	Base of natural logarithm, flapping hinge offset
E_d	Damping angle
ESHP	Engine shaft horsepower
F	Force
FC	Fuel count
F_q	Pitch input
F_R	Reaction force from ball race
$F_{S\text{ blade on hub}}$	Shear force of the blade on hub
$F_{T\text{ blade on hub}}$	Tangential force of the blade on hub
ft	Foot
FTE	Flight Test Engineer

FTM	Flight Test Manual
F_x	Force in the x direction
g	Gravity
GCA	Ground controlled approach
GW	Gross weight
H	Rotor hub force, $\perp$ to shaft
h	Height of hub above CG
h'	Longitudinal distance between the rotor shaft and the CG
h'_{lat}	Lateral distance between the rotor shaft and the CG
H-V	Height -Velocity
h_{AGL}	Height above ground level
H_{P_o}	Observed pressure altitude
HQR	Handling Qualities Rating
h_{TR}	Height of the tail rotor above the CG
Hz	Hertz (cycles per second)
$\ddot{h}$	Vertical acceleration
$\dot{h}$	Vertical velocity
$\dot{h}_{TD}$	Vertical velocity at touchdown
I	Moment of inertia
i	Imaginary index
I_f	Moment of inertia about flapping hinge
IGE	In ground effect
IMC	Instrument Meteorological Conditions
in	Inch
I_R	Rotor rotational moment of inertia
i_s	Stabilizer incidence
i_{TR}	Tail rotor inclination out of vertical plane
I_{xx}	Moment of inertia about x axis, roll moment of inertia
I_{xz}	Product of inertia about x z axes
I_{yy}	Moment of inertia about y axis, pitch moment of inertia
I_{zz}	Moment of inertia about z axis, yaw moment of inertia
K	Spring constant
KIAS	Knots indicated airspeed
kn	Knot

KTAS	Knots true airspeed
L	Net moment about x axis, Roll moment, Lift, Length
$L_{A_{1s}}$	Roll moment due to lateral cyclic pitch angle
lb	Pound
$L_{B_{1s}}$	Roll moment due to longitudinal cyclic pitch angle
L_{CG}	Roll moment about CG
$L_{\delta_{LAT}}$	Roll moment due to lateral control
LDO	Lateral-directional oscillation
$L_{\delta_{PED}}$	Roll moment due to pedal control
L_{Equiv}	Equivalent length
L_{f+t}	Roll moment due to the fuselage/tail
L_H	Roll moment due to rotor hub forces
$\overline{L}_H$	Average roll moment due to rotor hub forces for b blades
ln_e	Natural logarithm
L_p	Roll moment due to roll rate
L_q	Roll moment due to pitch rate
L_{θ_C}	Roll moment due to collective pitch angle
$L_{\theta_{TR}}$	Roll moment due to tail rotor pitch angle
L_r	Roll moment due to yaw rate
L_t	Tail lift
l_t	Distance from the tail to the CG
L_u	Roll moment due to longitudinal velocity
L_v	Roll moment due to lateral velocity
$L_{v_{PED}}$	Pedal contribution to roll moment due to lateral velocity
L_w	Roll moment due to vertical velocity
M	Net moment about y axis, Pitch moment
m	Mass
M_A	Aerodynamic moment
M_α	Pitch moment due to angle of attack
$M_{A_{1s}}$	Pitch moment due to lateral cyclic pitch angle
$M_{B_{1s}}$	Pitch moment due to longitudinal cyclic pitch angle
M_{CG}	Moment about CG
$M_{CG_{f+t}}$	Pitch moment due to the aerodynamic forces on the fuselage/tail

$M_{\delta_{LONG}}$	Pitch moment due to longitudinal control
M_H	Pitch moment due to rotor hub force
$\overline{M_H}$	Average pitch moment due to rotor hub force for b blades
MilSpec	Military Specification
M_p	Pitch moment due to roll rate
M_q	Pitch moment due to pitch rate
M_{θ_C}	Pitch moment due to collective pitch angle
$M_{\theta_{TR}}$	Pitch moment due to tail rotor pitch angle
$M_{\dot{\theta}}$	Pitch moment due to time rate of change of pitch angle
M_S	Blade mass moment
MSL	Maneuvering stability level
M_u	Pitch moment due to longitudinal velocity
M_v	Pitch moment due to lateral velocity
M_W	Moment due to weight
M_w	Pitch moment due to vertical velocity
$M_{\dot{w}}$	Pitch moment due to vertical acceleration
N	Net moment about z axis, Yaw moment, Denominator of the fractional amplitude
n	Normal acceleration, Normal load factor
$N_{A_{1s}}$	Yaw moment due to lateral cyclic pitch angle
NAWCAD	Naval Air Warfare Center Aircraft Division
$N_{B_{1s}}$	Yaw moment due to longitudinal cyclic pitch angle
$N_{\delta_{LAT}}$	Yaw moment due to lateral control
$N_{\delta_{PED}}$	Yaw moment due to pedal control
N_{f+t}	Yaw moment due to the fuselage/tail
N_g	Engine gas generator speed
N_p	Yaw moment due to roll rate
N_{θ_C}	Yaw moment due to collective pitch angle
$N_{\theta_{TR}}$	Yaw moment due to tail rotor pitch angle
N_R	Main rotor speed
N_r	Yaw moment due to yaw rate
n_{SS}	Steady state normal acceleration
N_u	Yaw moment due to longitudinal velocity

N_v	Yaw moment due to lateral velocity
N_w	Yaw moment due to vertical velocity
OFT	Operational flight trainer
OGE	Out of ground effect
P	Period
p	Angular velocity about x axis, Roll rate
p_0	Initial angular velocity about x axis, Roll rate
PFLF	Power for level flight
PIO	Pilot Induced Oscillation
p_{ss}	Steady state angular velocity about x axis, roll rate
$\dot{p}$	Angular acceleration about x axis
Q	Engine torque, Torque
q	Angular velocity about y axis, Pitch rate
Q_0	Initial torque
Q_{MR}	Main rotor torque
q_{ss}	Steady state pitch rate
θ_{TR}	Tail rotor pitch angle
$\ddot{q}$	Second time derivative of pitch rate
$\dot{q}$	Angular acceleration about y axis, Pitch acceleration
R	Resultant aerodynamic force, Rotor radius
r	Angular velocity about z axis, Radius along blade, Yaw rate
r'	Radius along blade outboard of flapping hinge
r_0	Initial angular velocity about z axis, Yaw rate
$\dot{r}_0$	Initial angular acceleration about z axis
rad	Radian
Rate control effectiveness	Angular rate achieved one second following a unit step control displacement
$\dot{r}$	Angular acceleration about z axis, Yaw acceleration
s	Second
SAR	Search and rescue
SAS	Stability Augmentation System
SCAS	Stability Control Augmentation System
SHSS	Steady heading sideslip
T	Thrust

t	Time
T_0	Initial thrust
t_0	Initial time
$T_{1/2}$	Time to one-half amplitude
$T_{1/N}$	Time to decay to 1/N of maximum amplitude
T_b	Thrust of b blades
TCDB	Trim control displacement band
TCP	Trim control position
T_d	Time to double amplitude
T_o	Observed temperature
TOOC	Turns on one control
TPP	Tip path plane
T_{TR}	Tail rotor thrust
u	Translational velocity component along x axis
u_0	Initial velocity, Initial velocity along x axis
u_{NR}	Translational velocity component along the non-rotating x axis system
u_P	Downward velocity
USNTPS	U.S. Naval Test Pilot School
u_T	Transverse velocity
$\ddot{u}$	Time rate of change of linear acceleration along x axis
$\dot{u}$	Linear acceleration along x axis
V	Velocity, Free stream velocity, Relative velocity
v	Translational velocity component along y axis
VAR	Vibration Assessment Rating
VFR	Visual flight rules
V_H	Maximum level flight airspeed
V_{hor}	Velocity in the horizontal plane
v_i	Induced velocity, Induced velocity at hover
V_{Limit}	Limit velocity
V_{max}	Airframe limited level flight airspeed
VMC	Visual Meteorological Conditions
V_{NE}	Velocity never exceed

v_{NR}	Translational velocity component along the non-rotating y axis system
V_o	Observed airspeed
VSTOL	Vertical/short takeoff and landing
V_v	Vertical velocity
$\ddot{v}$	Time rate of change of linear acceleration along y axis
$\dot{v}$	Linear acceleration along y axis
W	Weight
w	Translational velocity component along z axis
w_0	Initial velocity component along z axis
W_{BALL}	Weight of the ball
w_{QS}	Quasi-static vertical velocity
w_{rotor}	TPP translational velocity along z axis
w_{ss}	Steady state translational velocity in the z direction
$\ddot{w}$	Time rate of change of linear acceleration along z axis
$\dot{w}$	Linear acceleration along z axis
X	Resultant force in x direction
x	Orthogonal direction along longitudinal axis of the aircraft; Distance along x axis, Displacement
x_0	Initial displacement
$X_{B_{1s}}$	Longitudinal force due to longitudinal cyclic pitch angle
X_δ	Longitudinal force due to control
$X_{\delta_{LONG}}$	Longitudinal force due to longitudinal control
X_q	Longitudinal force due to pitch rate
X_{θ_C}	Longitudinal force due to collective pitch angle
X_u	Longitudinal force due to longitudinal velocity
X_w	Longitudinal force due to vertical velocity
$\ddot{x}$	Time rate of change of velocity in x direction, Acceleration along x axis, Acceleration
$\dot{x}$	Time rate of change of x, Velocity, Velocity in x direction
Y	Resultant force in y direction
y	Orthogonal direction along lateral axis of the aircraft
$Y_{A_{1s}}$	Side force due to lateral cyclic pitch angle
$Y_{\delta_{LAT}}$	Side force due to lateral control

$Y_{\delta_{PED}}$	Side force due to pedal control
Y_{f+t}	Side force due to the fuselage/tail
Y_p	Side force due to roll rate
$Y_{\theta_{TR}}$	Side force due to tail rotor pitch angle
Y_R	Rotor side force
Y_r	Side force due to yaw rate
Y_u	Side force due to longitudinal velocity
Y_v	Side force due to lateral velocity
Y_{v_R}	Main rotor contribution to side force due to lateral velocity
$Y_{v_{TR}}$	Tail rotor contribution to side force due to lateral velocity
Z	Resultant force in z direction
z	Orthogonal direction along vertical axis of the aircraft
$Z_{B_{1s}}$	Vertical force due to longitudinal cyclic pitch angle
Z_{δ_C}	Vertical force due to collective control
Z_q	Vertical force due to pitch rate
Z_{θ_C}	Vertical force due to collective pitch angle
Z_u	Vertical force due to longitudinal velocity
Z_w	Vertical force due to vertical velocity
$\ddot{z}$	Acceleration along z axis

GREEK SYMBOLS

α (alpha)	Angle of attack
α_{BE}	Blade element angle of attack
α_{eff}	Effective angle of attack
α_s	Angle of attack, shaft referenced
α_T	Tail angle of attack
$\dot{\alpha}$	Time rate of change of angle of attack
β (beta)	Flapping angle, Sideslip angle
$\ddot{\beta}$	Flapping angle acceleration

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$\dot{\beta}$	Flapping angle rate
δ (delta)	Control
δ_C	Collective control
δ_{LAT}	Lateral control
δ_{LONG}	Longitudinal control
δ_{PED}	Pedal control
δ_{th}	Throttle control
$\dot{\delta}$	Rate of change of control
ϕ (phi)	Roll angle
$\ddot{\phi}$	Roll angle acceleration
$\dot{\phi}$	Rate of change of roll angle
γ (gamma)	Flight path angle, Lock number
γ_0	Initial flight path angle
$\dot{\gamma}$	Time rate of change of flight path angle
φ (psi)	Inflow angle
λ (lambda)	Characteristic root, Inflow ratio
λ_s	Spiral mode root
λ_{TR}	Tail rotor inflow ratio
μ (mu)	Advance ratio, Amplitude ratio
π (pi)	Mathematical constant
θ (theta)	Blade pitch angle, Pitch angle, Feathering angle, Shaft angle, Pitch attitude
θ_C	Collective pitch angle
θ_{max}	Maximum pitch angle
θ_{TR}	Tail rotor pitch angle
$\ddot{\theta}$	Pitch acceleration

$\dot{\theta}$	Rate of change of pitch angle, Time rate of change in θ
$\dot{\theta}_{max}$	Maximum time rate of change in θ
$\dot{\theta}_{QS}$	Quasi-static rate of change of pitch angle
ρ (rho)	Density
τ (tau)	Time constant
τ_R	Roll mode time constant
τ_s	Spiral mode time constant
Ω (Omega)	Rotor angular velocity
ω (omega)	Frequency, Forcing frequency, Forcing frequency
Ω_0	Initial rotor angular velocity
ω_d	Damped frequency
ω_n	Natural frequency
$\dot{\Omega}$	Rotor angular acceleration
ψ (psi)	Blade azimuth angle, Phase angle, Yaw angle, Yaw attitude
$\ddot{\psi}$	Yaw attitude acceleration
$\dot{\psi}$	Rate of change of yaw attitude
ζ (zeta)	Damping ratio, Lead-lag angle

APPENDIX II

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APPENDIX II

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EQUATIONS

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Equation		Page
$m\ddot{x} + B\dot{x} + Kx = 0$	*eq 3.1*	3.7
$\lambda^2 + \frac{B}{m}\lambda + \frac{K}{m} = 0$	*eq 3.2*	3.7
$\lambda_{1,2} = -\frac{B}{2m} \pm \sqrt{\left(\frac{B}{2m}\right)^2 - \frac{K}{m}}$	*eq 3.3*	3.8
$x(t) = C_1 e^{\lambda_1 t} + C_2 e^{\lambda_2 t}$	*eq 3.4*	3.8
$x(t) = Ae^{-\zeta\omega_n t} \sin \omega_n \sqrt{1-\zeta^2}\, t$	*eq 3.5*	3.9
$B_{CRIT} = 2m\left(\frac{K}{m}\right)^{\frac{1}{2}}$	*eq 3.6*	3.10
$\omega_n = \left(\frac{K}{m}\right)^{\frac{1}{2}}$	*eq 3.7*	3.12
$\zeta = \frac{B}{B_{CRIT}}$	*eq 3.8*	3.13

$\lambda^2 + 2\,\omega_n\,\zeta\lambda + \omega_n^2 = 0$	*eq 3.9*	3.13
$\lambda_{1,2} = -\zeta\omega_n \pm i\,\omega_n \left(1 - \zeta^2\right)^{\frac{1}{2}}$	*eq 3.10*	3.13
$\zeta = \sin E_d$	*eq 3.11*	3.14
$\cos E_d = \sqrt{1 - \zeta^2}$	*eq 3.12*	3.14
$\omega_d = \omega_n \sqrt{1 - \zeta^2}$	*eq 3.13*	3.14
$\zeta\omega_n = \dfrac{\ln_e N}{T_{1/N}}$	*eq 3.14*	3.14
$\dot{w} - Z_w\, w = 0$	*eq 3.15*	3.15
$w = C_1\, e^{\lambda t}$	*eq 3.16*	3.16
$\tau = -\dfrac{1}{\lambda} = -\dfrac{1}{Z_w}$	*eq 3.17*	3.17
$C_{1/2} = \dfrac{T_{1/2}}{P}$	*eq 3.18*	3.23
$T_{1/N} = \dfrac{\ln_e N}{\zeta\omega_n}$	*eq 3.19*	3.23

$$C_{1/N} = \frac{\ln_e N}{2\pi \frac{\zeta}{\sqrt{1 - \zeta^2}}}$$

eq 3.20 3.24

CHAPTER 4

$$\theta = A_0 - A_1 \cos\psi - B_1 \sin\psi - A_2 \cos 2\psi - B_2 \sin 2\psi \ldots - A_n \cos n\psi - B_n \sin n\psi$$

eq 4.1 4.19

$$\theta = \theta_C - A_1 \cos\psi - B_1 \sin\psi$$

eq 4.2 4.20

$$a_1 = a_{1s} + B_{1s}$$

eq 4.3 4.20

$$b_1 = b_{1s} - A_{1s}$$

eq 4.4 4.20

$$A_0 = A_{0s} = \theta_C$$

eq 4.5 4.20

$$\beta = a_0 - a_{1s} \cos\psi - b_{1s} \sin\psi - a_{2s} \cos 2\psi - b_{2s} \sin 2\psi \ldots - a_{ns} \cos n\psi - b_{ns} \sin n\psi$$

eq 4.6 4.21

$$\Delta\ CF = \Omega^2\ r\ m\ \Delta r$$

eq 4.7 4.24

$$\Delta M = \Delta CF\ r \sin\beta$$

eq 4.8 4.25

$$M = \int_0^R \Omega^2\ r^2\ m\ \beta\ dr = \Omega^2\ \beta \int_0^R m\ r^2\ dr$$

eq 4.9 4.25

$$K = \frac{M}{\beta} = \Omega^2 \int_0^R m\ r^2\ dr$$

eq 4.10 4.25

$$I_f = \int_0^R m\, r^2\, dr$$ *eq 4.11* 4.25

$$\omega_n = \sqrt{\frac{K}{I}} = \sqrt{\frac{\Omega^2 I_f}{I_f}} = \Omega$$ *eq 4.12* 4.25

$$\frac{\omega}{\omega_n} = \frac{\omega}{\Omega} = 1$$ *eq 4.13* 4.25

$$\Delta M = m\, \Delta r\, \Omega^2\, (r' + e)\, r'\, \beta$$ *eq 4.14* 4.26

$$M = \Omega^2\, \beta \int_0^{R-e} m\, (r' + e)\, r'\, dr'$$ *eq 4.15* 4.26

$$I_f = \int_0^{R-e} m\, r'^2\, dr'$$ *eq 4.16* 4.26

$$\frac{M_f}{g} = \int_0^{R-e} m\, r'\, dr'$$ *eq 4.17* 4.27

$$M = \Omega^2\, \beta\, (I_f + e\, \frac{M_f}{g})$$ *eq 4.18* 4.27

$$K = \frac{M}{\beta} = \Omega^2 \left(I_f + e\, \frac{M_f}{g} \right)$$ *eq 4.19* 4.27

$$I_f = m \int_0^{R-e} r'^2\, dr = \frac{mR^3}{3} \left(1 - \frac{e}{R} \right)^3$$ *eq 4.20* 4.27

