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Accepted for the Council.

Interim Vice Provost and
Dean of the Graduate School

DEVELOPMENT OF AN AUTOMATED HELICOPTER STABILITY & CONTROL FLIGHT TESTING DATA COLLECTION AND ANALYSIS SOFTWARE SYSTEM

A Thesis

Presented for the

Master of Science

Degree

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Alan Davis

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DEDICATION

This thesis is dedicated to:

Patti
Jessica and Felicia

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I would also like to extend my sincerest thanks to my wife and children who have tolerated the many long hours dedicated to this project

ABSTRACT

This thesis presents the software system developed to collect, present and analyze helicopter stability and control flight test data near real time inflight. The software system is designed to be relatively simple to operate. The functionality of the program is presented in detail with example screenshots of each window in the entire program. The procedures for use of the program are presented in a step-by-step fashion that could serve as the basis for a user's manual.

The data presentation and analysis techniques are discussed from both the user interface and the internal program functionality perspectives. The mathematics used in the computation of the engineering parameters are described in detail.

The flight test techniques and pilot procedures as well as the background theory that were used to determine the appropriate flight test techniques are also presented.

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NOMENCLATURE

$a0$ = *Coning angle.*

a_{1s} = *Longitudinal flapping angle (shaft referenced).*

A = *Amplitude constant affected by the initial conditions of the motion.*

$AFCS$ = *Automatic Flight Control System.*

b = *Number of blades.*

b_{1s} = *Lateral flapping angle (shaft referenced).*

B = *Viscous damping constant.*

$B_{Critical}$ = *Damping constant which satisfies the critical damping case.*

$C_{1/2}$ = *Cycles to one-half amplitude.*

C_2 = *Cycles to double amplitude.*

CG = *Center of gravity.*

e = *Base of natural logarithm.*

e = *Flapping hinge offset.*

f_s = *Sampling frequency.*

f_A = *Aliased frequency.*

Ft = *Feet.*

$f+t$ = *Fuselage and tail.*

FTE = *Flight Test Engineer.*

g = *Gravity.*

h = *Height of the rotor hub above the center of gravity.*

h_{TR} = *Height of the tail rotor above the CG.*

H = *Rotor hub force perpendicular to the mast.*

HQR = *Handling Qualities Rating.*

Hz = *Hertz.*

$\imath$ = *Imaginary index.*

I_{xx} = *Moment of inertia in the roll axis.*

I_{yy} = *Moment of inertia in the pitch axis.*

I_{zz} = *Moment of inertia in the yaw axis.*

IGE = *In ground-effect.*

k = *Spring constant.*

ℓ_m = *Longitudinal distance between the mast and the CG.*

ℓ_t = *Length of the tail.*

ln = *Natural logarithm.*

L = *Roll moment.*

L_p = *Roll moment due to roll rate.*

L_r = *Roll moment due to yaw rate.*

L_v = *Roll moment due to lateral velocity.*

$L_{\delta_{Lat}}$ = *Roll moment due to lateral control.*

$L_{\delta_{Ped}}$ = *Roll moment due to pedal control.*

LDO = *Lateral-Directional Oscillation.*

m = *Mass*

M = *Pitch moment.*

$M_{CG_{f\&t}}$ = *Pitch moment due to aerodynamic forces of fuselage and tail.*

M_S = *Blade mass moment.*

M_q = *Pitch moment due to pitch rate.*

M_u = *Pitch moment due to longitudinal velocity.*

M_w = *Pitch moment due to vertical velocity.*

$M_{\delta_{Coll}}$ = *Pitch moment due to collective control.*

$M_{\delta_{Long}}$ = *Pitch moment due to longitudinal control.*

N = *Yaw moment.*

N = *Number of samples.*

$N_{f\&t}$ = *Yaw moment due to aerodynamic forces of fuselage and tail.*

N_p = *Yaw moment due to roll rate.*

N_r = *Yaw moment due to yaw rate.*

N_v = *Yaw moment due to lateral velocity.*

$N_{\delta_{Lat}}$ = *Yaw moment due to lateral control.*

$N_{\delta_{Ped}}$ = *Yaw moment due to pedal control*

N_x = *Linear acceleration along the x-axis.*

N_y = *Linear acceleration along the y-axis.*

N_z = *Linear acceleration along the z-axis, normal acceleration, normal load factor.*

OGE = *Out of ground-effect.*

p = *Roll rate.*

p = *Roll rate acceleration.*

P = *Period of the oscillation.*

q = *Pitch rate.*

$\dot{q}$ = *Pitch rate acceleration.*

Q_{MR} = *Main rotor torque.*

r = *Yaw rate, radius along the blade.*

$\dot{r}$ = *Yaw rate acceleration.*

R = *Rotor radius.*

sec = *Seconds.*

ss = *Steady state.*

t = *Time.*

T = *Main rotor thrust.*

$T_{½}$ = *Time to one-half amplitude.*

T_2 = *Time to double amplitude.*

T_{TR} = *Tail rotor thrust.*

TPP = *Tip path plane.*

u = *Linear velocity along the x-axis.*

$\dot{u}$ = *Linear acceleration along the x-axis.*

u_0 = *Initial velocity component along the x-axis.*

USNTPS = *United States Naval Test Pilots School.*

v = *Linear velocity along the y-axis.*

$\dot{v}$ = *Linear acceleration along the y-axis.*

v_0 = *Initial velocity component along the y-axis.*

V = *Velocity, Free stream velocity.*

w = *Vertical velocity.*

$\dot{w}$ = Linear acceleration along the z-axis.

w_0 = *Initial velocity component along the z-axis.*

W = *Weight.*

$\dot{x}$ = *Linear displacement.*

X = *Longitudinal force.*

X_q = *Longitudinal force due to pitch rate.*

X_u = *Longitudinal force due to longitudinal velocity.*

X_w = *Longitudinal force due to vertical velocity.*

$X_{\delta_{Coll}}$ = *Longitudinal force due to collective control.*

$X_{\delta_{Long}}$ = *Longitudinal force due to longitudinal control.*

Y = *Lateral force.*

$Y_{f\&t}$ = *Side force due to aerodynamic forces of fuselage and tail.*

Y_p = *Lateral force due to roll rate.*

Y_r = *Lateral force due to yaw rate.*

Y_R = *Rotor side force.*

Y_v = *Lateral force due to lateral velocity.*

$Y_{\delta_{Lat}}$ = *Lateral force due to lateral control.*

$Y_{\delta_{Ped}}$ = *Lateral force due to pedal control.*

Z = *Vertical force*

Z_q = *Vertical force due to pitch rate.*

Z_u = *Vertical force due to longitudinal velocity.*

Z_w = *Vertical force due to vertical velocity.*

$Z_{\delta_{Coll}}$ = *Vertical force due to collective control.*

$Z_{\delta_{Long}}$ = *Vertical force due to longitudinal control*

α = *Angle of attack.*

β = *Sideslip angle.*

δ_{Coll} = *Collective control.*

δ_{Lat} = *Lateral control.*

δ_{Long} = *Longitudinal control*

δ_{Ped} = *Pedal control.*

Δ = *Change in ...*

ϕ = *Roll angle.*

γ = *Flight path angle, Lock number.*

γ_0 = *Initial flight path angle.*

λ = *Characteristic root.*

θ = *Pitch attitude.*

ρ = *Density.*

τ = *Time constant.*

ω_d = *Damped natural frequency.*

ω_n = *Natural frequency.*

Ω = *Rotor angular velocity.*

ψ = *Yaw angle, Blade azimuth angle.*

ζ = *Damping Ratio.*

Chapter 1

INTRODUCTION

Purpose

The objective of this effort was to develop an automated software system for recording, monitoring, and analyzing helicopter stability and control flight test data. The software development project had the following primary design goals:

1. Record the proper data parameters at the appropriate rate to thoroughly document the stability and control characteristics of any helicopter.
2. Have the capability to immediately analyze the data and output engineering values for real-time in-flight assessment.
3. Provide a relatively simple Windows® based graphical user interface that minimizes the technical expertise necessary for the user to collect, analyze and present the aircraft stability and control characteristics.
4. Must be constructed from commercial-off-the-shelf (COTS) products to minimize acquisition and development costs.

Background

Helicopter stability and control flight testing is inherently complicated, considering the complexity of the theoretical equations of motion governing the physics of helicopter motion. Adding to the theoretical complexity is the requirement to record multiple data parameters at a much higher rate than possible using a manual data acquisition system. Finally, the reduction and logical presentation of acquired data is a tedious and error prone process that is typically done after the flight test is complete, the automation of this process would provide test results to the flight test crew for immediate analysis. This would provide the crew with feedback on the quality of the particular test performed.

Additional potential benefits of this system include safety advantages and cost savings. An automated system would reduce the amount of time that the flight crew would need to focus their attention inside the cockpit recording data. This adds a measure of safety to the performance of

the flight test task. Cost savings could be realized by the reduction of lengthy flights required to gather quality flight test data or in the elimination of the requirement for expensive telemetry systems that might be utilized in lieu of this system.

The University of Tennessee Space Institute (UTSI) is currently on contract to design, engineer, and build a highly portable system that can be quickly installed in any helicopter to rapidly assess the aircraft's stability and control characteristics. This software development effort was in direct support of that project.

Chapter 2

DESCRIPTION OF HARDWARE

Overview

The hardware components necessary to compose a functional stability and control analysis system for this project, include the portable aircraft instrumentation sensor package and a laptop or notebook computer. The sensor package described below is the sensor package that the data acquisition software was written to accommodate, however, the data acquisition software can be modified to function with other sensor hardware components. This type of modification is not a built-in capability of the current program and therefore would require rewriting many of the core code modules and a recompilation of the program. This type of modification could only be realistically be completed by a user that is proficient in the G programming language.

Aircraft Instrumentation Sensor Package

The aircraft instrumentation sensor package is designed to be highly portable, allowing it to be easily transferred from aircraft to aircraft with minimal installation time. There are six components that comprise the sensor package, the main sensor box with solid state inertial instruments, four control position wire pull displacement transducers, and an optional battery power source for the instrumentation package. The sensor package outputs the following channels to the laptop computer:

- Pitch Attitude
- Roll Attitude
- Heading
- Longitudinal Cyclic Stick Position
- Lateral Cyclic Stick Position
- Pedal Position
- Collective Control Position
- Pitch Rate
- Roll Rate
- Yaw Rate
- Vertical Acceleration
- Longitudinal Acceleration
- Lateral Acceleration

With the inclusion of additional sensors and, some modifications to the software, additional data channels could be used in the data reduction process. Two pressure transducers could sense

static air pressure and total (static + dynamic) pressure. With these additional channels, airspeed and altitude information would be available for processing during data reduction. Also, strain gauges could be added to the flight controls to measure control input forces. Although these parameters are not necessary for the data reduction processes presented in this work, the data reduction for some flight tests could benefit from the addition of these parameters. The modifications necessary to make the software function with these additional data are substantial and would require a significant programming effort.

The main sensor box is very small having physical dimensions of only 13 x 8 x 6 inches (length, width, height) and weighs less than 8 pounds. This component contains a Crossbow Technology Inc. Digital Measuring Unit-Attitude & Heading Reference System (DMU-AHRS), and two power converters to provide proper output voltages.

The DMU-AHRS is a nine axis measurement system that outputs accurate acceleration, angular rates, attitude and magnetic orientation. [1] The technology used in the DMU-AHRS is a combination of silicon micro-electronic-mechanical system (MEMS) accelerometers, coriolis-effect angular rate sensors, magnetometers, and high performance digital signal processing. Algorithms are embedded in the unit which provide stabilized roll, pitch, and heading information within one degree of accuracy. [1]

The main sensor box should be mounted as close to the center of gravity (CG) of the aircraft as possible This will minimize any "lever effect". If it is not mounted at the center of gravity, then rotations around the center of gravity will cause the DMU accelerometers to measure an acceleration proportional to the product of the angular rate squared and the distance between the DMU and the CG. Also, the DMU-AHRS should not be exposed to large magnetic fields. This could permanently magnetize internal components of the DMU-AHRS and degrade its magnetic heading accuracy.

The control position sensors are Space Age Controls', part number 160-0830, wire pull displacement transducers. These 'string potentiometers' must be installed orthogonal to the flight controls, for proper system calibration. Figure 2-1 shows a typical installation of the system.

The total cost for the entire portable instrumentation system is estimated to be $33,500. [2]

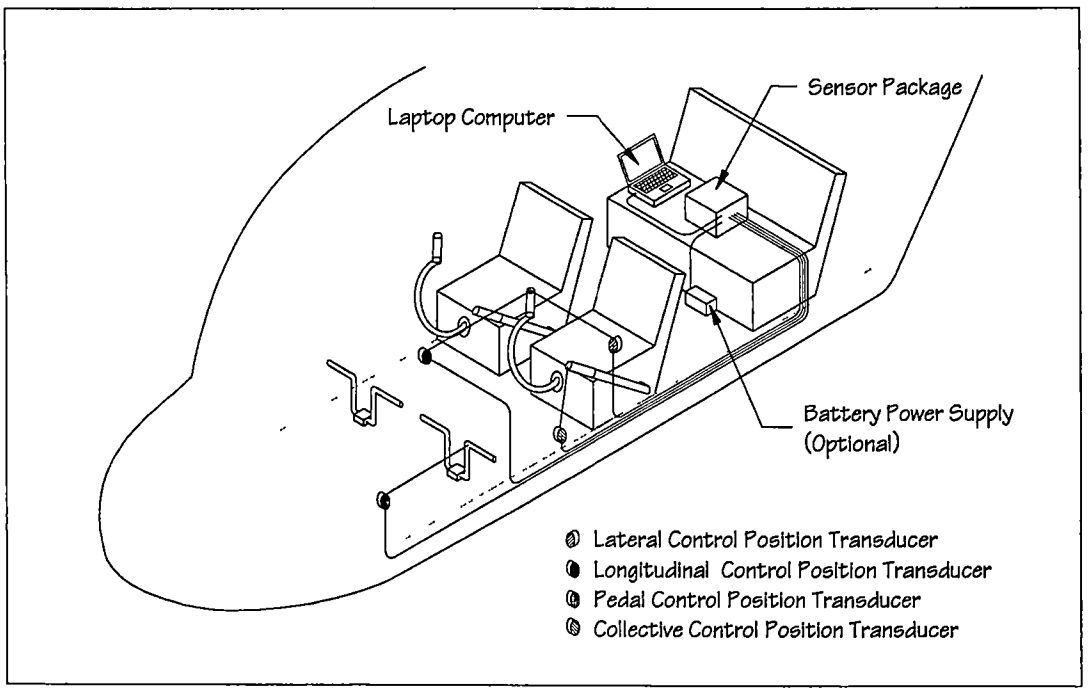


Figure 2-1 Typical Aircraft Instrumentation Sensor Package Installation

Laptop Personal Computer

General

The laptop personal computer is used to control the instrumentation package, collect the required data at the command of the flight crew, present the acquired data in a format that is comprehensible to the flight crew for immediate assessment, and store the data for future analysis and presentation. National Instruments' LabVIEW software was the host application that ran the Helicopter Stability and Control Analysis Software developed in this effort.

System Requirements

The requirements of the laptop computer are largely driven by the LabVIEW software application The Windows 95/NT version of LabVIEW runs on any system that supports Windows 95 or Windows NT 4.0. A minimum of 24 megabytes (MB) of random access memory (RAM) is required for this version to run effectively. 32 MB of RAM and at least 30 MB of swap space available is recommended The standard LabVIEW installation requires approximately 110 MB of hard disk space. A full installation requires approximately 150 MB. [3]

The laptop computer must have the capability to operate at a screen resolution of at least 1024 x 768 to properly view all functions of the software. Also, in order to successfully operate the Helicopter Stability and Control Analysis Software an integrated pointing device (trackball, touch-pad, mouse, or keyboard integrated pointing device) is necessary to perform some of the software functions. Since the operating environment is subject to significant levels of vibration, motion, and g forces, manipulating a touch-pad, trackball, or keyboard integrated pointing device can be difficult. Also, there is not usually adequate space for a traditional mouse and mouse pad. To overcome many of these obstacles a remote, handheld, pointing device is recommended. Specifically, the Fujitsu, part number FID-824, Palm Mouse (figure 2-2) is recommended. With this device, the user manipulates a navigation dome that responds to light thumb pressure to move their cursor. The degree of the dome's tilt angle determines how fast the cursor moves. When thumb pressure is released, the dome automatically returns to its center "home" position. Two other buttons function as right- and left-mouse buttons. The user can swap the function of these buttons and adjust cursor response using the Windows mouse control panel.

The final piece of hardware that is necessary to complete the system is the data acquisition card. This software is designed to work with the IOtech Daq/112B data acquisition PC card. This is a Type II Personal Computer Memory Card International Association (PCMCIA) notebook computer expansion card, which serves as the communication link between the software on the computer and the hardware in the aircraft instrumentation sensor package. This card is an analog to digital converter with built in data buffering to prevent data loss. The maximum sampling rate of this card is 100 kHz, with 12 bit resolution, on up to 16 channels.

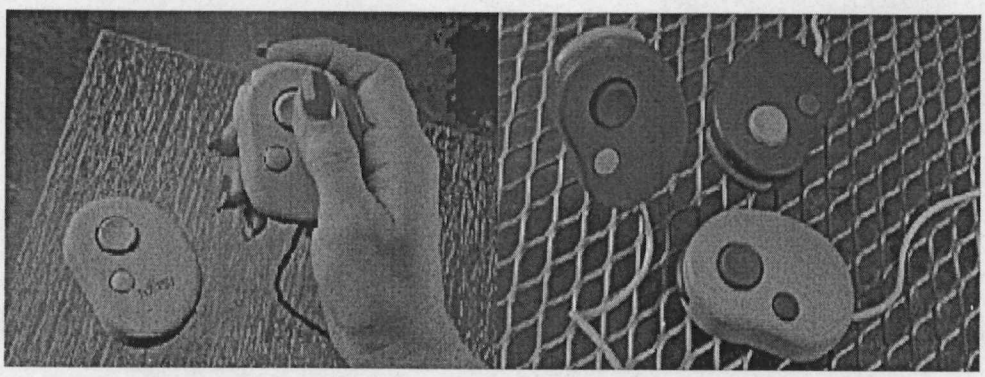


Figure 2-2 Recommended Pointing Device — Fujitsu Palm Mouse

Chapter 3

STABILITY AND CONTROL FLIGHT TESTING

General

Although an aircraft may be designed primarily from performance considerations, it must also possess acceptable handling characteristics. However, good handling characteristics can be achieved by artificial methods. [4] Many aircraft now are increasingly dependent upon automatic flight control systems (AFCS) for good stability characteristics and expanding flight envelopes. Modern flight dynamics is concerned not only with the stability and control of the basic aircraft but also with the often complex interaction between the aircraft and the flight control system. The flight control system comprises motion sensors, flight control computers, control actuators, and other control hardware. Stability and control flight testing is conducted on the aircraft together with the flight control system evaluated as a whole system and flight tested with applicable systems degraded to evaluate possible failure modes.

Flight Testing Definitions

Degrees of Freedom

An aircraft in flight is capable of motion in six degrees of freedom, meaning that the aircraft is free to move in three translational components and three rotational components. The rotational components of motion are equivalent to rolling, pitching, and yawing about the aircraft's center of mass, or more commonly referred to as center of gravity (CG). The translational components are the linear velocity components of forward, lateral and vertical velocity. A helicopter in flight has additional degrees of freedom when the rotor system is considered. The primary motions are the coning, lateral flapping, and longitudinal flapping of the rotor. Although these motions can be significant, it can be shown that the rotor tip path plane corresponds to stimuli well within one revolution (<0.3 seconds) and therefore can be strictly considered as a force and moment generator. [5]

For the purpose of describing the aircraft stability and control characteristics the aircraft will be considered a rigid body. This implies that all points in the airframe maintain fixed relative positions in space at all times. Despite the fact that in actuality there is considerable flexibility in the airframe of most aircraft, the interaction of flexibility effects with the aerodynamics greatly complicates the theoretical model of the motion of the aircraft. The complication added by considering the aeroelastic effects is impractical and sufficient accuracy can be determined using the rigid body model. The equations of motion of a rigid body can be separated into rotational equations and translational equations if the coordinate axis origin is chosen to be at the aircraft center of gravity. [6] For the aircraft to be in equilibrium along any straight unaccelerated flight path, the equation of static equilibrium must be satisfied for each degree of freedom. That is the sums of the forces and the sums of the moments about each axis must all be equal to zero.

Axis System Notation

If the aircraft is to fly steadily along any arbitrary flight path the forces acting on it must be in static equilibrium if the flight path is a straight one, and in dynamic equilibrium if the flight path is curved or accelerated in any way. [7] In order to discuss these equilibriums a standard system of nomenclature and notation must be used to refer to the aircraft orientation and motion. For this purpose the aircraft is represented by a set of orthogonal axes with the origin at the aircraft's center of gravity as shown in figure 3-1. This axis system is a right-hand system with the positive X, Y, and Z axes pointing forward, right, and down respectively. Table 3-1 defines each component of force, displacement, velocity, and inertia for each axis.

Rotor Angle Notation

Just as with the body reference axes system, the rotor angles are defined to aid in the discussion of rotor characteristics. Figure 3-2a depicts the rotor longitudinal angular relationships; and figure 3-2b presents the lateral rotor angles. The longitudinal tilt of the tip path plane in relation to the shaft is the longitudinal flapping angle, a_{1s}. The longitudinal flapping angle is positive when the blade flaps up over the front of the aircraft ($\psi = 180°$). The longitudinal cyclic pitch angle, B_{1s}, is the angle between the shaft plane and the control plane (swashplate plane). B_{1s} is positive when the longitudinal cyclic control is displaced aft. The

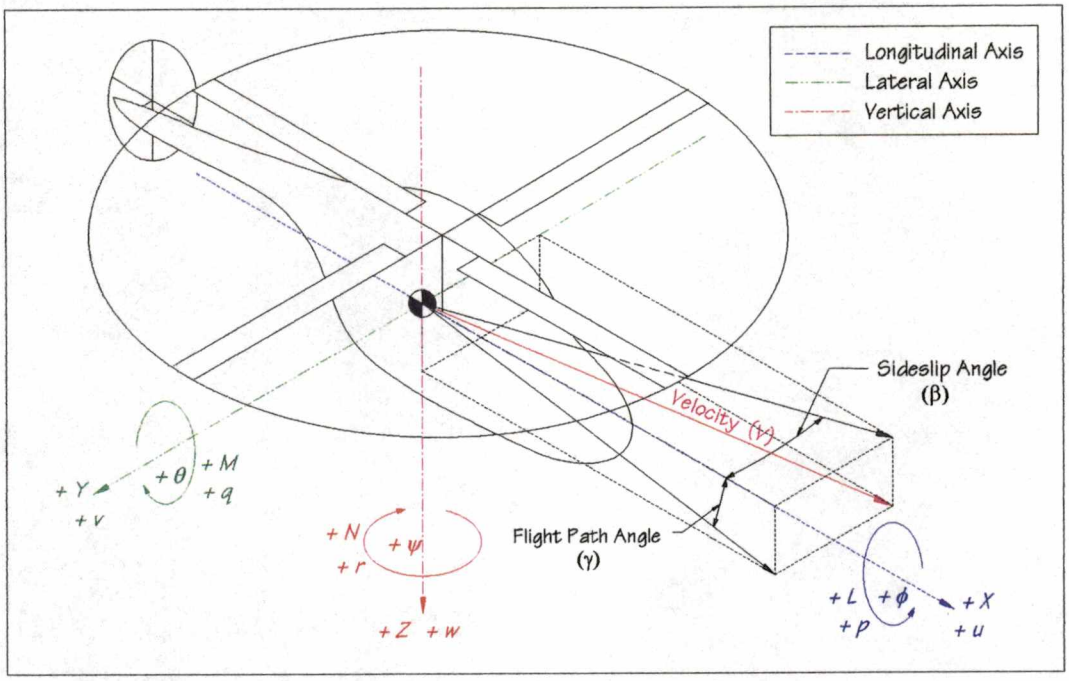


Figure 3-1 Aircraft Axis System Notation

Table 3-1 Axis System Notation Description

Axes / Measurement	x (Positive in forward direction)	y (Positive to starboard)	z (Positive in downward direction)
Force Components	X Longitudinal Force	Y Lateral Force	Z Vertical Force
Moment Components	L Rolling Moment	M Pitching Moment	N Yawing Moment
Angles of Rotation	ϕ Angle of Bank	θ Angle of Pitch	ψ Angle of Yaw
Linear Velocity	u Forward Velocity	v Lateral Velocity	w Vertical Velocity
Angular Velocity	P Rate of Roll	q Rate of Pitch	r Rate of Yaw
Moments of Inertia	I_{XX} In Roll	I_{YY} In Pitch	I_{ZZ} In Yaw

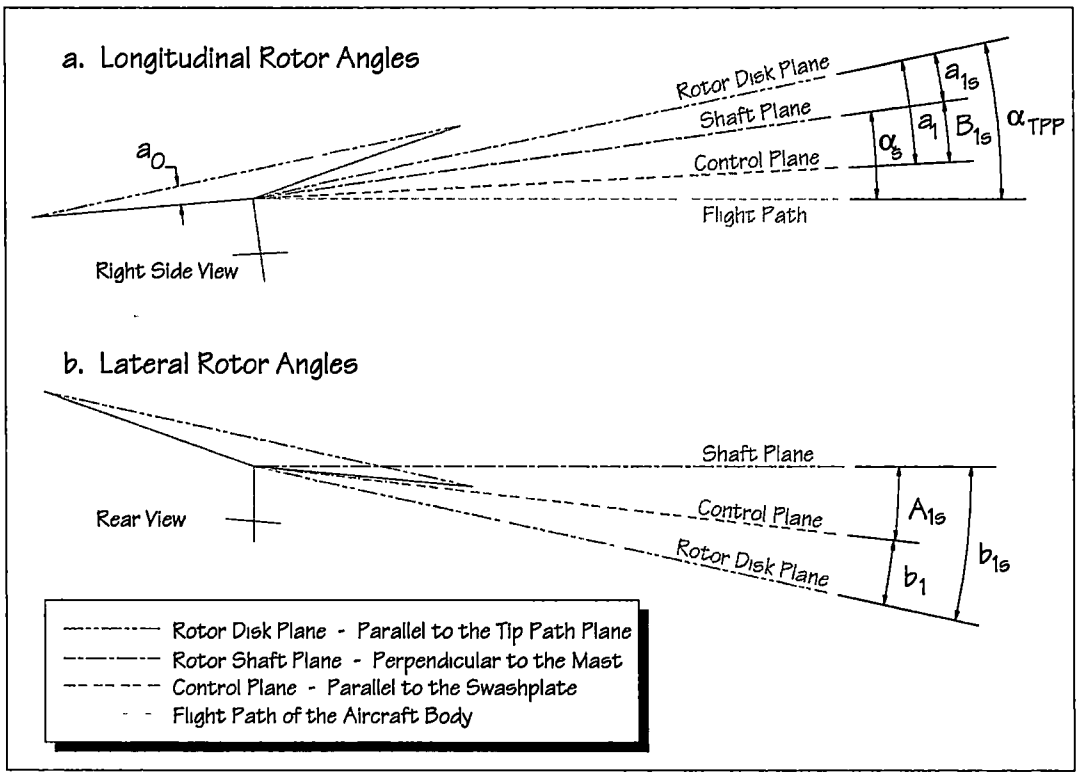


Figure 3-2 Rotor Angular Relationships

lateral tilt of the tip path plane in relation to the shaft is the lateral flapping angle, b_{1s} The lateral flapping angle is positive when the blade flaps down over the right side of the aircraft ($\psi = 90°$). The lateral cyclic pitch angle, A_{1s}, is the angle between the shaft plane and the control plane (swashplate plane). A_{1s} is positive when the lateral cyclic control is displaced right.

Control System Notation

The manipulation of the flight controls can be represented in several ways. For example the longitudinal cyclic control can be measured as a function of the displacement of the control stick (δ_{Long}) or the angular displacement of the swashplate (B_{1s}). The notation used throughout this manuscript for control position is referenced to the displacement of the flight controls in the cockpit where the control positions are measured. Positive control displacement is indicated by a right lateral cyclic, aft longitudinal cyclic, right pedal, and up collective as shown in figure 3-3.

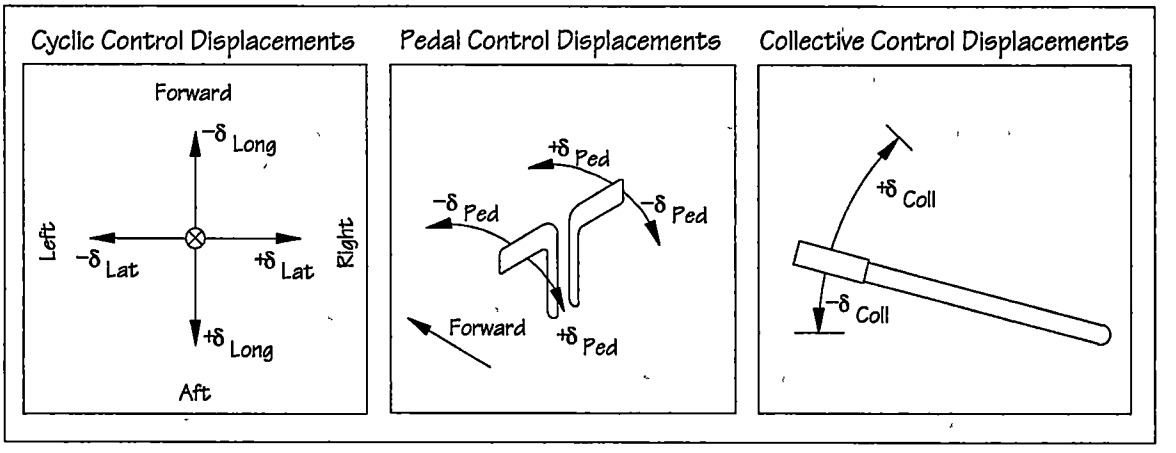


Figure 3-3 Control System Notation

Stability Derivative Notation

The notation used throughout this thesis for the dimensional stability derivatives are normalized with respect to mass for force derivatives and inertia for moment derivatives as follows:

$$X_{(\,)} = \frac{1}{m}\frac{\partial X}{\partial(\,)}\frac{1}{\sec} \qquad \textit{eq. 3-1}$$

$$Y_{(\,)} = \frac{1}{m}\frac{\partial Y}{\partial(\,)}\frac{1}{\sec} \qquad \textit{eq. 3-2}$$

$$Z_{(\,)} = \frac{1}{m}\frac{\partial Z}{\partial(\,)}\frac{1}{\sec} \qquad \textit{eq. 3-3}$$

$$L_{(\,)} = \frac{1}{I_{xx}}\frac{\partial L}{\partial(\,)}\frac{1}{ft\sec} \qquad \textit{eq. 3-4}$$

$$M_{(\,)} = \frac{1}{I_{yy}}\frac{\partial M}{\partial(\,)}\frac{1}{ft\sec} \qquad \textit{eq. 3-5}$$

$$N_{(\,)} = \frac{1}{I_{zz}}\frac{\partial N}{\partial(\,)}\frac{1}{ft\sec} \qquad \textit{eq. 3-6}$$

Where: $X_{()}$ = *Longitudinal force due to ( ) parameter*

$Y_{()}$ = *Lateral force due to ( ) parameter*

$Z_{()}$ = *Vertical force due to ( ) parameter*

m = *Mass*

$( )$ = *Parameter to evaluate the partial derivative with respect to.*

$L_{()}$ = *Roll moment due to ( ) parameter.*

$M_{()}$ = *Pitch moment due to ( ) parameter.*

$N_{()}$ = *Yaw moment due to ( ) parameter.*

I_{xx} = *Moment of inertia in the roll axis.*

I_{yy} = *Moment of inertia in the pitch axis.*

I_{zz} = *Moment of inertia in the yaw axis.*

Aircraft Equations of Motion

The aircraft equations of motion are the foundation on which the entire framework of flight dynamics is built and provide the essential key to a proper understanding of flying qualities. At their simplest, the equations of motion can describe a small perturbed motion about a trim condition. At their most complex they can be completely descriptive, embodying static stability, dynamic stability, aero-elastic effects, atmospheric disturbances, and control system dynamics for a given aircraft configuration. [8] Although it *may* be possible to fully describe the motion of an aircraft mathematically it is impractical to utilize the equations in this form. However, considerable insight into the airframe dynamics may be obtained by simplifying the set of governing equations of motion into a linearized form based on small perturbations about an initial trimmed flight condition. [9] It is these linearized equations of motion that are used as the basis for the development of the flight test techniques presented in this manuscript.

Aircraft Stability and Control

Stability and control are closely inter-related since the amount of stability possessed by an aircraft determines the control forces that the control system must generate to trim and maneuver the aircraft. Aircraft handling is generally concerned with two relatively distinct aspects of aircraft response to stimuli, the short term, or transient response, and the comparatively long term response. The ability of the pilot to handle the short term dynamics of an aircraft is critically dependant on the speed and stability of the response. In other words, the bandwidth of the human

pilot and the control bandwidth of the aircraft must be compatible and the stability margins of the dynamic modes must be adequate. Thus, the requirement for good short term handling tends to be the main consideration in studies of flying and handling qualities. Longer term handling is concerned with the establishment and maintenance of a steady flight condition, which is determined by static stability in particular and influenced by the long period dynamic modes. The dynamic modes associated with long term handling tend to be slow and the frequencies involved are relatively low. [8] Thus, their control is well within the bandwidth of the average human pilot even when the modes are marginally unstable (divergent). Stability and control flight testing measures the aircraft's stability and control characteristics which in turn determine the pilot's ability to position the aircraft in the desired attitude, altitude, speed and direction of flight.

Stability

Stability is the property of a body that causes it when disturbed from a condition of equilibrium or steady motion to develop forces or moments to restore the original condition. Stability can be positive, negative or neutral. Positive stability is characterized by an object's tendency to return to the initial condition. Positive stability is generally considered to be a desirable aircraft characteristic. Negative stability is conversely the object's tendency to move further from the initial position after any disturbance. Neutral stability is characterized by an object's tendency to remain at any new position or rate after a disturbance from the original condition. Stability can be further defined by examining both the initial tendency and the motion of the object over a period of time. These classifications of stability are defined static stability and dynamic stability.

Static Stability

Static stability is the initial and immediate tendency of an object after a disturbance from equilibrium. Static stability can be positive, negative, or neutral as defined above. This initial tendency can best be understood by considering a ball at rest on a surface. If the surface has a concave shape, figure 3-4a, and the ball is displaced (B) from the original position and released the initial motion of the ball will be in the direction of the original position (A). Negative static stability is demonstrated by a ball on a convex surface, figure 3-4b, which indicates any

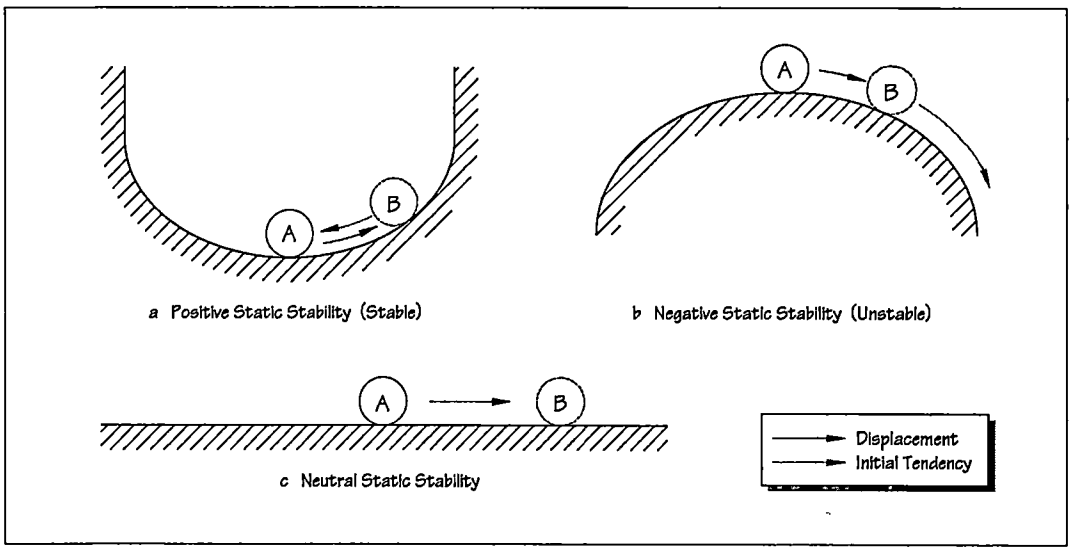


Figure 3-4 Static Stability

disturbance from the initial condition (A) will cause the ball to diverge from the initial position. Figure 3-4c illustrates a neutral static stability condition where disturbance from the initial position (A) results in a new equilibrium state at position B.

Dynamic Stability

Dynamic stability refers to the motion of an object as measured over a period of time. Similar to static stability, the nature of dynamic stability can be positive, negative, or neutral. However, when considering the dynamic stability of an object the static stability characteristics of the object must also be considered For example, an object may exhibit positive static stability while at the same time produce an unstable dynamic response. Additionally, the characteristics of the dynamic response may be oscillatory or non-oscillatory. An oscillatory response occurs when the motion of the object, after a disturbance, moves initially toward the original condition but progresses beyond (overshooting) the initial position. This type of response is only possible if the static stability is positive. There are six possible combinations of the above characteristics, three are oscillatory and three are non-oscillatory as discussed in Table 3-2 and graphically depicted in figure 3-5.