$\frac{M_f}{g} = m \int_0^{R-e} r' \, dr = \frac{mR^2}{2}\left(1 - \frac{e}{R}\right)^2$	*eq 4.21*	4.27
$\omega_n = \sqrt{\frac{K}{I_f}} = \Omega \sqrt{1 + \frac{eM_f}{g\, I_f}} = \Omega \sqrt{1 + \frac{\frac{3}{2}\frac{e}{R}}{1 - \frac{e}{R}}}$	*eq 4.22*	4.27
$\frac{\omega}{\omega_n} = \frac{\omega}{\Omega} = \sqrt{1 + \frac{\frac{3}{2}\frac{e}{R}}{1 - \frac{e}{R}}}$	*eq 4.23*	4.27
$I_f \ddot{\beta} + B\dot{\beta} + K\beta = 0$	*eq 4.24*	4.28
$\beta = -\frac{B}{2I_f} \pm \sqrt{\left(\frac{B}{2I_f}\right)^2 - \frac{K}{I_f}}$	*eq 4.25*	4.29
$B_{CRIT} = 2I_f \left(\frac{K}{I_f}\right)^{\frac{1}{2}}$	*eq 4.26*	4.29
$\zeta = \frac{B}{B_{CRIT}}$	*eq 4.27*	4.29
$M_A = \int_0^{R-e} r' \frac{\rho}{2} a\, c\, r' \dot{\beta}\, (r' + e)\, \Omega \, dr$	*eq 4.28*	4.29
$B = \frac{2M_A}{\partial\dot{\beta}} = \frac{c\rho a R^4}{8}\left(1 - \frac{e}{R}\right)^4 \Omega \left(\frac{1 + \frac{e}{3R}}{1 - \frac{e}{R}}\right)$	*eq 4.29*	4.29

$$\zeta = \frac{\gamma}{16}\left(1 - \frac{e}{R}\right)^4 \left(\frac{1 + \frac{e}{3R}}{1 - \frac{e}{R}}\right) \frac{1}{\omega_n / \Omega}$$

eq 4.30 4.30

$$\gamma = \frac{c\rho a R^4}{I_f}$$

eq 4.31 4.30

$$\psi = \tan^{-1}\left[\frac{2\zeta\left(\frac{\omega}{\omega_n}\right)}{1 - \left(\frac{\omega}{\omega_n}\right)^2}\right]$$

eq 4.32 4.31

$$\mu = \frac{1}{\sqrt{\left[\left(1 - \left(\frac{\omega}{\omega_n}\right)^2\right)^2 + 4\,\zeta^2\left(\frac{\omega}{\omega_n}\right)^2\right]}}$$

eq 4.33 4.32

$$\frac{b_{1s}}{a_{1s}} = -\cot\psi$$

eq 4.34 4.34

$$\frac{b_{1s}}{a_{1s}} = -\frac{\frac{12}{\gamma}\frac{e}{R}}{\left[1 + \frac{e}{3R}\right]}$$

eq 4.35 4.34

$$a_{1s} = -\left(\frac{16}{\gamma}\right)\left(\frac{q}{\Omega}\right)$$

eq 4.36 4.36

$$b_{1s} = -\left(\frac{q}{\Omega}\right)$$

eq 4.37 4.36

$$a_{1s} = \frac{\dfrac{-\dfrac{16}{\gamma}\left(\dfrac{q}{\Omega}\right)}{\left(1 - \dfrac{e}{R}\right)^2} + \left(\dfrac{p}{\Omega}\right)}{1 - \dfrac{\mu^2}{2}} + \frac{\dfrac{\dfrac{12}{\gamma}\,\dfrac{e}{R}}{\left(1 - \dfrac{e}{R}\right)^3}\left[\dfrac{-\dfrac{16}{\gamma}\left(\dfrac{p}{\Omega}\right)}{\left(1 - \dfrac{e}{R}\right)^2} - \left(\dfrac{q}{\Omega}\right)\right]}{1 - \dfrac{\mu^4}{4}}$$

eq 4.38 4.36

$$b_{1s} = \frac{\dfrac{-\dfrac{16}{\gamma}\left(\dfrac{p}{\Omega}\right)}{\left(1 - \dfrac{e}{R}\right)^2} + \left(\dfrac{q}{\Omega}\right)}{1 - \dfrac{\mu^2}{2}} + \frac{\dfrac{\dfrac{12}{\gamma}\,\dfrac{e}{R}}{\left(1 - \dfrac{e}{R}\right)^3}\left[\dfrac{-\dfrac{16}{\gamma}\left(\dfrac{q}{\Omega}\right)}{\left(1 - \dfrac{e}{R}\right)^2} - \left(\dfrac{p}{\Omega}\right)\right]}{1 - \dfrac{\mu^4}{4}}$$

eq 4.39 4.36

$$\mu = \frac{V}{\Omega R}$$

eq 4.40 4.36

$$\beta = a_0 - a_{1s} \cos \psi - b_{1s} \sin \psi$$

eq 4.41 4.37

$$a_0 = \frac{\gamma}{8}\left[\theta_C \left(1 + \mu^2\right) + \frac{4}{3}\lambda\right] - \frac{M_W}{I\Omega^2}$$

eq 4.42 4.38

$$a_{1s} = -B_{1s}\,\frac{1 + \frac{3}{2}\mu^2}{1 - \dfrac{\mu^2}{2}} + \frac{\frac{8}{3}\theta_C\,\mu + 2\,\lambda\,\mu}{1 - \dfrac{\mu^2}{2}}$$

eq 4.43 4.38

$$b_{1s} = A_{1s} + \frac{\frac{4}{3}\mu\, a_0}{1 + \dfrac{\mu^2}{2}}$$

eq 4.44 4.38

$$\lambda = \frac{V \sin \alpha_s - v_i}{\Omega R}$$

eq 4.45 4.38

$$-F_{S_{\text{hub on blade}}} + \int_e^R dT \cos\beta = \int_e^R \ddot{\beta}\,(r - e)\,dm$$

eq 4.46 4.45

$$F_{S_{\text{blade on hub}}} = T_b - \Omega^2 M_S \left(a_{1s} \cos \Omega t + b_{1s} \sin \Omega t\right)$$

eq 4.47 4.45

$$M_H(\psi) = -F_{S_{\text{ blade on hub}}} \left(e \cos \psi\right)$$

eq 4.48 4.46

$$L_H(\psi) = -F_{S_{\text{blade on hub}}} \left(e \sin \psi\right)$$

eq 4.49 4.46

$$\overline{M_H} = \frac{b}{2\pi} \int_0^{2\pi} M_H(\psi)\,d\psi$$

eq 4.50 4.46

$$\overline{L_H} = \frac{b}{2\pi} \int_0^{2\pi} L_H(\psi)\,d\psi$$

eq 4.51 4.46

$$\overline{M_H} = \frac{1}{2} eb\,\Omega^2 M_S\,a_{1s}$$

eq 4.52 4.46

$$\overline{L_H} = \frac{1}{2} eb\,\Omega^2 M_S b_{1s}$$

eq 4.53 4.46

$$\sum M_{CG} = Th \sin\ a_{1s} + \overline{M_H} = Th + \frac{1}{2} eb\,\Omega^2 M_S\,a_{1s}$$

eq 4.54 4.48

$$H = T \sin a_{1s} \cos \varphi$$

eq 4.55 4.50

CHAPTER 6

$$m\left[\left(-\Delta\dot{u} - w_0\Delta q\right) + \left(X_u\Delta u + X_w\Delta w\right) + \left(X_q\Delta q\right) + \left(-g\cos\gamma_0\Delta\theta\right)\right]$$
$$= -m\left[X_{B_{1s}}\Delta B_{1s} + X_{\theta_C}\Delta\theta_C\right] = -m\left[X_{\delta_{Long}}\Delta\delta_{Long} + X_{\delta_C}\Delta\delta_C\right]$$

eq 6.1 6.4

$$m\left[\left(-\Delta\dot{w} - u_0\Delta q\right) + \left(Z_u\Delta u + Z_w\Delta w\right) + \left(Z_q\Delta q\right) + \left(-g\sin\gamma_0\Delta\theta\right)\right]$$
$$= -m\left[Z_{B_{1s}}\Delta B_{1s} + Z_{\theta_C}\Delta\theta_C\right] = -m\left[Z_{\delta_{Long}}\Delta\delta_{Long} + Z_{\delta_C}\Delta\delta_C\right]$$

eq 6.2 6.4

$$I_{yy}\left[\left(-\Delta q\right) + \left(M_u\Delta u + M_w\Delta w\right) + \left(M_q\Delta q + M_w\Delta w\right)\right]$$
$$= -I_{yy}\left[M_{B_{1s}}\Delta B_{1s} + M_{\theta_C}\Delta\theta_C\right] = -I_{yy}\left[M_{\delta_{Long}}\Delta\delta_{Long} + M_{\delta_C}\Delta\delta_C\right]$$

eq 6.3 6.4

$$\Delta M = \left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\Delta a_{1s} + \left(h' + ha_{1s}\right)\Delta T + h\Delta H + \Delta M_{CG_{f+t}}\right]$$

eq 6.4 6.7

$$M_{B_{1s}} = \frac{1}{I_{yy}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial a_{1s}}{\partial B_{1s}} + \left(h' + ha_{1s}\right)\frac{\partial T}{\partial B_{1s}} + h\,\frac{\partial H}{\partial B_{1s}} + \frac{\partial M_{CG_{f+t}}}{\partial B_{1s}}\right]$$

eq 6.5 6.9

$$M_u = \frac{1}{I_{yy}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial a_{1s}}{\partial u} + \left(h' + ha_{1s}\right)\frac{\partial T}{\partial u} + h\,\frac{\partial H}{\partial u} + \frac{\partial M_{CG_{f+t}}}{\partial u}\right]$$

eq 6.6 6.11

$$M_q = \frac{1}{I_{yy}} \left[\left(Th + \frac{ebM_S\Omega^2}{2} \right) \frac{\partial a_{1s}}{\partial q} + \left(h' + ha_{1s} \right) \frac{\partial T}{\partial q} + h \frac{\partial H}{\partial q} + \frac{\partial M_{CG_{f+t}}}{\partial q} \right]$$

eq 6.7 6.13

$$M_w = \frac{1}{I_{yy}} \left[\left(Th + \frac{ebM_S\Omega^2}{2} \right) \frac{\partial a_{1s}}{\partial w} + \left(h' + ha_{1s} \right) \frac{\partial T}{\partial w} + h \frac{\partial H}{\partial w} + \frac{\partial M_{CG_{f+t}}}{\partial w} \right]$$

eq 6.8 6.15

$$\begin{bmatrix} X_u & X_w & -g & 0 \\ Z_u & Z_w & 0 & 0 \\ M_u & M_w & 0 & 0 \\ 0 & -\frac{1}{u_0} & 1 & -1 \end{bmatrix} \begin{bmatrix} \Delta u \\ \Delta w \\ \Delta\theta \\ \Delta\gamma \end{bmatrix} = - \begin{bmatrix} X_{B_{1s}} \\ Z_{B_{1s}} \\ M_{B_{1s}} \\ 0 \end{bmatrix} \Delta B_{1s} - \begin{bmatrix} X_{\theta_C} \\ Z_{\theta_C} \\ M_{\theta_C} \\ 0 \end{bmatrix} \Delta\theta_C$$

eq 6.9 6.26

$$\begin{bmatrix} X_{B_{1s}} & X_w & -g & 0 \\ Z_{B_{1s}} & Z_w & 0 & 0 \\ M_{B_{1s}} & M_w & 0 & 0 \\ 0 & -\frac{1}{u_0} & 1 & -1 \end{bmatrix} \begin{bmatrix} \Delta B_{1s} \\ \Delta w \\ \Delta\theta \\ \Delta\gamma \end{bmatrix} = - \begin{bmatrix} X_u \\ Z_u \\ M_u \\ 0 \end{bmatrix} \Delta u - \begin{bmatrix} X_{\theta_C} \\ Z_{\theta_C} \\ M_{\theta_C} \\ 0 \end{bmatrix} \Delta\theta_C$$

eq 6.10 6.26

$$\frac{\Delta B_{1s}}{\Delta u} \; \frac{\left[-g \left(Z_w M_u - M_w Z_u \right) \right]}{g \left(Z_w M_{B_{1s}} - M_w Z_{B_{1s}} \right)} = + \frac{M_u - M_w \frac{Z_u}{Z_w}}{-M_{B_{1s}} + M_w \frac{Z_{B_{1s}}}{Z_w}}$$

eq 6.11 6.28

$$\Delta M = \frac{\partial M}{\partial u} \Delta u + \frac{\partial M}{\partial w} \Delta w + \frac{\partial M}{\partial B_{1s}} \Delta B_{1s}$$

eq 6.12 6.28

$$\Delta M = I_{yy}\left[M_u \, \Delta u + M_w \, \Delta w + M_{B_{1s}} \, \Delta B_{1s} \right]$$ *eq 6.13* 6.29

$$\Delta w = -\frac{Z_u}{Z_w}\Delta u - \frac{Z_{B_{1s}}}{Z_w}\Delta B_{1s}$$ *eq 6.14* 6.29

$$\Delta M = 0 = I_{yy}\left(M_u - \frac{Z_u}{Z_w} M_w \right)\Delta u + I_{yy}\left(M_{B_{1s}} - \frac{Z_{B_{1s}}}{Z_w} M_w \right)\Delta B_{1s}$$ *eq 6.15* 6.29

$$\text{Static stability} = I_{yy}\left(M_u - \frac{Z_u}{Z_w} M_w \right)$$ *eq 6.16* 6.30

$$\left[X_u \, \Delta u + X_w \, \Delta w \right] + X_q \Delta q - g\Delta\theta + \left[X_{B_{1s}} \, \Delta B_{1s} + X_{\theta_C} \, \Delta\theta_C \right] = 0$$ *eq 6.17* 6.39

$$u_0 \, \Delta q + \left[Z_u \, \Delta u + Z_w \, \Delta w \right] + Z_q \Delta q + \left[Z_{B_{1s}} \, \Delta B_{1s} + Z_{\theta_C} \, \Delta\theta_C \right] = 0$$ *eq 6.18* 6.40

$$\left[M_u \, \Delta u + M_w \, \Delta w \right] + M_q \Delta q + \left[M_{B_{1s}} \, \Delta B_{1s} + M_{\theta_C} \, \Delta\theta_C \right] = 0$$ *eq 6.19* 6.40

$$-\frac{\Delta w}{u_0} + \Delta\theta - \Delta\gamma = 0$$ *eq 6.20* 6.40

$$g\,(\Delta n) + Z_w \, \Delta w + Z_{B_{1s}} \, \Delta B_{1s} = 0$$ *eq 6.21* 6.42

$$M_w \Delta w + M_{B_{1s}} \Delta B_{1s} + Mq \;\; g \; \frac{\Delta n}{u_0}$$ *eq 6.22* 6.42

$$q = g\,\frac{\Delta n}{u_0}$$

eq 6.23 6.42

$$\frac{\Delta B_{1s}}{\Delta q} = \frac{M_q\,Z_w - u_0\,M_w}{M_w\,Z_{B_{1s}} - Z_w\,M_{B_{1s}}}$$

eq 6.24 6.43

$$\frac{\Delta B_{1s}}{\Delta n} = \frac{g}{u_0}\,\frac{\left(M_q\,Z_w - u_0 M_w\right)}{\left(M_w\,Z_{B_{1s}} - Z_w\,M_{B_{1s}}\right)}$$

eq 6.25 6.43

$$q = \frac{g}{u_0}\left(n - \frac{1}{n}\right)$$

eq 6.26 6.44

$$g\Delta\left(n - \frac{1}{n}\right) + Z_w\,\Delta w + Z_{B_{1s}}\,\Delta B_{1s} = 0$$

eq 6.27 6.45

$$M_w\,\Delta w + M_{B_{1s}}\,\Delta_{B_{1s}} + M_q\,\frac{g}{u_0}\left(n - \frac{1}{n}\right) = 0$$

eq 6.28 6.45

$$\frac{dq}{dn} = \frac{g}{u_0}\left(1 + \frac{1}{n^2}\right) \quad \text{for coordinated turns}$$

eq 6.29 6.45

$$\frac{dq}{dn} = \frac{g}{u_0} \quad \text{for a symmetric pull up}$$

eq 6.30 6.45

$$\frac{\Delta B_{1s}}{\Delta n} = \frac{g}{u_0}\,\frac{\left(M_q\,Z_w - u_0 M_w\right)}{\left(M_w\,Z_{B_{1s}} - Z_w\,M_{B_{1s}}\right)}\left(1 + \frac{1}{n^2}\right)$$

eq 6.31 6.45

$$\omega_n = \left(\frac{K}{m}\right)^{\frac{1}{2}} = \left[-g \frac{\left(M_u - \frac{Z_u}{Z_w} M_w\right)}{\left(M_q - \frac{u_0}{Z_w} M_w\right)} \right]^{\frac{1}{2}}$$

eq 6.32 6.48

$$\Delta n = \frac{1}{g} u_0 \dot{\gamma} = -\frac{1}{g}\left(\dot{w} - u_0 q\right) = -\frac{1}{g} u_0 \left(\dot{\alpha} - \dot{\theta}\right)$$

eq 6.33 6.51

$$\ddot{q} + \left(-Z_w - M_q\right)\dot{q} + \left(Z_w M_q - u_0 M_w\right)q = M_{B_{1s}} \dot{B}_{1s} + \left(M_w Z_{B_{1s}} - Z_w M_{B_{1s}}\right)B_{1s} = F_q$$

eq 6.34 6.53

$$\lambda_{1,2} = -\left(\frac{-Z_w - M_q}{2}\right) \pm \left[\left(\frac{-Z_w - M_q}{2}\right)^2 - \left(Z_w M_q - u_0 M_w\right)\right]^{\frac{1}{2}}$$