Table 3-2 Static and Dynamic Stability Possibilities

Static Stability	Dynamic Stability	Description	Figure
Non-Oscillatory Responses			
Positive	Positive	A displaced object moves towards the original equilibrium without overshooting and settles down to the original equilibrium condition	3-5a
Neutral	Neutral	A displaced object remains at the new position of equilibrium indefinitely.	3-5b
Negative	Negative	A displaced object diverges continuously away from the original equilibrium condition.	3-5c
Oscillatory Responses			
Positive	Positive	A displaced object initially moves towards the original equilibrium, but overshoots it Motion continues away from equilibrium but only to a displacement that is less than the initial disturbance displacement, then the motion begins in the opposite direction toward equilibrium with another possible overshoot. This periodic motion continues to decay eventually returning to the original equilibrium condition	3-5d
Positive	Neutral	A displaced object initially moves towards the original equilibrium, but overshoots it. Motion continues away from equilibrium to a displacement that is exactly equal to the initial disturbance displacement, then the motion begins in the opposite direction toward equilibrium with another overshoot continuing again to the initial disturbance displacement. This periodic motion continues indefinitely.	3-5e
Positive	Negative	A displaced object initially moves towards the original equilibrium, but overshoots it. Motion continues away from equilibrium but only to a displacement that is greater than the initial disturbance displacement, then the motion begins in the opposite direction toward equilibrium with another overshoot. This periodic motion grows exponentially.	3-5f

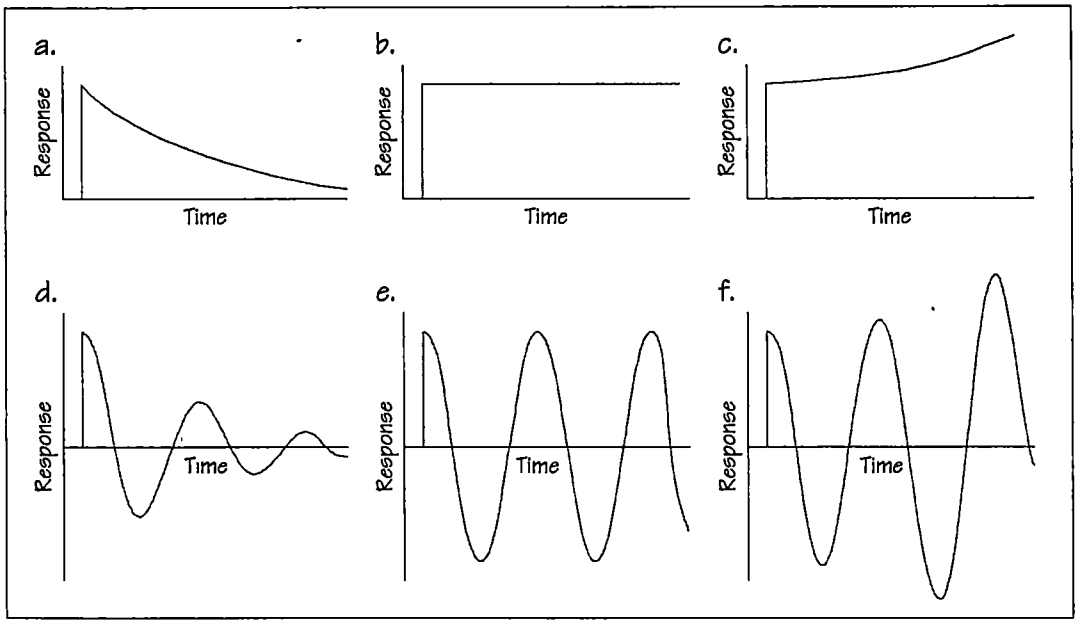


Figure 3-5 Dynamic Stability

Control

Controllability can be defined as the ability of the aircraft to respond to control movements. Two terms are used to define control further: *control power* is the measure of the total moment or force available to the pilot for maneuvering from a steady trimmed flight condition or for compensating for large gust disturbances; *control sensitivity* is the measure of either aircraft acceleration or steady velocity produced by a unit of control motion. [5]

The pilot's opinion of controllability is shaped by several factors. The most apparent of these factors are the initial response of the aircraft to a control input and the total attitude change which results [10] Although there is a connection between stability and controllability, the relationship is not direct For example, an aircraft that has strong positive stability is difficult to control because it is resistant to change from the initial condition and an aircraft that is strongly unstable will also be difficult to control because of it's tendency to diverge from the initial condition. [11] Maximum controllability (without an automatic flight control system) would occur in a an aircraft with neutral stability, in this case the aircraft can be controlled with little effort however there would not be any natural control force cues to indicate an off trim condition. The amount of controllability designed into an aircraft is dependent upon the mission of the aircraft, a transport

type aircraft is designed to be more stable than a attack aircraft indicating that there are trade-offs in the design process. As a general rule, pilots prefer controllability over stability, since it is the increased maneuverability that gives the pilot the feeling of confidence that they can properly position the aircraft [5]

Flying Qualities versus Stability and Control

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. [12] Flying qualities are a *qualitative* evaluation requiring the *subjective judgment* of a trained pilot. Flying and handling qualities are substantially dependent on, and are usually described in terms of, the stability and control characteristics of the aircraft. The dynamic behavior of the aircraft is significantly shaped by its stability and control properties, which in turn have their roots in the aerodynamics of the airframe. [8] An aircraft's stability and control characteristics are of course significant in the pilot's opinion of an aircraft's flying qualities; however there are several other factors including flight control system, the cockpit controls and displays, the environment, the pilot workload, and stress, which influence a pilot's evaluation of an aircraft's flying qualities. Aircraft stability and control characteristics are determined by open loop flight testing; whereas, closed loop flight testing techniques are used to determine an aircraft's flying qualities.

Closed Loop Flight Testing

Closed loop flight testing consists of performing mission relevant flight tasks and evaluating the resulting man-machine performance.[12] Specifically, a task, conditions, and performance standards are established and the pilot attempts to perform the task by manipulation of the flight controls. While performing the task he monitors the aircraft response through all available cues and makes corrective control inputs as necessary to meet the performance standards. This process is called closed loop or pilot-in-the-loop flight testing because the pilot monitors the aircraft via a feedback path and actively attempts to make corrective inputs to null existing discrepancies.

Open Loop Flight Testing

Open loop testing consists of measuring the aircraft's response to natural or artificial disturbances and is conducted without the pilot actively manipulating the flight controls. The resulting motion is used to determine and quantify the aircraft's stability and control characteristics. The data acquired from this type of testing will yield engineering parameters of time constants, natural frequency, damping, and stability characteristics. Typically, time histories of pertinent parameters are recorded and analyzed to determine the aircraft response characteristics, which are categorized into two groups, first order systems and second order systems as described in the Low Order Equivalent Systems section below.

Open Loop Test Methods and Techniques

Open loop dynamics data are obtained to explain specific closed loop data, to document the control system or aircraft response characteristics, to ensure safe operation of the aircraft at corners of the operational flight envelope 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. [12]

Trim

The pilot must establish stabilized initial conditions to provide adequate reference for time histories and comparison to off trim flight conditions. The initial conditions can be achieved by either trimming all control forces to zero or by holding the control fixed against the control force while maintaining equilibrium flight conditions. If the test requires precise control inputs to create the disturbance, the pilot trims all control forces to zero. If the test utilizes a release from an off trim condition to create the disturbance, the pilot stabilizes at the equilibrium conditions by holding the off trim control forces. [12]

Input Shape

Whether the input is a step, pulse, or doublet (figure 3-6) depends on several factors including the nature of the response mode to be excited, whether it is to be excited selectively, the data

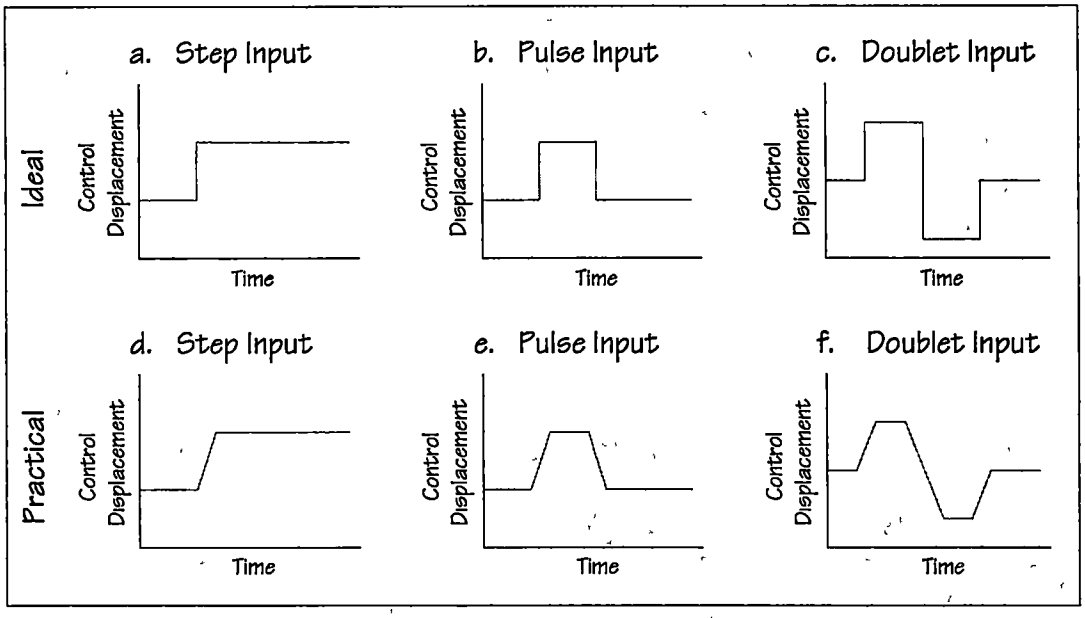


Figure 3-6 Open Loop Flight Testing Control Input Shapes

reduction technique, and sometimes on specific data constraints. For the pulse and doublet, the controls must be returned to the initial trim position. The controls are returned to trim to eliminate any forces or moments generated by the control displacement and to record the free response (without a forcing function) after the initial disturbance. For doublets, the test pilot makes a symmetrical input. Generally step inputs are used for first order responses; while steps, pulses, and doublets are used for second order responses. [12]

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, nonlinearalities of aerodynamics with airspeed might distort the data. Additionally, if the input is large, the aircraft can reach a test limit prior to observing the complete response. On the other hand, it is desirable to correspond open loop data to realistic input magnitudes used to perform closed loop mission representative 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.

Quickness of Input

The input frequency should maximize the disturbance at the response frequency. To approximate the natural frequency the pilot can make a series of sinusoidal, symmetrical control doublets. Start at a low frequency, and observe the response Increase the frequency until the largest response is observed. Once the input frequency is determined, this frequency is used for subsequent control inputs.

Excitation Methods

Natural Excitation — No conscious pilot introduced 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.

Artificial Excitation — The use of the control input shapes, amplitude and quickness discussed above are used to introduce a perturbation from the equilibrium flight condition. If an artificial excitation method is used, the disturbance should result in an aircraft response similar to a response following a natural disturbance.

The method of excitation can determine the type of response documented. Natural disturbances resulting in 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 or impossible.

Low Order Equivalent Systems

The physical motion of the helicopter in flight can be expressed mathematically by the equations of motion as discussed in the Aircraft Equations of Motion section above. These general equations can be simplified by maintaining specific variables constant during the conduct of a flight test. Often the complicated equations of motion can be further simplified by making one or more generally accepted assumptions. The result of this simplification is that the motion of the aircraft can usually be represented by a first or second order linear differential equation. A first order system would be representative of a motion that is only constrained by a damping

force A second order system is characterized by a motion that is influenced by a restoring force and a damper. If the helicopter is considered as a rigid body the inflight modes of motion can be depicted satisfactorily with these two descriptive systems.

First Order Responses

A mass whose motion is only restrained by viscous damping is a typical example of a first order response and is mathematically generically described by the following first order linear homogeneous differential equation.

$$\dot{x} - D_x \cdot x = 0 \qquad \textit{eq. 3-7}$$

Where: x = *Angular or linear acceleration.*

D_x = *Damping derivative.*

x = *Angular or translational velocity.*

Specifically in a helicopter the first order response solutions are represented in by following equations:

$$\dot{p} - L_p \cdot p = 0 \qquad \textit{eq. 3-8}$$

$$\dot{q} - M_q \cdot q = 0 \qquad \textit{eq. 3-9}$$

$$\dot{r} - N_r \cdot r = 0 \qquad \textit{eq. 3-10}$$

$$\dot{w} - Z_w \cdot w = 0 \qquad \textit{eq. 3-11}$$

Where: $\dot{p}$ = *Roll rate acceleration*

L_p = *Rolling moment due to roll rate*

p = *Roll rate.*

$\dot{q}$ = *Pitch rate acceleration.*

M_q = *Pitching moment due to pitch rate.*

q = *Pitch rate.*

$\dot{r}$ = *Yaw rate acceleration.*

N_r = *Yawing moment due to yaw rate*

r = *Yaw rate.*

$\dot{w}$ = *Linear acceleration along the z axis.*

Z_w = *Vertical force due to vertical velocity.*

w = *Vertical velocity.*

The generic characteristic equation has a real root λ which is a negative number and equal to the damping derivative. Solving this equation for the velocity yields the time response shown in equation 3-12 and graphically depicted figure 3-7.

$$x = C_1 \cdot e^{\lambda t} \qquad \textit{eq. 3-12}$$

Where: x = *Angular or translational velocity.*

C_1 =*Constant affected by the initial conditions of the motion.*

e = *Base of natural logarithm.*

λ = *Characteristic root.*

t = *Time.*

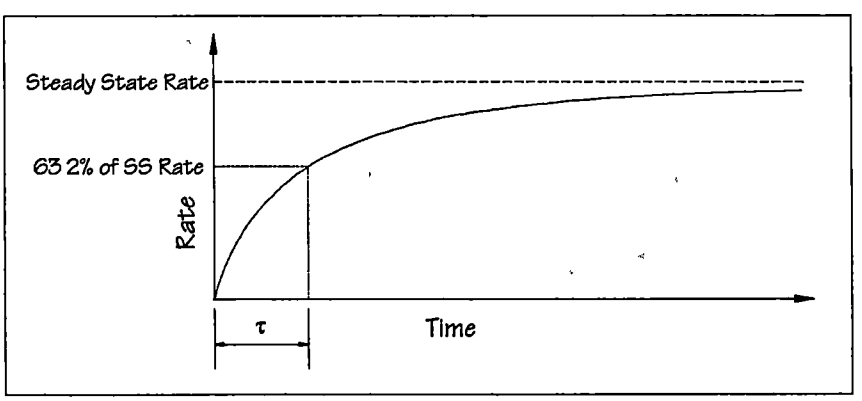


Figure 3-7 Typical Stable First Order Response

This time response graphic depiction can be characterized by determining the time required for the motion to reach 63.2% of the steady state rate, this time is referred to as the time constant, τ. The time constant, once determined from flight test data, is used to calculate the characteristic root (the damping derivative) of the mode tested by the following equations.

$$\lambda = -\frac{1}{\tau} \quad \textit{and...} \qquad \textit{eq. 3-13}$$

$$\lambda = L_p \quad \textit{or...} \qquad \textit{eq. 3-14}$$

$$\lambda = M_q \quad \textit{or...} \qquad \textit{eq. 3-15}$$

$$\lambda = N_r \quad or... \qquad \textit{eq. 3-16}$$

$$\lambda = Z_w \quad or... \qquad \textit{eq 3-17}$$

Where: λ = *Characteristic root/damping derivative (L_p, M_q, N_r, or Z_w).*

L_p = *Rolling moment due to roll rate (Roll Damping).*

M_q = *Pitching moment due to pitch rate (Pitch Damping).*

N_r = *Yawing moment due to yaw rate (Yaw Damping).*

Z_w = *Vertical force due to vertical velocity (Heave Damping).*

Second Order Responses

Many physical systems, including several important aircraft responses, exhibit the properties of a spring, mass, damper system. A simple illustration depicting the forces acting on a typical spring, mass, damper system are shown in figure 3-8.

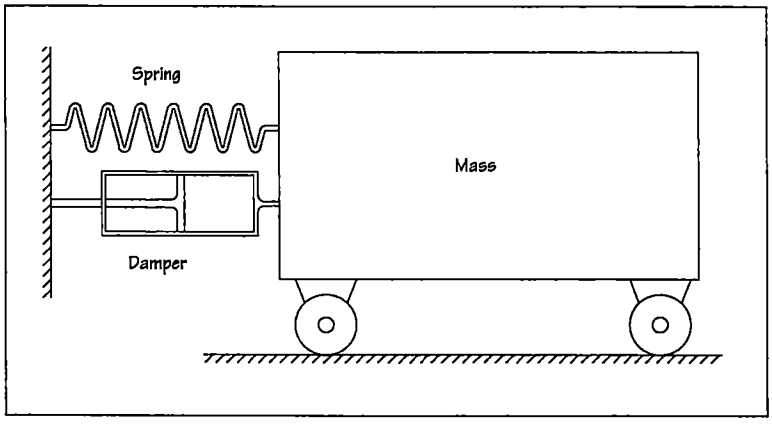


Figure 3-8 Spring Mass Damper System

In this system, the spring provides a linear restoring force that is proportional to linear displacement of the mass, and the damping device provides a damping force that is proportional to the velocity of the mass.

The mathematical model for response of this type system is a second-order differential in the form of equation 3-18.

$$F(t) = m\frac{d^2x}{dt^2} + B\frac{dx}{dt} + kx \qquad \textit{eq. 3-18}$$

Where: t = *Time.*

x = *Linear displacement.*

m = *Mass.*

B = *Viscous damping constant.*

k = *Spring constant.*

The function F(t) is called the forcing function. If the forcing function is 0, the response of the system is referred to as the free response. When the system is driven by a forcing function the response is referred to as the forced function. [13]

It is well known from the theory of ordinary differential equations that the solution of any linear, ordinary differential equation with constant coefficients is always of the form; [14]

$$x = Ae^{\lambda t} \qquad \textit{eq. 3-19}$$

Where: A = *Constant affected by the initial conditions of the motion.*

e = *Base of natural logarithm.*

λ = *Characteristic root.*

t = *Time.*

and substituting into the differential equation yields

$$\lambda^2 Ae^{\lambda t} + \frac{B}{m}\lambda Ae^{\lambda t} + \frac{k}{m}Ae^{\lambda t} = 0 \qquad \text{eq. } 3\text{-}20$$

or...

$$\lambda^2 + \frac{B}{m}\lambda + \frac{k}{m} = 0 \qquad \textit{eq. 3-21}$$

which is called the characteristic equation. The roots of this characteristic equation are called the characteristic roots of the system. The roots of equation 3-21 are...

$$\lambda_{1,2} = \frac{-B \pm \sqrt{B^2 - 4mk}}{2m} \qquad \textit{eq. 3-22}$$

or...

$$\lambda_{1,2} = -\frac{B}{2m} \pm \sqrt{\left(\frac{B}{2m}\right)^2 - \frac{k}{m}} \qquad \textit{eq. 3-23}$$

The type of motion that occurs if the system is displaced from equilibrium and released depends upon the physical constants of the problem (the mass, spring constant, and the damping constant). Three possibilities exist:

First if...

$$\frac{B}{2m} > \sqrt{\frac{k}{m}}$$

Then the roots are negative and real, indicating that the motion is stable, non-oscillatory, and will diminish over time. This type of response is referred to as an over damped motion.

Second if...

$$\frac{B}{2m} = \sqrt{\frac{k}{m}}$$

This represents the boundary between the over damped exponential motion and the damped sinusoidal motion. This response is referred to as the critically damped response. When this situation exists the motion will return to the trim condition as soon as physically possible without any over shoot.

Lastly if...

$$\frac{B}{2m} < \sqrt{\frac{k}{m}}$$

Then the roots are complex, indicating that the motion is oscillatory in the form of a damped sinusoidal having a frequency given by:

$$\omega = \sqrt{\frac{k}{m} - \left(\frac{B}{2m}\right)^2} \qquad \textit{eq. 3-24}$$

Where: ω = *Frequency.*

k = *Spring constant.*

m = *Mass.*

B = *Viscous damping constant.*

The level of dynamic stability of a second order system is generally expressed in terms of the damping ratio, ζ. This is the ratio of the real system damping constant to the damping constant which makes the system critically damped.

$$\zeta = \frac{B}{B_{Critical}} \qquad \textit{eq. 3-25}$$

Where: ζ = *Damping ratio.*

B = *Viscous damping constant.*

$B_{Critical}$ = *Damping constant which satisfies the critical damping case.*

The damping ratio can be expressed as a function of the exponential damping coefficient ($-\zeta\omega_n$) and the damped natural frequency as follows:

$$\zeta = \frac{-\zeta\omega_n}{\omega_d}\sqrt{\frac{1}{1+\left(\frac{\zeta\omega_n}{\omega_d}\right)^2}} \qquad \textit{eq. 3-26}$$

$$\zeta = -\frac{\zeta\omega_n}{\omega_d}\sqrt{\frac{\omega_d^{\,2}}{\omega_d^{\,2}+(\zeta\omega_n)^2}} \qquad \textit{eq. 3-27}$$

$$\zeta = \frac{\sqrt{\omega_n^{\,2}-\omega_d^{\,2}}}{\omega_d} \qquad \textit{eq. 3-28}$$

And the frequency of the oscillation given by equation 3-24 is referred to as the damped natural frequency, the undamped or natural frequency of the system is:

$$\omega_n = \frac{\omega_d}{\sqrt{1-\zeta^2}} \qquad \textit{eq. 3-29}$$

Where: ω_n = *Natural frequency.*

ω_d = *Damped natural frequency.*

ζ = *Damping ratio.*

Chapter 4

SOFTWARE DEVELOPMENT

Typographic Conventions

This manuscript uses standard typographic conventions when referring to the software functions and user interface. Table 4-1 shows examples of the conventions used.

Table 4-1 Typographic Conventions

Function	Examples
Keyboard keys are enclosed by angle brackets.	Press <Enter> Press <Tab> Press <F1>
Key combinations ("Hot-keys") are keyboard keys joined by a plus sign. This means to press and hold the first key and then press the second key.	Press <Alt> + <X> Press <Shift> + <F7> Press <Ctl> + <S>
Buttons and Tabs on the screen are enclosed by straight brackets.	Click [OK] Click [Cancel] Click [Help]
Labels and other text on the screen are enclosed by single quotes.	'Sample Rate' 'Pitch Rate' 'Display Length'
Pull-down or pop-up menus and submenus instructions are separated by the pipe (|) symbol.	Select File | Save Select Calibrate | DMU-AHRS Select Edit | Copy
Information to be typed by the user appears in bold face.	Type **A:\setup**

Software Definitions

Table 4-2 lists the computer software definitions used throughout this manuscript. These definitions are provided for the understanding of basic terminology associated with the use of personal computer software applications.

Table 4-2 Software Definitions

Term	Definition
Click	To tap on a mouse button, pressing it down and then immediately releasing it. Note that clicking a mouse button is different from pressing (or dragging) a mouse button, which implies that you hold the button down without releasing it. The phrase to click on means to select (a screen object) by moving the mouse pointer to the object's position and clicking a mouse button.
Control	An object in a window or dialog box that allow user adjustment or input. Examples of controls include push-buttons, sliders, check boxes, and drop-down lists.
Dialog Box	A box that appears on a display screen to present information or request input. Typically, dialog boxes are temporary -- they disappear once the user has entered the requested information.
Double Click	Some operations require a double click, meaning that the user must click a mouse button twice in rapid succession. The time interval between the clicks is determined by the setup of the host computer.
Field	A space allocated for a particular item of information. Most fields have certain attributes associated with them. For example, some fields are numeric whereas others are textual.
Focus	When an object has the focus, it can receive input from a user. In the Microsoft Windows interface, several applications can be running at any time, but only the application with the focus will have an active title bar and can receive user input. Within the active application only one control can receive user input by means of the keyboard or pointing device. The control with the focus will generally have a visual cue (highlight, blinking cursor, etc.) to indicate the focus.
Hot Key	A key sequence that executes a command.
Indicator	A window object that displays data or output.
Menu	A list of commands or options from which the user can choose. The user can choose an item from the menu by highlighting it and then pressing the Enter or Return key, or by simply pointing to the item with a mouse and clicking one of the mouse buttons.
Popup Menu	A menu that appears temporarily when the right mouse button is clicked on a selection. Once a selection is made from a pop-up menu, the menu automatically disappears.
Select	To choose an object so that you can manipulate it in some way. To select an object, move the pointer to the object and click a mouse button or press the <Tab> key repetitively until the control has the focus.
Window	An enclosed, rectangular area on a display screen in which the graphical user interface is displayed.

Development Environment

National Instruments' LabVIEW software is the host application that runs the Helicopter Stability and Control Analysis Software developed in this effort. LabVIEW was chosen for this role because of the powerful built-in data acquisition, signal processing, data analysis, data presentation and data storage capabilities. LabVIEW combines an easy-to-use graphical interface with the flexibility of a powerful programming language. It offers an intuitive environment, tightly integrated with measurement hardware that permits the programmer to rapidly produce solutions for data acquisition, data analysis, and data presentation

G is the programming language at the heart of LabVIEW. G, like C or BASIC, is a general-purpose programming language with extensive libraries of functions for any programming task. G includes libraries for data acquisition, General Purpose Interface Bus (GPIB) and serial instrument control, data analysis, data presentation, and data storage. It also includes conventional program debugging tools, breakpoints can be set, execution can be animated and/or single-stepped through the program to see how data passes through the program. This makes debugging and program development easier. G differs from those programming languages in one major aspect. Other programming languages are text-based while G is graphical.

G is a general-purpose programming system, but it also includes libraries of functions and development tools specifically designed for data acquisition and instrument control. G programs are called *virtual instruments* (*VIs*) because their appearance and operation can imitate actual instruments. However, VIs are similar to the functions of conventional programming languages.

A VI consists of an interactive user interface, a dataflow diagram that serves as the source code, and icon connections that set up the VI so that it can be called from higher level VIs. More specifically, VIs are structured as follows.

- The interactive user interface of a VI is called the *window* or *front panel*, because it simulates the panel of a physical instrument. The front panel can contain knobs, pushbuttons, graphs, and other controls and indicators. You enter data using a mouse and keyboard, and then view the results on the computer screen.

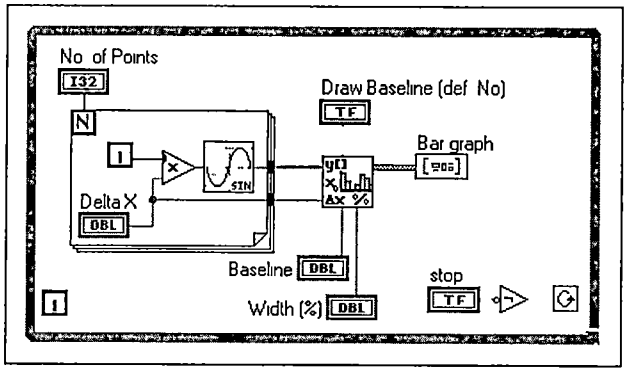


Figure 4-1 G Programming Block Diagram Example

- The VI receives instructions from a *block diagram*, which is constructed in G The block diagram is a pictorial solution to a programming problem The block diagram is also the source code for the VI An example block diagram is shown in figure 4-1
- VIs are hierarchical and modular They can be used as top-level programs, or as subprograms within other programs or subprograms A VI, when used within another VI, is called a *subVI* The *icon and connector* of a VI work like a graphical parameter list so that other VIs can pass data to a subVI

With these features, G makes use of the concept of modular programming The application can be divided into a series of tasks, which can be divided again until a complicated application becomes a series of simple subtasks A VI is built to accomplish each subtask and then those VIs are combined on another block diagram to accomplish the larger task Finally, the top-level VI contains a collection of subVIs that represent application functions

Because each subVI is executable individually, separate from the rest of the application, debugging is much easier Furthermore, many low-level subVIs often perform tasks common to several applications, so specialized set of subVIs are developed which can be reused in future applications [15]

Software Testing and Debugging

The software developed in this effort was tested and debugged continually during the development process The testing included two major parts, the user interface functionality and the processing of the data Functional testing of the user interface was straightforward This

included verify that each component of the interface completed the intended program task. For example, clicking a menu or button on the window should execute the appropriate task. This was completed in a systematic method to ensure the entire user interface was functional.

Data processing testing was more complicated and included several methods of providing input data to the software data processing routines. During the development of the this software the hardware components of this system were not available. This meant that data acquisition from the actual aircraft was not possible. Three methods were used to provide artificial data to the program First a math model was used to generate a pure signal that represented an appropriate system response (first order/second order). This technique was used to validate the data analysis functions and to ensure the data flow through the program was correct. Once the data flow was validated, the math model was modified to introduce a randomized scatter into the pure signal to qualitatively assess the robustness of the data analysis functions. Next a control on the user interface was added that allowed a simulated input signal to be fed to the to the program. This method removed the math model and provided data that often was not compliant with the theoretical math model. This allowed for further assessment of the robustness of the analysis functions. The final method of data input was to manually input data that were recorded from a different data recording device installed in another aircraft. This method was intended to validate the software data processing routines with real aircraft data. In actuality the recorded data that was available had a high level of noise and was ineffective in providing meaningful assessment of the software data processing capabilities. This testing did highlight the fact that if the signals received from the designed hardware are noisy, then there may need to be additional signal processing routines added to the software. This additional signal processing capability could easily be added to the data acquisition routines. Several implementation methods for adding signal processing could be used One method is to apply a set of predetermined filters to the necessary data channels in the design of the software code. With this approach the user is not burdened with understanding and applying the necessary filtering However, this method is totally inflexible and could be ineffective in some circumstances A much more flexible method would allow the user to apply and adjust the filters for each data channel. Although this method would be fully customizable to the specific application, it would require a comprehensive user

understanding of the data filtering requirements. Also, implementation of this method would require an extensive programming effort to develop a practical user interface to apply the desired data filtering.

Program Flow

During the preliminary design phase of the development of the Helicopter Stability and Control Flight Testing program a simple user interface was a primary design driver. The overall scheme of the software was envisioned to be 'Switchboard' window with a series of choices to select the desired flight test. Once selected, the flight test window would allow the user to collect the flight data, analyze it and allow the user to save the data if desired. This model is very simple as shown in figure 4-2.

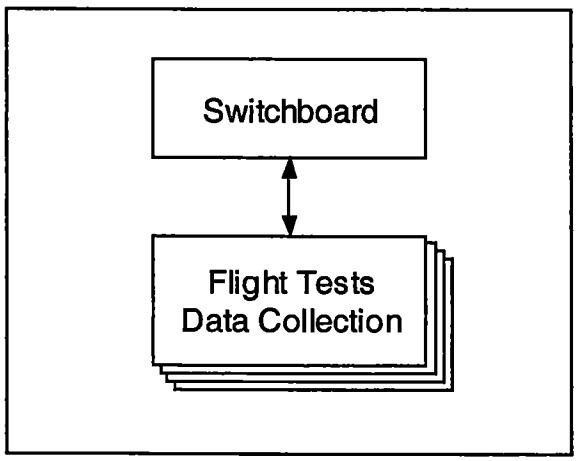


Figure 4-2 Initially Envisioned Program Block Diagram

The simplicity of this model proved to be impracticable It was impossible to design good interface windows that provided data collection, presentation and analysis in one window. It became clear that it was necessary to separate the data collection from the presentation and analysis. The final model that was implemented in the Helicopter Stability and Control Flight Testing program is shown in figure 4-3. Independent data presentation windows provided an opportunity to format the data in a fashion that was compatible with the desired printed data output. The second benefit of dividing the data collection and data presentation windows is the ability of collecting data in flight without having to spend time with detailed analysis inflight. Data can be collected and saved to disk, then after the flight the data can be reopened and

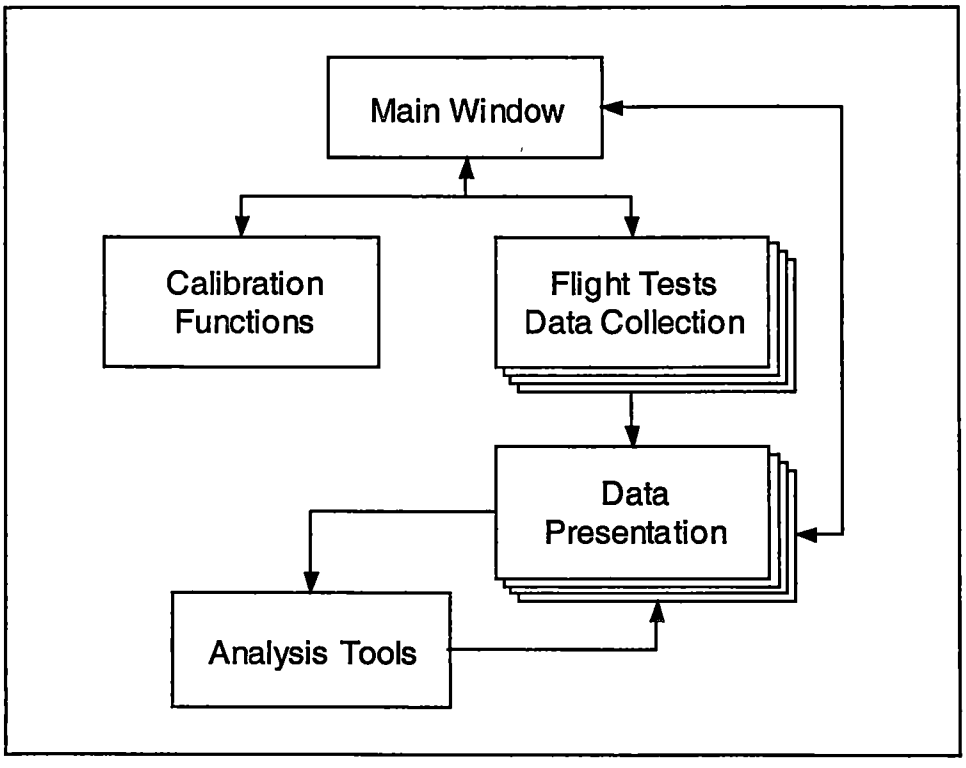


Figure 4-3 Final Program Block Diagram

analyzed on the ground. The saved data are opened in a data presentation window without the clutter of the data collection interface. The final advantage of dividing the data collection from the data presentation was that it allowed for greater used of code modularization. In several cases, major portions of the data presentation windows could be reused for multiple flight tests. Another example of modularization is in the analysis tools. There are two primary analysis tools that are called from any of the data presentation windows.

The Main Window

The main window for the Helicopter Stability and Control Flight Testing program is shown in figure 4-4. This window is presented when the program is initially started and whenever specific testing or analysis is not being conducted. The sample rate is adjustable with the knob at the top left. The indicators in this window provide a continuous, real-time, graphical representation of each of the data channels. The attitude, heading, and linear acceleration indicators at the top of the window provide only instantaneous values; whereas, the body rates and control position

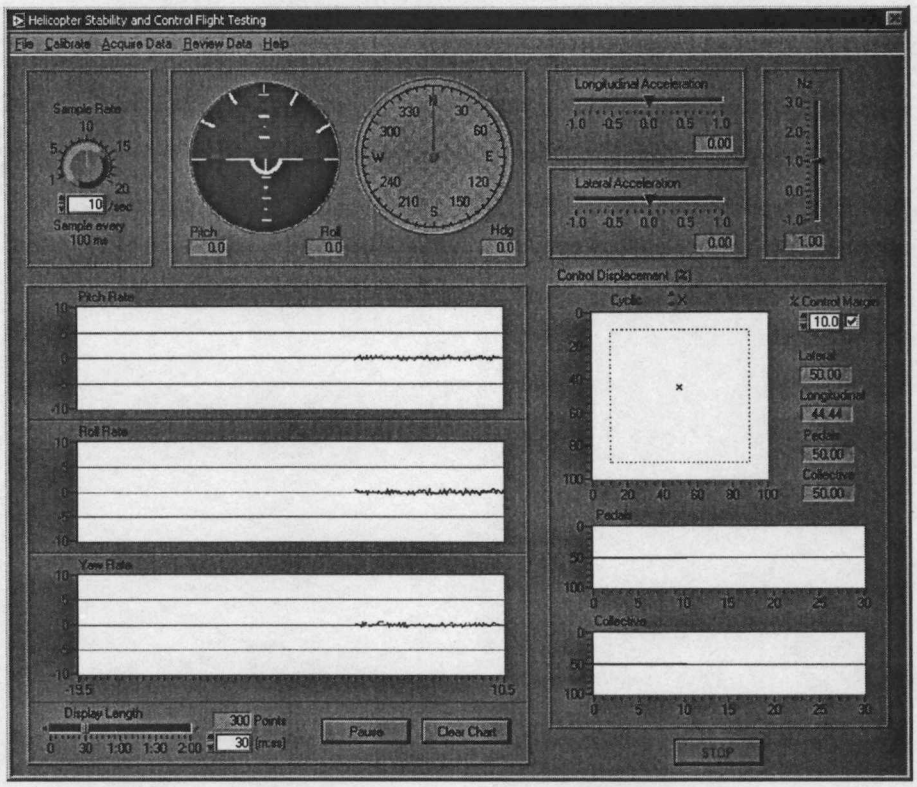


Figure 4-4 Main Window

graphs display short time history plots for these parameters. The length of the histories are adjustable with the 'Sample Length' control at the bottom left of the window.

The menu structure of the Main window provides access to the programs functionality. Navigation between the main window and the functional areas of the program is accomplished by this hierarchal menu system. The top level of this structure includes the File, Calibrate, Acquire Data, Review Data, and Help selections. The Calibrate, Acquire Data, and Review Data menu selections contain the primary functions of the software. System calibration functions are performed from the Calibrate menu, the Acquire Data menu allows access to each of the flight test data acquisition windows, and the Review Data menu accesses the data presentation and analysis windows. The complete menu structure is shown in figure 4-5.

The File menu allows the user to send an image of the Main window, with the current displayed data, to the printer. To configure the printer, the user can select 'Print Setup...'. Also, the application can be closed by selecting the 'Exit' command from the File menu.

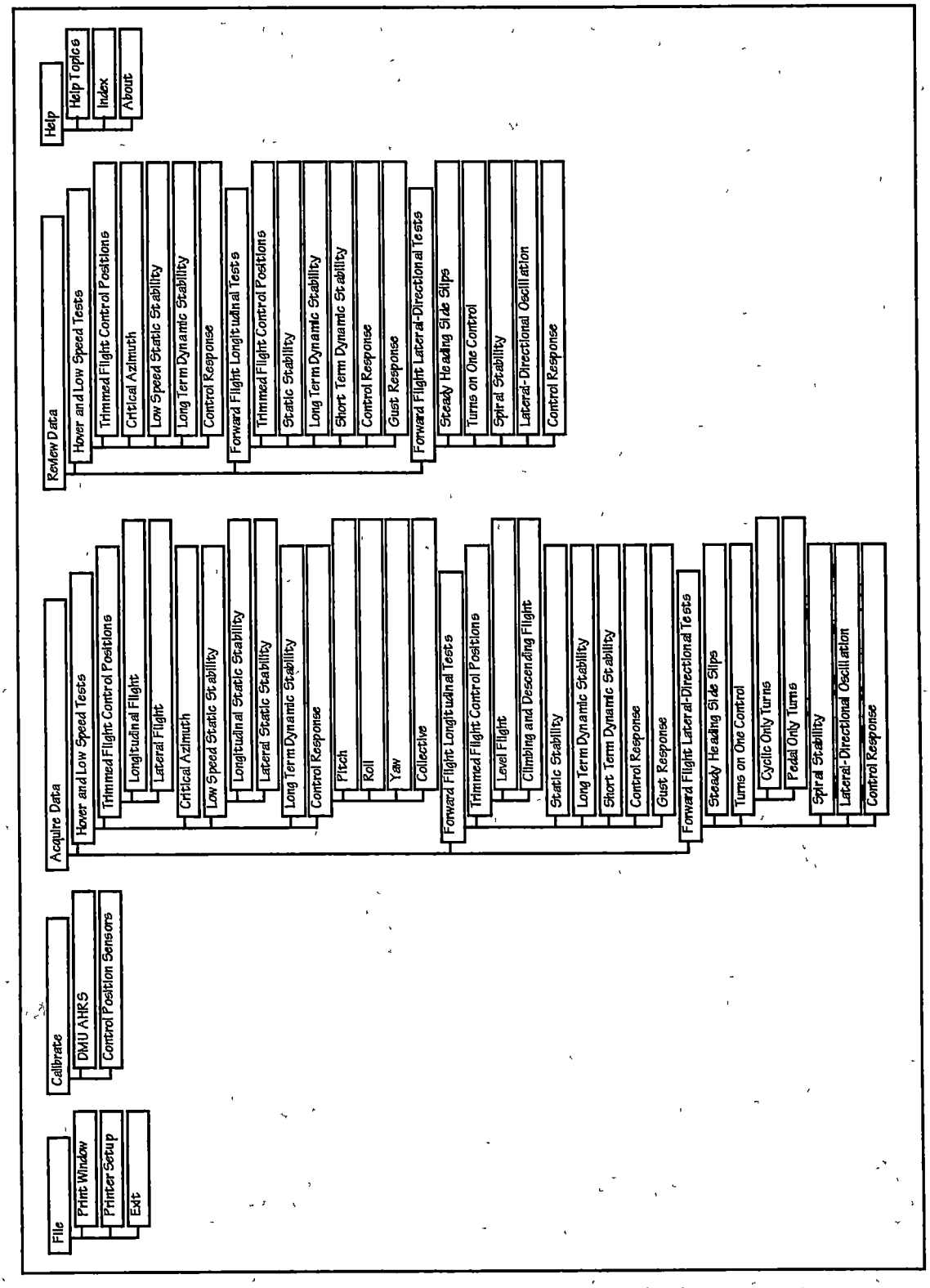


Figure 4-5 Hierarchical Menu Structure Block Diagram

The core functionality of the program is nested in the Acquire Data and the Review Data menus. The flight tests are organized by three categories within the Acquire Data menu selection: Hover and Low Speed Tests, Forward Flight Longitudinal Tests, and Forward Flight Lateral-Directional Tests.

The Help menu currently contains only one functional selection. The About menu selection opens a window that displays some descriptive information about the Helicopter Stability and Control Analysis Software. The Help Topics and Index selections are not functional in this version of the software.

System Calibration

The Helicopter Stability and Control Analysis Software and instrumentation sensor package hardware must be calibrated for each aircraft into which the hardware is installed. Once the calibration is completed the calibration data are automatically saved to the computer hard drive and automatically loaded into memory each time the program is run. If the hardware is moved back and forth between two or more test vehicles, calibration must be performed after each move, even if the hardware was previously installed in that particular vehicle. Calibration of the system is largely conducted through the software interface. The software controlled calibration functions are accessed through the 'Calibrate' menu selection from the main window. The components which require calibration are the DMU-AHRS and the control position wire pull displacement transducers.

DMU-AHRS

To access the control position sensor calibration dialog box (figure 4-6) select Calibrate | DMU-AHRS from the main window menu. There are three separate calibrations that may be conducted with this dialog box.

Calibration constants that are necessary to correctly scale the output data are stored internally in an EEPROM memory chip. These constants are the custom offset and sensitivity information determined for each AHRS unit. Each AHRS unit calibration constants are set at the factory and should not have to be changed. However, this information can be input into the configuration screen by selecting 'Calibration Constants'.

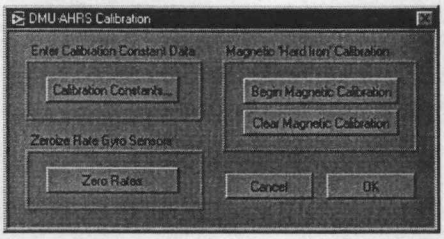


Figure 4-6 DMU-AHRS Calibration Dialog

The 'Zero Rates' button is used to zero the rate sensor outputs. This should be an essential part of your strategy in using the DMU effectively. The pitch, roll and yaw angles are calculated by integrating the output of the angular rate sensors. Rate sensors are subject to small offsets in the angular rate measurement. A constant offset error in angular rate will be integrated into an error in angle that increases with time -- angular drift. The DMU rate sensors should be zeroed to maintain the best accuracy.

The rate sensors will need to be zeroed more often when subject to large shocks or extremes of temperature. After pressing the 'Zero Rates' button the AHRS must be motionless for approximately 5 seconds. The DMU unit should be stationary during the zeroing process, but need not be level. Zeroing measures the bias in the output of the rate sensors when the DMU is in a condition of zero angular rate. [16] These values are used to minimize errors in the rate sensor measurements.

The DMU-AHRS uses magnetic sensors to compute heading. Ideally, the magnetic sensors would be measuring only earth's magnetic field to compute the heading angle. In actual applications, however, residual magnetism in the DMU-AHRS itself and in the aircraft will add to the magnetic field measured by the DMU. This unaccounted for magnetic field will create errors in the heading measurement if they are not determined and corrected. The DMU-AHRS can actually measure any extra constant magnetic field that is associated with the DMU installed in the aircraft and apply the appropriate correction. This process is called hard iron calibration. The *Hard Iron Calibration* will help correct for magnetic fields that are fixed with respect to the DMU. It cannot correct for time varying fields, or fields created by components that move with respect to the DMU. The DMU accounts for the extra magnetic field by making a series of measurements.

To start the hard iron calibration by clicking the 'Begin Magnetic Calibration' button. The DMU will use all subsequent measurements to model the magnetic environment. The pilot should make at least two complete pedal turns, with as much variation in pitch and roll as possible. After completing the pedal turns click the 'End Magnetic Calibration' button to end the hard iron calibration process. The DMU will calculate the hard iron magnetic fields and store these as calibration constants in the EEPROM. To clear the hard iron calibration constants, 'Clear Magnetic Calibration' button. The DMU will set the hard iron correction to zero. This is useful in determining the performance of the bare sensor. For best accuracy, conduct the calibration process with the DMU installed in the aircraft. Performing the calibration process with the DMU by itself, only corrects for the magnetism in the DMU itself. If the DMU is then installed in an aircraft, errors arising from the magnet fields associated with the aircraft will result.

Control Position Wire Pull Displacement Transducers

Calibration of the control position wire pull transducers is conducted with the rotors static. If the aircraft is equipped with hydraulically actuated flight controls it is highly recommended that the hydraulic system be energized to provide a more accurate system calibration.

To access the control position sensor calibration dialog box (figure 4-7) select Calibrate | Control Position Sensors from the main window menu. The use of this dialog box is straightforward. Simply place the control to be calibrated in the indicated position and click on the button corresponding to the position of the control. Calibration of the cyclic position wire pull transducers requires some further explanation. When the [Forward Ortho] button is clicked the actual position of the cyclic must be against the forward stop but it must also be aligned orthogonal to the longitudinal wire pull transducer. This method is particularly important if the longitudinal (or lateral) wire pull transducer is not centered within the lateral (or longitudinal) control travel as shown in figure 4-8. For the [Aft Ortho] position, the cyclic must be against the aft stop and again aligned orthogonal to the longitudinal wire pull transducer. The same basic method is used for the [Left Ortho] and [Right Ortho] positions, the control must be aligned orthogonal to the lateral wire pull transducer and against the left and right stops respectively. The

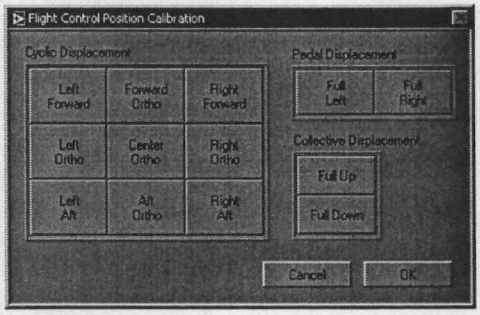


Figure 4-7 Flight Control Position Wire Pull Transducer Calibration Dialog

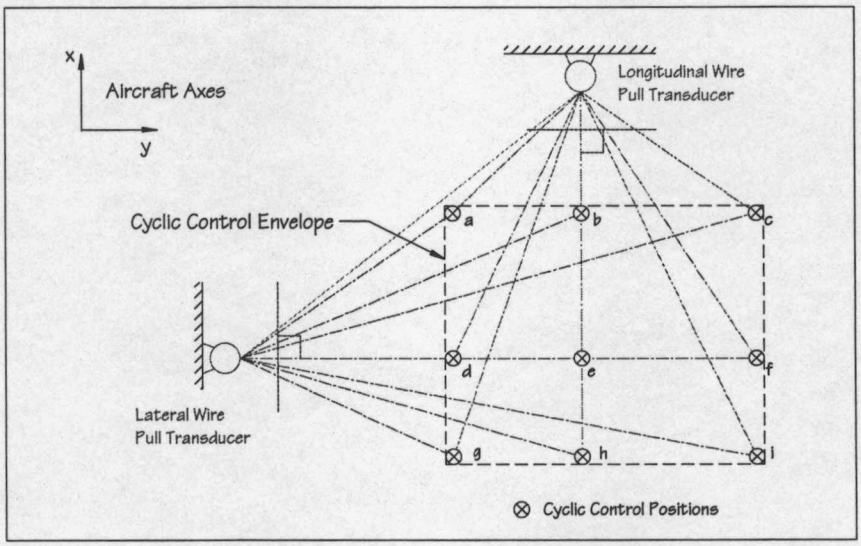


Figure 4-8 Proper Cyclic Control Positioning for Wire Pull Transducer Calibration

cyclic [Center] position is also marked with the cyclic positioned orthogonal to both the lateral and longitudinal wire pull transducers. The cyclic corner positions ([Left Forward], [Right Forward], [Left Aft], and [Right Aft]) are marked at the extents of the cyclic control envelope.

Procedures

1. Position the appropriate flight control in the correct position. The order in which the control positions are marked is not important; however, a systematic method should be used to avoid missing a calibration point. Calibration requires all indicated control positions be marked.
2. Click the button associated with the position of the control.
3. Repeat this procedure for each of the control positions indicated.
4. Click [OK] to save the calibration data to the hard disk.

Program Notes:

The positions of the other controls are ignored while one control is calibrated. For example collective and pedals positions are immaterial when calibrating the cyclic travel.

Clicking the [Cancel] button discards any collected control position calibration data and closes the Flight Control Position Calibration window.

Several important details should be mentioned about figure 4-8. First, notice that the wire extending from the lateral wire pull transducer is perpendicular to the aircraft's x-axis at calibration points d, e, and f. At points b, e, and h, the longitudinal wire pull transducer wire is perpendicular to the y-axis. This defines the meaning of the expression "orthogonal to the wire pull transducer". This positioning of the cyclic control does not place the control in the center of the control envelope but is important to the proper determination of the cyclic position.

Next, figure 4-8 indicates that the lateral wire pull transducer is positioned to the left of the cyclic envelope and the longitudinal wire pull transducer is positioned forward of the cyclic. This configuration is not necessary for this program to properly function, these transducers can be positioned on either side of the control envelope. The important constraints in mounting the cyclic wire pull transducers is that they are: 1) at very nearly the same height as the point where the wires attach to the cyclic control, and 2) within the bounds of the opposite axis control envelope. The second requirement means that the lateral wire pull transducer should not be mounted any further forward or aft than the longitudinal cyclic control limits, and the longitudinal transducer must not be mounted left or right beyond the lateral control bounds.

Finally, an important detail shown in figure 4-8 is that at the corner of the control envelope closest to both transducers (point a), the two wires do not reach a point where they progress *over-center*. An over-center condition would be where the cyclic control could be positioned to a point where the wires reach a maximum amount or retraction and then begin to extend again as the control is moved toward the corner of the control envelope. The cyclic control should never cross a line that connects the two points of origin on the wire pull transducers to arrive at any extent of the control envelope.

Mathematics of Control Position Determination

The determination of the collective and pedal position is based on a ratio of the current control position to the total control travel. This control position percent travel value is computed as follows:

$$Pos_{Coll} = \frac{Pot_{Coll_i} - Pot_{Coll_{Min}}}{Pot_{Coll_{Max}} - Pot_{Coll_{Min}}} \qquad \textit{eq. 4-1}$$

$$Pos_{Ped} = \frac{Pot_{Ped_i} - Pot_{Ped_{Min}}}{Pot_{Ped_{Max}} - Pot_{Ped_{Min}}} \qquad \textit{eq. 4-2}$$

Where: Pos_{Coll} = *Collective position as a percent of total travel.*

Pot_{Coll_i} = *Instantaneous collective wire pull transducer (WPT) value.*

$Pot_{Coll_{Min}}$ = *Minimum collective WPT value.*

$Pot_{Coll_{Min}}$ = *Maximum collective WPT value.*

Pos_{Ped} = *Pedal position as a percent of total travel.*

Pot_{Ped_i} = *Instantaneous pedal WPT value.*

$Pot_{Ped_{Min}}$ = *Minimum pedal WPT value.*

$Pot_{Ped_{Min}}$ = *Maximum pedal WPT value.*

The determination of the cyclic position is considerably more complicated because the two-dimensional movement of the cyclic control. Although the cyclic position is measured by two wire pull transducers, the readings from these transducers does not directly correspond to lateral or longitudinal cyclic control position. Each transducer actually measures the radius of a circle which is centered on the position of the respective wire pull transducer. If arcs are circumscribed at the distances of the two radii the intersection of the two arcs locates the position of the cyclic control as shown in figure 4-9.

The calculations used to determine this position are based on the equation for a circle (equation 4-3).

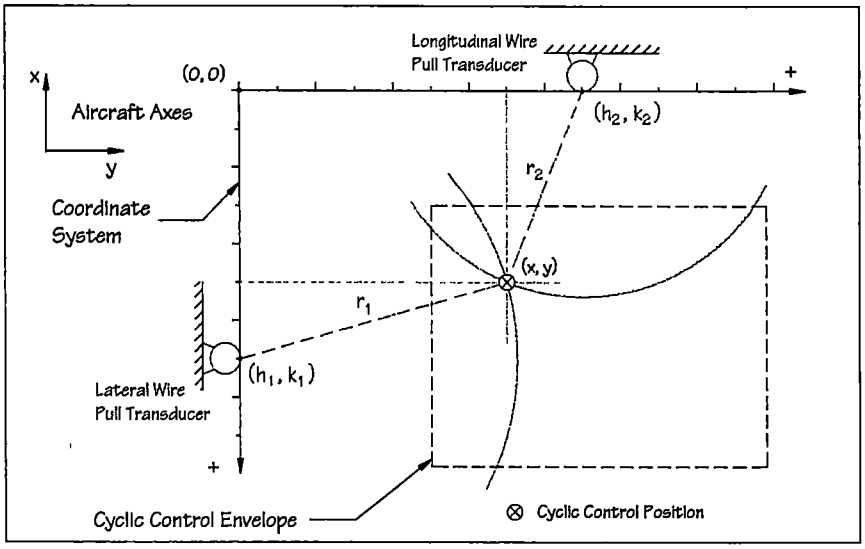


Figure 4-9 Determination of Cyclic Control Position

The general form for the equation of a circle is:

$$r = \sqrt{(x-h)^2 + (y-k)^2} \qquad \textit{eq 4-3}$$

Where: r = *Radius of the circle*

x = *Abscissa or horizontal coordinate*

y = *Ordinate or vertical coordinate.*

h = *Horizontal displacement of the center of the circle from the origin*

k = *Vertical displacement of the center of the circle from the origin.*

The two circles that are circumscribed around the two wire pull transducers are:

$$r_1 = \sqrt{(x-h_1)^2 + (y-k_1)^2} \qquad \textit{eq. 4-4}$$

$$r_2 = \sqrt{(x-h_2)^2 + (y-k_2)^2} \qquad \textit{eq. 4-5}$$

The x coordinates for the given radii are expressed by solving for x as follows:

$$x_1 = h_1 + \sqrt{r_1^2 + 2k_1y_1 - k_1^2 - y_1^2} \qquad \textit{eq 4-6}$$

$$x_2 = h_2 + \sqrt{r_2^2 + 2k_2y_2 - k_2^2 - y_2^2} \qquad \textit{eq. 4-7}$$

The y coordinates for the given radii are expressed by solving for y as follows:

$$y_1 = k_1 + \sqrt{r_1^{\,2} + 2h_1 y_1 - h_1^{\,2} - x_1^{\,2}} \qquad \textit{eq. 4-8}$$

$$y_2 = k_2 + \sqrt{r_2^{\,2} + 2h_2 y_2 - h_2^{\,2} - x_2^{\,2}} \qquad \textit{eq. 4-9}$$

Since the intersection of the two arcs is the point of interest, $x_1 = x_2$ and $y_1 = y_2$. Then a Cartesian coordinate system is defined where the origin is at a point where $h_1 = 0$, and $k_2 = 0$ as shown in figure 4-9. Substituting and simplifying:

$$x = \frac{-4h_2 r_2^{\,2} + 4h_2 k_1^{\,2} + 4h_2 r_1^{\,2} + 4h_2^{\,3} + \ldots}{2\left(4h_2^{\,2} + 4k_1^{\,2}\right)} \qquad \textit{eq. 4-10}$$

$$4 \times \sqrt{\begin{array}{l} -\left(k_1^{\,6}\right) + \left(-h_2^{\,4} k_1^{\,2}\right) + \left(2k_1^{\,4} r_1^{\,2}\right) + \left(2k_1^{\,4} r_2^{\,2}\right) + \ldots \\ \left(2h_2^{\,2} k_1^{\,2} r_1^{\,2}\right) + \left(2h_2^{\,2} k_1^{\,2} r_2^{\,2}\right) + \left(2k_1^{\,2} r_1^{\,2} r_2^{\,2}\right) - \ldots \\ \left(k_1^{\,2} r_1^{\,4}\right) - \left(k_1^{\,2} r_2^{\,4}\right) - \left(2h_2^{\,2} k_1^{\,4}\right) \end{array}}$$

and:

$$y = \sqrt{r_2^{\,2} - (x - h_2)^2} \qquad \textit{eq. 4-11}$$

Where:
- x = *Lateral Cartesian coordinate cyclic position.*
- y = *Longitudinal Cartesian coordinate cyclic position.*
- h_1 = *Lateral position of the lateral wire pull transducer (WPT).*
- h_2 = *Lateral position of the longitudinal WPT.*
- k_1 = *Longitudinal position of the lateral WPT.*
- k_2 = *Longitudinal position of the longitudinal WPT.*
- r_1 = *Instantaneous lateral WPT value.*
- r_2 = *Instantaneous longitudinal WPT value.*

Then, the control position is expressed as a percentage of the total axis travel by:

$$Pos_{Lat} = \frac{x - Pot_{Lat_{Min}}}{Pot_{Lat_{Max}} - Pot_{Lat_{Min}}} \qquad \textit{eq. 4-12}$$

$$Pos_{Long} = \frac{y - Pot_{Long_{Min}}}{Pot_{Long_{Max}} - Pot_{Long_{Min}}} \qquad \textit{eq. 4-13}$$

Where: Pos_{Lat} = *Lateral cyclic control position.*

x = *Lateral Cartesian coordinate cyclic position.*

$Pot_{Lat_{Min}}$ = *Minimum lateral cyclic control WPT value.*

$Pot_{Lat_{Min}}$ = *Maximum lateral cyclic control WPT value.*

Pos_{Long} = *Longitudinal cyclic control position.*

y = *Longitudinal Cartesian coordinate cyclic position.*

$Pot_{Long_{Min}}$ = *Minimum longitudinal cyclic control WPT value.*

$Pot_{Long_{Min}}$ = *Maximum longitudinal cyclic control WPT value.*

Data Acquisition

The data acquisition device for this project is the IOtech Daq/112B data acquisition (DAQ) PCMCIA notebook computer expansion card. This card is an analog to digital converter with built in data buffering to prevent data loss. The maximum sampling rate is 100 kHz, with 12 bit resolution, on up to 16 channels. For each flight test the software automatically selects the appropriate data channels to properly document the test being performed. The setup of the DAQ device for each test is simplified to the extent that the only user configuration necessary is the selection of the sample rate. The sample rate parameter for each test defaults to a value that is typically adequate to achieve acceptable test results.

The samples of a signal obtained from the data acquisition card constitute the time domain representation of the signal. This representation gives the amplitudes of the signal at the instant of time during which it had been sampled. When the signal that is being sampled represents a non-oscillatory response, the sample rate needs only to be rapid enough to adequately define the shape of the signal. If, however the signal represents an oscillatory function, the sampling rate must be more carefully selected to avoid misrepresenting the signal as discussed below. In any case the data collected from the DAQ system is always in the form of time domain information; that is, the amplitude of the signal at one instant or a series of equally spaced time intervals. However, in many cases it is desirable to know the frequency content of a signal rather than the amplitudes of the individual samples. The representation of a signal in terms of its individual frequency components is the frequency domain representation of the signal.

The algorithm used to transform samples of the data from the time domain into the frequency domain is the discrete Fourier transform or DFT. The DFT establishes the relationship between the samples of a signal in the time domain and their representation in the frequency domain.

When the DFT is applied to N samples of a time domain representation of the signal, the result is also a sequence of length N samples, but the information it contains is of the frequency domain representation. The relationship between the N samples in the time domain and the N samples in the frequency domain is explained below.

If the signal is sampled at a sampling rate of f_s Hz, then the time interval between the samples (that is, the sampling interval) is Δt, where $\Delta t = \frac{1}{f_s}$.

When the sample signals are denoted by x[i], $0 < i < N-1$ (for N samples). The discrete Fourier transform, given by.

$$X_k = \sum_{i=0}^{N-1} x_i e^{-\frac{j2\pi ik}{N}} \qquad \textit{eq. 4-14}$$

The resulting output is the frequency domain representation of x[i] Both the time domain x and the frequency domain X have a total of N samples. Analogous to the time spacing of Δt between the samples of x in the time domain, the frequency spacing is:

$$\Delta f = \frac{f_s}{N} = \frac{1}{N\Delta t} \qquad \textit{eq. 4-15}$$

Δf is also known as the frequency resolution. To increase the frequency resolution (smaller Δf) the number of samples (N) must be increased, (with f_s constant) or the sampling frequency, f_s must be decreased (with N constant).

A/D converters (ADCs) are an integral part of DAQ process. One of the most important parameters of an analog input system is the rate at which the DAQ board samples an incoming signal. The sampling rate determines how often an analog-to-digital (A/D) conversion takes place. A fast sampling rate acquires more points in a given time and can therefore often form a better representation of the original signal than a slow sampling rate Sampling too slowly may result in a poor representation of the analog signal. Figure 4-10 shows an adequately sampled signal, as well as the effects of under sampling. The effect of under sampling is that the signal

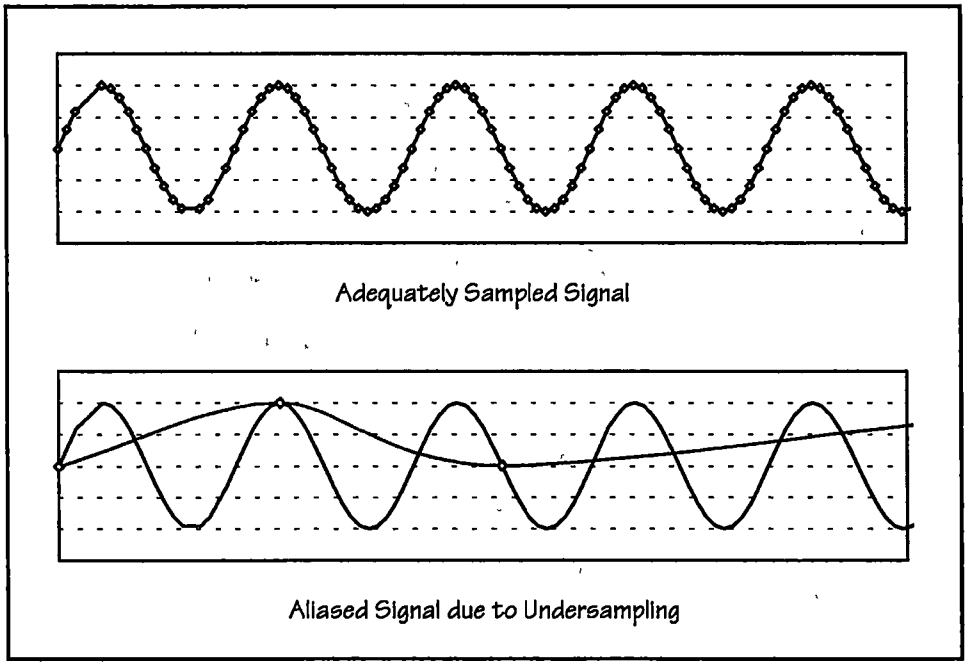


Figure 4-10 Effect of Sampling Rate on Digital Sampling of an Analog Signal

appears as if it has a different frequency than it truly does. This misrepresentation of a signal is called an alias.

According to the Nyquist theorem, to avoid aliasing sampling must occur at a rate greater than twice the maximum frequency component in the signal being acquired. For a given sampling rate, the maximum frequency that can be represented accurately, without aliasing, is known as the Nyquist frequency. The Nyquist frequency is one-half the sampling frequency. Signals with frequency components above the Nyquist frequency will appear aliased between DC and the Nyquist frequency. The alias frequency is the absolute value of the difference between the frequency of the input signal and the closest integer multiple of the sampling rate. The next two figures illustrate this phenomenon. For example, assume f_s, the sampling frequency, is 100 Hz. Also, assume the input signal contains the following frequencies —25 Hz, 70 Hz, 160 Hz, and 510 Hz. These frequencies are shown in figure 4-11.

In figure 4-11, the frequencies below the Nyquist frequency ($f_s/2 = 50$ Hz) are sampled correctly. Frequencies above the Nyquist frequency appear as aliases. For example, F1 (25 Hz) appears at the correct frequency, but F2 (70 Hz), F3 (160 Hz), and F4 (510 Hz) have aliases at

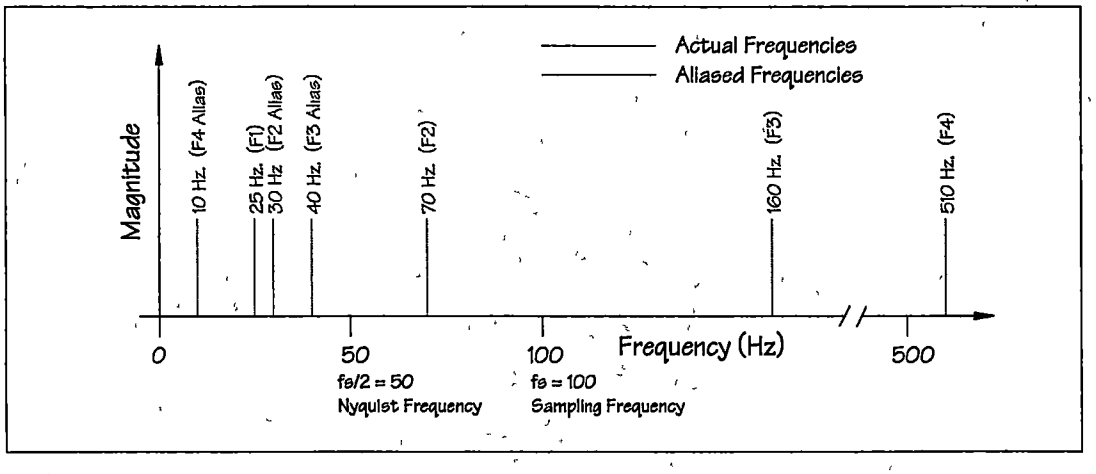


Figure 4-11 Frequency Aliasing

30 Hz, 40 Hz, and 10 Hz, respectively. To calculate the alias frequency, use the following equation:

$$f_A = \left|(I \cdot f_s) - f_{input}\right| \qquad \textit{eq. 4-16}$$

Where: f_A = *Aliased Frequency.*

$I \times f_s$ = *Closest integer multiple of the sampling frequency, f_s.*

f_{input} = *Actual frequency.*

For example,

$$\textit{Alias F2} = |(1 \times 100) - 70| = 30\ Hz$$

$$\textit{Alias F3} = |(2 \times 100) - 160| = 40\ Hz$$

$$\textit{Alias F4} = |(5 \times 100) - 510| = 10\ Hz$$

The obvious question is, "How fast should I sample?" The first thought may be to sample at the maximum rate available on the DAQ card. However, if you sample very fast over long periods of time, you may not have enough memory or hard disk space to hold the data. Figure 4-12 shows the effects of various sampling rates. In case a, the sine wave of frequency f is sampled at the same frequency, f_s (samples/sec) = f (cycles/sec), or at 1 sample per cycle. The reconstructed waveform appears as an alias at DC. As you increase the sampling to 7 samples/4 cycles, as in case b, the waveform increases in frequency, but aliases to a frequency less than the original signal (3 cycles instead of 4). The sampling rate in case b is $f_s = 7/4\,f$. If you increase the sampling rate to $f_s = 2f$, the digitized waveform has the correct frequency (same number of

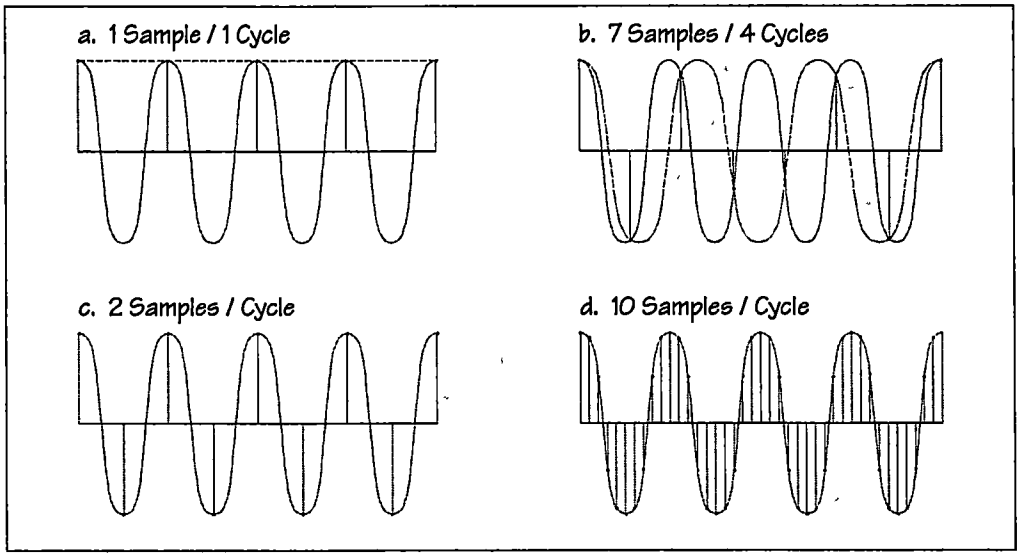


Figure 4-12 Example Sampling Rates

cycles), and can be reconstructed as the original sinusoidal wave, as shown in case c. For time-domain processing, it may be important to increase your sampling rate so that the samples more closely represent the original signal. By increasing the sampling rate to well above f, say to $f_s = 10f$, or 10 samples/cycle, you can accurately reproduce the waveform, as shown in case d.

Data Analysis

The data collected during the flight testing falls into two major categories. These are static data and dynamic data. Static data are relatively constant for the sampling period, whereas dynamic data are continuously changing during the sampling period. Typically static data are plotted and stand alone representing an aircraft characteristic. Some interpretation may be necessary, for example determining the sign of the slope of a line, but these data generally do not require complicated mathematical analysis. The specific methods for analyzing static data are discussed with the data reduction techniques for each test. The analysis of dynamic data usually require significantly more complex mathematical methods to determine the engineering results.

First Order Dynamic Response Data Analysis

Program Operation

The computer method for analyzing a dynamic response is a two step process. First, the acquired data are trimmed to remove extraneous data that was recorded before and after the

desired response sequence. Then the relevant data are passed to a first order response data analysis window.

The flight tests which acquire dynamic data have an [Analyze Data] button. Clicking this button will pass the accumulated data to the appropriate analysis routine. This process begins by sending the data to the Trim Data Plot window (figure 4-13). The Trim Data Plot window is opened with the time history data for the chosen motion displayed.

The following procedures are used to operate this window:

Procedures

1. Drag the green triangle at the top of the graph to the beginning and the red triangle to the end of the relevant data. Alternatively, enter time values corresponding to the beginning and end of the relevant data, in the 'Begin Selection' and the 'End Selection' controls at the bottom of the graph. The 'Begin Selection' control should be set to a point where the rate first begins to increase. The 'End Selection' control should be set to the highest rate point. If there is any data where the rate begins to decrease, these data must be trimmed with the 'End Selection' control, or some of the analysis tools will fail.
2. Click [Trim Selection] to trim the unwanted data from before the beginning and after the end of the selected data.
3. Click [Continue] to close the Trim Data Plot window and pass the selected data to the Analyze First Order Response window or the Analyze Second Order Response window as appropriate. Clicking the [Continue] button also trims the data from before the 'Begin Selection' and after the 'End Selection' settings.

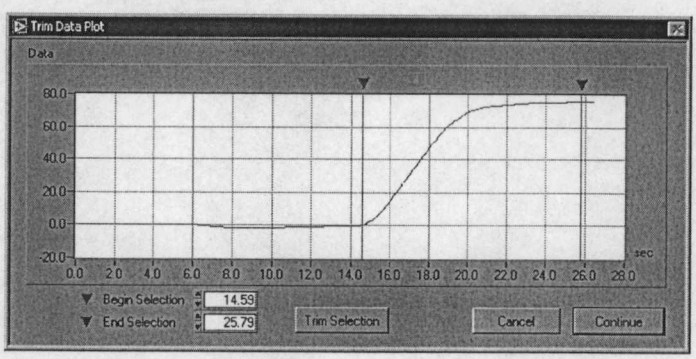


Figure 4-13 Trim Data Plot Window

The Analyze First Order Response window (figure 4-14) is opened with the data that was passed to it displayed. The top left area of this window shows information about the sample data that was passed to the window in the 'Sample Information' section. This information includes the rate at which the sample was taken and the length of the trimmed data in seconds. The calculated engineering data results are also presented at the top of this window in the 'Test Results' section. The Test Results are constantly calculated as the analysis parameters ('Steady State', or 'Start Time', and 'End Time') are adjusted.

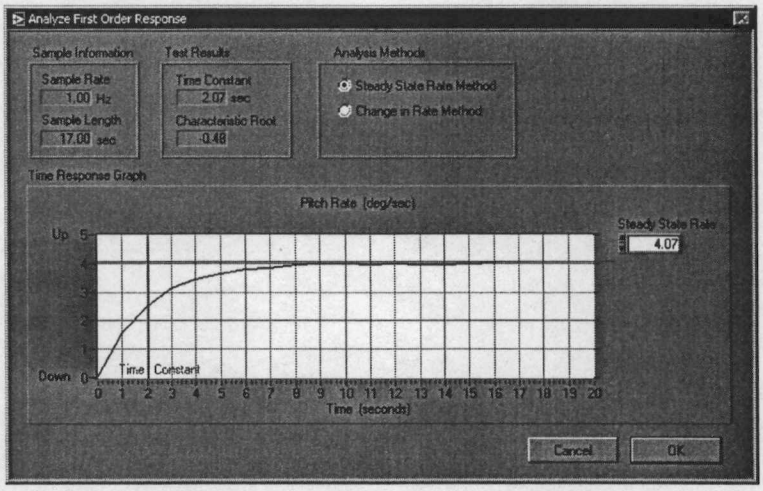


Figure 4-14 Analyze First Order Response Window

The last control at the top of this window is the 'Analysis Method' selection control. This control allows the user to choose one of two mathematical analysis methods. The two methods are the 'Steady State Rate Method' and the 'Change in Rate Method'. The 'Steady State Method' is the simpler of the two methods; however, this method can only be utilized if a steady state rate is achieved in the flight test. If a steady state rate is not achieved during flight testing the 'Change in Rate Method' must be utilized.

Procedures

1. Select the desired 'Analysis Method' by clicking on the appropriate option in the 'Analysis Methods' section. It is normally easier to use the 'Steady State Rate Method' if the data clearly illustrates that the steady state was achieved. The 'Change in Rate Method' can be utilized to analyze nearly any data but is more difficult to properly operate.