eq 6.35 6.53

CHAPTER 7

$$\dot{v} - Y_v \Delta v - Y_r \Delta r + u_0 \Delta r - Y_p \Delta p - g\phi = Y_{A_{1s}} \Delta A_{1s} + Y_{\theta_{TR}} \Delta\theta_{TR}$$
$$= Y_{\delta_{LAT}} \Delta\delta_{LAT} + Y_{\delta_{PED}} \Delta\delta_{PED}$$

eq 7.1 7.4

$$\dot{p} - L_v \Delta v - L_r \Delta r - L_p \Delta p = L_{A_{1s}} \Delta A_{1s} + L_{\theta_{TR}} \Delta\theta_{TR}$$
$$= L_{\delta_{LAT}} \Delta\delta_{LAT} + L_{\delta_{PED}} \Delta\delta_{PED}$$

eq 7.2 7.5

$$\dot{r} - N_v \Delta v - N_p \Delta p - N_r \Delta r = N_{A_{1s}} \Delta A_{1s} + N_{\theta_{TR}} \Delta\theta_{TR}$$
$$= N_{\delta_{LAT}} \Delta\delta_{LAT} + N_{\delta_{PED}} \Delta\delta_{PED}$$

eq 7.3 7.5

$$\Delta Y = T\,\Delta b_{1s} + b_{1s}\,\Delta T + \Delta Y_R + \Delta T_{TR} + \Delta Y_{f+t}$$ *eq 7.4* 7.9

$$\Delta L = (Th + \frac{1}{2}\,ebM_S\Omega^2)\Delta b_{1s} + hb_{1s}\Delta T + h\Delta Y_R + h_{TR}\Delta T_{TR} + \Delta L_{f+t}$$ *eq 7.5* 7.9

$$\Delta N = Th'\Delta b_{1s} + h'b_{1s}\Delta T + h'\Delta Y_R - l_t\Delta T_{TR} + \Delta N_{f+t} + \Delta Q_{MR}$$ *eq 7.6* 7.9

$$N_v = \frac{1}{I_{zz}}\left[Th'\frac{\partial b_{1s}}{\partial v} + h'b_{1s}\frac{\partial T}{\partial v} + h'\frac{\partial Y_R}{\partial v} - l_t\frac{\partial T_{TR}}{\partial v} + \frac{\partial N_{f+t}}{\partial v} + \frac{\partial Q_{MR}}{\partial v}\right]$$ *eq 7.7* 7.11

$$N_v \approx \frac{1}{I_{zz}}\left[-l_t\frac{\partial T_{TR}}{\partial v} + \frac{\partial N_{f+t}}{\partial v}\right]$$ *eq 7.8* 7.11

$$N_r = \frac{1}{I_{zz}}\left[Th'\frac{\partial b_{1s}}{\partial r} + h'\,b_{1s}\frac{\partial T}{\partial r} + h'\frac{\partial Y_R}{\partial r} - l_t\frac{\partial T_{TR}}{\partial r} + \frac{\partial N_{f+t}}{\partial r} + \frac{\partial Q_{MR}}{\partial r}\right]$$ *eq 7.9* 7.14

$$N_r \approx \frac{1}{I_{zz}}\left[-l_t\frac{\partial T_{TR}}{\partial r} + \frac{\partial N_{f+t}}{\partial r}\right]$$ *eq 7.10* 7.15

$$N_p = \frac{1}{I_{zz}}\left[Th'\frac{\partial b_{1s}}{\partial p} + h'b_{1s}\frac{\partial T}{\partial p} + h'\frac{\partial Y_R}{\partial p} - l_t\frac{\partial T_{TR}}{\partial p} + \frac{\partial N_{f+t}}{\partial p} + \frac{\partial Q_{MR}}{\partial p}\right]$$ *eq 7.11* 7.16

$$N_{\theta_{TR}} = \frac{1}{I_{zz}}\left[Th'\frac{\partial b_{1s}}{\partial \theta_{TR}} + h'b_{1s}\frac{\partial T}{\partial \theta_{TR}} + h'\frac{\partial Y_R}{\partial \theta_{TR}} - l_t\frac{\partial T_{TR}}{\partial \theta_{TR}} + \frac{\partial N_{f+t}}{\partial \theta_{TR}} + \frac{\partial Q_{MR}}{\partial \theta_{TR}}\right]$$ *eq 7.12* 7.18

$$N_{\theta_{TR}} \approx \frac{1}{I_{zz}}\left[\ -l_t \frac{\partial T_{TR}}{\partial \theta_{TR}}\ \right]$$

eq 7.13 7.18

$$N_{\delta_{PED}} = \left(\frac{\partial \theta_{TR}}{\partial \delta_{PED}}\right) N_{\theta_{TR}}$$

eq 7.14 7.18

$$N_{A_{1s}} = \frac{1}{I_{zz}}\left[\ Th' \frac{\partial b_{1s}}{\partial A_{1s}} + h'b_{1s} \frac{\partial T}{\partial A_{1s}} + h' \frac{\partial Y_R}{\partial A_{1s}} - l_t \frac{\partial T_{TR}}{\partial A_{1s}} + \frac{\partial N_{f+t}}{\partial A_{1s}} + \frac{\partial Q_{MR}}{\partial A_{1s}}\right]$$

eq 7.15 7.20

$$N_{\delta_{LAT}} = \left(\frac{\partial A_{1s}}{\partial \delta_{LAT}}\right) N_{A_{1s}}$$

eq 7.16 7.21

$$L_v = \frac{1}{I_{xx}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial b_{1s}}{\partial v} + hb_{1s}\frac{\partial T}{\partial v} + h\frac{\partial Y_R}{\partial v} + h_{TR}\frac{\partial T_{TR}}{\partial v} + \frac{\partial L_{f+t}}{\partial v}\ \right]$$

eq 7.17 7.22

$$L_r = \frac{1}{I_{xx}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial b_{1s}}{\partial r} + hb_{1s}\frac{\partial T}{\partial r} + h\frac{\partial Y_R}{\partial r} + h_{TR}\frac{\partial T_{TR}}{\partial r} + \frac{\partial L_{f+t}}{\partial r}\ \right]$$

eq 7.18 7.24

$$L_p = \frac{1}{I_{xx}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial b_{1s}}{\partial p} + hb_{1s}\frac{\partial T}{\partial p} + h\frac{\partial Y_R}{\partial p} + h_{TR}\frac{\partial T_{TR}}{\partial p} + \frac{\partial L_{f+t}}{\partial p}\ \right]$$

eq 7.19 7.26

$$L_p \approx \frac{1}{I_{xx}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial b_{1s}}{\partial p} + h_{TR}\frac{\partial T_{TR}}{\partial p}\ \right]$$

eq 7. 20 7.26

$$L_{A_{1s}} = \frac{1}{I_{xx}}\left[\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial b_{1s}}{\partial A_{1s}} + hb_{1s}\frac{\partial T}{\partial A_{1s}} + h\frac{\partial Y_R}{\partial A_{1s}} + h_{TR}\frac{\partial T_{TR}}{\partial A_{1s}} + \frac{\partial L_{f+t}}{\partial A_{1s}}\right]$$

eq 7.21 7.28

$$L_{\delta_{LAT}} = \left(\frac{\partial A_{1s}}{\partial \delta_{LAT}}\right)L_{A_{1s}}$$

eq 7.22 7.29

$$L_{\theta_{TR}} = \frac{1}{I_{xx}}\left[h_{TR}\frac{\partial T_{TR}}{\partial \theta_{TR}}\right]$$

eq 7.23 7.30

$$L_{\delta_{PED}} = \left(\frac{\partial \theta_{TR}}{\partial \delta_{PED}}\right)L_{\theta_{TR}}$$

eq 7.24 7.30

$$Y_v = \frac{1}{m}\left[T\frac{\partial b_{1s}}{\partial v} + b_{1s}\frac{\partial T}{\partial v} + \frac{\partial Y_R}{\partial v} + \frac{\partial T_{TR}}{\partial v} + \frac{\partial Y_{f+t}}{\partial v}\right]$$

eq 7.25 7.32

$$Y_r = \frac{1}{m}\left[T\frac{\partial b_{1s}}{\partial r} + b_{1s}\frac{\partial T}{\partial r} + \frac{\partial Y_R}{\partial r} + \frac{\partial T_{TR}}{\partial r} + \frac{\partial Y_{f+t}}{\partial r}\right]$$

eq 7.26 7.33

$$Y_p = \frac{1}{m}\left[T\frac{\partial b_{1s}}{\partial p} + b_{1s}\frac{\partial T}{\partial p} + \frac{\partial Y_R}{\partial p} + \frac{\partial T_{TR}}{\partial p} + \frac{\partial Y_{f+t}}{\partial p}\right]$$

eq 7.27 7.35

$$Y_p \approx \frac{1}{m}\left[T\frac{\partial b_{1s}}{\partial p} + \frac{\partial T_{TR}}{\partial p}\right]$$

eq 7.28 7.35

$$Y_{A_{1s}} = \frac{1}{m}\left[T\frac{\partial b_{1s}}{\partial A_{1s}} + b_{1s}\frac{\partial T}{\partial A_{1s}}\right]$$

eq 7.29 7.36

$$Y_{A_{1s}} \approx g \left[\frac{\partial b_{1s}}{\partial A_{1s}} \right]$$ *eq 7.30* 7.36

$$Y_{\delta_{LAT}} = \left(\frac{\partial A_{1s}}{\partial \delta_{LAT}} \right) Y_{A_{1s}}$$ *eq 7.31* 7.37

$$Y_{\theta_{TR}} = \frac{1}{m} \left[\frac{\partial T_{TR}}{\partial \theta_{TR}} \right]$$ *eq 7.32* 7.38

$$Y_{\delta_{PED}} = \left(\frac{\partial \theta_{TR}}{\partial \delta_{PED}} \right) Y_{\theta_{TR}}$$ *eq 7.33* 7.38

$$-Y_v u_0 \Delta\beta \; -g\Delta\phi = Y_{A_{1s}} \Delta A_{1s} + Y_{\theta_{TR}} \Delta\theta_{TR} = Y_{\delta_{LAT}} \Delta\delta_{LAT} + Y_{\delta_{PED}} \Delta\delta_{PED}$$ *eq 7.34* 7.46

$$-L_v u_0 \Delta\beta = L_{A_{1s}} \Delta A_{1s} + L_{\theta_{TR}} \Delta\theta_{TR} = L_{\delta_{LAT}} \Delta\delta_{LAT} + L_{\delta_{PED}} \Delta\delta_{PED}$$ *eq 7.35* 7.46

$$-N_v u_0 \Delta\beta = N_{A_{1s}} \Delta A_{1s} + N_{\theta_{TR}} \Delta\theta_{TR} = N_{\delta_{LAT}} \Delta\delta_{LAT} + N_{\delta_{PED}} \Delta\delta_{PED}$$ *eq 7.36* 7.46

$$\frac{\Delta\delta_{PED}}{\Delta\beta} = -u_0 \left(\frac{L_{\delta_{LAT}} N_v - L_v N_{\delta_{LAT}}}{L_{\delta_{LAT}} N_{\delta_{PED}} - L_{\delta_{PED}} N_{\delta_{LAT}}} \right)$$ *eq 7.37* 7.50

$$\frac{\Delta\delta_{PED}}{\Delta\beta} \approx \left(-\frac{u_0}{N_{\delta_{PED}}} \right) N_v$$ *eq 7.38* 7.50

$$\frac{\Delta\delta_{LAT}}{\Delta\beta} = -u_0\left(\frac{N_{\delta_{PED}} L_v - N_v L_{\delta_{PED}}}{L_{\delta_{LAT}} N_{\delta_{PED}} - L_{\delta_{PED}} N_{\delta_{LAT}}}\right)$$

eq 7.39 7.50

$$\frac{\Delta\delta_{LAT}}{\Delta\beta} \approx \frac{u_0}{L_{\delta_{LAT}}}\left(L_v - L_{\delta_{PED}}\frac{N_v}{N_{\delta_{PED}}}\right)$$

eq 7.40 7.50

$$\frac{\Delta\delta_{LAT}}{\Delta\beta} = \left(-\frac{u_0 L_v}{L_{\delta_{LAT}}}\right) - \left[\frac{L_{\delta_{PED}}}{L_{\delta_{LAT}}}\left(\frac{\Delta\delta_{PED}}{\Delta\beta}\right)\right]$$

eq 7.41 7.50

$$L_{\delta_{PED}} \approx \frac{1}{I_{xx}} h_{TR}\left(\frac{\partial T_{TR}}{\partial\delta_{PED}}\right)$$

eq 7.42 7.52

$$-Y_v u_0\Delta\beta + (u_0 - Y_r)\Delta r - g\phi = Y_{A_{1s}}\Delta A_{1s} + Y_{\theta_{TR}}\Delta\theta_{TR} = Y_{\delta_{LAT}}\Delta\delta_{LAT} + Y_{\delta_{PED}}\Delta\delta_{PED}$$

eq 7.43 7.56

$$-L_v u_0\Delta\beta - L_r\Delta r = L_{A_{1s}}\Delta A_{1s} + L_{\theta_{TR}}\Delta\theta_{TR} = L_{\delta_{LAT}}\Delta\delta_{LAT} + L_{\delta_{PED}}\Delta\delta_{PED}$$

eq 7.44 7.56

$$-N_v u_0\Delta\beta - N_r\Delta r = N_{A_{1s}}\Delta A_{1s} + N_{\theta_{TR}}\Delta\theta_{TR} = N_{\delta_{LAT}}\Delta\delta_{LAT} + N_{\delta_{PED}}\Delta\delta_{PED}$$

eq 7.45 7.56

$$L_v u_0\Delta\beta + L_r\frac{\Delta r}{u_0\Delta\beta}u_0\Delta\beta + L_{\delta_{LAT}}\Delta\delta_{LAT} = 0$$

eq 7.46 7.60

$$L_v u_0\Delta\beta + L_r\frac{N_v}{(-N_r)}u_0\Delta\beta + L_{\delta_{LAT}}\Delta\delta_{LAT} = 0$$

eq 7.47 7.61

$$\frac{\Delta\beta}{\Delta\delta_{LAT}} = \frac{1}{u_0} \frac{L_{\delta_{LAT}}}{\left[\left(-L_v\right) - L_r \frac{N_v}{\left(-N_r\right)}\right]}$$ *eq 7.48* 7.62

$$\frac{\Delta\beta}{\Delta\delta_{LAT}} = \frac{1}{u_0} \frac{\left(-N_r\right) L_{\delta_{LAT}}}{\left(L_v N_r - L_r N_v\right)}$$ *eq 7.49* 7.63

$$\frac{\Delta r}{\Delta\delta_{LAT}} = \frac{\Delta r}{u_0\Delta\beta} \quad \frac{u_0\Delta\beta}{\Delta\delta_{LAT}} = \frac{N_v L_{\delta_{LAT}}}{\left(L_v N_r - L_r N_v\right)}$$ *eq 7.50* 7.63

$$L_v\, u_o\Delta\beta \quad + \quad L_r\Delta r \quad + \quad L_{\delta_{PED}}\Delta\delta_{PED} \quad = \quad 0$$ *eq 7.51* 7.64

$$N_v u_0\Delta\beta \quad + \quad N_r\, \Delta r \;+ \quad N_{\delta_{PED}}\Delta\delta_{PED} \quad = \quad 0$$ *eq 7.52* 7.64

$$\frac{\left(u_0\Delta\beta\right)}{\Delta r} = \frac{L_r}{\left(-L_v\right)}$$ *eq 7.53* 7.65

$$N_v \frac{L_r}{\left(-L_v\right)} \Delta r \quad - \left(-N_r\right) \Delta r + N_{\delta_{PED}}\Delta\delta_{PED} \quad = \quad 0$$ *eq 7.54* 7.66

$$\frac{\Delta r}{\Delta\delta_{PED}} \quad = \quad \frac{\left(-L_v\right) N_{\delta_{PED}}}{L_v N_r - L_r N_v}$$ *eq 7.55* 7.66

$$\frac{u_0\Delta\beta}{\Delta\delta_{PED}} \quad = \quad \frac{L_r N_{\delta_{PED}}}{L_v N_r - L_r N_v}$$ *eq 7.56* 7.66

$$\frac{\Delta r}{\Delta \delta_{PED}} = \frac{\left(-L_v\right) N_{\delta_{PED}} \left[1 - \left(\frac{L_{\delta_{PED}}}{N_{\delta_{PED}}} \right) \left(\frac{N_v}{L_v} \right) \right]}{L_v N_r - L_r N_v}$$ *eq 7.57* 7.69

$$\frac{u_0 \Delta \beta}{\Delta \delta_{PED}} = \frac{L_r N_{\delta_{PED}} \left[1 - \left(\frac{L_{\delta_{PED}}}{N_{\delta_{PED}}} \right) \left(\frac{N_r}{L_r} \right) \right]}{L_v N_r - L_r N_v}$$ *eq 7.58* 7.69

$$W_{BALL} \sin\phi = \frac{W_{BALL}}{g} u_0 \omega \cos\phi = \frac{W_{BALL}}{g} u_0 r$$ *eq 7.59* 7.71

$$\phi \approx \sin\phi = \frac{u_0 r}{g}$$ *eq 7.60* 7.71

$$p = \frac{\left(\frac{\Delta L}{I_{xx}} \right)}{\left(-L_p\right)} \left(1 - e^{-\frac{t}{\tau_R}} \right)$$ *eq 7.61* 7.72

$$\lambda_s = -\frac{1}{\tau_s} = \frac{-\left(L_v N_r - L_r N_v\right)}{-L_v \left(1 - \frac{u_0}{g} N_p \right) - N_v L_p \left(\frac{u_0}{g} \right)}$$ *eq 7.62* 7.75

$$r = r_0 \, e^{\lambda_s t} = r_0 \, e^{-\frac{t}{\tau_s}}$$ *eq 7.63* 7.75

$$\Delta \ddot{\psi} + \left(-N_r\right) \Delta \dot{\psi} + \left(N_v u_0\right) \Delta \psi = 0$$ *eq 7.64* 7.77