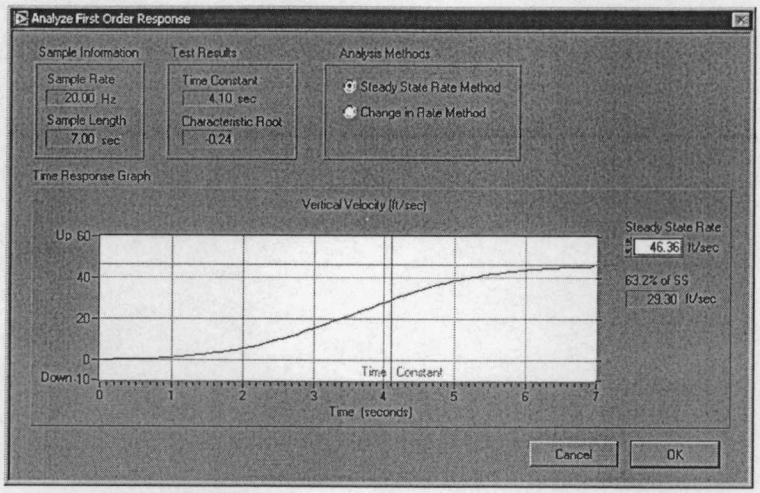


Figure 4-15 Determination of the Time Constant from the Steady State Rate

Steady State Rate Method (Figure 4-15)

2. Set the 'Steady State' by setting the numeric value or by dragging the red line at the right edge of the graph. The steady state rate is the value at which the rate stabilizes. This is the rate where the forces or moments are balanced with the damping.

Change in Rate Method (Figure 4-16)

3. Set the 'Start Time' and the 'End Time' values by setting the numeric values or by dragging the red lines at the top edge of the graph. As the 'Start Time' and the 'End Time' settings are adjusted the program will continuously display the midpoint (Mid) between these two points. Also, while adjusting these parameters the Time

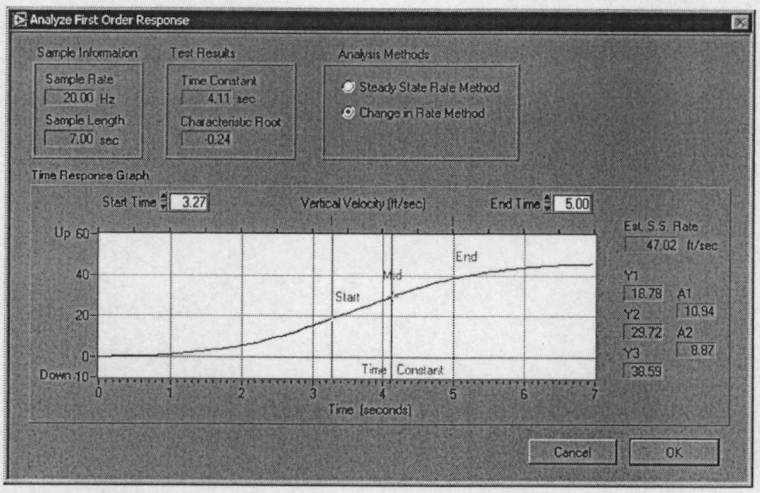


Figure 4-16 Determination of the Time Constant by the Change in Rate

Constant will be calculated and displayed on the graph. To properly adjust these analysis parameters requires some experimentation.

a. If the plot consists of a concave up portion and a concave down portion the 'Start Time' should be set to a point after the end of the concave up segment.
b. The 'End Time' setting should be initially set to encompass the majority of the concave down portion of the plot.
c. Make small adjustments to either or both the 'Start Time' and the 'End Time' values until the midpoint and the time constant indicators are as close to each other as possible.

Both Methods

4. When the analysis parameters are satisfactorily adjusted to adequately define the time constant, click the [OK] button to pass the engineering data results back to the originating flight test data presentation window.

The Analyze First Order Response window utilizes a body axis rate to determine the time constant of the mode of motion. This method is not practical for the analysis of the spiral response because the roll rates for this response are generally very low. Therefore, for this test, the program uses a variation of the first order analysis methods shown above. This technique uses the aircraft's roll attitude time history trace to determine the time constant. The window used for analysis is shown in figure 4-17.

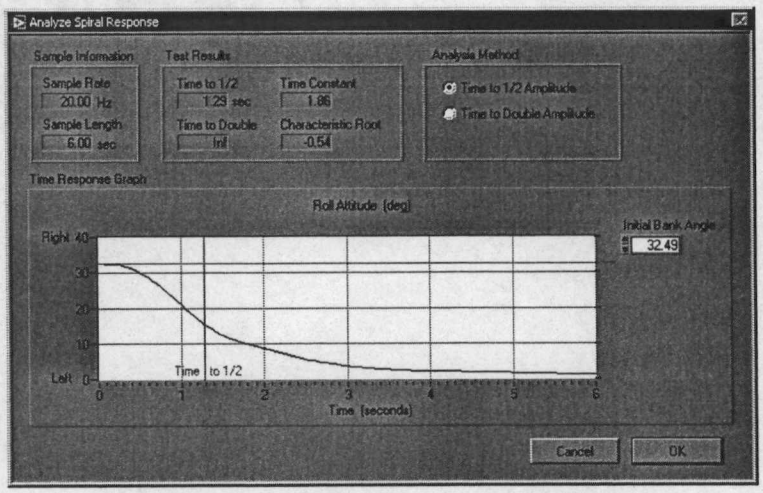


Figure 4-17 Analyze Spiral Response Window

This window is very similar to the Analyze First Order Response window above. This method determines the time required for the bank angle to either half or double from the initial trim value. The calculated engineering data results are presented at the top of this window in the 'Test Results' section. The Test Results are constantly calculated as the analysis parameter ('Initial Bank Angle') is adjusted. The 'Analysis Method' selection control at the top of this window is used to select either the time to one-half or the time to double.

Procedures

1. Select the desired 'Analysis Method' by clicking on either 'Time to 1/2 Amplitude' or 'Time to Double Amplitude'.
2. Set the 'Initial Bank Angle' setting by setting the numeric value or by dragging the red line at the right edge of the graph. This is the equilibrium flight condition bank angle from which the test began. As this parameter is adjusted the time to half or double amplitude is continually computed and displayed.
3. When the 'Initial Bank Angle' is satisfactorily adjusted to adequately define the time constant, click the [OK] button to pass the engineering data results back to the originating flight test data presentation window.

Analysis Mathematics

The mathematics required to calculate the desired engineering data are derived from the equations of a first order response presented in chapter 3. The procedures used are entirely different for the three analysis methods.

Steady State Rate Method

First, 63.2% of the user set steady state rate is determined. This value is the ordinate value of a point along the plot whose abscissa corresponds to the time constant value. The time constant is determined by performing a cubic spline interpolation along the plot at the calculated ordinate. The interpolant used for this function is determined by finding the second derivative of the curve at each point along the plot. Once the time constant is determined the characteristic root is calculated by finding the negative inverse of the time constant as follows:

$$\lambda = -\frac{1}{\tau} \qquad \textit{eq. 4-17}$$

Where: λ = *Characteristic root.*

τ = *Time constant.*

Change in Rate Method

The values for the parameters indicated in figure 4-18 are determined as follows. First, the value for the midpoint (t2) between the 'Start Time' and the 'End Time' values is calculated. Then the values Y1, Y2, and Y3 are determined by performing a cubic spline interpolation along the curve at t1, t2, and t3. Finally, A1 and A2 are calculated as the difference between Y1 & Y2, and Y2 & Y3 respectively.

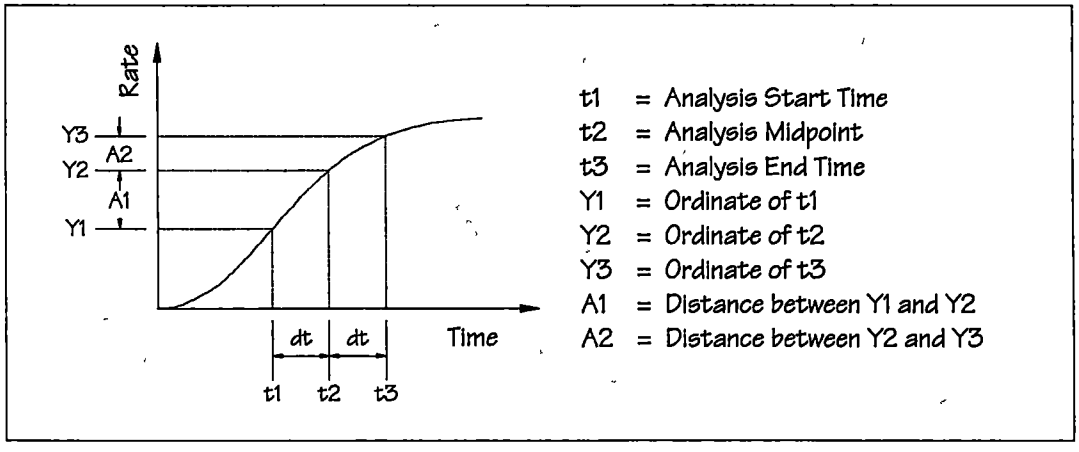


Figure 4-18 First Order Response Analysis Parameters used in the Change in Rate Method

Once all the indicated values have been measured or calculated the time constant is calculated as follows:

$$dt = t2 - t1 = t3 - t2 \qquad \textit{eq. 4-18}$$

$$A1 = Y2 - Y1 \qquad \textit{eq. 4-19}$$

$$A2 = Y3 - Y2 \qquad \textit{eq. 4-20}$$

$$\tau = \frac{dt}{\ln_e\left(\frac{A1}{A2}\right)} \qquad \textit{eq. 4-21}$$

Where: dt = *Change in time.*

$t1$ = *Analysis start time.*

$t2$ = *Analysis midpoint time.*

$t3$ = *Analysis end time.*

$Y1$ = *Ordinate value corresponding to the analysis start time.*

$Y2$ = *Ordinate value corresponding to the analysis midpoint time.*

$Y3$ = *Ordinate value corresponding to the analysis end time.*

$A1$ = *Change in rate over first period of time.*

$A2$ = *Change in rate over second period of time.*

τ = *Time constant.*

Once the time constant is determined the characteristic root is calculated by finding the negative inverse of the time constant as follows:

$$\lambda = -\frac{1}{\tau} \qquad \textit{eq. 4-22}$$

Where: λ = *Characteristic root.*

τ = *Time constant.*

Spiral Mode Method

First, 50% or 200% of the user set initial bank angle is determined. This value is the ordinate value of a point along the plot whose abscissa corresponds to the time to half or double amplitude The time to half or double amplitude is determined by performing a cubic spline interpolation along the plot at the calculated ordinate. The interpolant used for this function is determined by finding the second derivative of the curve at each point along the plot. Then the time constant is calculated by equation 4-23 or 4-24. Once the time constant is determined the characteristic root is calculated by finding the negative inverse of the time constant as shown in equation 4-25.

$$\tau = \frac{T_{1/2}}{\ln(2)} \qquad \textit{eq. 4-23}$$

$$\tau = \frac{T_{2x}}{\ln(0.5)} \qquad \textit{eq. 4-24}$$

$$\lambda = -\frac{1}{\tau} \qquad \textit{eq. 4-25}$$

Where: τ = *Time constant.*

$T_{½}$ = *Time for the bank angle to subside to one-half the initial value.*

T_{2x} = *Time for the bank angle to increase to double the initial value.*

λ = *Characteristic root.*

Second Order Dynamic Response Data Analysis

The analysis techniques used to determine the desired engineering data results are divided into two sub-categories. The difference in the techniques is based on the amount of damping present in the system being analyzed. In terms of the damping ratio, systems with a damping ratio of 0.5 or less are referred to as *lightly damped* in this manuscript. Damping ratios of 0.5 or greater are considered *heavily damped.* The computer data analysis routines that are coded into the software will adequately quantify second order responses with damping ratio values between approximately – 0.3 and +1.3. The lightly damped oscillation method works well for systems that have a damping ratio between – 0.3 and + 0.5. Correspondingly, the heavily damped oscillation method works well for systems that have a damping ratio between + 0.5 and + 1.3. Realistically, systems that have a damping ratio that is less than – 0.3 are relatively unlikely. However, if this type of response is encountered in flight testing, precisely quantifying the response is not as important as determining how quickly the response diverges. Also, systems that have a damping ratio of greater than 1.3 are so over-damped that quantifying this is also unnecessary. During flight testing, systems that have approximately 0.8 or greater damping ratios will appear deadbeat to the pilot.

Program Operation

The same basic two step process used to analyze a first order response is used to analyze a second order response. First, the acquired data is trimmed to remove unneeded data that was

recorded before and after the desired response sequence. Then the relevant data is passed to the second order response analysis window.

The flight tests which acquire dynamic data have an [Analyze Data] button. Clicking this button will pass the accumulated data to the appropriate analysis routine. This process begins by sending the data to the Trim Data Plot window (figure 4-19). The Trim Data Plot window is opened with the time history data for the chosen motion displayed.

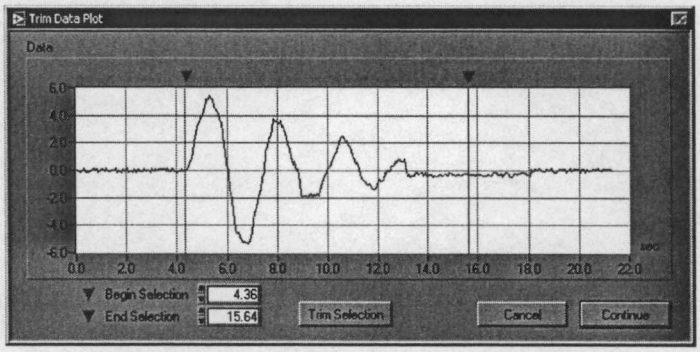


Figure 4-19 Trim Data Plot Window

The following procedures are used to operate this window:

Procedures

1. Drag the green triangle at the top of the graph to the beginning and the red triangle to the end of the relevant data. Alternatively, enter time values corresponding to the beginning and end of the relevant data, in the 'Begin Selection' and the 'End Selection' controls at the bottom of the graph.
2. Click [Trim Selection] to trim the unwanted data from before the beginning and after the end of the selected data.
3. Click [Continue] to close the Trim Data Plot window and pass the selected data to the Analyze First Order Response window or the Analyze Second Order Response window as appropriate. Clicking the [Continue] button also trims the data from before the 'Begin Selection' and after the 'End Selection' settings.

The Analyze Second Order Response window is opened with the data that was passed displayed. The top left area of this window shows information about the sample data that was passed to the window in the 'Sample Information' section. This information includes the rate at

which the sample was taken and the length, in seconds, of the trimmed data. The calculated engineering data results are also presented at the top of this window in the 'Test Results' section. The Test Results are constantly calculated as the analysis parameters ('Trim Point', 'Threshold %', and 'Peak Width') are adjusted. At the top right is the 'Analysis Method' selection with options for 'Lightly Damped Oscillation', and 'Heavily Damped Oscillation'. The analysis method defaults to the Lightly Damped Oscillation setting for all tests except the Longitudinal Short Term Dynamic Stability which defaults to Heavily Damped Oscillation since this is typically the character of this mode. The user can change this setting at any time during the analysis process to better assess the mode of motion. As a rule-of-thumb, if the number of clearly identifiable overshoots is two or more, the lightly damped method should be used to quantify the response. Accordingly, if there is only one or no identifiable overshoots, the heavily damped oscillation method will work better.

The 'Time Response Graph' shows the time domain information of the recorded motion data on the dark blue data plot. The analysis tools presented on this graph depend on the analysis method selected. With the Lightly Damped Oscillation method selected, the red curves on this graph represent the computed exponential decay (or growth) envelope of the motion and are always symmetrical about the trim line. These curves are recomputed continuously as the analysis parameters are adjusted and should roughly enclose the recorded motion data plot. When the Heavily Damped Oscillation method is selected four orange lines are shown that indicate the time of maximum amplitude and selected percentages of the maximum amplitude.

The 'Frequency Response Graph' shows the Fourier transform calculated frequency domain information of the recorded motion data. The red line on this graph indicates the most significant frequency that is present in the recorded data.

Procedures for the Lightly Damped Oscillation Method (Figure 4-20)

1. Adjust the 'Trim Point' setting by either editing the numeric value or by dragging the small red indicator on the right side of the 'Time Response Graph'. The trim point should be near the mean value of the oscillatory motion and should normally correspond to the initial trimmed state.

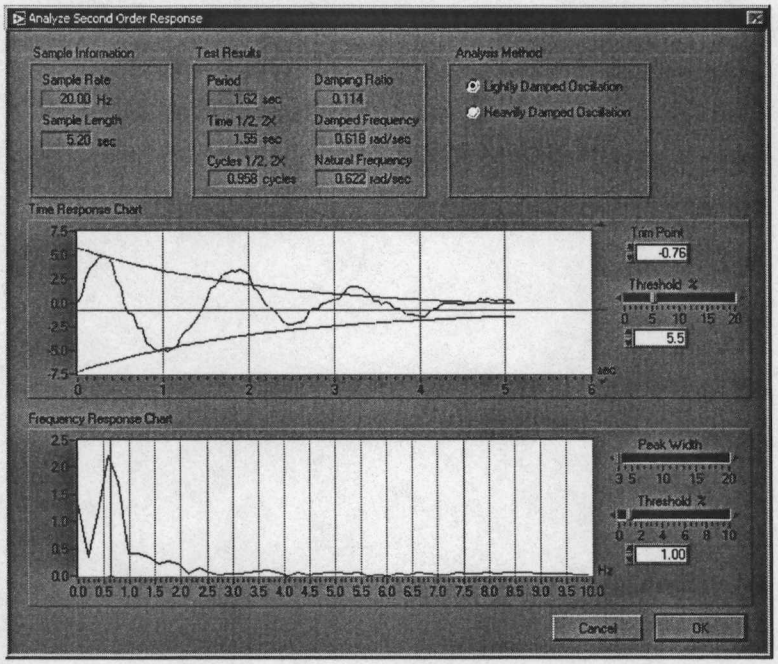


Figure 4-20 Lightly Damped Second Order Response Analysis Window

2. Adjust the Time Response Graph's 'Threshold %' setting by setting the numeric value or by dragging the slider control as necessary. The threshold setting adjusts the minimum detectable displacement from the trim value that will be recognized as a lobe of the oscillation. The higher the 'Threshold %' setting the larger the magnitude of the displacement must be to be detected. This value is a percentage of the range of the Y-scale.
3. Set the Frequency Response Graph's 'Peak Width' setting by dragging the slider control. The function that finds the frequency peaks depends on an algorithm that fits a quadratic polynomial to sequential groups of data points. The number of data points used in the curve fit is specified by this setting. The default value for this setting is three, which is good for most data, however if the data is erratic a higher setting may be necessary.
4. Adjust the Frequency Response Graph's 'Threshold %' setting by setting the numeric value or by dragging the slider control as necessary. This threshold setting adjusts the minimum magnitude for a detectable frequency as a percentage of the Y-scale.

5. When the analysis parameters are satisfactorily adjusted to adequately define the motion decay (or growth) and frequency content, click the [OK] button to pass the engineering data results back to the originating flight test data presentation window.

Procedures for the Heavily Damped Oscillation Method (Figure 4-21)

1. Adjust the 'Trim Point' setting by either editing the numeric value or by dragging the small red indicator on the right side of the 'Time Response Graph'. The trim point should be near the mean value of the oscillatory motion, if there was any, and should correspond to the final steady state condition.
2. The 'Peak Width' and 'Threshold %' settings are disabled with this analysis method. The oscillation frequency is calculated solely on the basis of the 'Time Response Graph'.
3. When the 'Trim Point' is satisfactorily adjusted to the steady state value, click the [OK] button to pass the engineering data results back to the originating flight test data presentation window.

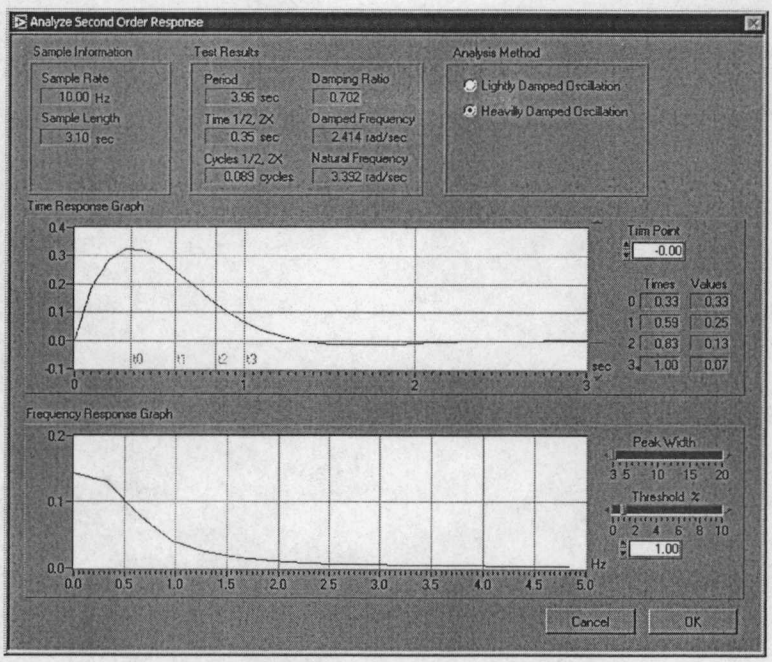


Figure 4-21 Heavily Damped Second Order Response Analysis Window

Analysis Mathematics

Analysis Mathematics for the Lightly Damped Oscillation Method

The mathematics required to calculate the desired engineering data are derived from the equations of a second order response presented in chapter 3.

First, the coordinates of each peak and valley of the time domain data are extracted. Then the absolute value of the difference between the trim value and the ordinate value of each of these coordinates is computed. This result replaces each ordinate of each original coordinate. These new coordinates are passed to a exponential curve fit routine which returns the amplitude and damping coefficients of the descriptive equation through these points. The form of this equation is.

$$y = Ae^{-\zeta\omega_n x} \qquad \textit{eq. 4-26}$$

Where: y = *Ordinate value.*

x = *Abscissa value.*

A = *Amplitude coefficient.*

$-\zeta\omega_n$ = *Exponential decay coefficient.*

e = *Base of natural log*

Then, the original data is passed to a Fast Fourier Transform (FFT) or Discrete Fourier Transform (DFT) routine that returns the frequency content of the recorded motion. The frequency domain data are searched for the most dominant frequency which is returned as the damped natural frequency (ω_d).

Next, the damping coefficient and the damped natural frequency are passed to equation 4-27 to compute the damping ratio.

$$\zeta = -\frac{\zeta\omega_n}{\omega_d}\sqrt{\frac{1}{1+\left(\frac{\zeta\omega_n}{\omega_d}\right)^2}} \qquad \textit{eq. 4-27}$$

Where: ζ = *Damping ratio*

$\zeta\omega_n$ = *Exponential damping coefficient.*

ω_d = *Damped natural frequency.*

Once the damping ratio has been determined the natural frequency can be determined by dividing the exponential damping coefficient by –1 times the damping ratio.

$$\omega_n = \frac{-\zeta\omega_n}{-\zeta} \qquad \textit{eq. 4-28}$$

Where: ω_n = *Natural frequency.*

$\zeta\omega_n$ = *Exponential damping coefficient.*

ζ = *Damping ratio.*

The period of the oscillation is equal to the inverse of the damped natural frequency.

$$P = \frac{1}{\omega_d} \qquad \textit{eq. 4-29}$$

Where: P = *Period of the oscillatory motion.*

ω_d = *Damped natural frequency.*

The time required for the oscillation to subside to one-half amplitude, or the time required for growth to double amplitude are computed as follows:

$$T_{1/2} = \frac{\ln\left(\frac{1}{2}\right)}{\zeta\omega_n} \qquad \textit{eq. 4-30}$$

$$T_2 = \frac{\ln(2)}{\zeta\omega_n} \qquad \textit{eq. 4-31}$$

Where: $T_{1/2}$ = *Time to one-half amplitude.*

T_2 = *Time to double amplitude.*

ζ = *Damping ratio.*

ω_n = *Natural frequency.*

And finally, the number of cycles required for the oscillation to subside to one-half amplitude, or required for growth to double amplitude are computed as follows:

$$C_{1/2} = \frac{T_{1/2}}{P} \qquad \textit{eq. 4-32}$$

$$C_2 = \frac{T_2}{P} \qquad \textit{eq. 4-33}$$

Where: $C_{½}$ = *Cycles to one-half amplitude.*

C_2 = *Cycles to double amplitude.*

$T_{½}$ = *Time to one-half amplitude.*

T_2 = *Time to double amplitude.*

P = *Period.*

Analysis Mathematics for the Heavily Damped Oscillation Method

First, the maximum amplitude of the response is measured and the time at which the maximum amplitude occurs is determined. For this analysis, the time of maximum amplitude is the analysis start time (t_0). Then, 75%, 40%, and 20% of the maximum amplitude are calculated. These values are passed to a cubic spline interpolation to determine the times required to reach each of these percentages. These times are referred to as t_1, t_2, and t_3 respectively. A graphic example of this is shown in figure 4-22. In this example, $t_0 = 0.2920$, $t_1 = 0.5304$, $t_2 = 0.7505$, and $t_3 = 0.9277$ seconds.

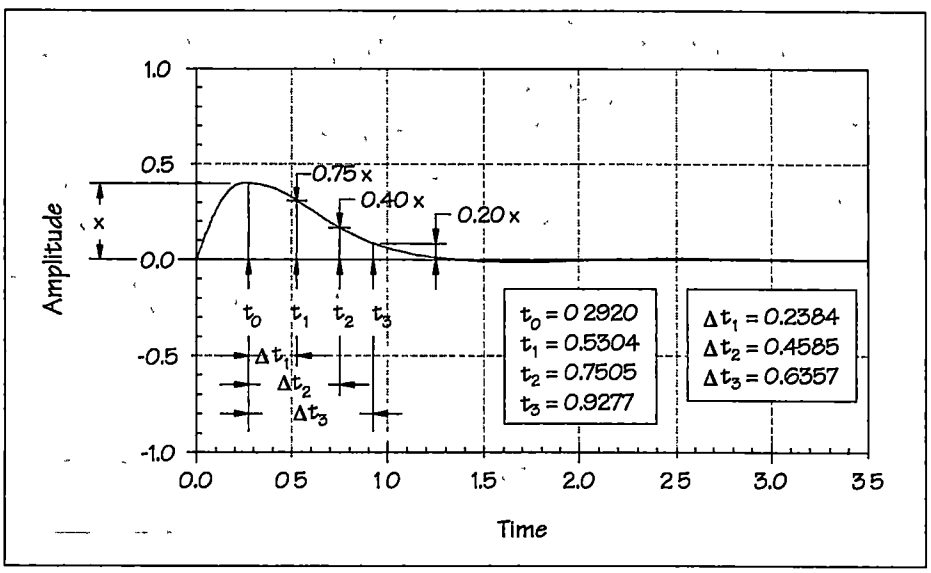


Figure 4-22 Heavily Damped Response Example Plot

Then the following values are computed:

$$\Delta t_1 = t_1 - t_0 \qquad \textit{eq. 4-34}$$

$$\Delta t_2 = t_2 - t_0 \qquad \textit{eq. 4-35}$$

$$\Delta t_3 = t_3 - t_0 \qquad \textit{eq. 4-36}$$

$$Y_1 = \frac{\Delta t_2}{\Delta t_1} \qquad \textit{eq. 4-37}$$

$$Y_2 = 2\frac{\Delta t_3 - \Delta t_2}{\Delta t_2 - \Delta t_1} \qquad \textit{eq. 4-38}$$

$$Y_3 = \frac{\Delta t_3}{2\Delta t_1} \qquad \textit{eq. 4-39}$$

Where: t_0 = *Time at which the response reaches maximum amplitude.*

t_1 = *Time at which the response subsides to 75% of the maximum amplitude.*

t_2 = *Time at which the response subsides to 40% of the maximum amplitude.*

t_3 = *Time at which the response subsides to 20% of the maximum amplitude.*

Δt_1 = *Elapsed time from maximum amplitude to 75% of maximum amplitude.*

Δt_2 = *Elapsed time from maximum amplitude to 40% of max amplitude.*

Δt_3 = *Elapsed time from maximum amplitude to 20% of max amplitude.*

$Y_\#$ = *Damped Response Analysis Chart entry values.*

The values calculated for Y_1, Y_2, and Y_3 are used for the entry points into the Damped Response Analysis Chart. Reading these values across to the appropriate curve, then down to determine the damping ratio. Small variations between the charted damping ratios are typical. The three damping ratio values are averaged. The averaged damping ratio the chart is reentered and the intersection with each of the four upper curves is read to the right. The values read out referred to as Y_4, Y_5, Y_6, and Y_7. Finally, the natural frequency of the response is computed with the following equations:

$$\omega_n = \frac{2 \times Y_4}{\Delta t_3} \qquad \textit{eq. 4-40}$$

$$\omega_n = \frac{Y_5}{\Delta t_2} \qquad \textit{eq. 4-41}$$

$$\omega_n = \frac{Y_6}{\Delta t_3} \qquad \textit{eq. 4-42}$$

$$\omega_n = \frac{Y_7}{3\Delta t_1} \qquad \textit{eq. 4-43}$$

Where: ω_n = *Natural frequency of the response.*

Δt_1 = *Elapsed time from maximum amplitude to 75% of maximum amplitude.*

Δt_2 = *Elapsed time from maximum amplitude to 40% of max amplitude.*

Δt_3 = *Elapsed time from maximum amplitude to 20% of max amplitude.*

$Y_{\#}$ = *Output values from the Damped Response Analysis Chart.*

Equations 4-40 through 4-43 will each produce a value for the natural frequency. Typically there will be some variance between these values; therefore the four values are averaged to find the approximate natural frequency. Once the damping ratio and the natural frequency have been determined, the damped natural frequency can be calculated as follows:

$$\omega_d = \omega_n \sqrt{1-\zeta^2} \qquad \textit{eq. 4-44}$$

Where: ω_d = *Damped natural frequency.*

ω_n = *Natural frequency.*

ζ = *Damping ratio.*

The period of the oscillation is equal to the inverse of the damped natural frequency.

$$P = \frac{1}{\omega_d} \qquad \textit{eq. 4-45}$$

Where: P = *Period of the oscillatory motion.*

ω_d = *Damped natural frequency.*

The time required for the oscillation to subside to one-half amplitude is computed as follows:

$$T_{1/2} = \frac{\ln\left(\frac{1}{2}\right)}{\zeta\omega_n} \qquad \textit{eq. 4-46}$$

Where: $T_{1/2}$ = *Time to one-half amplitude.*

ζ = *Damping ratio.*

ω_n = *Natural frequency.*

And finally, the number of cycles required for the oscillation to subside to one-half amplitude is computed as follows·

$$C_{½} = \frac{T_{½}}{P} \qquad \textit{eq. 4-47}$$

Where. $C_{½}$ = *Cycles to one-half amplitude.*

$T_{½}$ = *Time to one-half amplitude.*

P = *Period.*

For example the trace shown in figure 4-22 the following values were measured from the plot:

$t_0 = 0.2920\ sec$ $t_1 = 0\ 5304 sec$ $t_2 = 0\ 7505\ sec$ $t_3 = 0.9277\ sec$

$\Delta t_1 = 0.2384\ sec$ $\Delta t_2 = 0.4585\ sec$ $\Delta t_3 = 0\ 6357\ sec$

Then the chart entry values are found with equations 4-37, 4-38, and 4-39.

$Y_1 = 1\ 923$ $Y_2 = 1.610$ $Y_3 = 1.333$

Following the red example lines on figure 4-23 the damping ratios are found to be:

$\zeta = 0.754$ $\zeta = 0.770$ $\zeta = 0.781$

These values are averaged to find $\zeta \approx 0.768$.

Then by following the green example lines on figure 4-23 the chart output values are found to be.

$Y_4 = 1\ 19$ $Y_5 = 1\ 72$ $Y_6 = 2.42$ $Y_7 = 2.72$

Finally, the natural frequency is found by equations 4-40 through 4-43 to be:

$\omega_n = 3.744\ rad/sec$ $\omega_n = 3\ 751\ rad/sec$ $\omega_n = 3\ 807\ rad/sec$ $\omega_n = 3\ 803\ rad/sec$

These values are averaged to find $\omega_n \approx 3\ 776\ rad/sec$

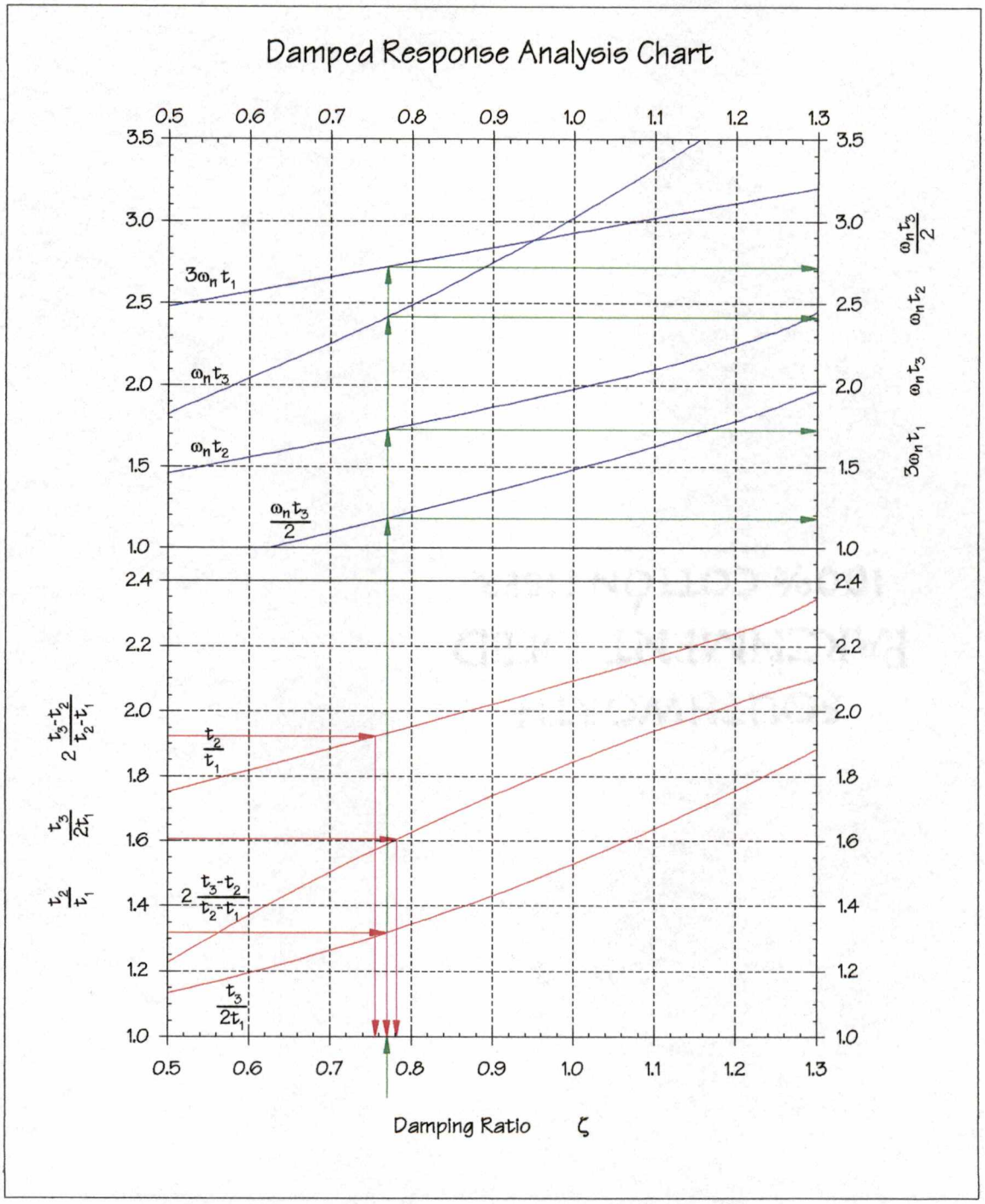


Figure 4-23 Heavily Damped Response Parameter Determination Chart

Display of Characteristic Roots on a Complex Plane

On each data presentation window there is a 'Show Root Locus' button. Once the response has been analyzed and either the time constant (for a first order response) or the natural frequency and the damping ratio (for a second order response) have been determined, the 'Show Root Locus' button will be enabled. Clicking the 'Show Root Locus' button opens the Root Locus Plot window (figure 4-24) and plots the roots of the characteristic mode of motion. This window presents the character of the response motion in a graphical format. The left half of this plane represents increasing stability or convergence, the right half indicates increasing instability or divergence. Characteristic roots on the real axis are indicative of an aperiodic first order response. Whereas, a complex conjugate pair, as shown in figure 4-24, indicate an oscillatory response. Increasing the radial distance from the origin represents an increase in the frequency of the oscillation.

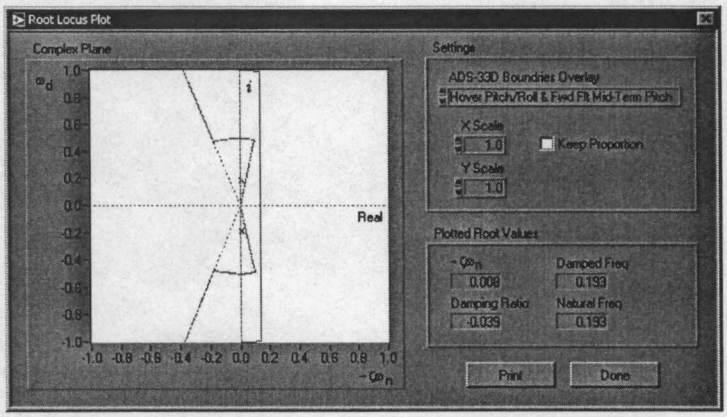


Figure 4-24 Root Locus Plot Window

The 'Settings' section of this window allows the user to overlay the standards established in the ADS-33D handling qualities specification and to rescale the plot as desired. The ADS-33D boundaries plotted, separate the Levels 1, 2, and 3 handling qualities as defined in the specification. There are three options available in the 'ADS-33D Boundary Overlay' dropdown list control. The 'None' selection clears the ADS-33 overlays from the plot. The 'Hover Pitch/Roll & Fwd Flt Mid-Term Pitch' selection displays the boundaries defined in ADS-33D Figure 3(3.3). These boundaries are used for Pitch and roll oscillations during fully attended operations at a hover and low speeds. This overlay is also used for forward flight pitch

oscillations categorized as mid-term by the specification. The mid-term oscillations of the specification are referred to as long term responses in this manuscript. The final option available is the 'Lateral-Directional Oscillatory Requirements' selection. This selection displays the boundaries defined in ADS-33D Figure 9(3.4), which is used to classify the forward flight lateral-directional oscillatory responses [17].

The 'X Scale', 'Y Scale', and 'Keep Proportional' settings allow the user rescale the either or both the X and/or the Y axes. If the 'Keep Proportional' option is checked then both axes will rescale with the 'X Scale' setting. When this window is initially opened, the plot is automatically scaled to ensure the roots are displayed, but the user can rescale the plot as desired.

The 'Plotted Root Values' section of this window displays the characteristic engineering parameters of the root plotted. If the plotted root was from a second order system then the parameters: $-\zeta\omega_n$ (exponential damping coefficient), damping ratio, damped natural frequency, and the natural frequency are displayed as shown in figure 4-24. For a first order response only the characteristic root is shown.

Clicking the 'Print' button sends a copy of this plot to the printer. Clicking the 'Done' button returns the program operation to the data presentation window that called the show root locus plot.

Printing Data

The collected and analyzed data for each of the flight tests may be printed by selecting Print Data from any data presentation window File menu. The format of the printed data is similar to the onscreen display of the data and is consistent with the data presentation methods recommended by the Helicopter Stability and Control Flight Test Manual developed by the United States Naval Test Pilots School. An example of the printed data is shown in Appendix A.

Data Storage

All acquired data can be stored to disk from any of the test data presentation windows. The data can be saved with a proprietary file header information via the File | Save menu selection; or simple raw data may be exported to a text file via the File | Export Data... menu selection.

Saved data can also be read in from the disk by either choosing either the File | Open or the File | Import Data... menu selections.

File Extensions

Each flight test can save data to the disk. To identify which test a given data file was generated by and to simplify the opening of data for subsequent review, each test has a unique file extension. A file extension is the three letter suffix that is appended to the user assigned file name. The default file extensions for each flight test are shown in table 4-3.

Table 4-3 Default File Extensions

Flight Test	Extension
Trimmed Flight Control Positions	.TCP
Critical Azimuth	.CAZ
Hover Static Stability	.HSS
Forward Flight Longitudinal Static Stability	.LSS
Maneuvering Stability	.MAN
Steady Heading Sideslips	.SHS
Hover Long Tern Dynamic Stability	.HLT
Long Tern Dynamic Stability	.LTD
Short Term Dynamic Stability	.STD
Lateral-Directional Oscillation	.LDO
Spiral Stability	.SPL
Control Response	.CR
Turns on One Control	.T1C

File Formats

Data saved to the disk via the File | Save menu selection is saved as an ASCII text file with 15 – 20 lines of header information prefixed at the beginning of the file. This file format is compatible with any text editor, and may be imported into many spreadsheet or mathematical analysis applications; however, the header information may cause some problems with importing the data. If problems arise with this process then the File | Export Data... command should be

used to export the data without the header. Then the data should be accessible in many other programs.

The storage space requirement for the data files is generally minimal, but may vary with the sample rate and the length of the sample. A typical data file is normally smaller than 50 – 100 kilobytes in size. If all data parameters are recorded at a sample rate of 10 hertz, for 30 seconds the file size is less than 100 kilobytes. The file size will grow dramatically at high sample rates. At the maximum sample rate of 100 kHz the file size can easily exceed 10 megabytes. An example printout of the raw data file is presented in Appendix B.

Program Operation

The 'File' menu (figure 4-25) in each of the data presentation windows allows the user to save the current data, open previously saved data, export the current data, import external data, print the current data, setup the printer, and exit the data presentation window.



Figure 4-25 Typical File Menu

After successfully collecting data for any of the flight tests the software will automatically ask the user to save the collected data for future analysis and/or presentation. If the user elects to save the data, a dialog box (figure 4-26) is presented that allows test condition information to be saved with the data.

Then the Save File dialog box (figure 4-27) allows the user to enter the desired file name and path. The data is then written to the user chosen file name and disk location, in a tab delimited text format which is readable in any text editor, word processor, or spreadsheet program. The data files can also be reopened by the Helicopter Stability and Control Analysis Software for further analysis or presentation.

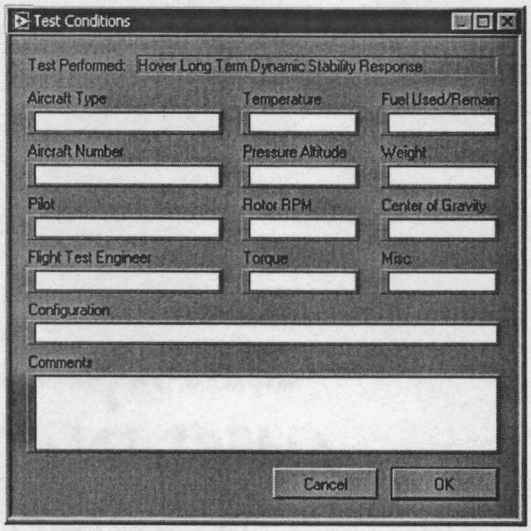


Figure 4-26 Test Conditions Dialog

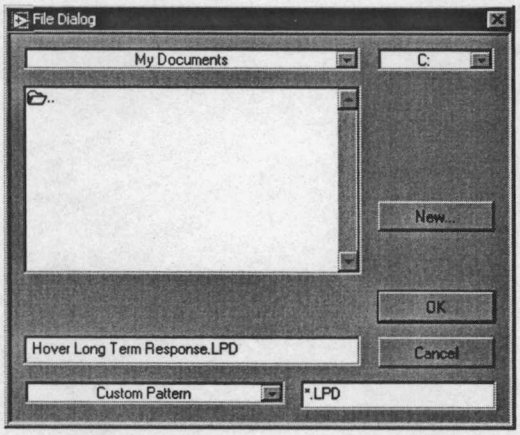


Figure 4-27 Save File Dialog

Chapter 5

HOVER AND LOW SPEED FLIGHT TESTING

General

The helicopter possesses the relatively unique ability to hover and fly at low speeds that are impossible for conventional aircraft. This unique capability necessitates flight testing in a regime that is not necessary for conventional aircraft. While many of the test methods and techniques are derived from conventional aircraft flight test methods some tests are unique to the flight testing of hover capable aircraft. A primary difficulty in conducting low speed flight testing is developing a reliable method to accurately determine the aircraft's speed relative to the air mass. The results of many of the hover and low speed tests are tightly integrated with the actual speed of the aircraft in the airmass, inaccurate or imprecise accounting of the airspeed will quickly contaminate the data and results of this testing. This chapter discusses some basic hover testing theoretical background, test techniques, and procedures for the operation of the Helicopter Stability and Control Analysis Software.

Whether the helicopter is in hover or forward flight, it is governed by the same equations of motion, but the fact that some of the derivatives are quite different strongly affects the flight characteristics of the aircraft in the two flight regimes. It is therefore useful to study the low speed and forward flight characteristics separately. One of the advantages of starting with the hover regime is that the coupling between the longitudinal and the lateral-directional equations is weak enough to be ignored, at least for the purpose of initial assessment. [5]

Safety Considerations

Typically, this testing is conducted before the forward flight 'up and away' testing is conducted and as always an incremental buildup testing procedure should be used. Since most hover and low speed testing will be conducted at very low altitudes above the ground, this testing adds a degree of inherent danger and therefore should be approached with an additional measure of planning and safety.

Theory

The hovering helicopter dynamics as a whole are very complicated and are often nonlinear; however, valuable indications of important handling characteristics can be determined by linear analysis of the aircraft response to small disturbances. From a rigorous standpoint, the set of six equations of motion (one for each degree of freedom of the helicopter) should be augmented with three more equations representing the coning, longitudinal flapping, and lateral flapping of the rotor. However the time constant for the flapping of conventional rotor blades corresponds to one-quarter to one-half of a rotor revolution. This rapid response justifies the *quasi-static* assumption which eliminates blade motion as separate degrees of freedom and simulates replacing the rotor with an imaginary device at the top of the mast, which essentially produces forces and moments instantaneously in response to changes in flight condition or control inputs [5] The dynamics of a helicopter are generally dominated by the forces exerted on the aircraft by the rotor, therefore the following quasi-static rotor characteristics (table 5-1) are presented to indicate the rotor response to various control and external inputs.

This list of quasi-static rotor characteristics gives a good indication of the expected response of the aircraft in the given circumstances, furthermore it is worth noting that besides the control derivatives the blowback/blow-side (M_u and L_v) are typically the most dominate terms for low speed dynamics of helicopters. Both of these derivatives are statically stable for the helicopter in low speed flight. However it is important to note, while both of these derivatives contribute to a measure of static stability, the contribution to the dynamic stability may cause the motion to be unstable in the longer term.

Longitudinal Equations of Motion for a Hovering Helicopter

The longitudinal equations of motion include two force equations (longitudinal and vertical force) and one moment equation (pitch moment). The equations of motion presented below are simplified by three significant assumptions. First the effects of coupling between the longitudinal axis and the lateral and vertical axes are minimal. Second, the derivatives X_w, $X_{\delta coll}$, Z_u, Z_q, $Z_{\delta long}$, M_w, and $M_{\delta coll}$, are set equal to zero since the first order contributions of these derivatives (which are proportional to u_0) are all zero in the hovering regime. Also, X_q is small, having a negligible

Table 5-1 Hovering Flight Quasi-Static Rotor Characteristics [12]

Perturbation	Rotor Response	Rotor Force Applied to the Aircraft	Derivative
Collective Increment $+\Delta\delta_{Coll}$	Increased Coning $+\frac{\partial a_0}{\partial \delta_{Coll}}$	Thrust Increase $+\Delta T$	$-Z_{\delta_{Coll}}$ Collective Control Sensitivity
Right Lateral Cyclic Increment $+\Delta\delta_{Lat}$	Tip Path Plane Tilts Right $+\frac{\partial b_{1s}}{\partial \delta_{Lat}}$	Right Rolling Moment about the x-Axis $+\Delta L$	$+L_{\delta_{Lat}}$ Lateral Control Sensitivity
Forward Longitudinal Cyclic Increment $+\Delta\delta_{Long}$	Tip Path Plane Tilts Forward $-\frac{\partial a_{1s}}{\partial \delta_{Long}}$	Pitch Down Moment about the y-Axis $-\Delta M$	$-M_{\delta_{Long}}$ Longitudinal Control Sensitivity
Downward Increment in Vertical Velocity $+\Delta w$	Increased Blade Angle of Attack, Increased Coning $+\frac{\partial a_0}{\partial w}$	Thrust Increase $+\Delta T$	$-Z_w$ Heave Damping
Forward Increment in Horizontal Velocity $+\Delta u$	Rotor Tilts Aft (Blowback) $+\frac{\partial a_{1s}}{\partial u}$	Nose Up Pitching Moment about the y-Axis due to; Aft tilted thrust vector, Hub moments due to hinge offset (if present), and Aft rotor H-force. $+\Delta M$	$+M_u$ Speed Stability
Right Lateral Increment in Horizontal Velocity $+\Delta v$	Rotor Tilts Left (Blow-side) $-\frac{\partial b_{1s}}{\partial v}$	Left Rolling Moment about the y-Axis due to; Left tilted thrust vector, Hub moments due to hinge offset (if present), and Left rotor H-force. $-\Delta L$	$-L_v$ Lateral Speed Stability
Nose Up Pitch Rate Increment $+\Delta q$	Rotor Lags Shaft by Angle Proportional to Pitch Rate $-\frac{\partial a_{1s}}{\partial q}$	Longitudinal thrust component damps pitch rate, Hinge offset hub moment damps pitch rate.	$-M_q$ Pitch Damping
Right Roll Rate Increment $+\Delta p$	Rotor Lags Shaft by Angle Proportional to Roll Rate $-\frac{\partial b_{1s}}{\partial p}$	Lateral thrust component damps roll rate, Hinge offset hub moment damps roll rate.	$-L_p$ Roll Damping

effect on dynamic motions. [12] Finally, the equations are linearized based upon the small perturbation assumption.

$$\dot{q} - M_u \Delta u - M_q \Delta q = M_{\delta_{Long}} \Delta \delta_{Long} \qquad \textit{eq. 5-1}$$

$$\dot{u} - X_u \Delta u - X_q \Delta q + g \Delta \theta = X_{\delta_{Long}} \Delta \delta_{Long} \qquad \textit{eq. 5-2}$$

$$\dot{w} - Z_w \Delta w = Z_{\delta_{Coll}} \Delta \delta_{Coll} \qquad \textit{eq. 5-3}$$

Where:

- $\dot{q}$ = *Pitch rate acceleration.*
- $\dot{u}$ = *Linear acceleration along the x-axis.*
- $\dot{w}$ = *Linear acceleration along the z-axis.*
- X_q = *Longitudinal force due to pitch rate.*
- X_u = *Longitudinal force due to longitudinal velocity.*
- M_q = *Pitch moment due to pitch rate.*
- M_u = *Pitch moment due to longitudinal velocity.*
- Z_w = *Vertical force due to vertical velocity.*
- Δu = *Change in linear velocity along the x-axis.*
- Δw = *Change in vertical velocity.*
- Δq = *Change in pitch rate.*
- g = *Gravity.*
- $\Delta \theta$ = *Change in pitch angle.*
- $M_{\delta_{Long}}$ = *Pitch moment due to longitudinal control.*
- $X_{\delta_{Long}}$ = *Longitudinal force due to longitudinal control.*
- $Z_{\delta_{Coll}}$ = *Vertical force due to collective control.*
- $\Delta \delta_{Long}$ = *Change in longitudinal control.*
- $\Delta \delta_{Coll}$ = *Change in collective control.*

Lateral-Directional Equations of Motion for a Hovering Helicopter

The lateral-directional derivatives Y_u, Y_r, and L_r, are set equal to zero because they are proportional the 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, the tail rotor thrust inputs could introduce rolling moments and side force. Control

coupling depends on the center of gravity location, tail rotor height, and control system. It is convenient to simplify the discussion by assuming pedal inputs produce only yawing moment and lateral control inputs only rolling moments. [12] The lateral-directional equations include one force equation (lateral or side force) and two moment equations (roll moment and yaw moment).

$$\dot{v} - Y_v \Delta v - Y_p \Delta p = Y_{\delta_{Lat}} \Delta \delta_{Lat} + Y_{\delta_{Ped}} \Delta \delta_{Ped} \qquad \textit{eq. 5-4}$$

$$\dot{p} - L_v \Delta v - L_p \Delta p = L_{\delta_{Lat}} \Delta \delta_{Lat} + L_{\delta_{Ped}} \Delta \delta_{Ped} \qquad \textit{eq. 5-5}$$

$$\dot{r} - N_v \Delta v - N_r \Delta r = N_{\delta_{Lat}} \Delta \delta_{Lat} + N_{\delta_{Ped}} \Delta \delta_{Ped} \qquad \textit{eq. 5-6}$$

Where:

- $\dot{v}$ = *Linear acceleration along the y-axis.*
- $\dot{p}$ = *Roll rate acceleration.*
- $\dot{r}$ = *Yaw rate acceleration.*
- Y_v = *Lateral force due to lateral velocity.*
- Y_p = *Lateral force due to roll rate.*
- L_v = *Roll moment due to lateral velocity.*
- L_p = *Roll moment due to roll rate.*
- N_v = *Yaw moment due to lateral velocity.*
- N_r = *Yaw moment due to yaw rate.*
- Δv = *Change in linear velocity along the y-axis.*
- Δp = *Change in roll rate.*
- Δr = *Change in yaw rate.*
- $Y_{\delta_{Lat}}$ = *Lateral force due to lateral control.*
- $Y_{\delta_{Ped}}$ = *Lateral force due to pedal control.*
- $L_{\delta_{Lat}}$ = *Roll moment due to lateral control.*
- $L_{\delta_{Ped}}$ = *Roll moment due to pedal control.*
- $N_{\delta_{Lat}}$ = *Yaw moment due to lateral control.*
- $N_{\delta_{Ped}}$ = *Yaw moment due to pedal control.*
- $\Delta\delta_{Lat}$ = *Change in lateral control.*
- $\Delta\delta_{Ped}$ = *Change in pedal control.*

Test Procedures

Trimmed Flight Control Positions

The purpose of this test is to determine the control margins and handling qualities in low speed forward, sideward and rearward flight. This test is used to determine if the control inputs necessary to maneuver the helicopter at low speeds are in the usual sense (forward cyclic for forward airspeed, right cyclic for right airspeed, etc.). Cyclic control displacement is an indicator of the aircraft's low speed static stability. Cross coupling is evaluated by looking at the amount of off axis control input necessary for steady trimmed flight at increasing speeds in one direction. Control margin, or percent of control travel remaining, is also evaluated during this test. Monitoring of the control positions for a given data point must include any transient fluctuations required to maintain the desired data point As a general rule of thumb 10% control travel remaining is a good planning figure to allow for unexpected gusts and maneuver recovery. Control margins are often used to set operating limits.

In addition to the quantitative information acquired during this test, the test pilot should make a qualitative assessment of the workload required to maintain each desired data point. This is typically done by assigning a subjective Handling Quality Rating (HQR) according to the Cooper-Harper rating scale (Appendix C) for each data point. Qualitative assessments can also be made of the vibration level at each data point as well as the field-of-view.

Test Method

The test is conducted by flying at steady, trimmed, level flight at a constant speed. Incremental speeds are flown with data recorded at each steady speed. Ambient wind conditions for this test should be three knots or less. The method to determine test vehicle airspeed should be as accurate as possible. The most accurate method requires a pace vehicle with a calibrated airspeed system. Alternate methods include; onboard low-airspeed indicator systems and external ground based telemetry systems. The first airspeed increment from hover should be 2 – 3 knots, then increments of 5 knots are typically used. The testing begins at a hover and proceeds to the faster speeds until the desired speed is reached, a control position test limit is reached, or the test

pilot determines that another speed increment could not be safely completed. This process is repeated for each of the cardinal relative aircraft azimuths.

Pilot Procedures

1. Establish a zero drift hover on the test heading at the test altitude. Record the initial trim conditions.
2. Establish the first airspeed increment, aircraft heading, and altitude with all control forces trimmed to zero, record data.
3. Assign a HQR value for each data point as desired.
4. Repeat steps 2 and 3 for each data point.

Program Operation

Trimmed flight control position testing is accomplished by selecting Acquire Data | Hover and Low Speed Tests | Trimmed Flight Control Positions | Longitudinal Flight (or Lateral Flight) from the menu on the main window (figure 5-1). The Low Speed Trimmed Flight Control Positions Data Collection window (figure 5-2) is used to accumulate the necessary flight test data. The top half of this window displays a 30 second time history of the position of each flight control. The lower half of the window is used to collect data for each desired trim condition.

Data Acquisition Procedures

1. Enter the length of time that the control positions will be monitored on each data point in the 'Sample Length' field. The 'Sample Length' setting can be set from 0 seconds (instant sample) to 10 seconds, the default is 3 seconds. This setting can be changed for subsequent data points as desired.

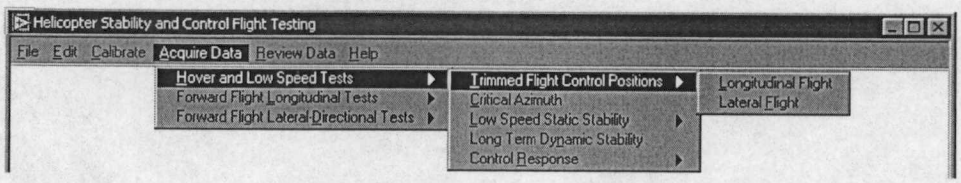


Figure 5-1 Low Speed Trimmed Flight Control Positions Menu Selection

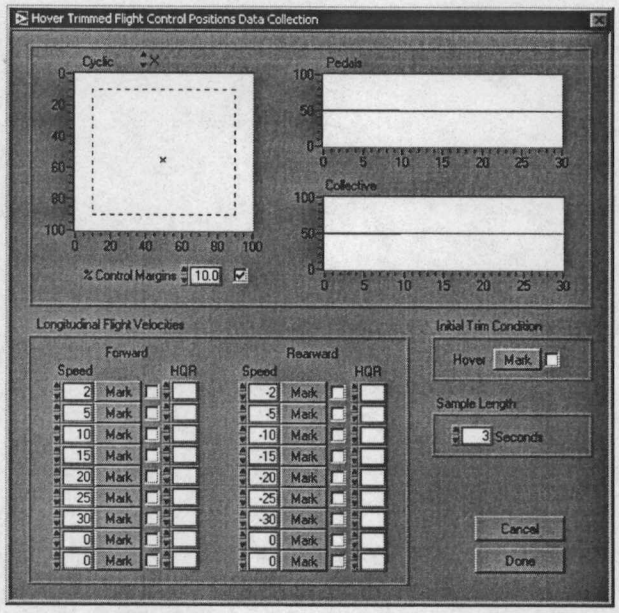


Figure 5-2 Low Speed Trimmed Flight Control Positions Data Collection Window

2. With the aircraft at a stable hover select the [Mark] button associated with 'Hover' to record the initial trim condition. After [Mark] is pressed, the pilot should keep the aircraft stabilized for the duration of the 'Sample Length' setting as the software will monitor the control positions for movement during this period.
3. With the aircraft trimmed on the next data point and the 'Speed' set to the data point being flown, select the [Mark] button corresponding to the speed being flown, to record data. Again the data will be sampled for the duration set in the 'Sample Length' setting.
4. Select or enter a handling quality rating in the 'HQR' field as desired.
5. Repeat steps 3 and 4 for each desired data point.
6. When the all of the desired data points have been collected, select the [Done] button to close the Low Speed Trimmed Flight Control Positions Data Collection window and open the Low Speed Trimmed Flight Control Positions Data Presentation window.

Program Notes:

The default values for the speeds can be changed to any values of interest. If the speed is changed from the default value, it must be set before the [Mark] button is pressed for that speed.

After a data point is marked, the [Mark] button mnemonic changes to [Clear]. If the user wishes to discard the data collected for that data point the [Clear] button can be pressed. The

data for that speed will be discarded, the button mnemonic changes back to [Mark], and new data can be acquired.

Although there are nine available data points for each direction (2, 5, 10, 15, 20, 25, 30, 35, & 40 knots) it is not necessary to complete all entries. For example the user might choose to test 5, 10, 20, 30, & 40 forward and 5, 10, & 20 rearward.

Selecting the [Cancel] button discards all collected data and closes the Low Speed Trimmed Flight Control Positions Data Collection window and returns to the main window.

Selecting the [Done] button closes the Low Speed Trimmed Flight Control Positions Data Collection window and opens the Low Speed Trimmed Flight Control Positions Data Presentation window.

Data Reduction and Presentation

The Low Speed Trimmed Flight Control Positions Data Presentation window is shown in figure 5-3. This window displays the pertinent attitude and control positions for the recorded data. The information on the lower right can be filled in by the user to save relevant test condition information with the data. Once this information has been entered by the user it will be automatically reentered each time a data presentation window is opened. Then only the information that has changed since the last test performed needs to be updated.

Analysis of this information is primarily based on the slope of the cyclic control position plot. In the example data presented in figure 5-3 the longitudinal cyclic stick position plot indicates increasing forward stick displacement with increasing forward airspeed. This plot also indicates increasing aft stick displacement with increasing rearward speed. Both of these indications signify that the control movements are in the usual sense and indicate positive low speed longitudinal static stability.

The following questions should be answered based on the data presented in this window:

1. Were the control movements required in the usual sense (forward cyclic for forward airspeed, right cyclic for right lateral flight, etc.)?
2. Were the control margins reached or exceeded?
3. Was there objectionable coupling between pitch, roll or yaw?

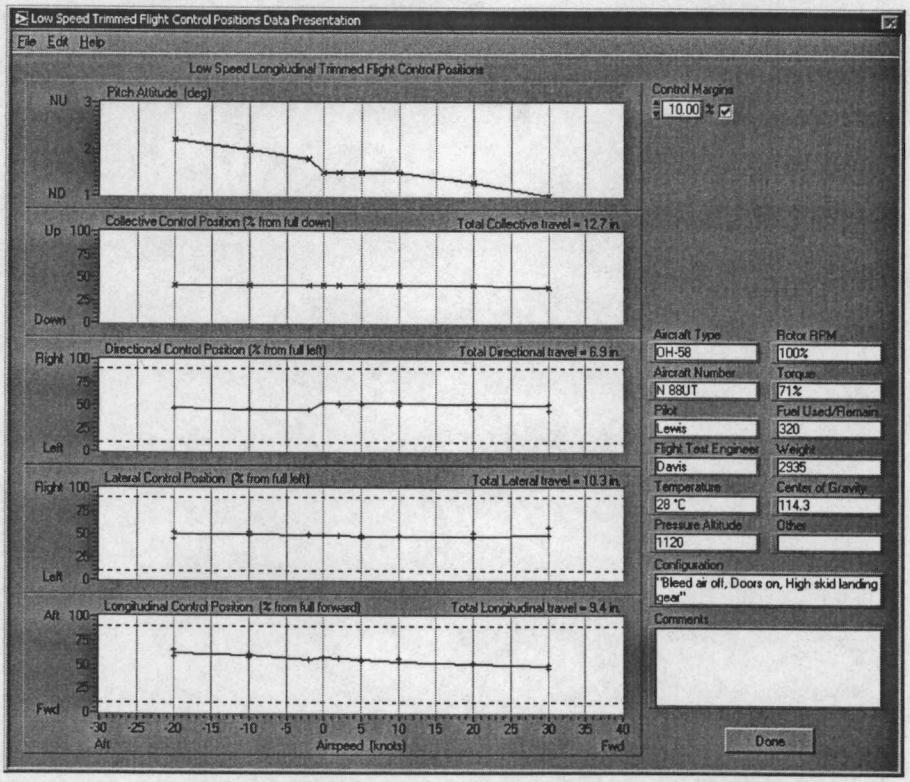


Figure 5-3 Low Speed Trimmed Flight Control Positions Data Presentation Window

4. Were the control position changes with power or airspeed excessive?
5. Were the aircraft attitude changes with power or airspeed excessive?

Critical Azimuth Determination

The purpose of this test is an evaluation of the control margins and pilot workload with changes in airspeed and relative azimuth. The techniques and considerations for trimmed flight control positions apply to critical azimuth testing. Critical azimuth data are typically collected in increments of 30 or 45 degrees although increments of 15 degrees or less are possible. Also, critical azimuth data can be collected for one airspeed or for a series of airspeeds as desired.

Test Methods

Critical azimuth testing is often conducted in conjunction with trimmed flight control positions testing and the same test methods are used. The test is conducted by flying at steady, trimmed, level flight at a constant speed. Data are recorded at the desired relative wind azimuth increments. Ambient wind conditions for this test should be three knots or less. The testing is

conducted to determine pilot workload and control position margins at varying relative wind azimuths. Caution should be exercised as control margins are approached. After each desired relative azimuth is documented for one airspeed the process may be repeated for an airspeed increment.

Pilot Procedures

1. Establish a constant airspeed on the desired test relative azimuth at the test altitude with all control forces trimmed to zero, record data.
2. Assign a HQR value for each data point as desired.
3. Repeat steps 1 and 2 for each data point.

Program Operation

The Critical Azimuth Data Collection window (figure 5-4) is used to accumulate the necessary flight test data. To access this window select Acquire Data | Hover and Low Speed Tests | Critical Azimuth from the menu on the main window. The top half of this window displays a 30 second time history of the position of each flight control. The lower half of the window is used to collect data for each desired azimuth increment.

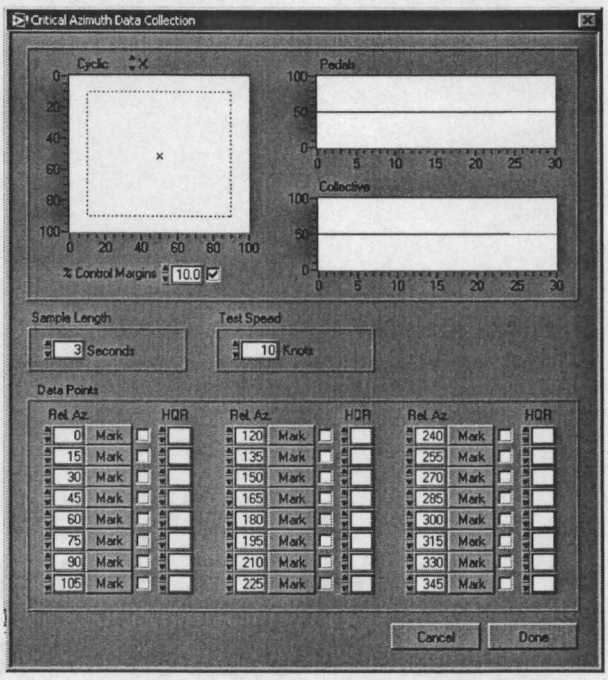


Figure 5-4 Critical Azimuth Data Collection Window

Data Acquisition Procedures

1. Enter the length of time that the control positions will be monitored on each data point in the 'Sample Length' field. The 'Sample Length' setting can be set from zero seconds (instant sample) to 10 seconds, the default is 3 seconds. This setting can be changed for subsequent data points as desired.
2. Enter the 'Test Speed' in knots. This is the speed that that is intended to be flown for each of the desired test azimuths.
3. With the aircraft stable and trimmed at the test speed and with the relative wind from the appropriate azimuth, select the [Mark] button associated with that azimuth to record data. After [Mark] is pressed, the pilot should keep the aircraft stabilized for the duration of the 'Sample Length' setting as the software will monitor the control positions for movement during this period.
4. Select or enter a handling quality rating in the HQR field as desired.
5. Repeat steps 4 and 5 for each desired data point.
6. When the all of the desired data points have been collected, select the [Done] button to close the Critical Azimuth Data Collection Window and open the Critical Azimuth Data Presentation Window.

Program Notes:

The default values for the relative azimuths can be changed to any values of interest. If the relative azimuth is changed from the default value, it must be set before the [Mark] button is pressed for that azimuth.

Although there are 24 preset relative azimuths it is not necessary to complete all entries. Also, test azimuths can be completed in any order at the discretion of the flight crew.

After a data point is marked, the [Mark] button mnemonic changes to [Clear]. If the user wishes to discard the data collected for that data point the [Clear] button can be pressed. The data for that speed will be discarded, the button mnemonic changes back to [Mark], and new data can be acquired.

Selecting the [Cancel] button discards all collected data and closes the Critical Azimuth Data Collection window and returns function to the main window.

Selecting the [Done] button closes the Critical Azimuth Data Collection window and opens the Critical Azimuth Data Presentation window.

Data Reduction and Presentation

The Critical Azimuth Data Presentation window is shown in figure 5-5. This window displays the control positions plotted against the relative azimuths for which data were recorded. The information on the lower right can be filled in by the user to save relevant test condition information with the data. The information entered by the user will be automatically reentered each time a data presentation window is opened. Information that has changed since the last test performed can be updated.

In the top right corner of this window there is a button labeled [Show Polar Plot]. Clicking this button displays the Polar Plot window (figure 5-6). This window provides a different graphical presentation of the same data. The options available for user selection are which data to plot and the scale of the plot. Data that is available for presentation on this plot is the HQR value, lateral, longitudinal, and pedal control positions.

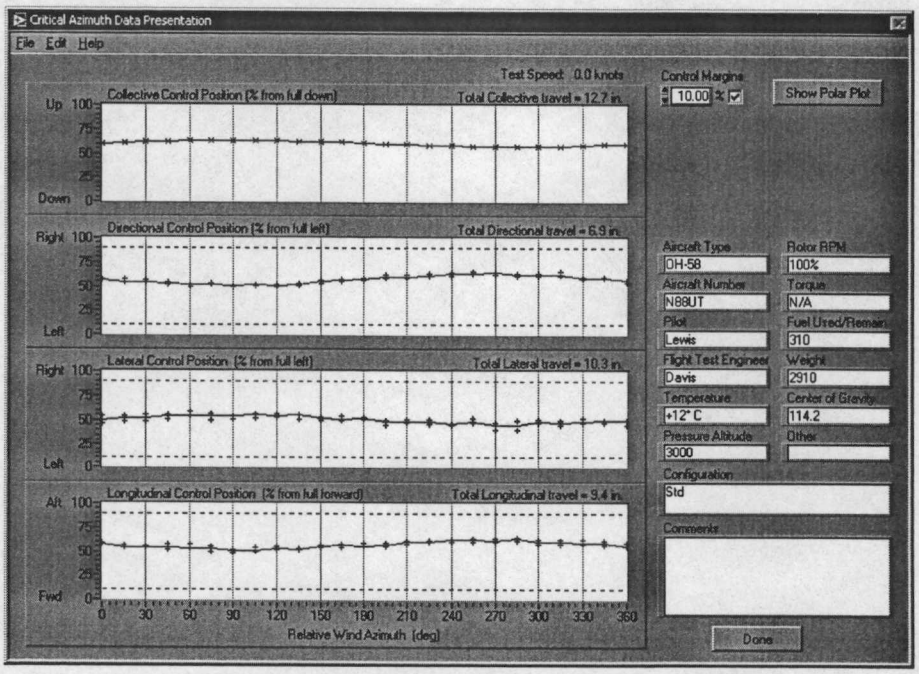


Figure 5-5 Critical Azimuth Data Presentation Window

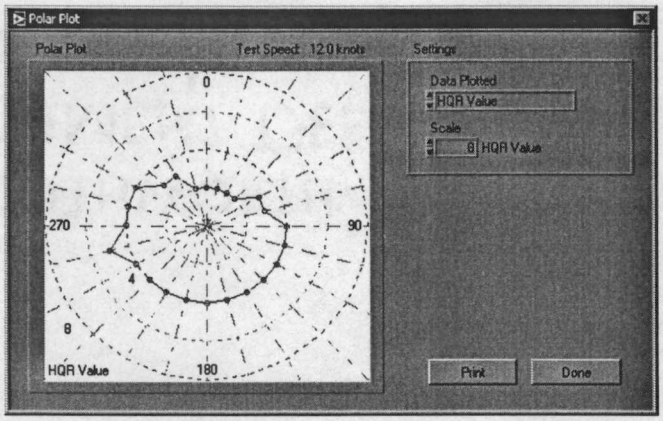


Figure 5-6 Polar Plot of Critical Azimuth Data

Critical azimuth determination is an extension of low speed 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, excessively large or frequent control inputs, or high vibration levels. [12] Critical azimuth data analysis should include:

1. Were the control margins reached or exceeded?
2. Were the control position changes with variation in relative azimuth excessive?
3. Which azimuths produced the highest HQRs?
4. Were the vibration levels objectionable?
5. Was the field-of-view adequate?

Low Speed Static Stability

The purpose of this test is to evaluate the static stability or total restoring moment generated by off trim conditions. This test is used to characterize the low speed static stability of the aircraft as positive, neutral, or negative. Positive static stability is indicated by increasing cyclic stick displacement in the direction of increasing airspeed. No variation in cyclic position at an off trim steady condition indicates neutral static stability. Negative lateral or longitudinal static stability would require a displacement opposite to the direction of flight. Positive directional static stability is indicated by directional pedal control displacement opposite to the direction of sideward flight (etc. left pedal for right lateral flight). Pedal inputs in the same direction as the aircraft motion indicates negative directional stability.

Although the direction of control displacement is an indicator of the sign of the static stability, neither the linear displacement nor the percentage of total travel, for a given airspeed change, is a measure of the strength of the stability. The amount of control displacement is a function of the gearing of the physical flight control linkages. Control margins are also evaluated during this test.

Test Method

This test is conducted by establishing a steady, unaccelerated, trimmed hover and then translating directionally without varying the collective control position. The control positions are compared to the trim condition control positions. Ambient wind conditions for this test should be calm. The method to determine test vehicle airspeed should be as accurate as possible. The most accurate method requires a pace vehicle with a calibrated airspeed system. Alternate methods include; onboard low-airspeed indicator systems, and external ground based telemetry systems. Airspeed increments of 2 – 3 knots are typically used. Low speed static stability is typically evaluated at airspeeds below effective translational lift (12 – 15 knots maximum) The testing begins at a hover and proceeds to the faster speeds until the desired speed is reached, a control position test limit is reached, or the test pilot determines that another speed increment could not be safely completed. For longitudinal static stability forward and rearward flight is conducted, for lateral and directional static stability flight is conducted with a relative wind from 90° and 270°.

Pilot Procedures

1. Establish a zero drift hover on the test heading at the test altitude. Record the initial trim conditions.
2. Accelerate to the desired airspeed (2 – 3 knot increment) with the collective fixed at the zero airspeed trim position, stabilize the airspeed and heading , record data.
3. Repeat step 2 until 10 – 15 knots is reached.

Program Operation

The Critical Azimuth Data Collection window (figure 5-7) is used to accumulate the necessary flight test data. To access this window select Acquire Data |

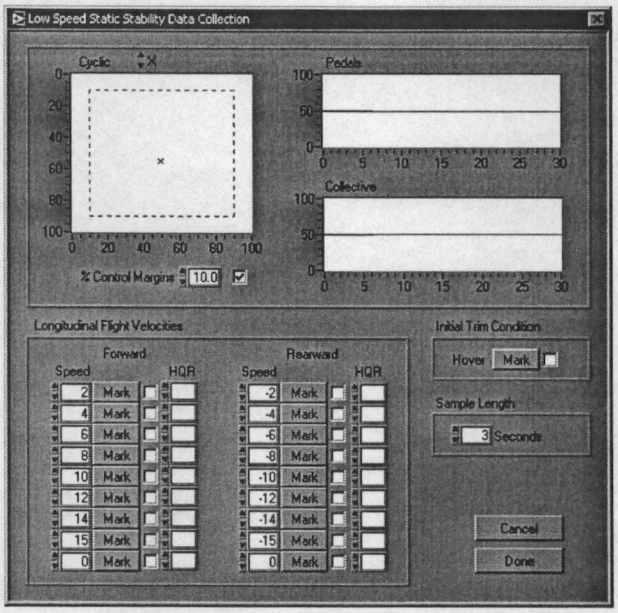


Figure 5-7 Low Speed Static Stability Data Collection Window

Hover and Low Speed Tests | Low Speed Static Stability | Longitudinal Static Stability (or Lateral-Directional Static Stability) from the menu on the main window. The top half of this window displays a 30 second time history of the position of each flight control. The lower half of the window is used to collect data for each desired off-trim condition.