$$\frac{\left| \overline{\Delta\phi} \right|}{\left| -\overline{\Delta\beta} \right|} = \frac{\sqrt{(1)^2 + (2)^2}\left(\left| L_v u_0 \right| \right)}{\sqrt{\omega_n^4 + L_p^2 \omega_n^2}}$$ *eq 7.65* 7.79

$$\angle \frac{\Delta\phi}{(-\Delta\beta)} = \tan^{-1}\left[\frac{-(-L_p)}{-\omega_n} \right]$$ *eq 7.66* 7.80

$$\angle \frac{\Delta\phi}{(-\Delta\beta)} = \tan^{-1} \frac{-\left[\frac{(-L_p)}{\omega_n} - 2\zeta \right]}{-\left[1 + \zeta \frac{(-L_p)}{\omega_n} \right]}$$ *eq 7.67* 7.81

$$\zeta = \frac{1}{2\omega_n}\left[-N_r - Y_v - \left(\frac{g}{u_0} - N_p \right) \frac{L_v u_0}{\left(\omega_n^2 + L_p^2 \right)} \right]$$ *eq 7.68* 7.81

$$\zeta = \left(-\frac{1}{2} \right)\left(\sqrt{\frac{-gL_v}{\left| L_p \right|^3}} \right)$$ *eq 7.69* 7.89

$$\Delta t = t_2 - t_1 = t_3 - t_2$$ *eq 7.70* 7.110

$$A_1 = p_2 - p_1$$ *eq 7.71* 7.110

$$A_2 = p_3 - p_2$$ *eq 7.72* 7.111

$$\tau = \frac{\Delta t}{\ln_e\left(\frac{A_1}{A_2}\right)}$$

eq 7.73 7.111

$$\tau = \frac{\left(t_4 - t_2\right)}{\ln_e\left(\frac{\Delta p_A}{\Delta p_B}\right)}$$

eq 7.74 7.111

$$\omega_d = \frac{2\pi}{P}$$

eq 7.75 7.113

$$\omega_n = \frac{\omega_d}{\sqrt{1 - \zeta^2}} = \frac{\frac{2\pi}{P}}{\sqrt{1 - \zeta^2}}$$

eq 7.76 7.113

CHAPTER 8

$$\dot{u} - X_u \Delta u - X_q \Delta q + g\Delta\theta = X_{B_{1s}} \Delta B_{1s} = X_{\delta_{LONG}} \Delta\delta_{LONG}$$

eq 8.1 8.4

$$\dot{q} - M_u \Delta u - M_q \Delta q = M_{B_{1s}} \Delta B_{1s} = M_{\delta LONG} \Delta\delta_{LONG}$$

eq 8.2 8.5

$$\dot{w} - Z_w \Delta w = Z_{\theta_C} \Delta\theta_C = Z_{\delta_C} \Delta\delta_C$$

eq 8.3 8.5

$$\dot{v} - Y_v \Delta v - Y_p \Delta p = Y_{A_{1s}} \Delta A_{1s} + Y_{\theta_{TR}} \Delta\theta_{TR} = Y_{\delta_{LAT}} \Delta\delta_{LAT} + Y_{\delta_{PED}} \Delta\delta_{PED}$$

eq 8.4 8.6

$$\dot{p} - L_v \Delta v - L_p \Delta p = L_{A_{1s}} \Delta A_{1s} + L_{\theta_{TR}} \Delta\theta_{TR} = L_{\delta_{LAT}} \Delta\delta_{LAT} + L_{\delta_{PED}} \Delta\delta_{PED}$$

eq 8.5 8.6

$$\dot{r} - N_v \Delta v - N_p \Delta p - N_r \Delta r = N_{A_{1s}} \Delta A_{1s} + N_{\theta_{TR}} \Delta \theta_{TR}$$

$$= N_{\delta_{LAT}} \Delta \delta_{LAT} + N_{\delta_{PED}} \Delta \delta_{PED}$$

eq 8.6 8.7

$$M_u u + M_{\delta_{LONG}} \delta_{LONG} = 0$$

eq 8.7 8.27

$$\frac{\Delta \delta_{LONG}}{\Delta u} = -\frac{M_u}{M_{\delta_{LONG}}} = -\frac{\text{Speed Stability Derivative}}{\text{Sensitivity}}$$

eq 8.8 8.27

$$\text{Period} = 2\pi \sqrt{\frac{L}{g}}$$

eq 8.9 8.33

$$u \cong \dot{x} \cong L\dot{\theta}$$

eq 8.10 8.33

$$\Delta M = 0 = \frac{\partial M}{\partial u} u + \frac{\partial M}{\partial \dot{\theta}} \dot{\theta}$$

eq 8.11 8.35

$$\dot{\theta} = \frac{\frac{\partial M}{\partial u}}{\frac{\partial M}{\partial \dot{\theta}}} u = -\frac{M_u}{M_{\dot{\theta}}} u$$

eq 8.12 8.35

$$L_{Equiv} = \frac{u}{\dot{\theta}} = -\frac{M_{\dot{\theta}}}{M_u}$$

eq 8.13 8.35

$$\text{Period} = 2\pi \sqrt{\frac{L_{Equiv}}{g}} = 2\pi \sqrt{\frac{M_{\dot{\theta}}}{gM_u}}$$

eq 8.14 8.35

$$\dot{q} - M_q q = M_{B_{1s}} B_{1s} = M_{\delta_{LONG}} \delta_{LONG}$$

eq 8.15 8.37

$$\dot{p} - L_p p = L_{A_{1s}} A_{1s} + L_{\theta_{TR}} \theta_{TR} = L_{\delta_{LAT}} \delta_{LAT} + L_{\delta_{PED}} \delta_{PED} \approx L_{\delta_{LAT}} \delta_{LAT}$$

eq 8.16 8.37

$$\dot{u} = \frac{B}{m}\, u = \frac{1}{m}\,\frac{\partial X}{\partial \delta}\,\delta$$

eq 8.17 8.39

$$\dot{u} - X_u\, u = X_\delta\, \delta$$

eq 8.18 8.39

$$\tau = -\frac{1}{X_u}$$

eq 8.19 8.39

$$q(t) = -\frac{M_{B_{1s}}\, B_{1s}}{M_q}\left(1 - e^{M_q t} \right)$$

eq 8.20 8.40

$$\dot{q}(t) = M_{B_{1s}}\, B_{1s} e^{M_q t}$$

eq 8.21 8.40

$$\theta\,(t) = \int_0^t q\, dt = -\frac{M_{B_{1s}}\, B_{1s}}{M_q}\left[t + \frac{1}{M_q}\left(1 - e^{M_q t} \right) \right]$$

eq 8.22 8.40

$$\dot{q}\,(0) = M_{B_{1s}}\, B_{1s}$$

eq 8.23 8.42

$$q_{ss} = -\frac{M_{B_{1s}}\, B_{1s}}{M_q}$$

eq 8.24 8.42

$$\tau = -\frac{1}{M_q}$$

eq 8.25 8.42

CHAPTER 10

$$\frac{d\Omega}{dt} = \dot{\Omega} = -\frac{1}{I_R} Q$$ *eq 10.1* 10.2

$$dT = 4\rho\pi \; r dr \; v_i^2$$ *eq 10.2* 10.3

$$C_Q = \frac{Q}{\rho A_D \Omega^2 R^3}$$ *eq 10.3* 10.4

$$\frac{Q}{Q_0} = \frac{\Omega^2}{\Omega_0^2}$$ *eq 10.4* 10.4

$$t = \frac{I_R \Omega_0}{Q_0} \left(\frac{\Omega_0}{\Omega} - 1 \right)$$ *eq 10.5* 10.5

$$\frac{1}{I_{zz}} \left(\frac{\partial N}{\partial \theta_{TR}} \right) \theta_{TR} = \frac{1}{I_{zz}} \left(\frac{\partial N}{\partial \delta_{PED}} \right) \delta_{PED} = \dot{r}_0$$ *eq 10.6* 10.9

$$T_{TR} \; l_t = Q_{MR}$$ *eq 10.7* 10.9

$$T_{TR} \left(\cos i_{TR} \right) l_t = Q_{MR}$$ *eq 10.8* 10.9

$$N_{\delta_{PED}} \approx - \frac{1}{I_{zz}} \left(\frac{\partial T_{TR}}{\partial \delta_{PED}} \right) l_t$$ *eq 10.9* 10.11

$$- \frac{Q_{MR}}{I_{zz}} = \frac{N}{I_{zz}} = \dot{r} - N_r r$$ *eq 10.10* 10.11

$\dot{r}_0 = -\frac{Q_{MR}}{I_{zz}}$	*eq 10.11*	10.15
$\delta_{PED} = -\frac{Q_{MR}}{N_{\delta_{PED}}}$	*eq 10.12*	10.15
$L_H = \left(\frac{ebM_S\Omega^2}{2}\right) b_{1s}$	*eq 10.13*	10.17
$M_H = \left(\frac{ebM_S\Omega^2}{2}\right) a_{1s}$	*eq 10.14*	10.18
$\frac{\partial a_{1s}}{\partial B_{1s}} = -\frac{\left(1 + \frac{3}{2}\mu^2\right)}{\left(1 - \frac{\mu^2}{2}\right)}$	*eq 10.15*	10.35
$M_{CG} = \left(Th + \frac{ebM_S\Omega^2}{2}\right) a_{1s}$	*eq 10.16*	10.36
$L_{CG} = \left(Th + \frac{ebM_S\Omega^2}{2}\right) b_{1s}$	*eq 10.17*	10.42
$\dot{h}_{TD} = \sqrt{2gh_{AGL}}$	*eq 10.18*	10.54
$\frac{d\Omega}{dt} = -\frac{1}{I_R}\left(\frac{\Omega^2}{\Omega_0^2}\right) Q_0$	*eq 10.19*	10.55

$$\frac{T}{T_0} = \frac{\Omega^2}{\Omega_0^2}$$

eq 10.20 10.56

$$\ddot{z} = \ddot{h} = \frac{W - T}{m} = g\left(1 - \frac{T}{W}\right)$$

eq 10.21 10.56

$$\dot{h} = gt^2\left(\frac{Q_0}{I_R\Omega_0}\right)\left(1 + \frac{tQ_0}{I_R\Omega_0}\right)^{-1}$$

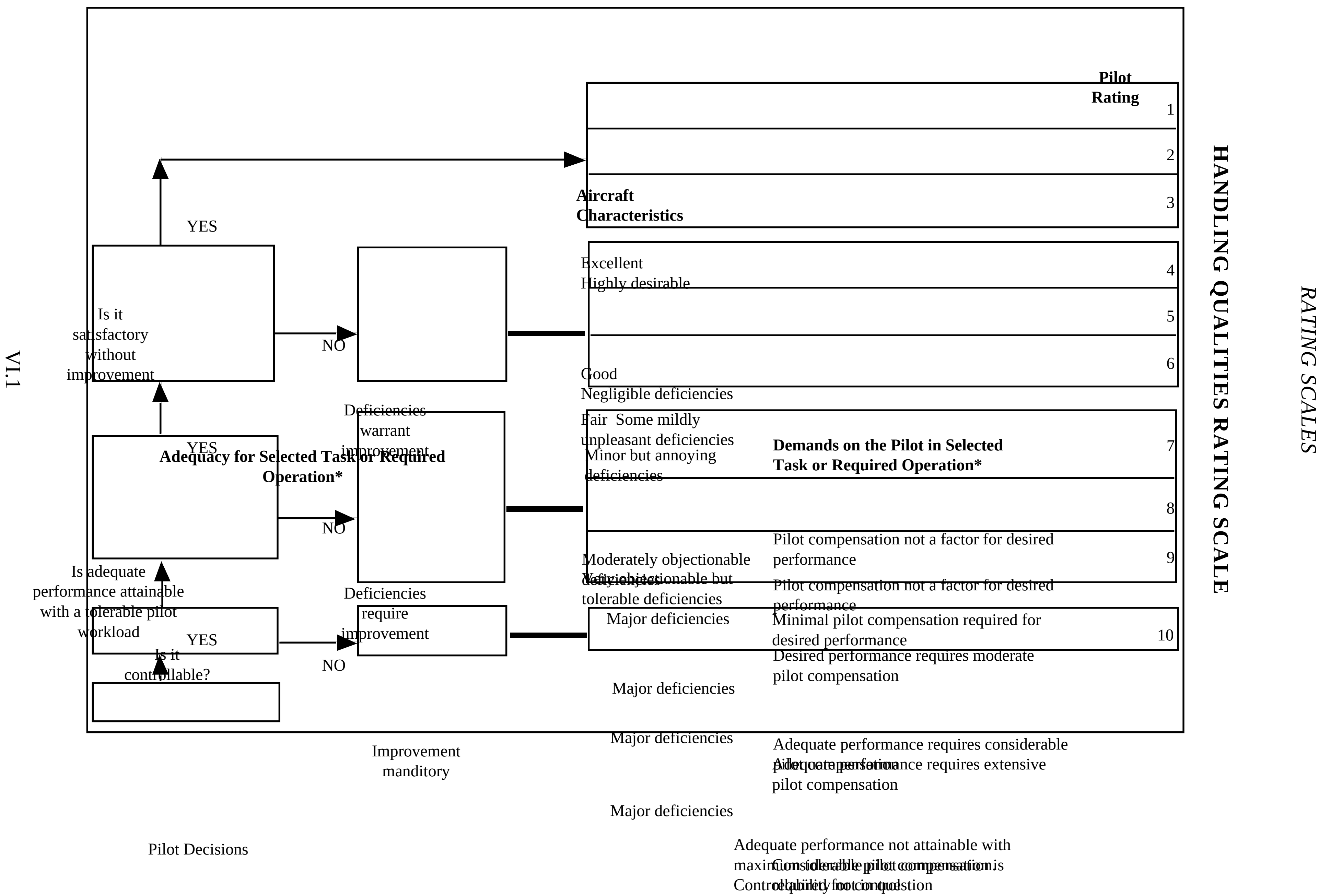
eq 10.22 10.57

$$\Delta h = \frac{gt^2}{2}\left[1 - \frac{2I_R\Omega_0}{Q_0 t} + \frac{2\left(I_R\Omega_0\right)^2}{\left(Q_0 t\right)^2}\ln\left(1 + \frac{Q_0 t}{I_R\Omega_0}\right)\right]$$

eq 10.23 10.57

APPENDIX VI

RATING SCALES



HANDLING QUALITIES RATING SCALE
RATING SCALES
VI.1
Pilot Rating
1
2
3
4
5
6
7
8
9
10
Aircraft Characteristics
Demands on the Pilot in Selected Task or Required Operation*
Adequacy for Selected Task or Required Operation*
Pilot Decisions
Is it satisfactory without improvement
Is adequate performance attainable with a tolerable pilot workload
Is it controllable?
YES
YES
YES
NO
NO
NO
Deficiencies warrant improvement
Deficiencies require improvement
Improvement manditory
Excellent Highly desirable
Good Negligible deficiencies
Fair Some mildly unpleasant deficiencies
Minor but annoying deficiencies
Moderately objectionable deficiencies
Very objectionable but tolerable deficiencies
Major deficiencies
Major deficiencies
Major deficiencies
Major deficiencies
Pilot compensation not a factor for desired performance
Pilot compensation not a factor for desired performance
Minimal pilot compensation required for desired performance
Desired performance requires moderate pilot compensation
Adequate performance requires considerable pilot compensation
Adequate performance requires extensive pilot compensation
Adequate performance not attainable with maximum tolerable pilot compensation. Controllability not in question
Considerable pilot compensation is required for control

SAMPLE PILOT COMMENT CARD

COMMENT CARD

1. Response Characteristics - Predictability, Quickness
   Attitude
   Translation
   Height control
   Precision vs. Gross Maneuvering

2. Control Characteristics
   Control forces and displacements
   Initial vs. final feel
   Harmony (pitch/roll)

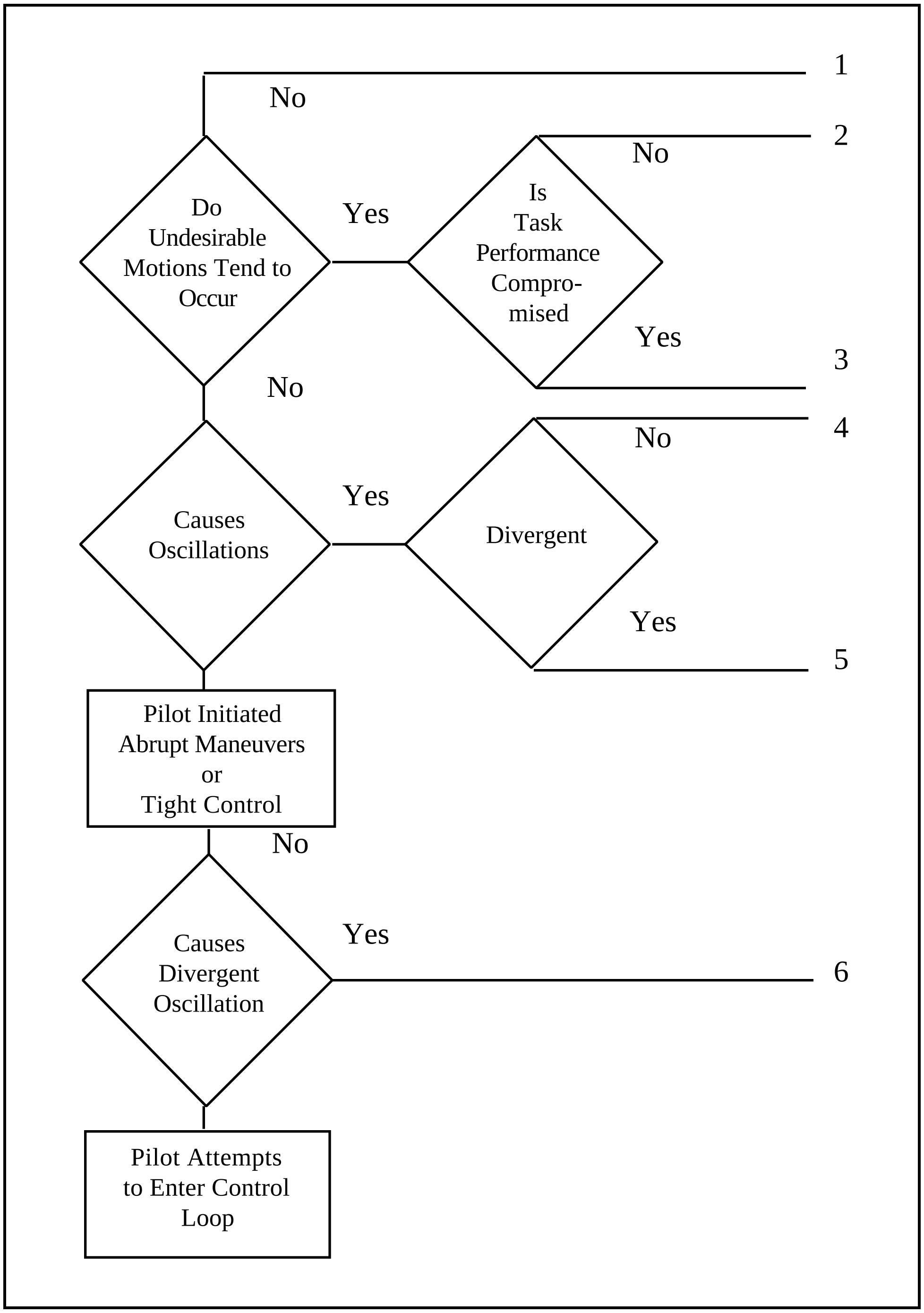
3. Special Control Techniques?