Data Acquisition Procedures

1. Enter the length of time that the control positions will be monitored on each data point in the 'Sample Length' field. The 'Sample Length' setting can be set from 0 seconds (instant sample) to 10 seconds, the default is 3 seconds. This setting can be changed for subsequent data points as desired.
2. With the aircraft at a stable hover select the [Mark] button associated with Hover to record the initial trim condition. After [Mark] is pressed, the pilot should keep the aircraft stabilized for the duration of the Sample Length setting as the software will monitor the control positions for movement during this period.
3. With the aircraft stabilized on the off-trim data point and the 'Speed' set to the data point being flown, select the [Mark] button corresponding to the speed being flown, to record data. Again the data will be sampled for the duration set in the 'Sample Length' setting.
4. Repeat step 3 for each desired data point.

5. When the all of the desired data points have been collected, select the [Done] button to close the Low Speed Static Stability Data Collection window and open the Low Speed Static Stability Data Presentation window.

Program Notes:

The default values for the relative azimuths can be changed to any values of interest. If the relative azimuth is changed from the default value, it must be set before the [Mark] button is pressed for that azimuth.

After a data point is marked, the [Mark] button mnemonic changes to [Clear]. If the user wishes to discard the data collected for that data point the [Clear] button can be pressed. The data for that speed will be discarded, the button mnemonic changes back to [Mark], and new data can be acquired.

Selecting the [Cancel] button discards all collected data and closes the Low Speed Static Stability Data Collection window and returns function to the main window.

Selecting the [Done] button closes the Low Speed Static Stability Data Collection window and opens the Low Speed Static Stability Data Presentation window.

Data Reduction and Presentation

Analysis of this information is primarily based on the slope of the cyclic control (or pedal control) position plot. An example of the Low Speed Static Stability Data Presentation window is presented as figure 5-8. This example is a lateral-directional data set but is representative of the longitudinal data as well. In the example data presented in figure 5-8 the lateral cyclic stick position plot indicates increasing right stick displacement with increasing right lateral airspeed. This plot also indicates increasing left stick displacement with increasing left lateral speed. Both of these indications signify positive low speed lateral static stability. The positive slope of the lateral cyclic control position plot indicates positive lateral static stability. This same example shows an increase in left pedal associated with an increase in right lateral airspeed and an increase in right pedal with an increase of left lateral flight speed. Both of these indications denote positive low speed directional static stability. Note that for directional stability the *negative* slope of the pedal control position plot indicates positive lateral static stability. Finally, for longitudinal

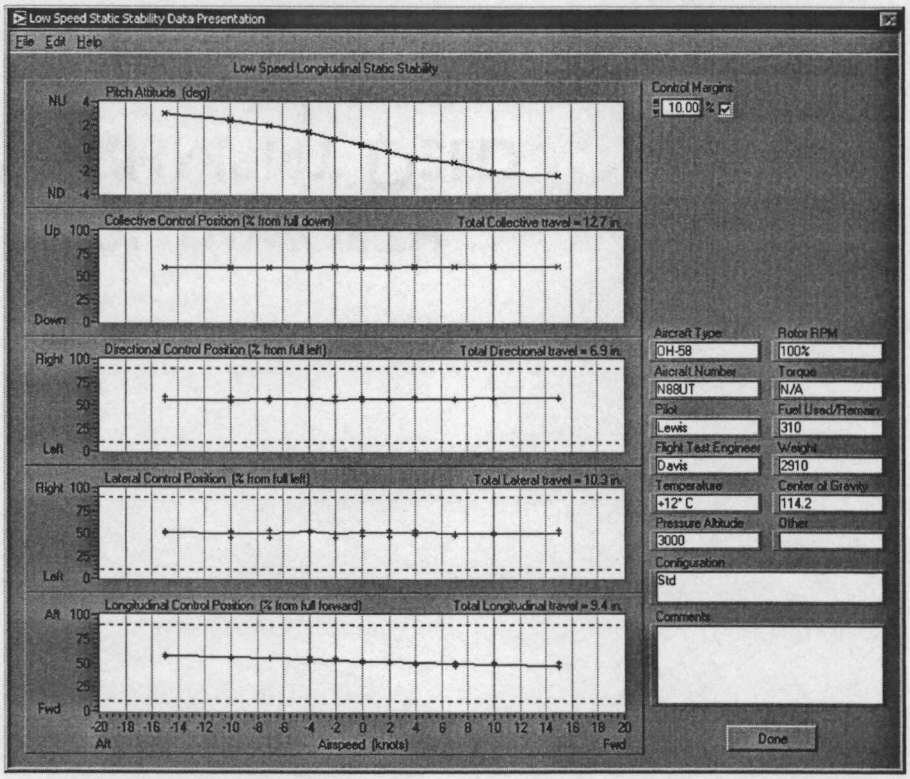


Figure 5-8 Low Speed Static Stability Data Presentation Window

static stability an increasing stick displacement (in the same direction as the speed change) with an increase in airspeed is the indication of positive low speed longitudinal static stability. Graphically this would be plotted as a negative sloping line for positive static stability. A summary of the static stability graphical indications are presented in table 5-2.

The information on the lower right can be filled in by the user to save relevant test condition information with the data. Once this information has been entered by the user it will be automatically reentered each time a data presentation window is opened. Then only the information that has changed since the last test performed needs to be updated.

Analysis of low airspeed static stability data should consider the following questions: [12]

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 aircraft characteristics suitable for the intended mission.

Table 5-2 Summary of Static Stability Graphing Indications

Test Axis	Aircraft Motion	What control input was required?	Slope of the control position curve	Static Stability
Longitudinal	Increasing Forward Speed	Increasing Forward Cyclic	Negative	Positive
		Neutral	Zero	Neutral
		Increasing Aft Cyclic	Positive	Negative
	Increasing Aft Speed	Increasing Aft Cyclic	Negative	Positive
		Neutral	Zero	Neutral
		Increasing Forward Cyclic	Positive	Negative
Lateral	Increasing Right Lateral Speed	Increasing Right Cyclic	Positive	Positive
		Neutral	Zero	Neutral
		Increasing Left Cyclic	Negative	Negative
	Increasing Left Lateral Speed	Increasing Left Cyclic	Positive	Positive
		Neutral	Zero	Neutral
		Increasing Right Cyclic	Negative	Negative
Directional	Increasing Right Lateral Speed	Increasing Left Pedal	Negative	Positive
		Neutral	Zero	Neutral
		Increasing Right Pedal	Positive	Negative
	Increasing Right Lateral Speed	Increasing Right Pedal	Negative	Positive
		Neutral	Zero	Neutral
		Increasing Left Pedal	Positive	Negative

Low Speed Long Term Dynamic Stability

The purpose of this test is to evaluate the long term response of the hovering helicopter as a function of time following a disturbance. The long term dynamic response is typically fully coupled in all three axes. This mode is generally a low frequency lightly damped oscillation that is typically damped by the pilot or by the Automatic Flight Control System (AFCS). The damping ratio is often slightly negative which results in a oscillatory divergent response. If the aircraft exhibits negative low speed static stability in any axis, the dynamic response in that axis will be aperiodic and divergent and will likely dominate the overall response. In this case the recovery from the test maneuver will have to be made quickly as the divergence will likely occur within a very short time span.

Test Method

This test is conducted by establishing a steady, unaccelerated, trimmed hover and fixing the flight controls while the aircraft is permitted to respond to natural external stimuli. This is an unsatisfactory technique if no response occurs, however this indicates a desirable aircraft characteristic. [12] If no natural excitation occurs, an artificial stimulus must be applied to initiate a response. Normally a longitudinal cyclic input is applied to produce an off-trim pitch attitude of about five degrees, then the controls are returned and fixed in the trim positions and the aircraft response is recorded. Since the aircraft response is typically coupled, an alternate technique may be used. This technique involves attempting to maintain the trim attitude in one or two axes while allowing one or more axes to respond naturally. For example the test pilot may maintain heading while the pitch and roll are allowed to respond. Another technique might maintain heading and roll attitude while the pitch response is monitored. This technique is difficult to perform and will often cause some contamination of the examined parameters. The advantage of this technique is a simplification of the data reduction process. Although this technique minimizes the data reduction and may be useful for generalizing one aspect of the low speed characteristics, it is not recommended for accurately documenting the aircraft's low speed long term dynamic response.

Pilot Procedures

1. Establish a zero drift, unaccelerated, hover with all control forces trimmed to zero.
2. Fix the flight controls in the trimmed positions and monitor and record the aircraft response to any external stimuli.

Or, if no natural response is obtained;

3. Introduce a small off-trim attitude change (5 – 10° change of pitch or roll), and immediately return the flight controls to the trim positions as accurately as possible. Record the aircraft response.

Program Operation

The Low Speed Long Term Dynamic Stability Data Collection window (figure 5-9) is displayed during the hover dynamic stability flight testing. To access this window select

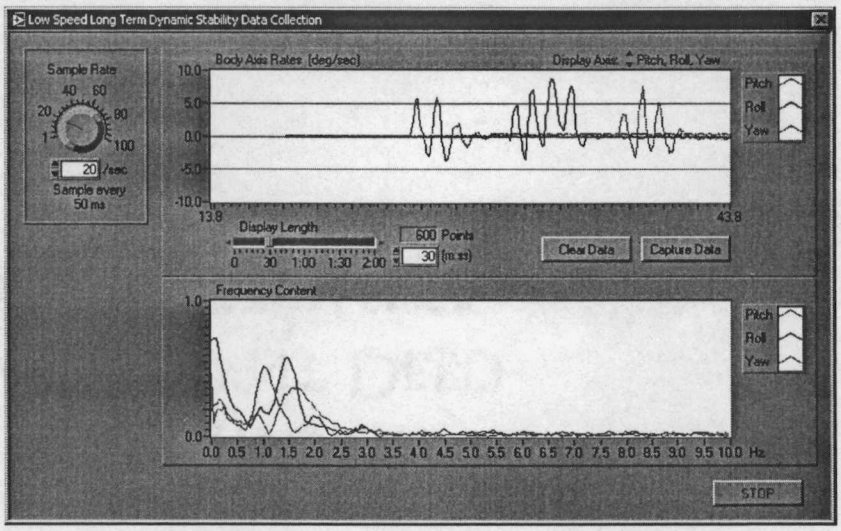


Figure 5-9 Low Speed Long Term Dynamic Stability Data Collection Window

Acquire Data | Hover and Low Speed Tests | Long Term Dynamic Stability from the menu on the main window. Since the data collected for this test is sampled over a period of time the user can select the rate of acquisition of the data samples. This is done with the 'Sample Rate' knob at the top left of the window. This setting adjusts the number of samples per second that will be acquired in the range of 1 sample per second to 100 samples per second. The time interval between each data sample is displayed below the 'Sample Rate' knob. In the example shown in figure 5-9 the sample interval is 50 milliseconds. The default setting for the sample rate is 20 samples per second and is generally suitable for this test.

Two continuously updating graphs are displayed in this window. The top graph is a time history of the body axis rates in degrees per second, the bottom graph displays the frequency content present for the selected body axis rates. When this window is initially opened both graphs display continuous streaming data for all three axes. The user may select specific axes to monitor by selecting the desired axes with the 'Display Axis' selection control. Since the hover long term dynamic response is typically coupled between all three axes it is recommended that this selection remain on 'Pitch, Roll, Yaw' for this test. The user can also set the length of time that is displayed on the time history plot, this is done by adjusting the 'Display Length' slider control. The valid range for this control, on this test, is from ten seconds to two minutes. The default length for this test is 30 seconds. The number of data points held in memory can be

calculated by multiplying the sample rate times the display length for example in figure 5-9 $20 \frac{\text{samples}}{\text{sec}}$ x 30 sec = 600 samples. Data that is older than the display length is 'expired' and can not be recovered. The display length must be long enough to capture the characteristic motion of the aircraft without the time expiring.

The bottom graph displays the frequency information for each axis displayed in the top graph. The frequency information is determined by a Fourier transformation of the time domain data. The range of frequencies that can be determined are directly related to the sample rate. The highest frequency that can be determined is one-half of sample rate in accordance with the Nyquist Theorem.

The low speed long term dynamic stability test procedures are conducted while this window is displayed, then before the data on the time history plot expires, the [Capture Data] button must be pressed. When the [Capture Data] button is pressed the time history data from the aircraft body attitude, body rates, accelerations, and stick positions are sent to the data presentation window.

Data Acquisition Procedures

1. Set the 'Sample Rate' as desired.
2. Set the 'Display Length' as desired.
3. Perform the Pilot Procedures above to stimulate the long period response.
4. When the desired response data is shown in the time history graph select the [Capture Data] button to close the Low Speed Long Term Dynamic Stability Data Collection window and open the Low Speed Long Term Dynamic Stability Data Presentation window.

Program Notes:

If the data on the time history graph is allowed to scroll off the window or 'expire', the data is lost forever. To prevent this from happening increase the 'Display Length' setting. Setting the 'Display Length' to an overly long setting will cause the data presentation window to display superfluous data. The default settings for the sample rate and the display length are normally

sufficient to properly document most typical helicopter responses but some experimentation with these settings may be necessary.

The 'Sample Rate' setting should *not* be changed during a flight test. All data for a specific test must be acquired at a constant sampling rate or the results will be inaccurate.

The [Clear Data] button clears both the time history and the frequency content graphs. It is not necessary to clear data before beginning an attempt to acquire flight test data, but it may be desirable to clear extraneous data before a "for-record test" is performed. This is especially recommended if the display length is set to a long length and there are several minutes of unnecessary or extraneous data preceding the test.

Data Reduction and Presentation

The time history data of the aircraft body attitudes, body rates, accelerations, and stick positions are available for analysis and presentation in the Low Speed Long Term Dynamic Stability Data Presentation window (figure 5-10). The left side of this window presents the relevant time histories, while the right side contains the analysis tools, engineering data results, and general test information The time histories display the scaled raw data from which the engineering data results are calculated. The time histories that are displayed for this test are pitch rate, pitch attitude, roll rate, roll attitude, and longitudinal stick position, although the user may select other plots as desired. To display plots of different parameters the user can click on the graph title and choose from a list of available plots as shown in figure 5-11.

When this window is initially opened, the 'Engineering Data Results' section is blank. These data are determined by means of a process that is initiated by clicking the [Analyze Response] button. This analysis can be done when the data is collected or the data can be saved and the analysis can be performed at a later time.

Although it is recognized that the long term dynamic response of the hovering helicopter is a third order (or higher) function, that is coupled between all axes, the analysis routine in this program will isolate only one axis at a time and return the coefficients of a second order function representative of the motion about that axis. Each axis may be examined, but this analysis is done in isolation from the motion of the other axes. This analysis is done by choosing the desired

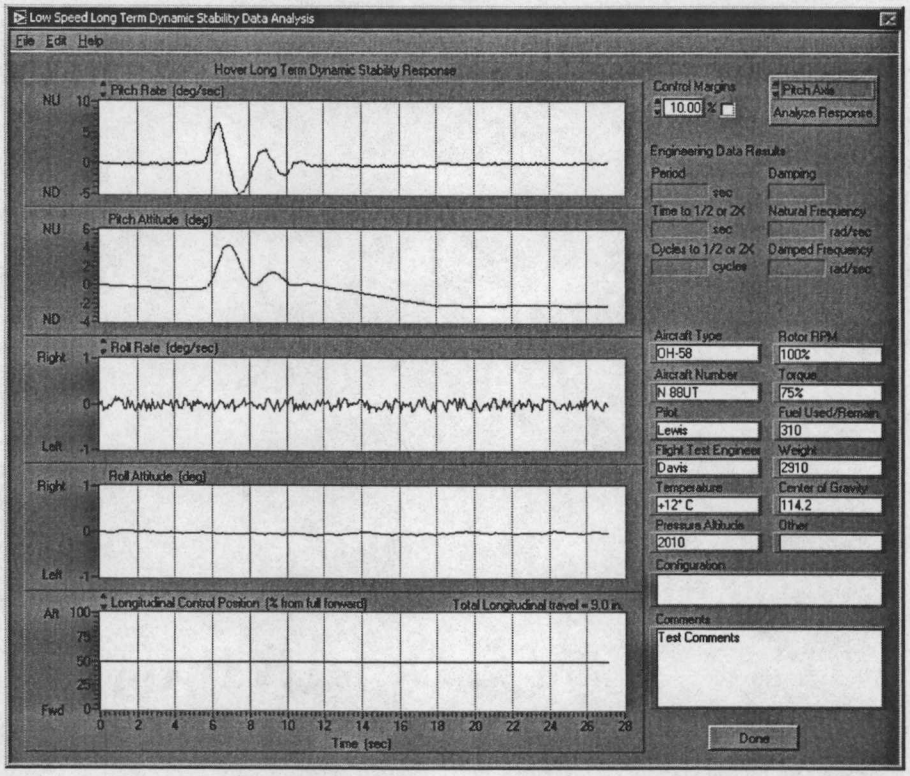


Figure 5-10 Low Speed Long Term Dynamic Stability Data Presentation Window

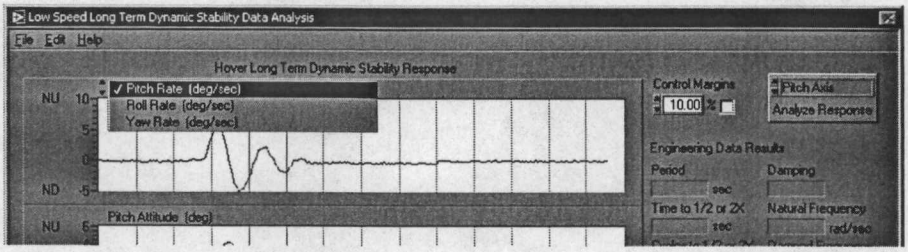


Figure 5-11 Selecting Alternative Time History Data Plots

axis from the selection at the top of the [Analyze Response] button and then clicking the [Analyze Response] button. The results of the analysis are returned and filled into the 'Engineering Data Results' section of the window. For more information on the response analysis process see the Second Order Dynamic Response Data Analysis section in Chapter 4.

The information on the lower right can be entered by the user to save relevant test condition information with the data. Once this information has been entered by the user it will be automatically reentered each time a data presentation window is opened. Then only the information that has changed since the last test performed needs to be updated.

Data Reduction

The engineering data results for the characteristic motion about the axis of interest can be determined by the following process:

1. Select the desired axis from the selection control at the top of the [Analyze Response] button.
2. Click the [Analyze Response] button.
3. Complete the procedures outlined in the Second Order Dynamic Response Data Analysis section in Chapter 4.

The following characteristic parameters are returned from this analysis:

- Period of the Motion
- Time to half or double Amplitude
- Cycles to half or double Amplitude
- Damping Ratio
- Natural Frequency
- Damped Natural Frequency

4. Once the engineering data parameters have been determined the user may opt to view the data on a complex plane by clicking the [Show Root Locus] button. For more information on the Root Locus plot see the Display of Characteristic Roots on a Complex Plane section in Chapter 4.

The data shown in the Low Speed Long Term Dynamic Stability Data Presentation window can also be visually analyzed for several important characteristics. Some of the attributes that may be determined are:

1. Is the response in each axis convergent or divergent?
2. Is the response in one axis? Is the response coupled between two or all three axes?
3. What is the phase delay between the axes for coupled responses?
4. What are the relative magnitudes between coupled axes?

Analysis of low airspeed long term dynamic stability data should also address the following questions:

1. How easy was it to excite the long term response?
2. How easy was it for the pilot to suppress the long term response?
3. Are the aircraft characteristics suitable for the intended mission?

Control Response

The purpose of this test is to evaluate the aircraft response to a flight control input. The aircraft response is determined by the control sensitivity, damping, and response lags in the examined axis. Control sensitivity is the acceleration per unit control input. Damping is the acceleration per unit of velocity. And the response lags are a consequence of the flight control system characteristics and rotor system type.

Several characteristics contributing to desirable response qualities are listed below. [12]

1. Pilots want a vehicle responsive enough to achieve some level of attitude change within a certain time after a control input (without being overly responsive).
2. Pilots want a reasonably steady state rate for a given input size.
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.

Each of these qualities are directly related to control sensitivity, damping, or both. The resulting motion due to a given control input can be characterized as a first order system.

Test Method

This test is conducted by establishing a steady, unaccelerated, trimmed out-of-ground-effect hover, applying control step input, and monitoring the aircraft response. An incremental buildup technique should be used to approach the maximum desired control input magnitude. Control fixtures are generally used to restrain control inputs. The control input should be quick and precise. If a fixture is used, the control should be firmly held against the fixture but not forcefully. Recovery is initiated when a steady state rate is achieved or a predetermined test flight limit is reached.

Test limits on bank angle can prevent achieving a steady state roll rate. If the test helicopter can achieve high roll rates and/or the test program is restricted to low roll attitudes, the recovery may occur prior to achieving a steady state roll rate. In this case a smaller control input may be evaluated that allows the aircraft to achieve a steady state rate, or a 'false start' technique may be used. In this technique, the test pilot establishes a bank attitude in the direction opposite to the

test input. Then the control step input is made from the banked control position. The use of the false start technique permits a greater roll attitude change and a longer time to elapse prior to recovery at the test attitude limit. This technique may permit the helicopter to achieve the steady state rate for the desired control input.

The data reduction process uses the steady state rate to determine aircraft characteristics, but does *not* critically depend on discerning a steady state rate. There are alternate analysis methods included in the program that do not require a steady state to determine the desired aircraft characteristics.

Pilot Procedures

1. Establish a zero drift, unaccelerated, OGE hover with all control forces trimmed to zero.
2. Set the control fixture for the desired displacement and position it for the desired input direction. The control input size and direction are verbally and visually verified between the pilot and copilot/flight test engineer. Start with small inputs and increase in size incrementally until desired input is achieved.
3. Apply the control input against the fixture and hold the controls fixed. Recover when a steady state rate is obtained or a predetermined test flight limit is reached.
4. Record the aircraft response.
5. When the maneuver is complete, announce the recovery so the control fixture can be quickly removed.
6. Repeat steps 1 through 5 for increasing control displacements until the maximum desired deflection is attained.
7. Qualitatively assess the predictability of the control response.

Program Operation

The Control Response Data Collection window (figure 5-12) is displayed during low speed control response flight testing. To access this window select Acquire Data | Hover and Low Speed Tests | Control Response | (Pitch, Roll, Yaw, or Collective) from the menu on the main window. Since the data collected for this test is sampled over a period of time the user can select the rate of acquisition of the data samples. This is done with the 'Sample

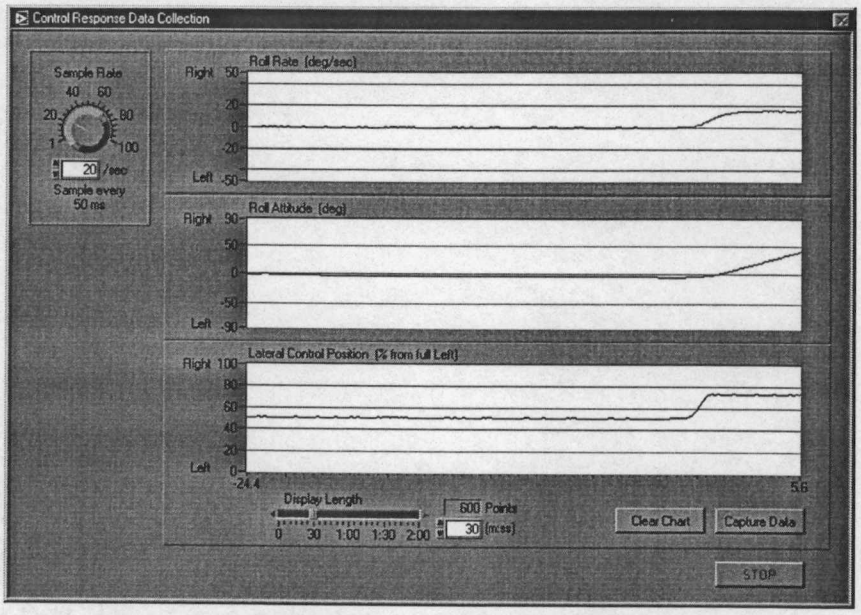


Figure 5-12 Control Response Data Collection Window

Rate' knob at the top left of the window. This setting adjusts the number of samples per second that will be acquired in the range of 1 sample per second to 100 samples per second. The time interval between each data sample is displayed below the 'Sample Rate' knob. The default setting for the sample rate is 20 samples per second and is generally suitable for this test.

Three continuously updating time history plots are displayed in this window. The length of time that is displayed on these graphs can be adjusted with the 'Display Length' slider control at the bottom of the graphs. The valid range for this control, on this test, is from ten seconds to two minutes. The default length for this test is 30 seconds. Also displayed with the 'Display Length' slider control is the 'Points' indicator which displays the number of data samples that are held in memory. This value is calculated by multiplying the sample rate (in samples per second) times the display length (in seconds). Only data displayed on the screen are useable. The 'Display Length' defines the time interval at which the data expire and are no longer available for determining the characteristic motion. The display length must be adjusted to capture the characteristic motion of the aircraft without allowing necessary data to expire.

The top graph is a time history of the body axis rates in degrees per second, or in the case of collective control response this graph displays the aircraft's vertical velocity in feet per second. The data presented on this graph are the basis for determining the engineering data results.

Ideally the data presented on this graph will attain a steady state rate during the performance of the flight test technique. The middle graph displays the appropriate body attitude or in the case of collective control response this is a plot of normal acceleration. The bottom graph shows the appropriate control position versus time plot.

The control response test procedures are conducted while this window is displayed. Before the data on the time history plots expire, the [Capture Data] button must be pressed. When the [Capture Data] button is pressed the time history data from the aircraft body attitude, body rates, accelerations, and stick positions are sent to the Control Response Data Presentation window.

Data Acquisition Procedures

1. Set the 'Sample Rate' as desired.
2. Set the 'Display Length' as desired.
3. Perform the Pilot Procedures above to test the control response.
4. When the desired response data is shown in the time history graph select the [Capture Data] button to close the Control Response Data Collection window and open the Control Response Data Presentation window.

Program Notes:

If the data on the time history graph is allowed to scroll off the window or 'expire', the data are no longer available. To prevent the loss of necessary data increase the 'Display Length' setting. Setting the 'Display Length' to an overly long setting will cause the data presentation window to display unnecessary data. The default settings for the sample rate and the display length are normally sufficient to properly document typical helicopter responses but some experimentation with these settings may be necessary to achieve the desired results.

The 'Sample Rate' setting should *not* be changed during a flight test. All data for a specific test must be acquired at a constant sampling rate or the results will be inaccurate.

The [Clear Data] button clears all three time history graphs. It is not necessary to clear data before beginning an attempt to acquire flight test data, but it may be desirable to clear extraneous data before a "for-record test" is performed. This is especially recommended if the display length is set to a long length and there are several minutes of unnecessary data preceding the test.

Data Reduction and Presentation

The time history data of the aircraft body attitudes, body rates, accelerations, and stick positions are available for analysis and presentation in the Control Response Data Presentation window (figure 5-13). The left side of this window presents the relevant time histories, while the right side contains the analysis tools, engineering data results, and general test information. The time histories display the scaled raw data from which the engineering data results are calculated.

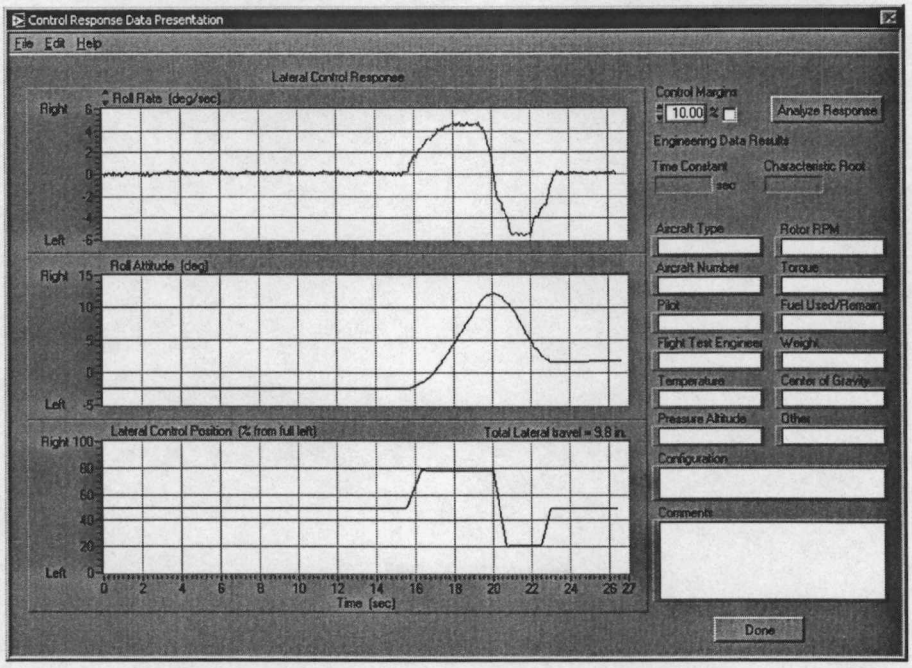


Figure 5-13 Control Response Data Presentation Window

When this window is initially opened, the 'Engineering Data Results' section is blank. These data are determined by means of a process that is initiated by clicking the [Analyze Response] button. This analysis can be done when the data are collected or the data can be saved and the analysis can be performed at a later time. Analysis is done by clicking the [Analyze Response] button. The results of the analysis are returned and filled into the 'Engineering Data Results' section of the window. For more information on the response analysis process see the First Order Dynamic Response Data Analysis section in Chapter 4.

The information on the lower right can be filled in by the user to save relevant test condition information with the data. Once this information has been, initially entered, it will be

automatically reentered each time a data presentation window is opened. Only the information that has changed since the last test performed needs to be updated.

Data Reduction

The engineering data results for the characteristic motion about the axis of interest can be determined by the following process:

1. Click the [Analyze Response] button.
2. Complete the procedures outlined in the First Order Dynamic Response Data Analysis section in Chapter 4.

The following characteristic parameters are returned from this analysis:

- Time Constant
- Characteristic Root

3. Once the engineering data parameters have been determined the user may opt to view the data on a complex plane by clicking the [Show Root Locus] button. For more information on the Root Locus plot see the Display of Characteristic Roots on a Complex Plane section in Chapter 4.

The data shown in the Control Response Data Presentation window can also be visually analyzed for several important characteristics. Some of the attributes that may be determined are the steady state rate, maximum rate, initial acceleration, maximum acceleration, and acceleration delay.

Analysis of low airspeed control response data should also address the following questions:

1. Was the response predictable?
2. Was the response consistent?
3. Was the steady state rate adequate?
4. Are the aircraft characteristics suitable for the intended mission?

Chapter 6

FORWARD FLIGHT LONGITUDINAL FLIGHT TESTING

General

This chapter discusses the helicopter longitudinal stability and control characteristics in forward flight. For the purposes of this discussion the term forward flight is defined as translational flight generally along the aircraft's x-axis at airspeeds (relative to the air mass) from approximately 30-40 knots to the aircraft's maximum designed never exceed speed (V_{NE}). The forward flight longitudinal characteristics include the pitch moments about the y-axis, and the linear force components along the x and z axes. The theoretical background for longitudinal stability and control flight testing, forward flight longitudinal flight test techniques, and procedures for the operation of the Helicopter Stability and Control Analysis Software are presented in this chapter.

Theory

The forward flight characteristics of the helicopter are governed by the static stability, dynamic stability, and flight control properties of the aircraft. The rotor produces many of the most dominant forces that control the helicopter dynamics, therefore table 6-1 presents the quasi-static rotor characteristics to indicate the rotor response to control inputs and external disturbances.

The following physical descriptions of the controlling forces and moments are presented to better understand the overall reactions of the helicopter in forward flight.

Longitudinal Static Stability

The longitudinal static stability can be measured by determining the stick displacement (or force required) versus aircraft speed. The static stability is said to be positive when cyclic stick displacement (increasing displacement in the direction of the increase in airspeed) is required to maintain an increase in airspeed. The helicopter longitudinal static stability characteristics are influenced by many aerodynamic factors, but are dominated by the effects of the main rotor and

Table 6-1 Forward Flight Longitudinal Quasi-Static Rotor Characteristics [12]

Perturbation	Rotor Response	Rotor Force Applied to the Aircraft
Collective Increment $+\Delta\delta_{Coll}$	Increased Coning, Increased Lateral Flapping due to Coning, Increased Longitudinal Flapping, Nose up $+\frac{\partial a_0}{\partial \delta_{Coll}}, +\frac{\partial b_{1s}}{\partial a_{1s}}, +\frac{\partial a_{1s}}{\partial \delta_{Coll}}$	$+\Delta T$, Thrust Increase $+\Delta M_H$, Nose Up $+\Delta M_{CG}$, Nose Up
Forward Longitudinal Cyclic Increment $+\Delta\delta_{Long}$	Tip Path Plane Tilts Forward $-\frac{\partial a_{1s}}{\partial \delta_{Long}}$	$-\Delta T$, Thrust Decrease $-\Delta M_H$, Nose Down $-\Delta M_{CG}$, Nose Down
Downward Increment in Vertical Velocity $+\Delta w$	Increased Blade Angle of Attack, Increased Coning, Increased Lateral Flapping due to Coning, Increased Longitudinal Flapping, Nose up $+\frac{\partial a_0}{\partial w}, +\frac{\partial b_{1s}}{\partial a_{1s}}, +\frac{\partial a_{1s}}{\partial \delta_{Coll}}$	$+\Delta T$, Thrust Increase $+\Delta M_H$, Nose Up $+\Delta M_{CG}$, Nose Up
Forward Increment in Horizontal Velocity $+\Delta u$	Rotor Tilts Aft (Blowback) $+\frac{\partial a_{1s}}{\partial u}$	$+\Delta T$, Thrust Increase (low u_0) $-\Delta T$, Thrust Decrease (high u_0) $+\Delta H$, Aft, Nose Up $+\Delta M_H$, Nose Up $+\Delta M_{CG}$, Nose Up
Nose Up Pitch Rate Increment $+\Delta q$	Rotor Lags Shaft by Angle Proportional to Pitch Rate $-\frac{\partial a_{1s}}{\partial q}$	$\Delta T = 0$ $+\Delta H$, Aft, Nose Up $-\Delta M_H$, Nose Down (Damping) $-\Delta M_{CG}$, Nose Down (Damping)

the effects of the horizontal tail surfaces. The main rotor creates pitching moments as a result of the attributes of speed stability and angle of attack stability.

Speed Stability

Considering the aircraft in trimmed level flight at a constant forward velocity, the rotor is producing an equal amount of lift on the left and right halves of the rotor disk despite the fact that there is a relative velocity differential between the advancing and retreating halves of the disk. This lift equilibrium is accomplished by cyclic feathering and rotor flapping. The rotor blades that are on the advancing side of the disk, experiencing an increased relative velocity flap, up to

reduce the angle of attack and the retreating blades flap down to increase the angle of attack. Typically, on U.S. built helicopters, the advancing rotor blades are on the right side in forward flight.

A rotor system that is suddenly subjected to an increase of airspeed (i.e. caused by a headwind gust) experiences an increased relative velocity on the advancing blades and a relative velocity decrease on the retreating blades The immediate result is a dissymmetry of lift condition that is manifested as a nose up pitching moment due to the mechanism of gyroscopic precession. The gyroscopic precession, or phase angle, is exactly 90° for rotating bodies with a flap hinge located at the axis of rotation. The 90° phase angle is reduced for rotor systems that have the flap hinge displaced (offset) in plane, from the axis of rotation. Rotor systems that have offset are affected by this reduction of the phase angle resulting in roll coupling with all pitch moments and vise-versa. For the purposes of this discussion, the phase angle will be assumed to be approximately 90°. The resulting aft tilt of the rotor disk has an aft thrust component that tends to decelerate the aircraft back towards the trim airspeed Therefore, speed stability has a stabilizing effect The speed stability is often referred to as blowback.

Angle of Attack Stability

Angle of attack stability is described by an initially trimmed rotor system at a constant forward airspeed that is affected by a sudden increase of angle of attack on all elements of all rotor blades. This can be caused by a collective control position change, an upward/downward gust through the rotor, a sudden change in the aircraft velocity vertical component, or a combination of these contributions. Considering the trimmed system that experiences a sudden increase of angle of attack, all blades encounter an equal angle of attack change, however, the lift increase is not equal for the advancing blades compared to the retreating blades. Although the entire rotor disk experiences an increase in lift the advancing half of the disk has a greater lift change that results in a nose up pitching moment due to the gyroscopic precession. The resulting aft tilt of the rotor produces an additional angle of attack increase which further increases the nose up pitching moment. This increasing nose up pitching moment with increasing angle of attack indicates that angle of attack stability is a destabilizing attribute.

The fuselage and tail also contribute to the angle of attack stability. Most fuselages are unstable with respect to angle of attack. At increased angles of attack, the fuselage typically contributes a nose up pitching moment. The opposite is true of the horizontal tail, increased angle of attack generally produces a nose down moment and a stabilizing effect. The relative strengths of these contributions vary with the physical design of each helicopter and with airspeed. At low airspeeds the contributions of the fuselage and tail are negligible, and at higher forward speeds some helicopters gain sufficient stability from the horizontal tail to overcome the unstable tendencies of the rotor.[18] The overall angle of attack stability for a given helicopter in a specific configuration is the resultant sum of the rotor, fuselage, and tail moments due to the angle of attack change.

Longitudinal Dynamic Stability

Dynamic stability examines the time rate of change of the motion of the aircraft. When considering the longitudinal characteristics of a helicopter in forward flight over a period of time following a disturbance, the effects of speed stability and angle of attack stability become coupled. That is to say that a disturbance in one mode will create a disturbance of the other mode. Additionally, the effects of aerodynamic damping must be included in discussions of dynamic responses. Helicopters in forward flight have dynamic characteristics that vary widely not only between aircraft type; but also, between same type at varying gross weight, center of gravity, configuration, and forward airspeed. [18] Two oscillatory dynamic modes typically exist as discussed below.

One can observe both longitudinal modes of motion in forward flight by suddenly deflecting the longitudinal cyclic control to a new position and holding it there. The initial response of the helicopter is to pitch in the appropriate direction corresponding to the control displacement and to develop a change in G forces. This short term behavior subsides within 1 to 2 seconds and involves primarily pitch angle (θ), angle of attack (α), and G force or normal acceleration (N_z). Velocity and altitude do not change appreciably during this short time. [18] If the control deflection (assume an aft step input) is held fixed at the displaced position, at the end of the short term motion the nose will be above the initial position and the airspeed will begin to change.

Because of the aircraft deceleration, the speed stability of the rotor will create a nose down moment. As the nose of the aircraft pitches down the component of weight of the aircraft along the flight path increases, tending to accelerate the aircraft. Thus there is a slowly changing relationship between pitch moments and x-axis forces, which causes long period oscillations involving pitch attitude (θ), velocity (u), flight path angle (γ), and altitude. [18] If the long period motion is damped, the aircraft approaches a new equilibrium state – in this example, at a lower velocity. Figure 6-1 shows a typical time history plot of the pertinent parameters of the longitudinal modes of motion described above.

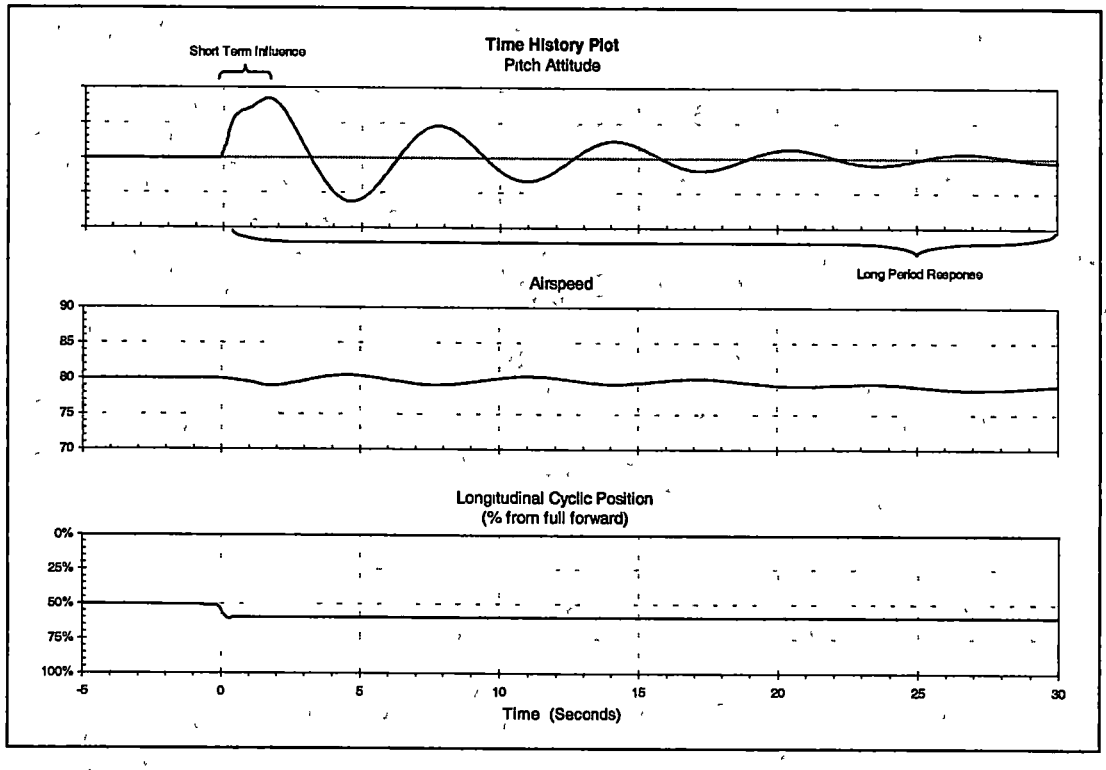


Figure 6-1 Example Time History Plot of Longitudinal Modes of Motion

Short Term Response

As discussed in Chapter 3 the short term response characteristics of a helicopter in forward flight influence the handling qualities in performing maneuvering tasks requiring pitching motion. The pilot should be able to make changes in pitch attitude and flight path quickly and easily. Instabilities in the short period response of the aircraft would make the aircraft very difficult to control. The short period response is typically oscillatory and convergent and may be heavily

damped This response is typical of a spring-mass-damper system with major contributions from the parameters presented in table 6-2

The spring or restoring force is a consequence of the rotor system design, the two design parameters that control the restoring force are the mast height and the flapping hinge offset The mechanism for the mast height to provide a restoring force is that the pitching moment from the thrust of the main rotor is directly proportional to the height of the thrust vector above the aircraft center of gravity The restoring force due to hinge offset is due to the moment created by the centrifugal force of a rotor system that is rotating out of plane with the rotor hub The system damping is primarily a result of the aerodynamic viscous effects

Ordinarily, the speed remains essentially constant while the pilot makes short term pitch attitude changes, therefore changes in airspeed are neglected when discussing the short period response For a constant speed the short period response is primarily governed by three aircraft

Table 6-2 Analogy of Short Term Response and Spring Mass Damper System [12]

Spring Mass Damper System	**Helicopter Pitch Rate Response**
Variable	Q
Mass Normalized Damping Constant	$\left(-Z_w - M_q\right)$
Mass Normalized Spring Constant	$\left(Z_w M_q - u_0 M_w\right)$
Mass Normalized Forcing Function	$\left[M_{\delta_{Long}} \delta_{Long} + \left(M_w Z_{\delta_{Long}} - Z_w M_{\delta_{Long}}\right)\right]\delta_{Long}$

Where q = *Pitch rate*

Z_w = *Vertical force due to vertical velocity (Heave damping)*

M_q = *Pitch moment due to pitch rate (Pitch rate damping)*

u_0 = *Initial velocity*

M_w = *Pitch moment due to vertical velocity (Angle of attack stability)*

$M_{\delta_{Long}}$ = *Pitch moment due to longitudinal control*

$Z_{\delta_{Pong}}$ = *Vertical force due to longitudinal control*

δ_{Long} = *Longitudinal control*

stability derivatives. These derivatives include the angle of attack derivative (M_w), the pitch rate damping (M_q), and the vertical or heave damping (Z_w) derivatives.

It can be shown that the solution for the two roots of the short period mode are as follows:

$$\lambda_{1,2} = -\left(\frac{-Z_w - M_q}{2}\right) \pm \sqrt{\left(\frac{-Z_w - M_q}{2}\right)^2 - \left(Z_w M_q - u_0 M_w\right)} \quad \textit{eq. 6-1}$$

Where: $\lambda_{1,2}$ = *Characteristic roots.*

M_q = *Pitch moment due to pitch rate.*

M_w = *Pitch moment due to vertical velocity.*

Z_w = *Vertical force due to vertical velocity.*

u_0 = *Initial velocity component along the x-axis.*

The short term response of the helicopter, like the response of a spring mass damper system, is aperiodic if the characteristic roots are real and is oscillatory if the roots are complex numbers. If $\left(\frac{-Z_w - M_q}{2}\right)^2 \rangle \left(Z_w M_q - u_0 M_w\right)$, λ_1 and λ_2 are real roots and the time response is a combination of first order (non-oscillatory) exponential responses. Since the heave damping derivative (Z_w) and the pitch damping derivative (M_q) are both inherently negative, the term $Z_w M_q$ is positive. If the angle of attack derivative (M_w) is negative and large enough for $\left(\frac{-Z_w - M_q}{2}\right)^2 \langle \left(Z_w M_q - u_0 M_w\right)$, λ_1 and λ_2 are a complex conjugate pair and the time response is sinusoidal When the angle of attack derivative (M_w) is positive, 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 the term $Z_w M_q - u_0 M_w$ [12]

The parameter, $Z_w M_q - u_0 M_w$, might be termed the maneuvering stability level (MSL). When this parameter is positive, both 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 [12]

Long Term Response

The long period response of a helicopter is similar to the phugoid response of a conventional airplane with one significant difference. The airplane phugoid response generally does not involve angle of attack changes whereas the angle of attack parameter is primary contributor to the helicopter long period response.

The long period response of a helicopter can also be modeled by a spring-mass-damper system and can be represented by a second order differential equation. The parameters of the spring-mass-damper system are presented in table 6-3.

Table 6-3 Analogy of Long Term Response and Spring Mass Damper System [12]

Spring Mass Damper System	Helicopter Speed Response
Variable	u
Mass Normalized Damping Constant	$\left[X_u + g\frac{\left[-Z_w^2 M_u - M_w\left(u_0 M_u - M_q Z_u - Z_w Z_u\right)\right]}{\left(Z_w M_q - M_w u_0\right)^2}\right]$
Mass Normalized Spring Constant	$\left[g\frac{\left(M_w Z_u - M_u Z_w\right)}{\left(Z_w M_q - M_w u_0\right)}\right]$
Mass Normalized Forcing Function	$\left[g\frac{\left(-Z_{\delta_{Long}} M_w + M_{\delta_{Long}} Z_w\right)}{\left(Z_w M_q - M_w u_0\right)}\right]\delta_{Long}$

Where: u = *x-axis aircraft velocity.*

g = *Gravity.*

Z_w = *Vertical force due to vertical velocity (Heave damping).*

Z_u = *Vertical force due to forward velocity.*

M_q = *Pitch moment due to pitch rate (Pitch rate damping).*

M_u = *Pitch moment due to forward velocity (Speed stability)*

M_w = *Pitch moment due to vertical velocity (Angle of attack stability)*

u_0 = *Initial velocity.*

$M_{\delta_{Long}}$ = *Pitch moment due to longitudinal control.*

$Z_{\delta_{Pong}}$ = *Vertical force due to longitudinal control.*

δ_{Long} = *Longitudinal control.*

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 responses. Similarly, the effective stiffness for the helicopter speed 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. [12]

Longitudinal Equations of Motion for a Helicopter in Forward Flight

The longitudinal equations for forward flight include two force equations (longitudinal and vertical force) and one moment equation (pitch moment). The equations of motion presented below assume that the initial flight path is near level ($\cos\gamma_0 \approx 1$) and that there is minimal coupling with the lateral-directional motion of the aircraft. [12]

$$-\Delta\dot{u} - w_0\Delta q + X_u\Delta u + X_w\Delta w + X_q\Delta q + -g\cos\gamma_0\Delta\theta\ldots = X_{\delta_{Long}}\Delta\delta_{Long} + X_{\delta_{Coll}}\Delta\delta_{Coll} \qquad \textit{eq. 6-2}$$

$$\left(-\Delta\dot{u} - w_0\Delta q\right) + \left(X_u\Delta u + X_w\Delta w\right) + \left(X_q\Delta q\right) + \left(-g\cos\gamma_0\Delta\theta\right)\ldots = \left(X_{\delta_{Long}}\Delta\delta_{Long} + X_{\delta_{Coll}}\Delta\delta_{Coll}\right) \qquad \textit{eq. 6-3}$$

$$\left(-\Delta\dot{w} - u_0\Delta q\right) + \left(Z_u\Delta u + Z_w\Delta w\right) + \left(Z_q\Delta q\right) + \left(-g\sin\gamma_0\Delta\theta\right)\ldots = \left(Z_{\delta_{Long}}\Delta\delta_{Long} + Z_{\delta_{Coll}}\Delta\delta_{Coll}\right) \qquad \textit{eq. 6-4}$$

$$\left(-\Delta q\right) + \left(M_u\Delta u + M_w\Delta w\right) + \left(M_q\Delta q + M_w\Delta w\right)\ldots = \left(M_{\delta_{Long}}\Delta\delta_{Long} + M_{\delta_{Coll}}\Delta\delta_{Coll}\right) \qquad \textit{eq. 6-5}$$

Where:

- m = *Mass.*
- I_{yy} = *Moment of inertia about the y-axis.*
- $\Delta\dot{u}$ = *Change in linear acceleration along the x-axis.*
- Δw = *Change in linear acceleration along the z-axis.*
- Δu = *Change in linear velocity along the x-axis.*
- Δw = *Change in vertical velocity.*
- u_0 = *Initial velocity component along the x-axis.*
- w_0 = *Initial velocity component along the z-axis.*

Δq = *Change in pitch rate.*

X_u = *Longitudinal force due to longitudinal velocity.*

X_w = *Longitudinal force due to vertical velocity.*

X_q = *Longitudinal force due to pitch rate.*

g = *Gravity.*

γ_0 = *Initial flight path angle.*

$\Delta\theta$ = *Change in pitch attitude.*

Z_u = *Vertical force due to longitudinal velocity.*

Z_w = *Vertical force due to vertical velocity.*

Z_q = *Vertical force due to pitch rate.*

M_u = *Pitch moment due to longitudinal velocity.*

M_w = *Pitch moment due to vertical velocity.*

M_q = *Pitch moment due to pitch rate.*

$X_{\delta_{Long}}$ = *Longitudinal force due to longitudinal control.*

$X_{\delta_{Coll}}$ = *Longitudinal force due to collective control.*

$Z_{\delta_{Long}}$ = *Vertical force due to longitudinal control.*

$Z_{\delta_{Coll}}$ = *Vertical force due to collective control.*

$M_{\delta_{Long}}$ = *Pitch moment due to longitudinal control.*

$M_{\delta_{Coll}}$ = *Pitch moment due to collective control.*

$\Delta\delta_{Long}$ = *Change in longitudinal control*

$\Delta\delta_{Coll}$ = *Change in collective control.*

Longitudinal Stability Derivatives

The general form of the equation for the longitudinal moment stability derivatives is presented as equation 6-6. This general equation is used for each pertinent flight or control variable (u, w, q, δ_{Long}, and δ_{Coll}) while the holding the remaining variables constant. The four terms in equation 6-6 can be examined individually and relative magnitudes can be assessed for each term, then the relative magnitudes can be summed to gauge the overall impact of the examined variable. The terms of this equation represent the major contributors to the longitudinal stability of the helicopter, and include ‹1› the rotor hub force (H) acting perpendicular to the mast,

‹2› the moments due to rotor thrust (T) acting perpendicular to the tip path plane, ‹3› the pitch moment due to the rotor hub force (M_H), and ‹4› the fuselage and tail moment ($M_{CGf\&t}$). The fuselage drag force is assumed to act through the center of gravity, and thus does not produce a pitching moment. [12]

$$M_{(\,)} = \frac{1}{I_{yy}}\left[\overbrace{\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial a_{1s}}{\partial(\,)}}^{\langle 1\rangle} + \overbrace{(\ell_m + ha_{1s})\frac{\partial T}{\partial(\,)}}^{\langle 2\rangle} + \overbrace{h\frac{\partial H}{\partial(\,)}}^{\langle 3\rangle} + \overbrace{\frac{\partial M_{CG_{f\&t}}}{\partial(\,)}}^{\langle 4\rangle}\right]$$

eq. 6-6

Where:

- $M_{(\,)}$ = *Pitch moment due to examined variable.*
- $(\,)$ = *Examined flight or control variable (u, w, q, δ_{Long}, or δ_{Coll})*
- I_{yy} = *Moment of inertia about the y-axis.*
- T = *Main rotor thrust.*
- h = *Height of the rotor hub above the center of gravity.*
- e = *Flapping hinge offset.*
- M_S = *Blade mass moment.*
- Ω = *Rotor angular velocity.*
- a_{1s} = *Longitudinal flapping angle (shaft referenced).*
- ℓ_m = *Longitudinal distance between the mast and the CG.*
- H = *Rotor hub force perpendicular to the mast.*
- $M_{CG_{f\&t}}$ = *Pitch moment due to aerodynamic forces of fuselage and tail.*

Test Procedures

Trimmed Flight Control Positions

The purpose of this test is to determine the variation in control positions and handling qualities with changes in power and airspeed. The entire forward flight airspeed envelope is typically evaluated and the full range of power settings from maximum continuous power to autorotation. Control margin, or percent of control travel remaining, is evaluated during this test.

In addition to the quantitative information acquired during this test, the test pilot should make a qualitative assessment of the workload required to maintain each desired data point. This is

typically done by assigning a subjective Handling Quality Rating (HQR) according to the Cooper-Harper rating scale for each data point.

Test Methods

Two basic procedures are used to evaluate the forward flight trimmed control positions. The level flight procedure measures the control inputs necessary for various airspeeds, and the climbing and descending procedure evaluates the control displacements with power changes.

The level flight test is conducted by flying at steady, trimmed, level flight at a constant speed. Incremental speeds are flown with data recorded at each steady speed. Airspeed increments of 10 knots are typically used. The testing begins at the slow speeds and proceeds to the faster speeds until the desired speed is reached, a control position test limit is reached, or the test pilot determines that another speed increment could not be safely completed.

For climbs and descents, one or more mission representative airspeeds are used. The helicopter is stabilized on constant airspeed and the power is varied from autorotation to climb. In this test the airspeed remains constant while the power is varied. A total of at least five power increments should be evaluated.

Pilot Procedures (Level Flight)

1. Establish a steady, coordinated, level flight at the desired test airspeed with all control forces trimmed to zero.
2. Record data.
3. Assign a HQR value for each data point as desired.
4. Establish the next airspeed increment and repeat steps 1 – 3 for each data point.

Pilot Procedures (Climbing and Descending Flight)

1. Establish a steady, coordinated, level flight at the desired test airspeed with all control forces trimmed to zero, record data.
2. Adjust the collective to achieve the desired power increment while maintaining airspeed constant. Retrim the control forces to zero and record data. The altitude should remain within ±1000 feet of the initial test altitude.
3. Assign a HQR value for each data point as desired.

4. Repeat steps 2 and 3 for each data point.

Program Operation

Forward flight trimmed control position testing is accomplished by selecting Acquire Data | Forward Flight Longitudinal Tests | Trimmed Flight Control Positions | Level Flight (or Climbing & Descending Flight) from the menu on the main window (figure 6-2). The Level Trimmed Flight Control Positions Data Collection (figure 6-3) and the Climbing & Descending Trimmed Flight Control Positions Data Collection (figure 6-4) windows are used to accumulate the necessary flight test data. The top half of each of these windows displays a 30 second time history of the position of each flight control. The lower half of the window is used to collect data for each desired trim condition.

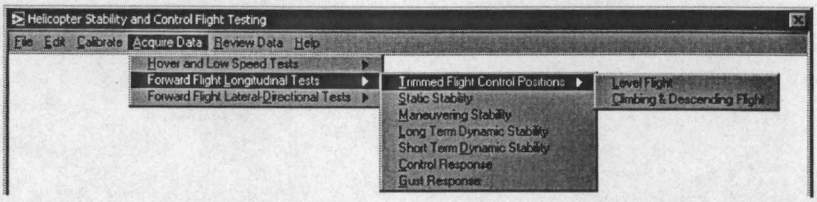


Figure 6-2 Forward Flight Trimmed Flight Control Positions Menu Selection

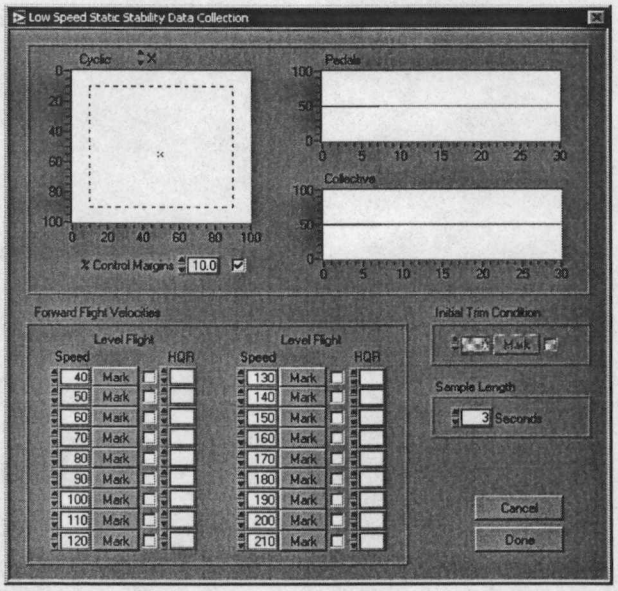


Figure 6-3 Level Trimmed Flight Control Positions Data Collection Window

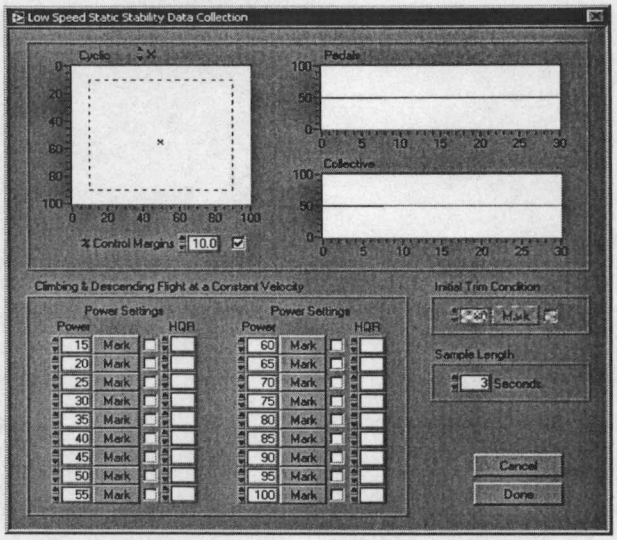


Figure 6-4 Climbing and Descending Trimmed Flight Control Positions Data Collection Window

Data Acquisition Procedures

1. Enter the length of time that the control positions will be monitored on each data point in the 'Sample Length' field. The 'Sample Length' setting can be set from 0 seconds (instant sample) to 10 seconds, the default is 3 seconds. This setting can be changed for subsequent data points as desired.
2. Choose a target airspeed (level flight) or power setting (climbing and descending flight) and adjust the 'Speed' or the 'Power' setting as necessary.
3. With the aircraft trimmed on the target speed or power setting for the data point being flown, select the [Mark] corresponding button, to record data. After [Mark] is pressed, the pilot should keep the aircraft stabilized for the duration of the 'Sample Length' setting as the software will monitor the control positions for movement during this period.
4. Select or enter a handling quality rating in the 'HQR' field as desired.
5. Repeat steps 2 – 4 for each desired data point.
6. When the all of the desired data points have been collected, select the [Done] button to close the Data Collection window and open the Data Presentation window.

Program Notes:

The default values for the speed/power settings can be changed to any values of interest. If the speed/power setting is changed from the default value, it must be set before the [Mark] button is pressed for that data point.

After a data point is marked, the [Mark] button mnemonic changes to [Clear]. If the user wishes to discard the data collected for that data point the [Clear] button can be pressed. The data for that speed will be discarded, the button mnemonic changes back to [Mark], and new data can be acquired.

In figure 6-3 there are eighteen different speed data points available for data collection, in figure 6-4 there are twelve power data points. It is not necessary to complete all data entries, the user can choose or change any of these values as desired.

Selecting the [Cancel] button discards all collected data and closes the Data Collection window and returns program operation to the main window.

Selecting the [Done] button closes the Data Collection window and opens the Data Presentation window.

Data Reduction and Presentation

The Level Flight and Climbing & Descending Trimmed Flight Control Positions Data Presentation windows are shown in figures 6-5 and 6-6. These windows display the pertinent attitude and control positions for the recorded data. The information on the lower right can be filled in by the user to save relevant test condition information with the data. Once this information has been entered by the user it will be automatically reentered each time a data presentation window is opened. Only the information that has changed since the last test performed needs to be updated.

Analysis of this information is primarily based on the slope of the cyclic control position plot. In the example data presented in figure 6-5 the longitudinal cyclic stick position plot indicates increasing forward stick displacement with increasing forward airspeed. This indication signifies that the control movements are in the usual sense and implies positive low speed longitudinal static stability.

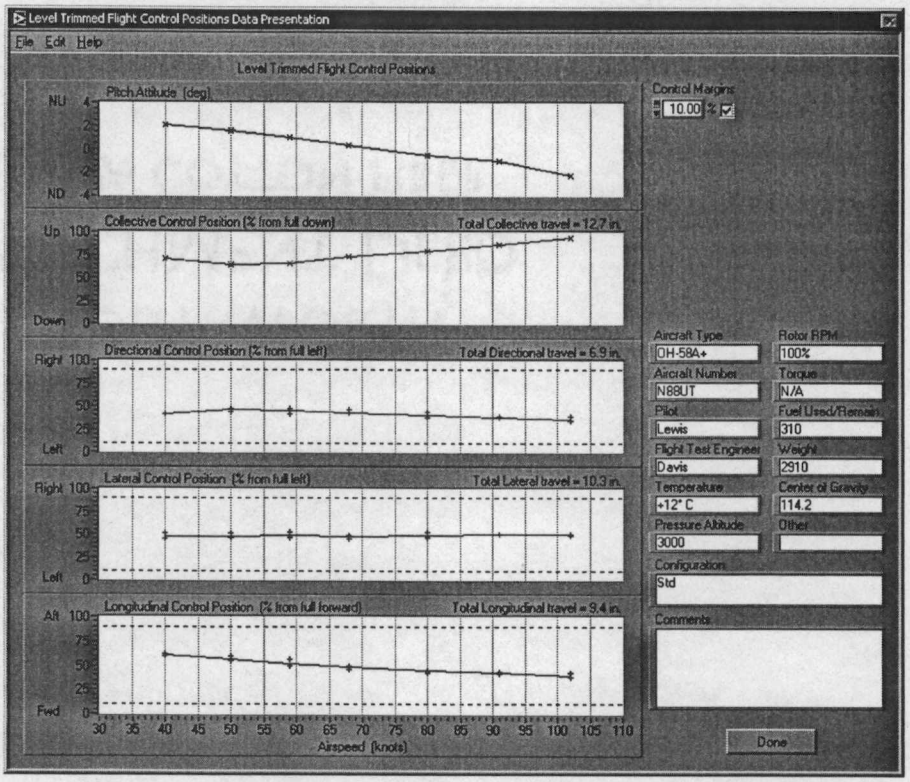


Figure 6-5 Level Trimmed Flight Control Positions Data Presentation Window

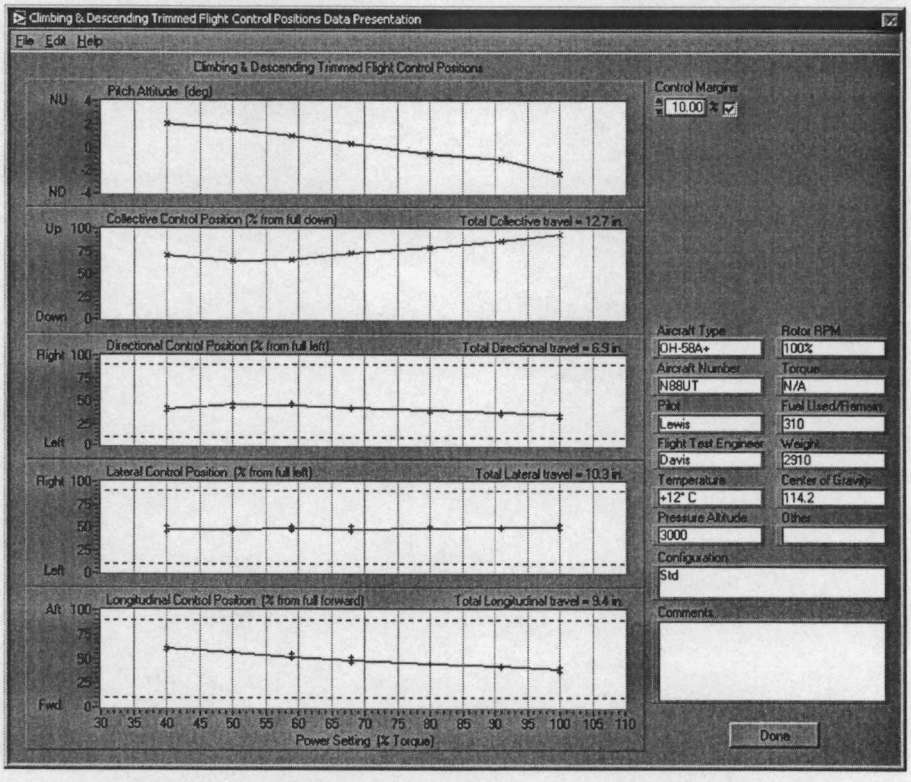


Figure 6-6 Climbing/Descending Trimmed Flight Control Positions Data Presentation Window

The following questions should be answered based on the data presented in these windows:

1. Were the control movements required in the usual sense (forward cyclic for increasing airspeed)?
2. Were the control margins reached or exceeded?
3. Was there objectionable coupling between pitch, roll or yaw?
4. Were the control position changes with power or airspeed excessive?
5. Were the aircraft attitude changes with power or airspeed excessive?

Longitudinal 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. Since the aircraft's forces/moments can not be directly measured in flight, the intent of this test is to measure the control displacements necessary to balance the pitching moments caused by airspeed variations from trim. [12] Positive static stability is indicated by increasing forward cyclic stick displacement with increasing airspeed and increasing aft stick with decreasing airspeeds.

Although the direction of cyclic displacement is an indicator of the sign of the static stability, neither the linear displacement nor the percentage of total travel, for a given airspeed change, is a measure of the strength of the stability. The amount of control displacement is a function of the gearing of the physical flight control linkages.

Test Method

This test is conducted by establishing a wings level, steady, unaccelerated, trimmed airspeed. With the collective control fixed and rotor speed constant, the airspeed is varied incrementally above and below the trim speed. To attain the off-trim speeds stabilized climbs or descents will result. The first airspeed increment above and below trim should be 2 – 3 knots, then increments of 5 knots are typically used. These airspeed increments are arbitrary. [12] It is much more important to be in stabilized flight than to establish the exact increment precisely. During this test an altitude band should be maintained; typically ±1000 feet is used.

Pilot Procedures

1. Establish a stabilized, wings level, unaccelerated, airspeed with the control forces trimmed to zero. Record the initial trim conditions.
2. Stabilize on each airspeed increment with the collective fixed at the trim airspeed position, record data.
3. Repeat step 2 until the desired airspeed range is covered.

Program Operation

The Longitudinal Static Stability Data Collection window (figure 6-7) is used to record the accumulated flight test data. To access this window select Acquire Data | Forward Flight Longitudinal Tests | Static Stability from the menu on the main window. The top half of this window displays a 30 second time history of the position of each flight control. The lower half of the window is used to collect data for each desired off-trim condition.

Data Acquisition Procedures

1. Enter the length of time that the control positions will be monitored on each data point in the 'Sample Length' field. The 'Sample Length' setting can be set from 0 seconds

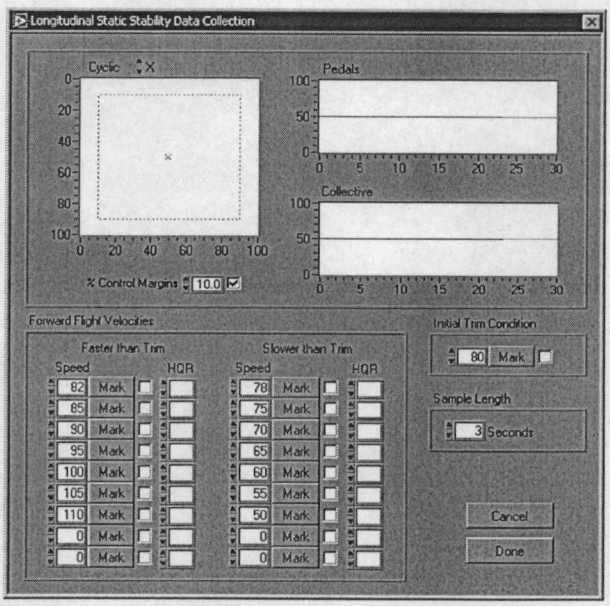


Figure 6-7 Longitudinal Static Stability Data Collection Window

(instant sample) to 10 seconds, the default is 3 seconds. This setting can be changed for subsequent data points as desired.

2. Set the trim speed in the 'Initial Trim Condition' speed field.
3. With the aircraft stabilized on the trim speed and control forces trimmed to zero, select the [Mark] button associated with the 'Initial Trim Condition'. After [Mark] is pressed, the pilot should keep the aircraft stabilized for the duration of the Sample Length setting as the software will monitor the control positions for movement during this period.
4. Restabilize the aircraft on the off-trim speed without retrimming the control forces.
5. With the aircraft stabilized on the off-trim data point and the 'Speed' set to the data point being flown, select the [Mark] button corresponding to the speed being flown, to record data. Again the data will be sampled for the duration set in the 'Sample Length' setting.
6. Repeat steps 4 – 5 for each desired data point.
7. When the all of the desired data points have been collected, select the [Done] button to close the Longitudinal Static Stability Data Collection window and open the Longitudinal Static Stability Data Presentation window.

Program Notes:

The default values for the initial trim speed and the off-trim speeds can be changed to any values of interest. If these values are changed from the default values, they must be set before the [Mark] button is pressed for that data point.

After a data point is marked, the [Mark] button mnemonic changes to [Clear]. If the user wishes to discard the data collected for that data point the [Clear] button can be pressed. The data for that speed will be discarded, the button mnemonic changes back to [Mark], and new data can be acquired.

Selecting the [Cancel] button discards all collected data and closes the Longitudinal Static Stability Data Collection window and returns function to the main window.

Selecting the [Done] button closes the Longitudinal Static Stability Data Collection window and opens the Longitudinal Static Stability Data Presentation window.

Data Reduction and Presentation

An example of the Longitudinal Static Stability Data Presentation window is presented as figure 6-8. Analysis of this information is primarily based on the slope of the cyclic control position plot. A longitudinal cyclic stick position plot indicates increasing forward stick displacement with increasing airspeed or increasing aft stick displacement with decreasing airspeed is indicative of positive longitudinal static stability. Graphically this would be plotted as a negative sloping line for positive static stability. A summary of the static stability graphical indications are presented in table 6-4.

The information on the lower right corner of the data presentation window can be filled in by the user to save relevant test condition information with the data. Once this information has been entered by the user it will be automatically reentered each time a data presentation window is opened. Only the information that has changed since the last test performed needs to be updated.

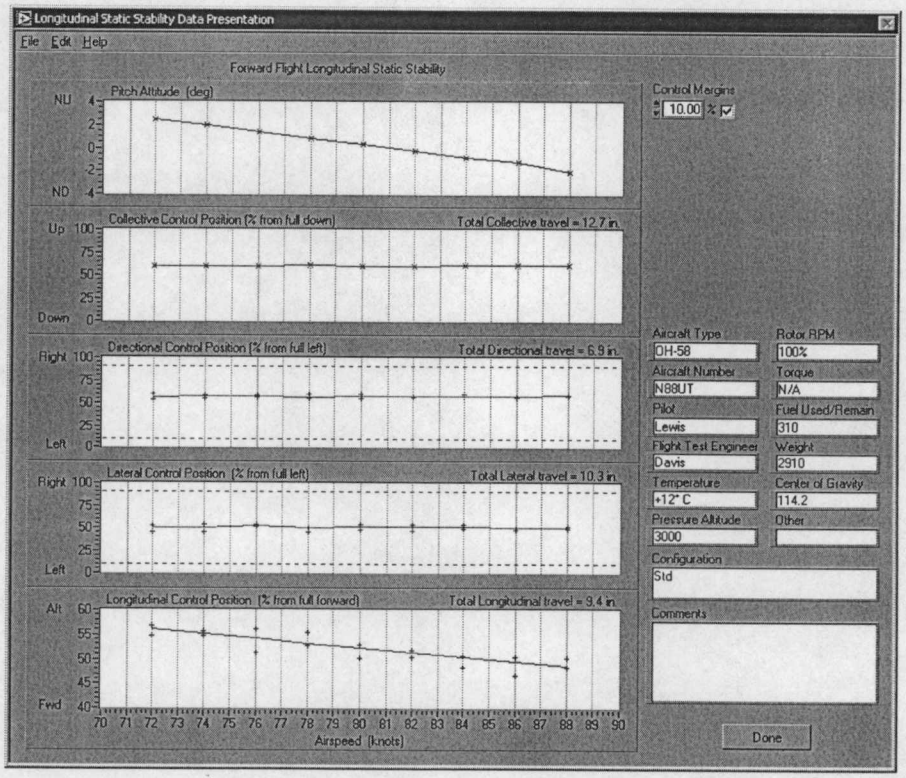


Figure 6-8 Longitudinal Static Stability Data Presentation Window

Table 6-4 Summary of Longitudinal Static Stability Graphing Indications

Aircraft Motion	What control input was required?	Slope of the control position curve	Static Stability
Increasing Forward Speed	Increasing Forward Cyclic	Negative	Positive
	Neutral	Zero	Neutral
	Increasing Aft Cyclic	Positive	Negative
Decreasing Forward Speed	Increasing Aft Cyclic	Negative	Positive
	Neutral	Zero	Neutral
	Increasing Forward Cyclic	Positive	Negative

Analysis of longitudinal static stability data should also consider the following questions:

1. Were control margins approached or exceeded?
2. Is there a speed band about trim within which there are no displacement (or force) cues to indicate the off-trim condition?
3. Are the aircraft characteristics suitable for the intended mission?

Maneuver Stability

The maneuver stability test is a variation on the longitudinal static 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 due to change in load factor is determined from the variation of the longitudinal control position with normal acceleration obtained during stabilized conditions with constant airspeed and collective. [12]

Three test techniques are used to create changes in the aircraft normal load factor: steady turns, symmetrical pull-ups, and symmetrical pushovers.