4. Task Performance? Workload?
   Tracking X-Y
   Tracking Y-Z
   Landing task

5. Effects of Turbulence?

6. Summary
   Good Features
   Objectionable Features
   Pilot Ratings (Tracking/Landing)

PIO RATING SCALE



TURBULENCE RATING SCALE

INTENSITY	AIRCRAFT REACTION	REACTION INSIDE AIRCRAFT
LIGHT		
Turbulence Chop	Turbulence that momentarily causes slight, erratic changes in altitude and/or attitude. Turbulence that causes slight, rapid and somewhat appreciable changes in altitude or attitude	Occupants may feel a slight strain against seat belts or shoulder straps. Unsecured objects may be displaced slightly. Food service may be conducted and little or no difficulty is encounter in walking
MODERATE		
Turbulence Chop	Turbulence that causes changes in altitude and/or attitude, but with the aircraft remaining in positive control at all times. It usually causes variations in indicated airspeed. Turbulence that causes rapid bumps or jolts without appreciable changes in aircraft altitude or attitude.	Occupants feel definite strains against seat belts or shoulder straps. Unsecured objects are dislodged. Food service and walking are difficult.
SEVERE		
Turbulence	Turbulence that causes large, abrupt changes in altitude and/or attitude. It usually causes large variations in indicated airspeed. Aircraft may be momentarily out of control	Occupants are forced violently against seat belts or shoulder straps. Unsecured objects are tossed about. Food service and walking are impossible.
EXTREME		
Turbulence	Turbulence in which the aircraft is violently tossed about and is practically impossible to control. It may cause structural damage.	
FREQUENCY	DEFINITION	
Occasional	Less than 1/3 of the time	
Intermittent	1/3 to 2/3 of the time	
Continuous	More than 2/3 of the time	

VIBRATION RATING SCALE

Degree of Vibration	Description	Pilot Rating
No Vibration		0
Slight	Not apparent to experienced aircrew fully occupied by their tasks, but noticeable if their attention is directed to it or if not otherwise occupied.	1 2 3
Moderate	Experienced aircrew are aware of the vibration but it does not affect their work, at least over a short period.	4 5 6
Severe	Vibration is immediately apparent to experienced aircrew even when fully occupied. Performance of primary task is affected or tasks can be done only with difficulty.	7 8 9
Intolerable	Sole preoccupation of aircrew is to reduce vibration.	10

Based on the Subjective Vibration Assessment Scale developed by the Aeroplane and Armament Experimental Establishment, Boscombe Down, England.

APPENDIX VII

MILITARY SPECIFICATION, HELICOPTER FLYING AND GROUND HANDLING QUALITIES; GENERAL REQUIREMENT FOR, MIL-H-8501A

APPENDIX VII

MILITARY SPECIFICATION, HELICOPTER FLYING AND GROUND HANDLING QUALITIES; GENERAL REQUIREMENT FOR, MIL-H-8501A

PARAGRAPH	CHARACTERISTIC	APPLICABLE FLIGHT TEST	FLIGHT CONDITIONS	SPECIFICATION REQUIREMENT
3.1.2	Operating conditions		Specification applies over all normal service loadings (GW & CG), Operating rotor speed range, All operational altitudes and temperatures	
3.2	Longitudinal Characteristics			
3.2.1	Control margin, Control power	Trimmed control positions in forward flight, low speed flight, and autorotation, Control response	30 kn rearward to V_{NE}, steady climbs and descents, autorotative flight from 0 kn to $V_{max\ auto}$	Steady smooth flight; Adequate control to produce 10% of maximum pitching moment available in hover; No objectionable shake, vibration, or roughness in helicopter or flight controls
3.2.2	Longitudinal control movement during hover	Low speed flight	Hovering flight (wind ≤ 3 kn)	Maintain hover over spot up to OGE altitude; Minimum cyclic control movement, < ±1.0 in cyclic movement

3.2.3	Longitudinal trimmability	Trimmed control positions in forward flight, low speed flight, and autorotation	30 kn rearward to V_{NE}, steady climbs and descents, autorotative flight from 0 kn to $V_{max\ auto}$	Trim control forces to zero; Positive self centering; Stick jump undesirable
3.2.4	Longitudinal force gradient	Mechanical characteristics	Rotors static electrical and hydraulic power applied; Verify results qualitatively in flight 30 kn rearward to V_{NE}, steady climbs and descents, autorotative flight from 0 kn to $V_{max\ auto}$	Force gradient for 1st inch of travel 0.5 to 2.0 lb/in; Force for 1st in ≥ breakout plus friction; No undesirable discontinuities; Positive slope with 1st in ≥ remaining slope
3.2.5	Longitudinal control requirements	Mission maneuvers	Trimmed level flight @ V_H to hover	Readily and safely perform a quick stop, hover, and level acceleration
3.2.6	Longitudinal limit control force	Mechanical characteristics	Operating envelope specified in Table I	Limit control force ≤ Table II (8.0 lb)
3.2.7	Longitudinal breakout plus friction	Mechanical characteristics	Rotors static, electrical and hydraulic power applied Verify results in flight	Breakout plus friction IAW Table II (minimum 0.5 lb maximum 1.5 lb)
3.2.8	Control force coupling, transient control forces	Mechanical characteristics, Control response, Mission maneuvers	Rapid longitudinal control displacement from trim	Free from objectionable transient forces; Forces shall not fall to zero; Lateral force coupling ≤ 20% longitudinal force; Directional force coupling ≤ 75% longitudinal force; No lateral or directional control force coupling with power operated controls

3.2.9	Longitudinal control response	Control response	30 kn rearward to V_{NE}, steady climbs and descents, autorotative flight from 0 kn to $V_{max\ auto}$	No objectionable or excessive decay in angular velocity after control input; Angular acceleration in proper direction within 0.2 s
3.2.10	Static longitudinal stability	Static longitudinal stability	All forward speeds and all trim and power conditions shown in Table I, constant collective	Positive control force and position stability; Between 15 to 50 kn forward and 10 to 30 kn rearward moderate instability permitted NTE 0.5 in displacement or 1.0 lb force
3.2.10.1	Static longitudinal stability	Static longitudinal stability	Same as above for most critical CG location	Same as above for most critical CG location
3.2.10.2	Longitudinal trim changes	Trimmed control positions in climb and descent	Climbs and descents, Airspeeds between zero to 1/2 $V_{min\ power}$; and V_{max}	No excessive trim changes; No more than 3.0 in displacement from initial trim position
3.2.11	Dynamic longitudinal stability	Longitudinal short term and long term response	Forward flight	Period: < 5 s, damp to $C_{1/2} \le 2$, no residual small amplitude oscillations; > 5 to < 10 s, at least lightly damped; >10 to < 20 s, $T_d \ge 10$ s
3.2.11.1	Maneuver stability	Maneuver stability, Control response	Lesser of the following: longitudinal step sufficient to generate $q \ge 0.2$ rad/s or, $n_z \ge 1.5$ g within 3 s or, 1 in aft step; Normal acceleration applies for speed < $V_{min\ power}$, angular velocity applies for all forward flight and hover	Time history of normal acceleration concave downward within 2 s; Time history of angular velocity concave downward within 2 s; Both remain concave downward until reaching maximum

3.2.11.2	Gust response	Cyclic pulse	Hover and forward flight; Control displacement IAW 3.2.11.1 for 0.5 s pulse	Normal acceleration shall not exceed ± 0.25 g within 10 s after aft longitudinal pulse
3.2.12	Maneuver stability	Maneuver stability, Control response	Hover and forward flight control displacement IAW 3.2.11.1	Normal acceleration increase with time following rapid rearward displacement of longitudinal control until maximum acceleration is reached
3.2.13	Longitudinal control power	Control response	Hover at maximum GW	One in step input produce angular displacement after 1.0 s $\geq \frac{45}{\sqrt[3]{W + 1000}}$ deg; Maximum step input produce angular displacement after 1.0 s $\geq \frac{180}{\sqrt[3]{W + 1000}}$ deg
3.2.14	Pitch rate damping	Control response	Hover	Pitch rate damping $\geq 8(I_y)^{0.7}$ ft-lb/rad/s
3.3	Directional and Lateral Characteristics			
3.3.1	Directional control power	Ground handling	Ground taxi (wheel gear); Water taxi (float gear); Wind 35 kn	Straight path without use of brakes; 360° turn pivoting on one wheel
3.3.2	Sideward flight	Low speed flight	Hover to 35 kn left and right	Steady flight; Controls and helicopter free from objectionable shake, vibration, or roughness
3.3.3	Lateral-directional control movement during hover	Low speed flight	Hover (wind ≤ 3 kn)	Maintain hover over spot up to OGE altitude; Minimum cyclic movement < ± 1 in lateral cyclic; Pedal movement < ± 1.0 in

3.3.4	Lateral control margin	Trimmed control positions in forward flight and low speed flight	30 kn rearward to V_{NE}, steady climbs and descents, autorotative flight from 0 kn to $V_{max\ auto}$; Hovering flight (wind ≤ 3 kn); Adverse lateral CG	Adequate control to produce 10% of maximum rolling moment available in hover
3.3.5	Directional control power	Control response	Hover at maximum GW (wind ≤ 3 kn)	One in step input produce angular displacement after 1.0 s ≥ $\frac{110}{\sqrt[3]{W + 1000}}$ deg; Maximum step input produce angular displacement after 1.0 s ≥ $\frac{330}{\sqrt[3]{W + 1000}}$ deg
3.3.6	Hovering turn	Mission maneuvers, Control response	Wind 35 kn; Maximum GW or takeoff power	Execute 360° turn; Maintain hover over spot; Adequate control margin; Yaw displacement following rapid, full pedal deflection at critical azimuth after 1.0 s ≥ $\frac{110}{\sqrt[3]{W + 1000}}$ deg
3.3.7	Directional control response	Control response	Hover, One in step input, Minimum GW	No tendency to over control; yaw displacement 1.0 s after input ≤ 50 deg/in
3.3.8	Directional control during autorotation	Mission maneuvers	Autorotation 0 kn to to $V_{max\ auto}$	Left and right coordinated turns possible

3.3.9	Static lateral-directional stability	Steady heading sideslips	50 kn to V_{NE}, up to full pedal displacement	Positive control fixed directional stability and effective dihedral; Stable gradient of control displacement versus sideslip; Linear gradient between sideslip angles of ± 15°; 10% longitudinal and lateral control effectiveness remaining
3.3.9.1	Cyclic only turns	Cyclic only turns	50 kn to V_{NE}; Lateral control input resulting in a bank of 30° in 6 s	Cyclic turns possible; No reversal of rolling velocity after small step displacement
3.3.9.2	Adverse yaw	Cyclic only turns	50 kn to V_{NE}	No objectionable adverse yaw
3.3.10	Lateral-directional trimmability	Trimmed control positions in forward flight, low speed flight, and autorotation	30 kn rearward to V_{NE}, steady climbs and descents, autorotative flight from 0 kn to V_{max} auto; Hovering flight (wind ≤ 3 kn)	Trim control forces to zero; Positive self-centering; Stick jump undesirable
3.3.11	Lateral-directional force gradient	Mechanical characteristics	Rotors static, electrical and hydraulic power applied; Verify results in flight 30 kn rearward to V_{NE}, steady climbs and descents, autorotative flight from 0 kn to V_{max} auto; Hovering flight (wind ≤ 3 kn)	Lateral force gradient 1st in of travel 0.5 to 2.0 lb/in; Force for 1st in ≥ breakout plus friction; Positive slope with 1st in ≥ remaining slope; Directional limit control force 15 lb; Linear directional force gradient; No undesirable lateral or directional gradient discontinuities
3.3.12	Lateral-directional limit control force	Mechanical characteristics	Operating envelope specified in Table I	Lateral cyclic limit force ≤ Table II (7.0 lb); Directional limit force ≤ Table II (15.0 lb)

3.3.13	Lateral-directional breakout plus friction	Mechanical characteristics	Verify results in flight	Breakout plus friction IAW Table II (Lateral minimum 0.5 lb, maximum 1.5 lb); (Directional minimum 3.0 lb, maximum 7.0 lb)
3.3.14	Control force coupling, transient control forces	Mechanical characteristics, Control response, Mission maneuvers	Rapid lateral and directional control displacements from trim	Free from objectionable transient forces; Forces shall not fall to zero; Longitudinal force coupling ≤ 40% lateral force; Directional force coupling ≤ 100% lateral force; Longitudinal force coupling ≤ 8% pedal force; Lateral force coupling ≤ 6% pedal force; No longitudinal control force coupling with power operated controls
3.3.15	Lateral control response	Control response	Level flight hover to V_{NE}	No tendency to over control; Maximum roll rate ≤ 20˚/s/in lateral control
3.3.16	Lateral-directional control response	Control response	30 kn rearward to V_{NE}, steady climbs and descents, autorotative flight from 0 kn to $V_{max\ auto}$	No objectionable or excessive delay in angular velocity after control input; Angular acceleration in proper direction within 0.2 s
3.3.17	Lateral trim changes	Trimmed control positions in climb and descent	Any combination of power and airspeed throughout flight envelope	No excessive trim changes with changes in collective ; No more than 2.0 in displacement from initial trim position

3.3.18	Lateral control power	Control response	Hover at maximum GW (wind ≤ 3 kn)	One in step input produce angular displacement after 0.5 s ≥ $\frac{27}{\sqrt[3]{W + 1000}}$ deg; Maximum step input produce angular displacement after 0.5 s ≥ $\frac{81}{\sqrt[3]{W + 1000}}$ deg
3.3.19	Roll and yaw rate damping	Control response	Hover	Roll rate damping ≥ $18(I_x)^{0.7}$ ft-lb/rad/s; Yaw rate damping ≥ $27(I_z)^{0.7}$
3.4	Vertical Characteristics			
3.4.1	Collective control movement during hover	Low speed flight	Hovering flight (wind < 3 kn); Up to OGE with constant rotor speed	Maintain altitude within ± 1 ft; Minimum movement of collective, < ± 1/2 in; No objectionable vertical oscillation due to governor lag
3.4.2	Collective control forces	Mechanical characteristics	Rotors static, electrical and hydraulic power applied; Verify results in flight	Limit control force ≤ Table II (7.0 lb); Breakout plus friction IAW Table II (minimum 1.0 lb, maximum 3.0 lb); Control remains fixed, no tendency to creep
3.4.3	Control force coupling	Mechanical characteristics	Rotors static, electrical power applied; Verify results in flight	No objectionable control force coupling, cyclic force ≤ 1 lb; No control force coupling with power operated controls
3.5	Autorotation, Rotor Characteristics, and Miscellaneous			

3.5.1	Rotor start and stop	Mission maneuvers, ground handling	Ground operation, Wind 45 kn (GW ≥1000 lb), Wind 35 kn (GW <1000 lb); Shipboard operation, wind 60 kn	Start and stop rotor
3.5.2	Ground operation	Mission maneuvers, ground handling	Ground operation, Takeoff rotor speed, Wind 45 kn (GW ≥1000 lb), Wind 35 kn (GW <1000 lb)	Maintain fixed position on level paved surface as power increased to takeoff power without use of wheel chocks
3.5.3	Ground taxi	Mission maneuvers, ground handling	Taxi and pivot	No damage to rotor coning stops or blade/ aircraft contact
3.5.4	Takeoff and landing	Mission maneuvers		Capable of satisfactory takeoff and landing
3.5.4.1	Vertical takeoffs and landings	Mission maneuvers	Wind 45 kn (GW ≥1000 lb), Wind 35 kn (GW <1000 lb)	Safe vertical takeoff and landing
3.5.4.2	Running takeoff	Mission maneuvers	Level, paved surface; Ground speed up to 35 kn, Wheel gear	Safe running takeoff
3.5.4.3	Running landing	Mission maneuvers, Autorotative landing	Level, paved surface; drift 3 kn any direction; drift 6 kn at 35 kn ground speed; Power on and autorotation	Safe running landing
3.5.4.4	Autorotative stopping distance	Autorotative landing	Wheel and skid gear; Level, paved surface; Touchdown speed 35 kn	Stopping distance ≤ 200 ft
3.5.4.5	Water landings (emergency flotation gear)	Mission maneuvers, Autorotative landing	Smooth water; 15 kn surface speed; 3 kn drift any direction; drift 5 kn at 15 kn; Power on and autorotation	Safe landing

3.5.5	Available delay time	Simulated sudden engine failures	Hover to V_{NE}; Sudden power reduction; Controls fixed	Safe transition from powered flight to autorotation; Collective control delay ≥ 2 s; Rotor speed not below minimum transient
3.5.5.1	Aircraft response to engine failure	Simulated sudden engine failures	Hover to V_{NE}; Sudden power reduction/application; Controls fixed	Allowable attitude change within 2 s: Pitch 10˚ Roll 10˚ Yaw 10˚, 20˚ below $V_{max\ R/C}$
3.5.6	Autorotative control forces	Autorotative flying qualities	Hover to V_{NE}; Sudden power reduction; Transition to autorotative flight	Limit control forces NTE Table II: Longitudinal 8.0 lb Lateral 7.0 lb Collective 7.0 lb Directional 15.0 lb
3.5.7	Autorotative landings	Autorotative landing	Wind < 3 kn; Power off	Landing speed 15 kn or less (0 kn desirable); Repeated safe operation
3.5.8(a)(1)	Power operated control system failures	Mechanical characteristics	Abrupt system failure; Verify results in flight	Controls free for 3 s, pitch, roll, and yaw rates ≤ 10˚/s; $n_z \le \pm 1/2$ g
3.5.8(a)(2)	Power operated control system failures	Mechanical characteristics	Abrupt system failure; Verify results in flight	Level flight at $\beta = 0°$ with limit control forces ≤ Directional 80 lb Collective 25 lb Longitudinal 25 lb Lateral 15 lb
3.5.8(b)	Flight with power operated control system OFF	Mechanical characteristics	30 kn rearward to V_{NE}; steady climbs and descents, autorotative flight from 0 kn to $V_{max\ auto}$; Hovering flight (wind ≤ 3 kn)	Trim control forces to zero

3.5.8(c)	Flight with power operated control system OFF	Mission maneuvers	Operating envelope	No tendency for collective to creep
3.5.8(d)	Flight with power operated control system	Mission maneuvers	From trimmed level flight at 40 kn and power operated control failure	Normal approach and landing without retrimming and limit control forces ≤ Directional 80 lb Collective 25 lb Longitudinal 25 lb Lateral 15 lb
3.5.8(e)	Primary power operated control system	System failures	Rotors static, hydraulic and electrical power applied; Verify results during ground run-up and in flight	No failure of primary power operated control system with engine or electrical system failure
3.5.8(f)	Trim system	Mechanical characteristics	Rotors static, hydraulic and electrical power applied	No failure of trim system with failure of power operated control system
3.5.8(g)	Independent power operated control systems	Mechanical characteristics	Rotors static, hydraulic and electrical power applied; Fail one control system; Verify results in flight	Safe operation; Rates of control motion not less than 50% of normal
3.5.9(a)	Automatic stabilization equipment (ASE) or stability augmentation system (SAS) failures	Mechanical characteristics, AFCS evaluations	Level flight, Abrupt disengagement or failure, Controls free for 3 s	Pitch, roll, yaw rates ≤ 10˚/s; $n_z \leq \pm 1/2$ g; No engagement transients
3.5.9(b)	Dual-independent ASE, SAS systems	Mechanical characteristics, AFCS evaluations	Level flight, Single system failure, Controls free for 3 s	Pitch, roll, yaw rates ≤ 10˚/s; $n_z \leq \pm 1/2$ g
3.5.9(c)	Control forces with ASE, SAS engaged	Mechanical characteristics	Rotors static, hydraulic and electrical power applied ASE, SAS systems engaged	Limit control forces NTE Table II: Longitudinal 8.0 lb Lateral 7.0 lb Collective 7.0 lb Directional 15.0 lb

3.5.9(d)	Flight with AFCS disengaged	Mission maneuvers	Normal level flight, Normal approach and landing	Level flight and safe landing under VMC conditions
3.5.9(e)	Control margin	Maneuvering stability, Steady heading sideslips	Steady level turn at V_{cruise} to maximum load factor, Steady sideslips power on and autorotation to sideslip limit	Use not more than 50% of available control moment in the unstable direction from trim level flight
3.5.10	Control system freeplay	Mechanical characteristics	Rotors static, hydraulic and electrical power applied, Verify results in flight	Total system freeplay $\leq \pm$ 0.2 in
3.5.11	Mechanical coupling	Mechanical characteristics, Mission maneuvers	Operating envelope	No adverse response due to mechanical coupling
3.5.11.1	Mechanical mixing	Mechanical characteristics, Mission maneuvers	Operating envelope	No adverse limitation in control power due to mechanical control mixing
3.6	Instrument Flight Characteristics			
3.6.1	Pilot workload	Mission maneuvers	Instrument flight ASE, SAS ON and OFF	No undue pilot effort; Stability and control characteristics ASE, SAS OFF as good as required under VMC

3.6.1.1	Control power and damping	Control response	Hovering flight 1 in step control input	Angular displacement after 1 s: Longitudinal: $\frac{73}{\sqrt[3]{W+1000}}$ deg Directional: $\frac{110}{\sqrt[3]{W+1000}}$ deg After 1/2 s: Lateral: $\frac{32}{\sqrt[3]{W+1000}}$ deg; Rate damping: Longitudinal $15(I_y)^{0.7}$ Directional $27(I_z)^{0.7}$ Lateral $25(I_x)^{0.7}$; Maximum control displacement shall produce 4 times the above angular displacement for longitudinal axis and 3 time for the lateral and directional axis
3.6.1.2	Longitudinal and lateral-directional dynamic stability	Longitudinal long and short term oscillations; Lateral-directional oscillations	Hover and level flight; controls fixed	Period: <5 s, $C_{1/2} \leq 1$; <10 s, $C_{1/2} \leq 2$; >10 s < 20 s, lightly damped; > 20 s $T_d \geq 20$ s; No undamped residual oscillations
3.6.2	Static lateral-directional stability	Steady heading sideslips	50 kn to V_{NE}; Up to full pedal displacement	Stable control force and position gradients; Linear gradient between sideslip of ±15°; 10% longitudinal and lateral control effectiveness remaining; Cyclic only turns possible (pedals free)

3.6.3	Static longitudinal stability	Static longitudinal stability	Steady level flight, Trim and power conditions IAW Table I	Positive control force and position stability
3.7	Vibration Characteristics			
3.7.1 (a)	Control vibration	Mission maneuvers, observe during all flight tests	Design flight envelope	No objectionable shake, vibration, or roughness; Maximum vibration acceleration at controls < 0.4 g for frequencies ≤ 32 cps; A_d of 0.008 in for frequencies >32 cps
3.7.1 (b)	Vibrations at crew and cabin stations	Mission maneuvers, observe during all flight tests	Design flight envelope	30 kn rearward to V_{cruise}: ≤ 0.15 g for frequencies ≤ 32 cps; A_d ≤ 0.003 in for frequencies > 32 cps; V_{cruise} to V_{max}: ≤ 0.2 g for frequencies ≤ 36 cps; A_d ≤ 0.003 in for frequencies > 36 cps; vibration velocity ≤ 0.039 fps for frequencies > 50 cps
3.7.2	Longitudinal vibratory control force	Control response, Mission maneuvers	Rapid longitudinal or lateral stick deflection	Control force vibration < 2 lb following rapid displacement
3.7.3	Mechanical instability	Mission maneuvers, observe during all flight tests	Design flight to include ground operations	No mechanical instabilities to include ground resonance

Table I

Power and Speed Conditions

Initial trim and power conditions	Speed range of interest
Hovering	0 to 30 kn
Level flight at 35 kn	15 to 60 kn
level flight at 80% V_{max}	60% V_{max} to V_{max}
Level flight at V_{max}	80 % V_{max} to V_{limit}
Climb at best rate of climb	$V_{max\ R/C}$ ± 15 kn
Partial power descent at 300 to 500 fpm	15 to 60 kn
Autorotation with trim as in "level flight at 80% V_{max}"	60% V_{max} to $V_{max\ auto}$
Autorotation at speed for minimum rate of descent	15 kn to trim speed + 20 kn

Table II

Limit Control Force

Control	Limit control force lb	Breakout plus friction lb	
		Minimum	Maximum
Longitudinal	8.0	0.5	1.5
Lateral	7.0	0.5	1.5
Collective	7.0	1.0 [1]	3.0
Directional	15.0	3.0 [1]	7.0
[1] May be measured with adjustable friction set			

APPENDIX VIII

AUTOMATIC FLIGHT CONTROL SYSTEM ASSESSMENT

APPENDIX VIII

AUTOMATIC FLIGHT CONTROL SYSTEM ASSESSMENT

References:	A	ETPS Ground School Notes Chapter E11
	B	MOD Defense Standard 00-970 Volume 2 Issue 1
	C	Military Specifications MIL-H-8501B and MIL-C-18244
	D	BCAR G1-2 Para 7 and BCAR 29

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The following information has been extracted substantially intact from the Empire Test Pilot's School Rotary Wing Flight Test Manual. Where appropriate, minor content and stylistic changes have been made to improve readability for students at USNTPS.

INTRODUCTION

1. The term "Automatic Flight Control System" (AFCS) covers a multitude of devices which may be used in various ways to alter or improve the stability and handling qualities of a helicopter, or to permit certain mission maneuvers to be flown automatically. For the purposes of this Appendix, the term "AFCS" will be taken to apply to all types of system unless specifically defined. There are three different levels of augmentation, although all three may well be considered in one aircraft system. These three levels are:

a. *Stability Augmentation System (SAS)*. This system uses rate gyros to provide rate damping and possibly integration of the signal to provide short term attitude hold. This type of system alone may be fitted to light helicopters (e.g. Gazelle, Jet Ranger).

b. *Automatic Stabilization Equipment (ASE)*. This uses vertical gyro or VRS information to provide a long term attitude hold. Rate damping may also be incorporated when the pilot is making a control input. Heading hold using compass information, although a Òhigher order functionÓ is normally a basic function of an ASE.

c. *Autopilot*. The term ÒautopilotÓ covers all those piloting functions additional to basic stability augmentation. Internal or external sensors may be used. Autopilot functions include altitude and airspeed holds and operational modes such as automatic transition and hover, capture and tracking of a navigation course (e.g. VOR) and coupled approach (e.g. ILS).

Other definitions may also be encountered. [Annex A gives a glossary of AFCS terminology.]

2. The type of testing required of an AFCS will depend on such aspects as the capabilities and complexities of the system, the stability characteristics of the basic aircraft, the missions of the aircraft, and the manner in which the aircraft will be operated to carry out these missions. It is therefore essential to understand not only the system, but also the operating requirements, before embarking on a test program. For example, many flight test hours could be wasted evaluating the automatic approach to a hover from entry conditions that are not mission relatable, although they are within the capability of the system. On the other hand, the AFCS of a helicopter that is to be certified for single pilot IFR must undergo considerable testing in all flight conditions.

AIMS OF THE TESTS

3. The aims of a full AFCS assessment are essentially the following:

a. To establish and analyze the correct operation of the system and of its individual components.

b. To assess the operating characteristics and ease of use of the system, the cockpit layout and all the controls, including engagement and disengagement, both independently and against the mission requirement.

c. To assess the aircraft handling characteristics when using the system.

d. To determine whether the equipment meets operational requirements.

e. To measure the performance of the system in all its modes, and assess the performance against the specification.

f. To assess the reliability and maintainability of the system, and to diagnose potential failures and evaluate their likely effect by carrying out a Failure Modes and Effects Analysis (FMEA).

g. To determine the effects of failures revealed by the FMEA, and particularly system runways, initially by simulation but then by flight testing.

GENERAL TEST METHOD

4. A full AFCS assessment will usually begin with an engineering evaluation and a cockpit assessment as the AFCS is developed. Increasing use is being made of simulators for even these early phases, particularly as digital systems become more common. In some cases, development hardware and software may be evaluated in the simulator, with the same boxes then being flight tested to verify the simulation. Flight testing will progress through engagement and disengagement of the various functions to performance measurement of the basic autostabilizer and autopilot modes; on a new type of helicopter, these tests may well be combined with stability and control tests of the basic aircraft. Subsequently, any specialized functions or maneuvers (such as transitions) will be assessed. Concurrently with the flight testing, the engineering evaluation will progress to an assessment of the effect of production component tolerances, both on performance and as part of the FMEA, so that finally, ground and flight testing of system failures and malfunctions can be carried out. Environmental and EMI testing will also be required.

5. Flying techniques adopted for longitudinal, lateral/directional and low airspeed stability and control tests are used for the basic data gathering sorties. From these results an assessment of the AFCS performance in stability augmentation and datum holding can be made. Mission relatable tests require a thorough knowledge and recent experience of the mission combined with a systematic and methodical approach. A cockpit assessment of the layout of the controls and indicators is important. During flight tests detailed observation of the behavior of the aircraft and the AFCS actuators is required. Hunting of actuators, oscillations in pitch and roll, tendency to overshoot required conditions, height variations, extremes of aircraft attitude, rate of power demand and indications to the pilot of loss of authority of the AFCS are a few of the characteristics

which should be assessed. Programmable control surfaces (such as horizontal stabilators) should be evaluated in exactly the same way, either as part of an overall AFCS assessment or separately.

ENGAGEMENT AND DISENGAGEMENT FACILITIES

6. **General.** It should be possible to engage and disengage the AFCS at any time in flight and on the ground. On engagement the AFCS should come into operation smoothly and complete disengagement of the AFCS should be possible quickly, safely and positively at any time independently of all other primary services unless it can be shown that the consequences of the AFCS remaining engaged in all circumstances are more desirable (see also Reference D - 29.672: Stability Augmentation Systems). The "emergency disengagement" facility should normally be on the cyclic control. The overall system engagement/disengagement facilities may not be the only method of disconnecting all or part of the AFCS. The pilot "flythrough" features should be verified as should the operation of any airspeed and/or weight-on-wheels (WOW) switches which affect AFCS operation. Ensure that the disengagement of attitude hold when the cyclic is moved (for example) is smooth and does not introduce any discontinuities into the apparent control response. Check that the heading hold is disengaged whenever the compass (or any other sensor) is aligned or slewed. Verify the operation of any airspeed switches (e.g. UH-60A FPS yaws, to hover-at 60 KIAS) at various combinations of sideslip and angle of attack to determine the effects of airspeed sensor pressure errors or sudden operation of an actuator. The consequences of an airspeed sensor malfunction will also have to be examined especially when it controls a high-authority device such as a movable stabilator. WOW switches are often used to override certain AFCS functions or to change their gain (e.g. CH-47D/HC Mk 1 pitch channel). Make sure that they do not introduce any undesirable characteristics such as hunting or transients especially during running landings or landings on sloping ground when the switches may not operate "cleanly" or simultaneously. Evaluate the effects of an airspeed switch malfunction.

7. **Test Methods.** Within the general requirements for engagement and disengagement, flight testing involves the operation of the various engagement and disengagement facilities both on the ground and throughout the flight envelope. Airborne testing is carried out by approaching incrementally the extremes of maneuver and speed,

starting from minimum power speed in level flight. The transient and steady state aircraft responses following the engagement or disengagement of the AFCS and any dangerous tendencies are recorded.

AFCS PERFORMANCE TESTING

8. Performance testing can be broken down into short term and long term performance of the basic augmentation functions and the evaluation of operational modes. The purpose of these tests is to measure the dynamic qualities of damping, datum holding, and residual oscillations. AFCS performance evaluation is conducted incrementally. Initial tests are done at a mid CG, light gross weight and at minimum power speed in level flight. All initial testing will be done in day visual meteorological conditions.

9. **Basic Augmentation Function.** This paragraph deals with the basic augmentation functions of an AFCS such as rate damping, heading hold and attitude hold. Testing in this category can be considered as short-term wherein the aircraft’s augmented response to deliberate control inputs and gusts is evaluated. Generally the system should be subjected to the full range of stability and control testing methods in order to discover the effect of the AFCS on the aircraft’s static and dynamic stability and control response in all axes. Several examples of AFCS modes of operation are mentioned below but it is emphasized that the list is by no means complete and that a thorough understanding of the operation of the system must be achieved before testing can begin.

a. *Static Stability Tests.* The primary method by which an AFCS affects static stability is through an attitude hold since rate damping is, by definition, a “dynamic” stability mode and generally operates through series actuators. Consider each axis in turn:

(1) Pitch. The performance of the AFCS in this respect will be strongly influenced by the variation of pitch attitude with speed. If q varies little over the speed range then an attitude hold system will contribute little to (and may actually detract from) static stability. In such a case an artificial airspeed hold augmentation (SH-60) or even a means of producing positive “cockpit” static stability by altering the length of control runs (CH-47 DASH) may be incorporated. Remember that both stick force and stick position cues are

important in determining static stability and that the operation of parallel actuators can affect these characteristics in different ways. Artificial pitch rate damping will increase maneuver stability but should not lead to an excessive stick force-per-g which may detract from agility. The effects of an attitude hold function on maneuver stability, especially during wind-up turns, should be carefully evaluated.

(2) Roll. Most ASE incorporates an angle of bank (AOB) hold which may operate over a limited range of roll attitude (Lynx) or over the full range (Chinook). Some systems confer positive spiral stability on the aircraft by requiring an into-turn cyclic deflection or force to hold an off-trim angle of bank while others are more like a rate demand system and will hold attitude (i.e. zero rate) when the cyclic is neutralized thus they produce perfect neutral spiral stability. The effect of an AFCS on static lateral stability (Lv) will depend upon the influence of yaw pedal movement on bank angle hold. The effects of attitude hold on sideways flight should be examined to see if it introduces discontinuities in the lateral cyclic/sideways speed relationship, especially close to the AOB at which the system may revert to rate damping only.

(3) Yaw. In many ways the artificial augmentation of the directional stability of a helicopter is the most complex function for an AFCS because the mode of operation of the system must change between hover and forward flight. Generally speaking, the AFCS will provide a “feet-off” heading hold (HH) in the low speed regime and a turn-coordination/heading hold (TC/HH) facility above a certain airspeed. Some systems retain the use of the yaw channel to provide simple HH in forward flight while others use TC in yaw and hold heading by adjusting AOB. In forward flight the disengagement of HH or the changeover from HH to TC can be a function of AOB, lateral cyclic displacement, yaw pedal displacement or the operation of a yaw force link or microswitch. The user friendliness of the chosen system should be carefully evaluated and mission related since it may force the pilot to employ a flying technique which is not ideal for the mission. The hover/forward flight mode change is normally a function of airspeed and should be evaluated in a variety of off-trim conditions with various degrees of sideslip, during turns, etc. The initiation of a turn in

forward flight can be problematic, especially for systems which achieve TC via sideslip transducers. Some aircraft (e.g. CH-47C) use a feed forward of the rate of lateral cyclic input into the yaw channel to initiate a yaw rate during turn entry. The extra yaw input is then washed out as the steady-state TC takes over again. The operation of such a system must be evaluated during slow and rapid entries into turns at various bank angles since imperfect operation can lead to adverse or proverse yaw. Any delay in the disengagement of HH during a turn may cause the aircraft to "hang-up" on its original heading as if directional stability was weak. Remember that any system which uses a lateral accelerometer in the yaw channel will be subject to the same sensations as the pilot in terms of sideforce cues and inherent sideslip.

b. *Dynamic Stability*. The fundamental requirement for an AFCS in terms of dynamic stability is to automatically suppress the nuisance modes of the natural aircraft response such as the long term (phugoid) or lateral/directional oscillation (dutch roll). Again the test techniques are similar to the dynamic stability tests which are applied to the unaugmented aircraft and comprise displacement of the aircraft from a trim condition, using either natural or artificial excitation, and observation of its response. The parameters which should be noted are as follows:

(1) Rate and precision of return to trim.

(2) Damping/number of overshoots.

(3) Cross-coupling.

The trim condition may be defined in terms of attitude, heading, airspeed, altitude or any other parameter which may be relevant to the particular AFCS function under scrutiny. The displacement method should also be tailored to the particular function and the following techniques may be employed:

(4) A rapid input with an immediate release to trim (RTT) which may expose problems with lack of damping or the excitation of undesirable response modes. This method may also "beat" switching functions thus revealing problems which could occur during rapid maneuvering.

(5) A slow input followed by a smooth RTT as soon as the required displacement is achieved.

(6) A slow input, holding the displacement for a number of seconds prior to RTT. The pilot should be aware of any build-up of control forces during this type of input since this may be indicative of the unrestricted operation of a parallel actuator which may result in a violent return to trim (and probable overshoot) when the controls are released.

(7) Very small, long period inputs which may be insufficient to disengage an attitude hold and could result in a force build-up as in (6).

(8) Steps, pulses, doublets and frequency sweeps of various magnitudes designed to expose any undesirable AFCS excitation.

(9) Open loop operation of trimmers to displace the AFCS datum without compensatory control movement.

The amount of attitude change from trim will depend upon the particular axis and the axis and flight envelope limits. Changes of ± 3°, ± 5°, and ± 10° are normally sufficient to evaluate short term performance in the pitch axis. Care should be taken at high airspeeds not to exceed VNE during the nosedown inputs. In the roll axis the same inputs should be used, followed by the angle of bank required for a standard rate turn (e.g. ± 15°), ± 30°, and the bank angle limit. In the yaw axis, the displacements will depend upon the sideslip envelope. Changes in heading should be made in increments up to the sideslip limit.

c. *Control Response*. The control response of the aircraft will be affected by AFCS rate damping and control shaping (e.g. control quickening). Control response should be evaluated in the normal way (i.e. using incremental step inputs) with particular emphasis on the smooth operation of any augmentation and a mission appropriate, predictable response. The disengagement of an attitude hold during maneuvering must not lead to jerkiness or unpredictability. Any rate damping will reduce the ultimate agility of the aircraft but this potential disadvantage may be more than outweighed by a qualitative improvement in handling qualities. Ideally the AFCS should completely suppress control cross-coupling, so check for off-axis response. Varying the size of the inputs will enable any limits of augmentation to be determined.

d. *Long Term Test Methods*. Long term performance testing involves the assessment of the AFCS during extended flights. Attitude, altitude, and airspeed holds are evaluated for their long term holding ability. These extended flights should be made in both smooth and gusty air. AFCS gains, which produce acceptable performance in smooth air, may cause a rough ride in gusty conditions. There may well be other features in the AFCS designed to solve this problem or to compensate for particular aircraft deficiencies (although they may not work as advertised). An example is the "Turbulence" mode in the SFIM 155D AFCS fitted to the SA365N; the helicopter suffers from a tendency to lateral/directional oscillation which is increased by turbulence. To try to damp this out, a yaw-roll precontrol signal, which is a function of yaw velocity and roll angle modulated for forward speed, is fed into the AFCS roll channel when this mode is selected. Ideally, AFCS gains should be optimized to provide a smooth, stable ride in calm air, but not be so tight that they cause the system to overwork in turbulence. An example might be a very tight heading hold. With a jump take-off, the heading may not vary more than ± 1°, which is exceptionally tight. But in turbulence, the high gains may cause the nose to return to trim too fast with the resultant high angular accelerations, causing the helicopter to snake through the air. An important point to check for during the extended flights is that the short term performance at the beginning and at the end of the flight is unchanged.

10. **Autopilot Modes.** There are a wide variety of “high-order” autopilot modes, the ultimate example of which is the full-authority digital FCS of the type which will be found in the RAH-66 Comanche. We shall confine ourselves to less esoteric modes which may be found in current generation helicopters. Autopilot modes usually operate through parallel actuators and thus move the cockpit controls so that the pilot can verify

correct response. Ideally such a mode should operate in conjunction with a flight director, not only so that the pilot can monitor the correct operation of the AFCS, but also to permit him to fly the mode manually (e.g. manual TF) should he desire or should there be an actuator failure.

a. *Altitude Hold.* Altitude hold should be evaluated in conditions which require power changes in order to exercise the system. The system's response may be optimized for a particular flight condition and so should be checked throughout the permitted speed and altitude range. Specific points for evaluation are:

(1) Engagements in transient conditions. The maximum allowable rate of climb or descent for engagement which results in an acceptable overshoot should be checked.

(2) Evaluate any measures for the protection of the engine/drivetrain from overtemp or overtorque. The absence of such devices may force a limit on gross weight, AOB, speed or turbulence level. Check that the collective actuator speed is matched to the characteristics of the engine control system such that, for example, an excessively rapid collective movement, which may cause a surge, is impossible.

(3) Ensure that it is impossible to excite an unstable mode of operation by making a series of rapid cyclic inputs. It may be necessary to instrument the drivetrain for this test.

(4) For systems which rely on the use of cyclic to hold altitude (e.g. Puma, SA 365N) it should be impossible to engage the system below minimum power speed. The pilot's choice of power setting will place considerable constraints on the available system performance and these should be investigated. Make sure that control margins are adequate especially at extreme CG positions.

(5) Check that the system can be temporarily disengaged, via a maneuver button, or permanently disengaged without removing the hand from the collective.

(6) Verify the effect of sideslip and other sources of pressure errors on a barometric altitude hold. Generally speaking there should be none since most devices use a sealed aneroid which measures only changes in pressure from the datum of the engagement altitude.

(7) Verify the effects of various surfaces on a radar altitude hold. If the system is unsmoothed it may only be appropriate to use it in the hover regime. Evaluate the effects of an underslung load if appropriate. Make sure there are no engagement errors which could endanger the aircraft when hovering close to the ground. The EMC/EMI characteristics of the sensor should be thoroughly investigated.

b. *Airspeed Hold*. The precise criteria for the engagement (either manual or automatic) of an airspeed hold must be thoroughly understood prior to testing. For example, the UH-60 incorporates a time delay, during which no cyclic inputs may occur, before it switches from pitch attitude hold to airspeed hold. Evaluate the system during climbs, descents, turns and power changes. There is normally a minimum speed for the engagement of an airspeed hold and so engagements at various pitch attitudes and rates of acceleration around this speed should be assessed. Evaluate the ease with which the airspeed datum can be altered and how rapidly the system captures the new value. Pressure errors should be assessed, especially during sideslips and if the system does not tap into the normal aircraft pilot-static circuit.

c. *Navigation Beacon Tracking*. There are many different systems for tracking courses defined by VORs or TACANs and so generalization is difficult. Some of the points which may be considered are:

(1) How accurately does the equipment track the course? An independent navaid (e.g. a map!) will be required to determine this. Does it track smoothly or continuously oscillate about the desired position? How does it cope with crosswind conditions? Is it accurate enough to allow an approach to be flown? Is the performance consistent irrespective of range

from the navaid?

(2) How does it behave when navigation signals become weak in the short term and in the long term? What happens during beacon passage in the cone of silence? Does the system gracefully return to heading hold for instance?

(3) Does the course intercept mode function sensibly? What is the maximum intercept angle? Does the system compute and execute the turn on to course properly without overshoots and/or excessive bank angles? What happens at beacon passage when a new outbound course is selected?

d. *ILS/MLS Tracking*. The evaluation of the automatic tracking of a landing aid is similar to that of a VOR/TACAN coupler except with an added dimension! There are some extra points to consider:

(1) How well does the system switch from altitude hold during the initial approach to glideslope capture? What happens if the glideslope signal fails?

(2) What happens at decision height? Is there a facility for leveling off at, say, a pilot-selected or default radar altitude?

(3) What happens if the pilot changes speed during the approach?

(4) What happens if you attempt to intercept the glideslope from above or below?

(5) Is there adequate indication of satisfactory system performance, especially during the critical stages of the approach?

e. *Automatic Transition and/or Hover Modes*. These modes are normally associated with maritime helicopters although they may have applications to future battlefield aircraft with the introduction of accurate area navaids such as GPS. The following items should be assessed in addition to appropriate elements of the altitude hold tests described above:

(1) System performance from a range of operationally relevant entry gates (i.e. speed/height combinations).

(2) The effects of wind strength, direction and shear.

(3) The accuracy of automatic flyover modes - how close to the marked flyover position is the final hover?

(4) The effects of the use of trim during coupled flight. The effectiveness of any automatic hover trim feature.

(5) The ease of pilot intervention and the cues to system malfunction.

(6) The ease with which the pilot can fly part of the transition manually (e.g. the performance of a manual turn while maintaining the automatic height/speed profile) and the subsequent re-engagement of the full automatic system.

(7) The accuracy of the automatic hover and the nature of the changeover from/to cable/doppler hover. The associated switching and advisory elements should be evaluated as should any drift introduced during the mode change.

f. *Movable Control Surfaces*. Programmable aerodynamic control surfaces are an integral part of the AFCS in many modern helicopters such as the H-60 series and the AH-64. Horizontal stabilators fulfill a variety of functions and contribute to attitude hold, rate damping, reduction of trim changes with power and even control augmentation. The yaw SAS of the MDH 520N operates a movable vertical surface to improve directional stability. Generally, for the purposes of testing, programmable surfaces may be regarded in the same way as any other high-authority AFCS element although the following specific aspects should be assessed:

(1) The stabilator will program with airspeed at a maximum rate which is limited by actuator response. Generally this is fairly slow (about 10 deg/sec) as dictated by the runaway case. Evaluate the system during rapid transitions to and from the hover to assess the impact of stabilator lag. Undesirable nose-down pitch moments may occur, especially at forward CG.

(2) If manual programming is normally permitted, evaluate the complete airspeed/stabilator angle envelope concentrating on FOV and control margins. Ensure that there are no undesirable transients when passing from manual to automatic mode, especially during dynamic maneuvers. The slew switches should be mounted on the normal flight controls.

(3) If manual programming is permitted in an emergency ensure that this does not impose an excessive workload and that the degradation in handling qualities is acceptable throughout the permitted flight envelope.

g. *Actuator Authority*. Most ASE incorporates parallel actuators which perform a trim follow-up function and keep the series actuators close to their null positions automatically by adjusting the positions of the flight controls at an appropriate, slow rate. The presence or absence of such a facility will dictate which of the following sets of testing should be performed:

(1) Observe whether the trim follow-up facility works correctly and permits the low-authority, high-speed series actuators to perform rate damping and control augmentation during maneuvering flight. Displace the aircraft from the trim condition and then make an additional control input to check for damping.

(2) Assess whether the control activity produced by parallel actuator action is acceptable or whether it produces unusual control forces during maneuvering flight.

(3) Make an instantaneous CG change by picking up or releasing an underslung load, releasing stores or winching to see how the system reacts.

(4) Assess the ease with which the pilot can accomplish nulling of the actuators using the manual trim facility (if fitted). Does the system readily go out of authority during normal flying necessitating constant attention to maintain effective stabilization?

MISSION TESTING

11. The results of the performance testing should lead to the recommendation of operating envelopes within which the ASE and autopilot modes may be used safety and effectively. The testing should now be directed towards an evaluation of the AFCS during typical mission tasks under realistic mission environments. If an AFCS is designed for a night over water mission, it should be tested at night and over water. Mission testing will be predominantly qualitative. The pilot must make the decision whether or not the AFCS is an aid toward mission completion. Important keys for this decision are:

a. Does the system do the job it was designed to do?

b. Pilot workload - does the system ease pilot workload or does it create an increased workload, even though it might well do the job?

c. Does the pilot have confidence that the system will aid him in completing his mission or would he be better off without it?

d. Does the system display consistent performance?

Quantitative data can be used to determine specification compliance point to possible reasons for problems. However only qualitative evaluation can decide whether or not the AFCS satisfies its primary requirement - that of easing pilot workload and improving operational effectiveness.

FAILURE MODE TESTING

12. The testing of malfunctions in an AFCS is an important step towards a helicopter release to service. Faults may be "passive" (i.e. no disturbance) or oscillatory (e.g. a position feedback failure) or a full hardover of an actuator. The use of rapid rate (but limited authority) series actuators in autostabilizers increases the likelihood of a malfunction causing a rapid divergence from a selected flight path. Reducing the rate of operation of actuators reduces the likelihood of runaways causing severe handling problems but slow acting actuators alone may be incapable of applying the stability augmentation required by helicopters. The height channel can use slow acting parallel actuators (e.g. Sea King), but in pitch, roll and yaw the AFCS will have rapid rate actuators and therefore any malfunctions causing runaways must be anticipated unless redundancy and self-monitoring are features of a particular system. This aspect is becoming more and more important as digital AFCS and Active Control Technology (ACT) start to become more common in the helicopter world, as indeed is true for the fixed wing community. The helicopter's ability to tolerate runaways of the AFCS is affected by its limited flight envelope in terms of "g", IAS, stress, and the operational requirement that most helicopter mission flying takes place at low level. Furthermore helicopter handling characteristics make the task of a rapid recovery from a runaway difficult.

13. **Pre-Test Study.** Preparation for AFCS failure testing should cover the following 3 aspects:

a. *Specification.* The specification must be interpreted in practical terms against a background knowledge of how the aircraft is likely to be used in service. This is particularly important when describing which criteria of pilot attentiveness to apply for a particular flight condition. Single pilot/2 pilot operation should be considered at this stage.

b. *Failure Analysis.* A detailed study of the AFCS is required to establish the effect of specific failures (e.g. gyro malfunctions, electronic faults etc.). The result of such failures may vary from automatic compensation by a duplex or triplex AFCS to a 3 axes simultaneous runaway as in the case of a failure of the No. 6 transistor in the Lynx ASE. The use of simulators to verify theoretical predictions of the effects of component failures is advantageous but time consuming; it is likely that the manufacturer will use simulation to develop safe characteristics on complex AFCSs. Methods of inducing failures (and the failures themselves) can be various and unexpected, especially if induced by electrical power supply interruptions or fluctuations. Electro-Magnetic Compatibility (EMC) testing is also vital, and may produce some unexpected results.

c. *Aircraft Limits.* A clear definition of the relevant aircraft limits is required. Normally the failure testing will be aimed at establishing that AFCS runaways do not cause a catastrophic failure. Thus it is likely that the sideslip, stress, g and IAS limits applicable for the trial will be beyond those pertaining for normal operations. Detailed structural analysis may be required to verify stress limitations. The ultimate g or stress limit may either occur during the runaway or during the subsequent recovery.

TEST METHODOLOGY

14. **Definitions.** The following definitions of AFCS failure testing terminology are adapted from those outlined in Reference B:

a. *Pilot Involvement.* This is the degree of pilot involvement in, or attentiveness to the flying task. Various levels may be distinguished:

(1) Active Flight. The pilot is continuously flying the aircraft via the flying controls.

(2) Attentive Flight. The pilot has to pay particular attention to flight control for short periods.

(3) Passive Flight. Minimum pilot attention is required.

The attentive and passive phases of flight can be further divided into "hands-on" and "hands-off" according to whether the pilot can release the flying controls for substantial periods. It may be the responsibility of the test team to judge which flight phase criteria are appropriate to the aircraft mission and the maneuvers which this encompasses.

b. *Times and Periods.* The times and periods pertinent to AFCS testing can be defined as follows:

(1) Rotorcraft Response Time. The period between failure occurrence and its recognition by the pilot by means of a suitable cue.

(2) Pilot Response Time. The time which elapses between recognition of the failure and the start of recovery action. Pilot response time comprises 2 elements:

(a) *Decision Time.* Decision time increases as the pilot relaxes his involvement level.

(b) *Reaction Time.* Not surprisingly reaction time is longer during "hands-off" flight phases.

(3) Intervention Time. Intervention time is the sum of rotorcraft response time and pilot response time.

The minimum acceptable pilot response times from reference B are summarized in Table 1 below.

Table 1
Defense Standard 00-970 Pilot Response Times

Flight Segment	Decision Time (Sec)	Reaction Time (Sec)	Pilot Response Time (Sec) (See Note 1)
Active	-	0.5 (See Note 2)	0.5
Attentive Hands on	1.0	0.5	1.5
Attentive Hands off	1.5	1.0	2.5
Passive Hands on	2.0	0.5	2.5
Passive Hands off	3.0	1.0	4.0

Notes: (1) Pilot Response Time = Decision Time + Reaction Time Intervention Time = Pilot Response Time + Rotorcraft Response Time.

c. *Cues*. Failure recognition is a function of the strength of the cues available to the pilot. These may take the form of an aircraft response in terms of angular rate or linear/angular acceleration; the operation of a visual or audio attention getter; or tactile cueing by movement of the flight controls. Malfunctions of rapid-acting actuators will generally produce obvious aircraft response cues. A failure of a slow parallel actuator may produce a dangerously insidious malfunction which may not be readily detected from aircraft response. An example of the latter is the malfunctioning of the Sea King collective parallel actuator which can slowly fly the aircraft into the sea. Inadequate cues may thus allow a catastrophic condition to develop without the pilot being aware that a malfunction has taken place.

15. **Causes of AFCS Failures.** AFCS failures can be caused by the malfunctioning of a wide variety of devices including external sensors, internal components and the actuators themselves. The severity of a particular failure will depend upon the influence of the malfunction on actuator position; the number of axes affected; actuator authority and speed; and whether or not provision has been made to reduce the impact of

the problem by monitors, comparitors, cut-outs or the “backing-off” of other lane(s) of a multiplex system. Simplex components which could cause full-authority, multi-axis hardovers should obviously have a very low probability of failure while the malfunctioning of a component which produces a benign single-axis response could be more easily tolerated. A detailed FMEA will reveal the likely outcomes and their chances of occurrence. Reference B gives guidance in this respect.

16. **Effects of AFCS Failures.** The effects of AFCS failures may be divided into 3 categories:

a. *Soft Failures.* The system in question merely ceases to function without causing any intrinsic aircraft response. There may not be any noticeable effect in the short term.

b. *Runaways.* An actuator is actively driven to an unwanted position. The extreme case is a full-scale, maximum-rate deflection known as a hardover. Runaways may occur in more than one axis simultaneously.

c. *Oscillatory Failures.* Oscillatory failures generally result from the failure of a component in a feedback loop and may be so severe that the pilot is only able to disengage the system using flight control-mounted devices.

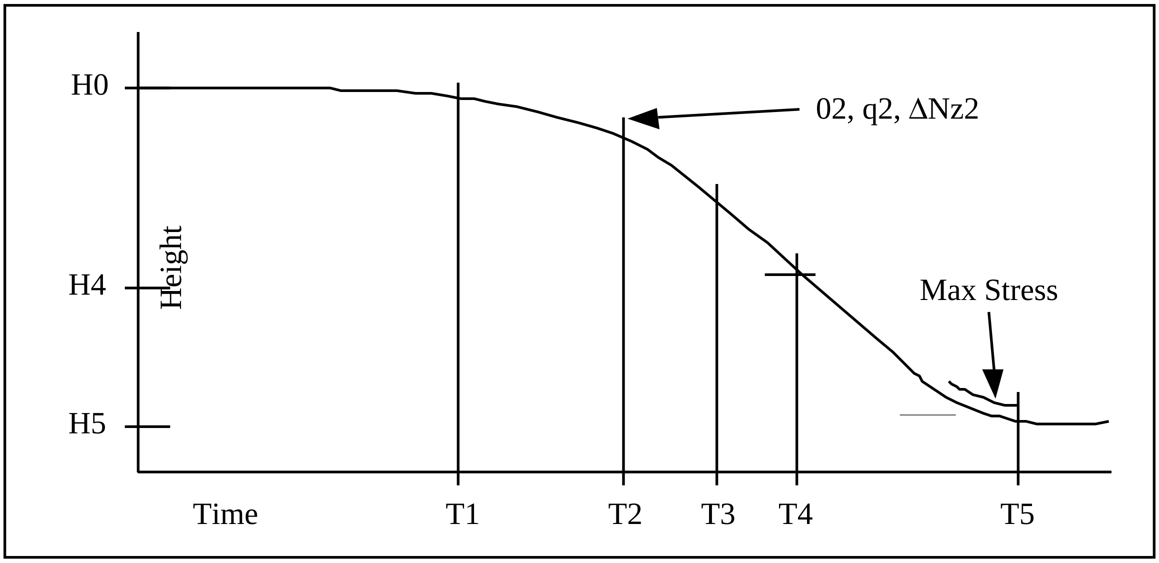
17. **Test Equipment.** It is highly desirable that the test aircraft be specially fitted with actuator position indicators if these are not standard equipment. The most important piece of equipment is a runaway or hardover box which can be used to inject artificial failures into the AFCS and cause actuator movement and consequent aircraft response. The box should be capable of driving each actuator to any position at rates varying from relatively slow (10 sec for full travel) to the maximum and holding the actuator in any position. The simulation of failures which could cause multi-axis runaways may also be necessary. Make sure that the simulated failure can be instantly removed in the event of an emergency.

18. **Test Technique.** The basic test technique comprises the injection of a simulated failure, pilot intervention after an appropriate intervention time and recovery to safe, controlled flight. This discussion will concentrate on the nose-down pitch case (which may well be the most critical, especially at high speed) and Figure 1 shows a simplified time history of such a condition.

It is essential to conduct an incremental build up to the critical failure cases and this takes place in several stages for each flight condition:

a. *Recovery from Unusual Attitudes.* The aircraft is deliberately placed in an unusual attitude representative of that likely at the intervention point and a recovery is executed. An incremental approach to this maneuver is required and real-time stress analysis is highly desirable since, at the most critical flight conditions, the aircraft is likely to exceed its normal flight envelope. Stress levels may have to be extrapolated to permit a limited number of demonstrations of AFCS failures out to the design limit of the aircraft.

b. *Simulated Failures.* A series of ramp and then step control inputs are made to demonstrate likely aircraft behavior after an AFCS failure. The maximum magnitude of the input can be related to the authority of the actuator in question and the ramp speed related to the actuator rate. A recovery is executed after a representative intervention time. A provisional profile for the demonstration runaway(s) can thus be established.



T1	Failure	T2 - T1 = Rotorcraft Response Time
T2	Recognition of Failure	T3 - T2 = Pilot Decision Time
T3	Initiation of Pilot Response	T4 - T3 = Pilot Reaction Time
T4	Recovery Initiation	T4 - T2 = Pilot Response Time
T5	Recovery Complete	T4 - T1 = Overall Intervention Time

$\theta 2$, q2, $\Delta Nz2$ Rotorcraft Response Cues
H0 - H4 = Height Loss Due to Failure
H4 - H5 = Height Loss Due to Recovery

Figure 1
Nose-Down Pitch Channel Runaway Time History

c. *AFCS Failures*. The final phase is the demonstration of AFCS failures using the runaway box. Again, the magnitude and speed of the actuator motions produced should be increased incrementally until they are representative of a maximum-rate, full-deflection hardover. An important consideration in determining the magnitude of the final test input is the method by which the actuator is nulled. If this is achieved automatically (see paragraph 10g) then it is unlikely that the series actuators will be displaced from the null position for a significant period and so a half-authority hardover may suffice. If, on the other hand, actuator nulling is achieved manually then the likelihood of the actuator being nulled when the actual failure occurs will depend on the ease with which the actuator can be reset, whether the offset is readily apparent to the pilot and the offset likely to occur as the aircraft carries out mission representative tasks. Obviously a displaced actuator has the potential to cause a more severe runaway (in the opposite direction to the offset) than one which is nulled. The final decision may well evolve from discussion about probabilities between the user and the manufacturer. Recommendations concerning aircrew training to minimize actuator offset may also be necessary.

d. *Flight Conditions*. Testing should begin in the most benign flight conditions, probably level flight at medium altitude and Vmin pwr, and progress incrementally to the most demanding such as high power climbs or flight at Vne. Any post-failure height loss should be carefully noted as this may have an impact on the operational effectiveness of the aircraft. For example the Chinook HC Mk 1 retains a recommended minimum height of 300 ft due to poor cueing of a nose-down DASH runaway. Testing should also build up to the most critical combinations of CG and loading. Remember that CG position may have an effect on actuator offset and thus runaway severity.

e. *Test Personnel*. Care should be taken to achieve realistic intervention times by avoiding the use of pre-warned pilots who have already experienced several malfunctions similar to the test condition. Ultimately the pilot should be required to perform routine mission tasks appropriate to the flight phase being investigated and then be subjected to unannounced AFCS failures. The use of several pilots will also provide a greater pool of opinion about the suitability of the AFCS for the role.

19. **Assessment Criteria.** There are several factors which may define the end point of AFCS failure testing:

a. The desired intervention time is reached for the most critical failure in a particular flight condition and flight phase.

b. The aircraft is unable to support the stress levels projected to occur at the test point.

c. A handling problem is encountered which would cause problems for the average operational pilot.

d. The minimum control margin is encroached, possibly as a result of the deflection of a series actuator.

e. The height loss encountered is sufficient to prevent useful exploitation of the aircraft in its intended mission while fulfilling the necessary intervention criteria.

20. **Test Results.** Possible outcomes of AFCS failure mode testing are:

a. A flight envelope restriction. This may be quite severe in the case of a single failure of a duplex system since the second failure would be unopposed.

b. AFCS modifications which could include the provision of better failure cueing devices, cut-outs, limiters or comparitors (e.g. barometric altimeter monitoring the gross errors of a radar altitude hold).

c. Operator advice concerning, for example, the nulling of actuators or the flight conditions wherein the AFCS may be engaged. The Chinook HC Mk 1 AFCS must be re-engaged in the same flight conditions as disengagement to avoid engagement transients due to the operation of the DASH.

21. **Training Facility.** The recovery from a runaway may be possible for a test pilot but too difficult for a newly trained squadron pilot. A requirement for training using runaway boxes in flight or in simulators may be specified in the final release. However the use of simulators should be treated with caution, because training is unlikely to be fully representative of an actual runaway inflight; for example, the sudden onset of harsh vibration in the real recovery may deter the pilot from pulling the stick back hard enough.

CONCLUSION

22. AFCS testing is a most challenging blend of the evaluations of aircraft systems and flying qualities. The test pilot must be completely open-minded and use an unbiased analytical approach to ensure accurate results. He should not prejudge system innovations; just because a system is new or it has a fancy display does not automatically make it good or bad. These strong initial subjective impressions can cause the pilot to overlook a bad feature or be overly critical in his assessment. Nevertheless all qualitative

opinion should be recorded as the trial develops, and the use of several pilots will frequently aid in producing the correct analysis of the system. The aircraft will be in service for many years and will invariably return for further evaluation of system modifications or the testing of previously unforeseen failure modes. An accurate and detailed report of the initial AFCS evaluation will ensure that these future trials can be conducted expeditiously.

ANNEX A

GLOSSARY OF HELICOPTER AFCS TERMINOLOGY

Note: These are generic terms only, and are constantly misquoted and misused. Be absolutely accurate in defining your terms and letting your reader know what you are saying.

Actuator - The means of mechanizing the desired output.

Actuator Trim - The means to adjust the actuator position, normally to either preposition it prior to a specific maneuver or to maintain it near the center of travel. May be done automatically or manually.

AFCS - Automatic Flight Control System. The portion of the flight controls that are moved automatically by a device other than the pilot.

Automatic Actuator Trim - The automatic retrimming of an actuator. Generally done to minimize the effects of a runaway in a simplex system.

ASE - Automatic Stabilization Equipment - generally an attitude hold system, it may incorporate rate damping when the flight controls are commanding an input.

Authority - The amount of the control movement that the AFCS can influence. Normally given as percentage of total control movement in each axis, it may also be useful to quote it as an equivalent control deflection.

Autopilot - Will hold an external condition such as altitude, radial or track.

Attitude Hold System - Will hold the commanded attitude in the relevant axes. Note that yaw is an axis, and heading hold is an attitude hold. Normally requires control position information to disable the attitude hold function to permit maneuvering. Some attitude holds may revert to rate damping during maneuvers commanded by the pilot.

Channel - The individual axis of the AFCS, i.e., pitch, roll or yaw, including all the elements of that part of the AFCS, i.e. airspeed hold channel through the cyclic.

Closed Loop - Pilot in the loop of aircraft control - actively involved in flying or maneuvering.

Comparator - The system of comparing signals, either between 2 computers, actuators or other devices to see if they are the same. Generally used as a failure protection device, i.e. to disconnect a channel when the output is in the same direction as the attitude change.

Control Augmentation - The movement of a flight control is sensed and fed forward into the AFCS in such a manner to cause an initial movement of the actuator in the desired direction. This is done to offset the sensation of a reduction in control response that would occur due to the AFCS damping the motion when it senses a disturbance.

Coupled - The ability of the system to perform a specific function using various sensors: i.e. a coupled ILS approach will require certain normal AFCS functions, as well as the use of ILS signals.

Cross-coupled Feedback - Feeding the position of one actuator into both its own computer and the other computers of a multiplexed system. This enables faster fault detection as well as failure protection.

Degraded Modes - The modes of operation other than fully operational. For example, with the pitch channel failed, it may be still possible to have a coupled turn, but not a coupled ILS approach.

Duplex - The duplication of capabilities: i.e. a duplex pitch channel system will have 2 independent pitch systems for redundancy. Note that this is a different definition than you may see elsewhere for duplex.

Failure Modes and Effects Analysis - The analysis of the complete AFCS and related mechanical, electrical and hydraulic systems to assess the effects that failure of the individual parts will have on the system. Generally will concentrate on the internal parts of the computer and go heavily into the effect of failure of certain components (diodes, transistors, etc.). The test pilot must review this to ensure that other items such as airspeed signals and the like are adequately considered.

Feed Forward - The insertion of a signal in the system somewhere downstream of any other signal, so as to ensure that the command has an effect (see Control Augmentation).

Fly-through - Enables the pilot to enter the loop and make a change without having to disengage and then re-engage the AFCS either as a segment or completely.

FMEA - Failure Modes and Effects Analysis (see above).

Gain - The ratio of output to input, which normally translates to the rate of movement of the actuator. The gain of a single channel of a duplex system may double if the other channel should fail, resulting in no perceivable difference to the pilot.

Hardover - The rapid movement of an actuator to the end of its travel, which would not have been commanded by the system given the conditions. Generally refers to a failure case, and is used as the worst condition due to the rapid rate of control surface movement.

Hover - Any time the relative speed is zero. Note that this can mean a zero ground speed hover, a zero airspeed hover, a hover alongside a moving ship, or a zero surface water speed hover. It is extremely important to define the exact terms in this case.

Inner Loop - The inner loop is used to make small adjustments to remove external disturbances. Typically, it does not move the cockpit controls. Series actuators are normally used.

Intervention Time - The time between the insertion of a failure in the system and the pilot taking control. It is comprised of:

a. Time for the failure to generate a response.

b. The time for the pilot to recognize the response.

c. The time to react to the response.

The intervention criteria used must be defined by the agency doing the testing, and as well, it must specify the criteria used to alert the pilot that something is wrong. ("G" onset, pitch or roll acceleration, airspeed change, etc.) and the time that it is expected to take for the pilot to react (i.e. hands on the controls, actively watching the controls/aircraft or busy with other tasks and not monitoring the aircraft/controls at all).

Lane - A British term, #1 lane could mean the #l pitch, roll and yaw channels if there was a second lane. Note that it can mean the same as channel if there is no redundancy.

Linear Variable Displacement Transducer (LVDT) - A transducer for determining control displacement or position, as well as the position of an actuator. A better solution than the conventional potentiometer.

Long Term - The motion or response of the helicopter over a long time period, generally longer than 5 seconds or so.

LVDT - Linear Variable Displacement Transducer. Sometimes also known as Linear Variable Differential Transformer. (see above).

Model Following - The use of a synthetic model of the aircraft/response in the memory of the computer and shaping the output to make the response follow this model.

Multiplex Systems - The use of more than one lane or channel. Simplex is the use of only one lane/channel, duplex uses 2, triplex 3, quadruplex uses 4, etc.

Open Loop - The pilot is not in the loop of flying or maneuvering the aircraft.

Operational Autopilot -An autopilot which will carry out a particular operational maneuver such as transition to a hover, or a coupled ILS. Generally built on top of an autopilot which is built on to of an attitude hold system and incorporates the use of the outer loop/parallel actuators.

Outer Loop - The outer loop of the system is used to make adjustments to ensure that a particular condition is maintained. It is typically done by moving the pilot's controls. Parallel actuators are often used.

Parallel Actuator - An actuator that is placed in parallel with the normal flight controls. Generally slow moving (low gain) and with high authority.

Quickening - The term typically used for control augmentation. Generally done by inserting a commanded input directly to the actuator before the computation has a chance to neutralize the response.

Response Command - To command a specific feature, i.e. rate, attitude, linear velocity and so on, regardless or with consideration of the aircraft/ environmental state. For example, in a rate command system, a 1" lateral cyclic displacement may always give a 10 degree/second roll rate, regardless of airspeed, or another system could give a constant stick force/G at speeds above 40 KIAS and give a constant attitude change per unit deflection below that speed.

Runaway - The movement of an actuator to an undesired position.

SAS - Stability Augmentation System.

Saturated - The inability of the system to work normally due to the relevant actuator being at full travel.

SCAS - Stability and Control Augmentation System. A system that incorporates control augmentation with another form of stability modification. One of the more widely misused terms, as it implies that there is a feed forward (control augmentation) when very often there is not.

Series Actuator - An actuator that moves in series with the normal flight controls. Normally of limited authority and with a high gain (rate of actuator movement).

Short Term - The aircraft response in the very short term following a control input or disturbance - usually within the time frame of 0 to 3 seconds.

Simplex - A system with at least part of the system relying on only one component. Note that if any part of a duplex system eventually relies upon a single component, anything from a gyro to an actuator, then it becomes a simplex system. For purposes of most discussions, this does not include "simple mechanical components" such as flight control tubes or swash plates.

Stability Augmentation System (SAS) - Generally used to describe a system which only dampens rates without regard to the pilot demand for maneuvering. Used to improve the basic flying qualities. Should not affect statics, nor should it move the flight controls. Normally an inner loop system. May incorporate rate integration for pseudo attitude hold.

Transparent - The quality of making the effects of the AFCS invisible to the pilot when he wishes to maneuver. Also known as "Fly-through."

Voting - In a triplex (or higher) system, or one that has a model following system, "voting" will compare outputs from the various systems and (hopefully) eliminate the odd man output, thus protecting against failures.