Test Method

The steady turn test technique is conducted by first attaining steady straight and level flight at a constant speed and control forces trimmed to zero. The aircraft is banked to the first desired bank angle with the collective fixed, while the pitch attitude is adjusted as necessary to maintain

airspeed. Control forces should not be retrimmed to zero at each bank angle. Data are recorded and the next bank angle is established. This process is continued until the maximum load factor or predetermined test limit is reached.

The symmetrical pull-up technique is performed from trimmed level flight. First the pitch attitude is increased to allow the airspeed to decrease, then the pitch attitude is decreased to allow the aircraft to accelerate. As the speed approaches the original trim speed the first desired aft cyclic increment is rapidly applied and held. The goal is to attain a level pitch attitude at the moment the airspeed returns to the trim speed. At that moment data are recorded. The process is repeated with incrementally larger magnitude inputs to achieve increased load factors until the maximum load factor or predetermined test limit is reached. This test technique requires considerable practice and proper timing to yield satisfactory results. Caution must be exercised with this method because an aircraft with neutral or unstable maneuvering stability characteristics will continue to increase in g when longitudinal control is held displaced from trim.

The symmetrical pushover technique is similar to the pull-up technique except that the cyclic control input is in the opposite direction. First, the airspeed in increased, then the aircraft is pitched up, then just prior to reaching the trim speed, a forward longitudinal cyclic increment is applied. As the speed reached the trim speed the aircraft should be in a level attitude and data are recorded. Additional caution must be exercised when performing this technique not to exceed the test g envelope, as many helicopters exhibit dangerous flight characteristics at low or negative load factors.

Pilot Procedures (Steady Turns Technique)

1. Establish a steady, coordinated, level flight at the desired test airspeed.
2. Trim all control forces to zero.
3. Fix the collective position.
4. Record the initial trim conditions.
5. Establish the first bank angle increment while the airspeed is maintained by adjusting the pitch attitude, do not adjust the collective or retrim the control forces.
6. Record data when the bank angle is established and the airspeed is stabilized.
7. Repeat steps 5 and 6 for each bank angle increment.

Pilot Procedures (Symmetrical Pull-up Technique)

1. Establish a steady, coordinated, level flight at the desired test airspeed.
2. Trim all control forces to zero.
3. Fix the collective position.
4. Record the initial trim conditions.
5. If a control fixture is used, establish the fixture increment and position the fixture as necessary.
6. Using cyclic control only decelerate below the trim airspeed.
7. Lower the nose of the aircraft below the trim attitude to accelerate toward the trim airspeed.
8. Prior to reaching the trim speed, rapidly displace the longitudinal control and hold against the fixture.
9. Record data at the moment where the airspeed stabilizes the trim speed, the aircraft attitude must also be at the trim attitude when data are recorded.
10. Repeat steps 5 – 9 for each data point.

Pilot Procedures (Symmetrical Pushover Technique)

1. Establish a steady, coordinated, level flight at the desired test airspeed.
2. Trim all control forces to zero.
3. Fix the collective position.
4. Record the initial trim conditions.
5. If a control fixture is used, establish the fixture increment and position the fixture as necessary.
6. Using cyclic control only accelerate above the trim airspeed.
7. Raise the nose of the aircraft above the trim attitude to decelerate toward the trim airspeed.
8. Prior to reaching the trim speed, rapidly displace the longitudinal control and hold against the fixture.
9. Record data at the moment where the airspeed reaches the trim speed, the aircraft attitude must also be at the trim attitude when data are recorded.

10. Repeat steps 5 – 9 for each data point.

Program Operation

Maneuver stability flight testing is accomplished by selecting Acquire Data | Forward Flight Longitudinal Tests | Maneuvering Stability from the menu on the main window. The Forward Flight Maneuvering Stability Data Collection window (figure 6-9) is used to accumulate the necessary flight test data. The top half of this window displays a 30 second time history of the position of each flight control. The lower half of the window is used to collect data for each desired test point.

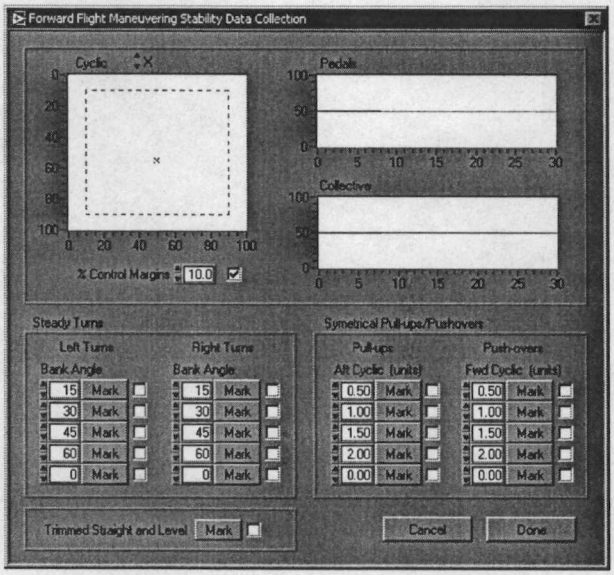


Figure 6-9 Maneuver Stability Data Collection Window

Data Acquisition Procedures

1. With the aircraft established in trimmed, steady, coordinated, level flight at the desired test airspeed select the [Mark] button associated with 'Trimmed Straight and Level' to record the initial trim condition..
2. Perform the Pilot Procedures outlined above to maneuver the aircraft into the proper position to record data.
3. Press the [Mark] button associated with the data point the pilot has maneuvered the aircraft for.
4. Repeat steps 2 and 3 for each desired data point.

5. When the all of the desired data points have been collected, select the [Done] button to close the Low Speed Trimmed Flight Control Positions Data Collection Window and open the Low Speed Trimmed Flight Control Positions Data Presentation Window.

Program Notes:

The default values for the bank angle and cyclic input can be changed to any values of interest. If a value is changed from the default value, it must be set before the [Mark] button is pressed for that test point.

After a data point is marked, the [Mark] button mnemonic changes to [Clear]. If the user wishes to discard the data collected for that data point the [Clear] button can be pressed. The data for that point will be discarded, the button mnemonic changes back to [Mark], and new data can be acquired.

Each test technique has five available test data points, four have preset default test values. The fifth is available for the user to set as desired. Although there are five available data points for test technique it is not necessary to complete all entries.

Selecting the [Cancel] button discards all collected data and closes the Forward Flight Maneuvering Stability Data Collection window and returns to the main window.

Selecting the [Done] button closes the Forward Flight Maneuvering Stability Data Collection Window and opens the Forward Flight Maneuvering Stability Data Presentation window.

Data Reduction and Presentation

The Forward Flight Maneuvering Stability Data Presentation window is shown in figure 6-10. This window displays the pertinent attitude and control positions for the recorded data. Analysis of this information is primarily based on the slope of the longitudinal cyclic control position plot. In the example data presented in figure 6-10 the longitudinal cyclic stick position plot indicates increasing aft stick displacement with increasing load factor. This plot also indicates increasing forward stick displacement with decreasing load factor in the pushover maneuvers. Both of these indications imply positive maneuver stability. A reversal of the slope of this curve would indicate an instability in the aircraft's maneuvering stability characteristics. This would occur in an aircraft that tends to 'dig-in' under increasing positive load factor.

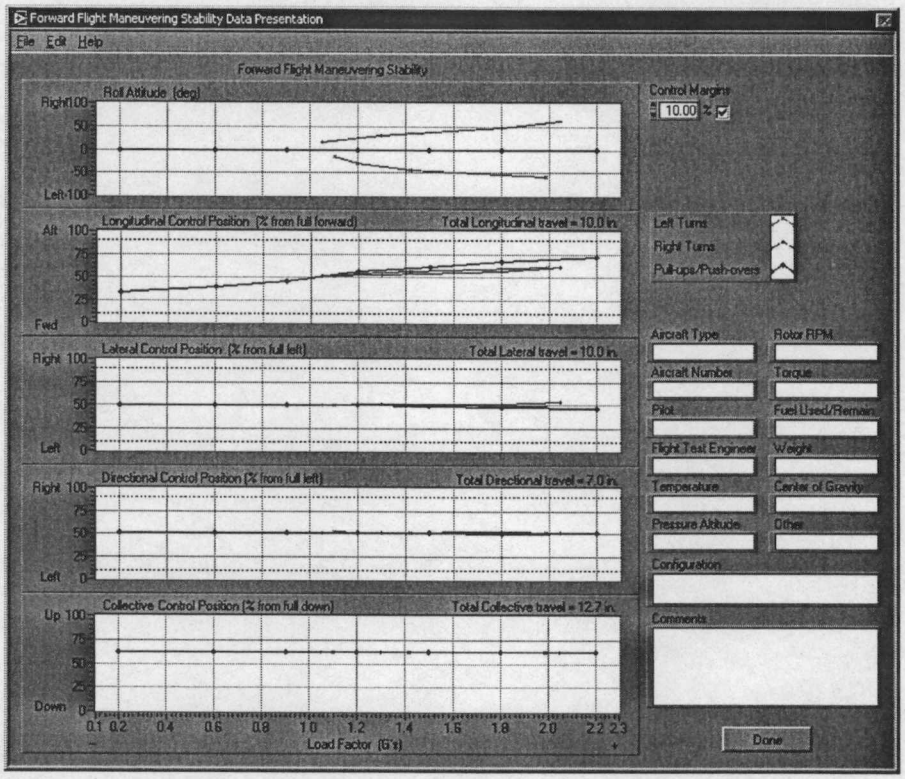


Figure 6-10 Maneuver Stability Data Presentation Window

The information on the lower right can be filled in by the user to save relevant test condition information with the data. Once this information has been entered by the user it will be automatically reentered each time a data presentation window is opened. Then only the information that has changed since the last test performed needs to be updated.

The following questions should be answered based on the data presented in this window:

1. Were the control movements required in the positive maneuver stability sense (aft cyclic for increasing load factor, forward cyclic for decreasing load factor)?
2. Did the longitudinal control displacements provide a physical cue to the pilot of normal acceleration and turn rate?
3. Was there any tendency to exceed the g envelope of the aircraft?

Long Term Dynamic Stability

The purpose of this test is to evaluate the long term longitudinal response of the helicopter as a function of time following a disturbance. The long term dynamic response is generally a low frequency lightly damped oscillation that is typically damped by the pilot or by the AFCS. The

intent of this test is to determine the longitudinal response; therefore, small off-axis control inputs may be applied to maintain the initial trim roll and yaw attitude. If the off-axis response is large the controls should be fixed and the full response documented.

Test Method

This test is conducted by establishing steady, unaccelerated, level flight with all control forces trimmed to zero. The flight controls are fixed while the aircraft is permitted to respond to natural external stimuli and the aircraft response is recorded. This is an unsatisfactory technique if no response occurs, however this indicates a desirable aircraft characteristic. [12] If no natural excitation occurs, an artificial stimulus must be applied to initiate a response. Longitudinal cyclic input is applied to accelerate or decelerate the aircraft approximately 10 knots from the trim speed, then the controls are returned and fixed in the trim positions and the aircraft response is recorded. If the response is moderately divergent the artificial stimulus must be small enough that the response can be adequately documented without exceeding a test limit. For both techniques it may be necessary to keep the aircraft level (laterally) by applying small lateral control inputs. This should only be done if the off axis response is small. If the cross-coupled response is large, the coupled response should also be documented.

Pilot Procedures

1. Establish steady, unaccelerated, level flight with all control forces trimmed to zero.
2. Fix the flight controls in the trimmed positions and monitor and record the aircraft response to any external stimuli.

Or, if no natural response is obtained;

3 Introduce a longitudinal cyclic input to accelerate or decelerate the aircraft approximately 10 knots from the trim speed and immediately return the flight controls to the trim positions as accurately as possible.
4. Record the aircraft response.

Program Operation

The Longitudinal Long Term Dynamic Stability Data Collection window (figure 6-11) is displayed during the forward flight longitudinal long term dynamic stability flight testing. To access this window select Acquire Data | Forward Flight Longitudinal Tests | Long Term Dynamic Stability from the menu on the main window. Since the data collected for this test is sampled over a period of time the user can select the rate of acquisition of the data samples. This is done with the 'Sample Rate' knob at the top left of the window. This setting adjusts the number of samples per second that will be acquired in the range of 1 sample per second to 100 samples per second. The time interval between each data sample is displayed below the 'Sample Rate' knob. In the example shown in figure 6-11 the sample interval is 100 milliseconds. The default setting for the sample rate is 10 samples per second and is generally suitable for this test.

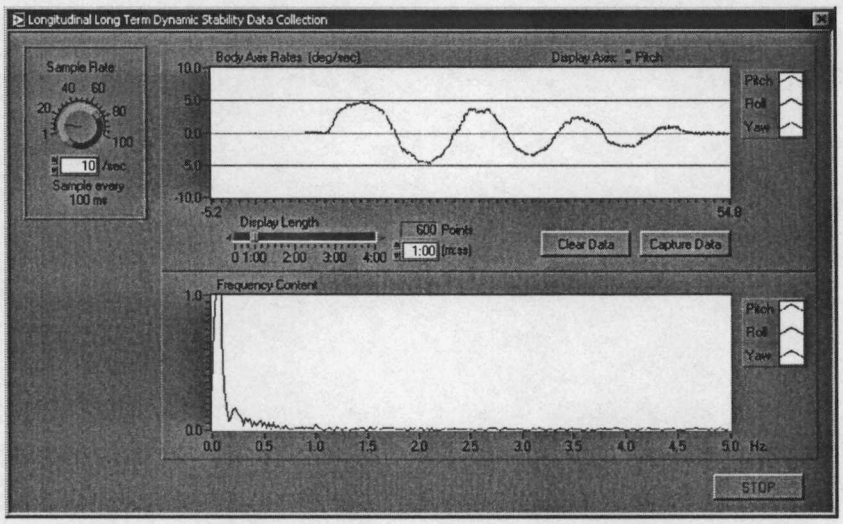


Figure 6-11 Longitudinal Long Term Dynamic Stability Data Collection Window

Two continuously updating graphs are displayed in this window. The top graph is a time history of the longitudinal pitch rate in degrees per second, the bottom graph displays the frequency content present for the body axis rates. The user may select specific axes to monitor by selecting the desired axes with the 'Display Axis' selection control. Since this test is primarily concerned with pitch rates, it is recommended that this selection remain on 'Pitch'. The user can also set the length of time that is displayed on the time history plot, this is done by adjusting the 'Display Length' slider control. The valid range for this control, on this test, is from thirty

seconds to four minutes. The default length for this test is 90 seconds. The number of data points held in memory can be calculated by multiplying the sample rate times the display length. Data that is older than the display length is 'expired' and can not be recovered. The display length must be long enough to capture the characteristic motion of the aircraft

The bottom graph displays the frequency information for the axis displayed in the top graph. The frequency information is determined by a Fourier transformation of the time domain data. The range of frequencies that can be determined are directly related to the sample rate. The highest frequency that can be determined is one-half of the sample rate in accordance with the Nyquist Theorem.

The long term dynamic stability test procedures are conducted while this window is displayed, then before the data on the time history plot expires, the [Capture Data] button must be pressed. When the [Capture Data] button is pressed the time history data from the aircraft body attitude, body rates, accelerations, and stick positions are sent to the data presentation window.

Data Acquisition Procedures

1. Set the 'Sample Rate' as desired.
2. Set the 'Display Length' as desired.
3. Perform the Pilot Procedures above to stimulate the long period response.
4. When the desired response data is shown in the time history graph select the [Capture Data] button to close the Low Speed Long Term Dynamic Stability Data Collection window and open the Low Speed Long Term Dynamic Stability Data Presentation window.

Program Notes:

If the data on the time history graph is allowed to scroll off the window or 'expire', the data is lost forever. To prevent this from happening increase the 'Display Length' setting. Setting the 'Display Length' to an overly long setting may cause the data presentation window to display unnecessary data. The default settings for the sample rate and the display length are normally

sufficient to properly document most typical helicopter responses but some experimentation with these settings may be necessary.

The 'Sample Rate' setting should not be changed during a flight test. All data for a specific test must be acquired at a constant sampling rate or the results will be inaccurate.

The [Clear Data] button clears both the time history and the frequency content graphs. It is not necessary to clear data before beginning an attempt to acquire flight test data, but it may be desirable to clear extraneous data before a "for-record test" is performed. This is especially recommended if the display length is set to a long length and there are several minutes of unnecessary data preceding the flight test.

Data Reduction and Presentation

The time history data of the aircraft body attitudes, body rates, accelerations, and stick positions are available for analysis and presentation in the Longitudinal Long Term Dynamic Stability Data Presentation window (figure 6-12). The left side of this window presents the relevant time histories, while the right side contains the analysis tools, engineering data results, and general test information. The time histories display the scaled raw data from which the engineering data results are calculated. The time histories that are displayed for this test are pitch rate, pitch attitude, airspeed, normal acceleration, and longitudinal stick position, although the user may select other plots as desired.

When this window is initially opened, the 'Engineering Data Results' section is blank. This analysis can be done when the data is collected or the data can be saved and the analysis can be performed at a later time. The data analysis is done by clicking the [Analyze Response] button. The process used to determine the engineering data is detailed in the Second Order Dynamic Response Data Analysis section in Chapter 4. The results of the analysis are returned and filled into the 'Engineering Data Results' section of the window.

The information on the lower right can be filled in by the user to save relevant test condition information with the data. Once this information has been entered by the user it will be automatically reentered each time a data presentation window is opened. Only the information that has changed since the last test performed needs to be updated.

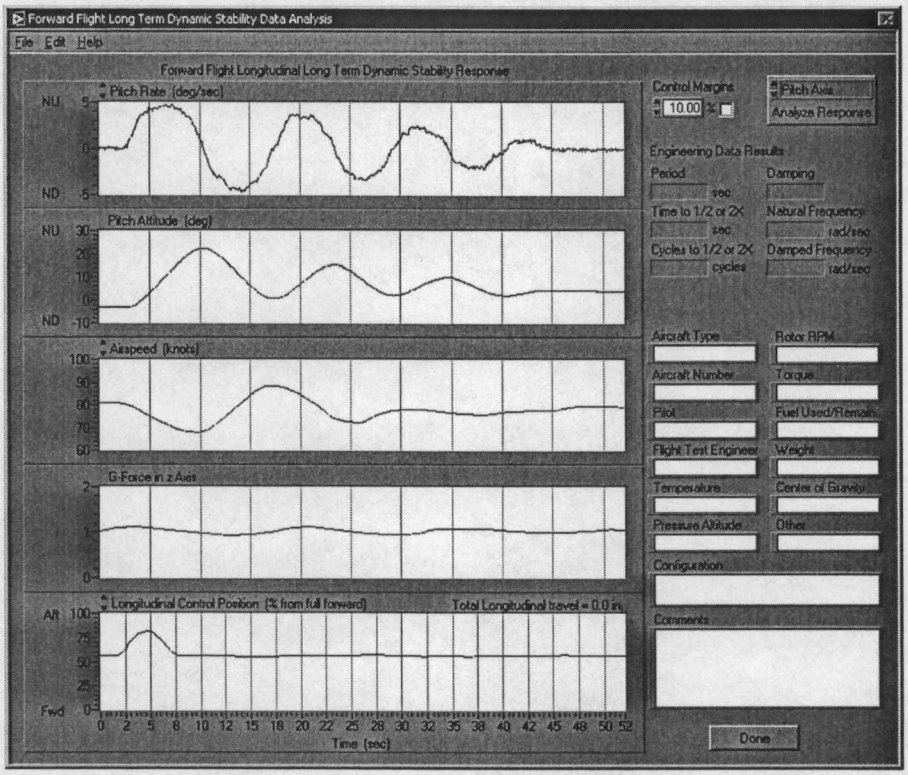


Figure 6-12 Longitudinal Long Term Dynamic Stability Data Presentation Window

Data Reduction

The engineering data results for the characteristic motion about the axis of interest can be determined by the following process:

1. Select the 'Pitch Axis' from the selection control at the top of the [Analyze Response] button.
2. Click the [Analyze Response] button.
3. Complete the procedures outlined in the Second Order Dynamic Response Data Analysis section in Chapter 4.

The following characteristic parameters are returned from this analysis:

- Period of the Motion
- Time to half or double Amplitude
- Cycles to half or double Amplitude
- Damping Ratio
- Natural Frequency
- Damped Natural Frequency

4. Once the engineering data parameters have been determined the user may opt to view the data on a complex plane by clicking the [Show Root Locus] button. For more information on the Root Locus plot see the Display of Characteristic Roots on a Complex Plane section in Chapter 4.

Analysis of longitudinal long term dynamic stability data should also address the following questions:

1. How easy was it to excite the long term response?
2. How easy was it for the pilot to suppress the long term response?
3. Are the aircraft characteristics suitable for the intended mission?

Short Term Dynamic Stability

The purpose of this test is to determine the characteristics of the short term longitudinal response due to longitudinal control inputs and to evaluate the pilot's ability to make small attitude changes. 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 under the classification of gust response. [12]

Test Method

This test is conducted by establishing steady, unaccelerated, level flight with all control forces trimmed to zero. A longitudinal control doublet input is used to excite the short term response. To determine the frequency of the doublet application a frequency sweep of the longitudinal control should be conducted. This is accomplished 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. [12]

Pilot Procedures

1. Establish steady, unaccelerated, level flight with all control forces trimmed to zero.
2. Conduct a longitudinal control input frequency sweep to determine the short term frequency by observing the maximum pitch attitude amplitude.
3. Make a doublet longitudinal control input at the determined short term frequency and record the aircraft response.

Program Operation

The Longitudinal Short Term Dynamic Stability Data Collection window (figure 6-13) is displayed during the forward flight longitudinal short term dynamic stability flight testing. To access this window select Acquire Data | Forward Flight Longitudinal Tests | Short Term Dynamic Stability from the menu on the main window. Since the data collected for this test is sampled over a period of time the user can select the rate of acquisition of the data samples. This is done with the 'Sample Rate' knob at the top left of the window. This setting adjusts the number of samples per second that will be acquired in the range of 1 sample per second to 100 samples per second. The time interval between each data sample is displayed below the 'Sample Rate' knob. In the example shown in figure 6-13 the sample interval is 20 milliseconds. The default setting for the sample rate is 50 samples per second and is generally suitable for this test.

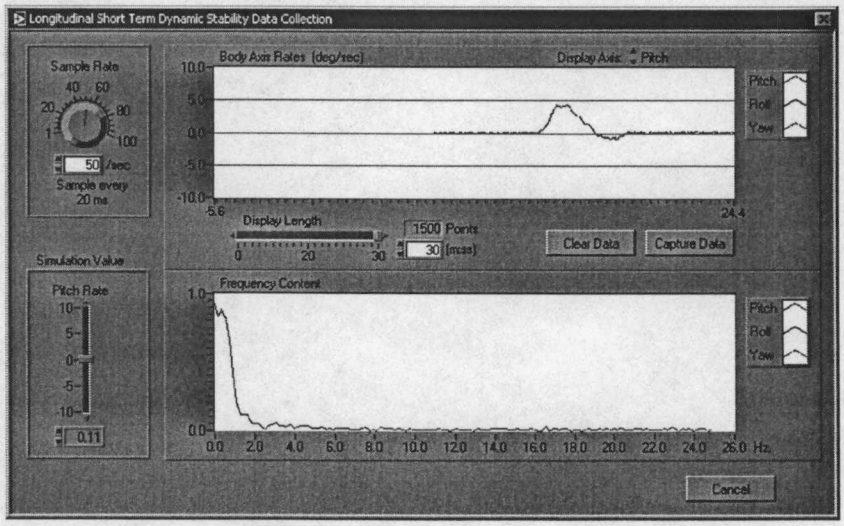


Figure 6-13 Longitudinal Short Term Dynamic Stability Data Collection Window

Two continuously updating graphs are displayed in this window. The top graph is a time history of the longitudinal control position as a percent of total travel, the bottom graph displays time history of the longitudinal pitch rate in degrees per second. The user may select specific axes to monitor by selecting the desired axes with the 'Display Axis' selection control. Since this test is primarily concerned with pitch rates, it is recommended that this selection remain on 'Pitch' for this test. The user can also set the length of time that is displayed on the time history plots, this is done by adjusting the 'Display Length' slider control. The valid range for this control, on this test, is from fifteen to forty-five seconds. The default length for this test is 30 seconds. The number of data points held in memory can be calculated by multiplying the sample rate times the display length for example in figure 6-13, $50 \frac{\text{samples}}{\text{sec}} \times 30 \text{ sec} = 1500$ samples. Data that is older than the display length is 'expired' and can not be recovered. The display length must be long enough to capture the characteristic motion of the aircraft without the expiring.

The longitudinal short term dynamic stability test procedures are conducted while this window is displayed, then before the data on the time history plot expires, the [Capture Data] button must be pressed. When the [Capture Data] button is pressed the time history data from the aircraft body attitude, body rates, accelerations, and stick positions are sent to the data presentation window.

Data Acquisition Procedures

1. Set the 'Sample Rate' as desired.
2. Set the 'Display Length' as desired.
3. Conduct a longitudinal control input frequency sweep to determine the short term frequency. Once the short term frequency is determined continue to oscillate the longitudinal control for at least two cycles at the determined frequency.
4. Make a doublet input at the determined short term frequency.

5. Select the [Capture Data] button to close the Low Speed Short Term Dynamic Stability Data Collection window and open the Low Speed Short Term Dynamic Stability Data Presentation window.

Program Notes:

The data necessary to determine the characteristics of the longitudinal short term dynamic response is gathered from two separate test steps. First, after the frequency sweep the frequency is determined. Then the response is measured following the doublet input. In order for the software to be able to extract the necessary information, at least two cycles of the longitudinal control input at the short term frequency and the response to the doublet input must be displayed on the time history graphs. The [Capture Data] button should be pressed as soon as the response to the doublet input subsides. If the data on the time history graph is allowed to scroll off the window or 'expire', the data is lost forever. To prevent this from happening increase the 'Display Length' setting. Setting the 'Display Length' to an overly long setting will cause the data presentation window to display unnecessary data. The default settings for the sample rate and the display length are normally sufficient to properly document most typical helicopter responses but some experimentation with these settings may be necessary.

The 'Sample Rate' setting should not be changed during a flight test. All data for a specific test must be acquired at a constant sampling rate or the results will be inaccurate.

The [Clear Data] button clears both the time history and the frequency content graphs. It is not necessary to clear data before beginning an attempt to acquire flight test data, but it may be desirable to clear extraneous data before a "for-record test" is performed. This is especially recommended if the display length is set to a long length and there are several minutes of unnecessary data preceding the test.

Data Reduction and Presentation

The time history data of the aircraft body attitudes, body rates, accelerations, and stick positions are available for analysis and presentation in the Longitudinal Short Term Dynamic Stability Data Presentation window (figure 6-14). The left side of this window presents the relevant time histories, while the right side contains the analysis tools, engineering data results,

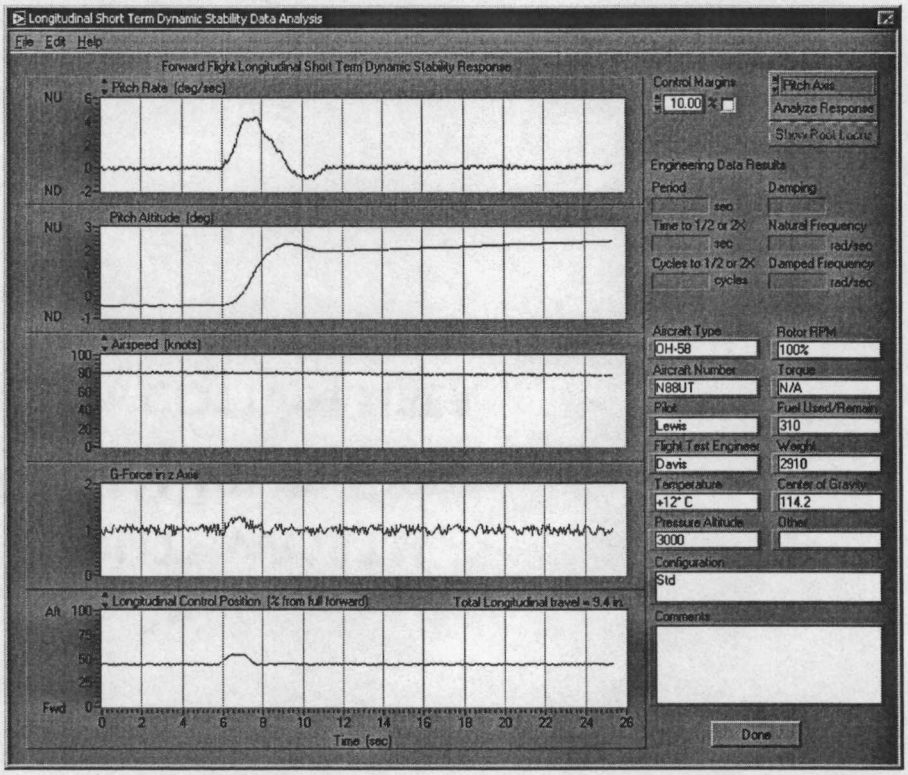


Figure 6-14 Longitudinal Short Term Dynamic Stability Data Presentation Window

and general test information. The time histories display the scaled raw data from which the engineering data results are calculated. The time histories that are displayed for this test are pitch rate, pitch attitude, airspeed, normal acceleration, and longitudinal stick position, although the user may select other plots as desired.

When this window is initially opened, the 'Engineering Data Results' section is blank. This analysis can be done when the data is collected or the data can be saved and the analysis can be performed at a later time. The data analysis is accomplished by clicking the [Analyze Response] button. The process used to determine the engineering data is discussed in the Second Order Dynamic Response Data Analysis section in Chapter 4. The results of the analysis are returned and filled into the 'Engineering Data Results' section of the window.

The information on the lower right can be filled in by the user to save relevant test condition information with the data. Once this information has been entered by the user it will be automatically reentered each time a data presentation window is opened. Only the information that has changed since the last test performed needs to be updated.

Data Reduction

The engineering data results for the characteristic motion about the axis of interest can be determined by the following process:

1. Select the 'Pitch Axis' from the selection control at the top of the [Analyze Response] button.
2. Click the [Analyze Response] button.
3. Complete the procedures outlined in the Second Order Dynamic Response Data Analysis section in Chapter 4.

The following characteristic parameters are returned from this analysis:

- Period of the Motion
- Time to half or double Amplitude
- Cycles to half or double Amplitude
- Damping Ratio
- Natural Frequency
- Damped Natural Frequency

4. Once the engineering data parameters have been determined the user may opt to view the data on a complex plane by clicking the [Show Root Locus] button. For more information on the Root Locus plot see the Display of Characteristic Roots on a Complex Plane section in Chapter 4.

Analysis of longitudinal short term dynamic stability data should also address the following questions:

1. Can the pilot make precise pitch attitude changes?
2 Are the aircraft characteristics suitable for the intended mission?

Control Response

The purpose of this test is to evaluate the aircraft response to a flight control input. The aircraft response is determined by the control sensitivity, damping, and response lags in the examined control axis. Control sensitivity is the acceleration per unit control input. Damping is the acceleration per unit of velocity. The response lags are the time from the control input to the aircraft response. The response lags are a consequence of the flight control system characteristics and rotor system type.

Several characteristics contributing to desirable response qualities are listed below. [12]

1. Pilots want a vehicle responsive enough to achieve some level of attitude change within a certain time after a control input (without being overly responsive).
2. Pilots want a reasonably steady state rate for a given input size.
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.

Each of these qualities are directly related to control sensitivity, damping, or both. The resulting motion due to a given control input can be characterized as a first order system.

Test Method

This test is conducted by establishing steady, unaccelerated, level flight with all control forces trimmed to zero. A control step input is applied, and the aircraft response is monitored. Control fixtures are generally used to restrain control inputs and an incremental buildup technique should be used to build up to the desired control input magnitude. The control input should be quick and as precise as possible. If a fixture is used, the control should be firmly held against the fixture but not forcefully. Recovery is initiated when a steady state pitch rate is achieved or a predetermined test flight limit is reached.

The data reduction process uses the steady state rate to determine aircraft characteristics, but does not critically depend on discerning a steady state rate. There are alternate analysis methods included in the program that do not require a steady state to determine the desired aircraft characteristics.

Pilot Procedures

1. Establish steady, unaccelerated, level flight with all control forces trimmed to zero.
2. Set the control fixture for the desired displacement and position it for the desired input direction. The control input size and direction are verbally and visually verified between the pilot and copilot/flight test engineer. Start with small inputs and increase in size incrementally until desired input is achieved.

3. Apply the control input against the fixture and hold the controls fixed. Recover when a steady state pitch rate is obtained or a predetermined test flight limit is reached.
4. Record the aircraft response.
5. When the maneuver is complete, announce the recovery so the control fixture can be quickly removed.
6. Repeat steps 1 – 5 for increasing control displacements until the desired deflection is attained.
7. Qualitatively assess the predictability of the control response.

Program Operation

The Control Response Data Collection window (figure 6-15) is displayed during forward flight control response flight testing. To access this window select Acquire Data | Forward Flight Longitudinal Tests | Pitch Control Response from the menu on the main window. Since the data collected for this test is sampled over a period of time the user can select the rate of acquisition of the data samples. This is done with the 'Sample Rate' knob at the top left of the window. This setting adjusts the number of samples per second that will be acquired in the range of 1 sample per second to 100 samples per second. The time interval between each data sample is displayed below the 'Sample Rate' knob. The default setting for the sample rate is 20 samples per second and is generally suitable for this test.

Three continuously updating time history plots are displayed in this window. The length of time that is displayed on these graphs can be adjusted with the 'Display Length' slider control at the bottom of the graphs. The valid range for this control, on this test, is from ten seconds to two minutes The default length for this test is 30 seconds. Also displayed with the 'Display Length' slider control is the 'Points' indicator which displays the number of data samples that are held in memory. This value is calculated by multiplying the sample rate (in samples per second) times the display length (in seconds). Data that is older than the display length is 'expired' and can not be recovered The display length must be long enough to capture the characteristic motion of the aircraft without expiring.

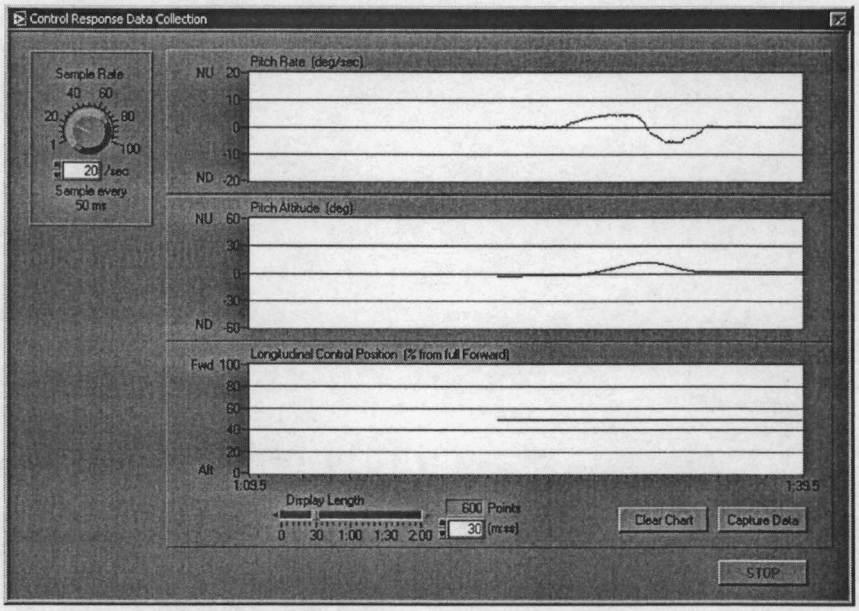


Figure 6-15 Longitudinal Control Response Data Collection Window

The top graph is a time history of the pitch rate in degrees per second. The data presented on this graph are the basis for determining the engineering data results. Ideally the data presented on this graph will attain a steady state rate during the performance of the particular flight test. The middle graph displays the pitch attitude. The bottom graph shows the longitudinal control position versus time.

The control response test procedures are conducted while this window is displayed, then before the data on the time history plots expire, the [Capture Data] button must be pressed. When the [Capture Data] button is pressed the time history data from the aircraft body attitude, body rates, accelerations, and stick positions are sent to the Control Response Data Presentation window.

Data Acquisition Procedures

1. Set the 'Sample Rate' as desired.
2. Set the 'Display Length' as desired.
3. Perform the Pilot Procedures above to test the control response.
4. When the desired response data is shown in the time history graph select the [Capture Data] button to close the Control Response Data Collection window and open the Control Response Data Presentation window.

Program Notes:

If the data on the time history graph is allowed to scroll off the window or 'expire', the data is lost forever. To prevent this from happening increase the 'Display Length' setting. Setting the 'Display Length' to an overly long setting will cause the data presentation window to display unnecessary data. The default settings for the sample rate and the display length are normally sufficient to properly document most typical helicopter responses but some experimentation with these settings may be necessary.

The 'Sample Rate' setting should not be changed during a flight test. All data for a specific test must be acquired at a constant sampling rate or the results will be inaccurate.

The [Clear Data] button clears all three time history graphs. It is not necessary to clear data before beginning an attempt to acquire flight test data, but it may be desirable to clear extraneous data before a "for-record test" is performed. This is especially recommended if the display length is set to a long length and there are several minutes of unnecessary data preceding the test.

Data Reduction and Presentation

The time history data of the aircraft body attitudes, body rates, accelerations, and stick positions are available for analysis and presentation in the Control Response Data Presentation window (figure 6-16). The left side of this window presents the relevant time histories, while the right side contains the analysis tools, engineering data results, and general test information. The time histories display the scaled raw data from which the engineering data results are calculated.

When this window is initially opened, the 'Engineering Data Results' section is blank. These data are determined by means of a process that is initiated by clicking the [Analyze Response] button. This analysis can be done when the data is collected or the data can be saved and the analysis can be performed at a later time. The results of the analysis are returned and filled into the 'Engineering Data Results' section of the window. For more information on the response analysis process see the First Order Dynamic Response Data Analysis section in Chapter 4.

The information on the lower right can be filled in by the user to save relevant test condition information with the data. Once this information has been entered by the user it will be

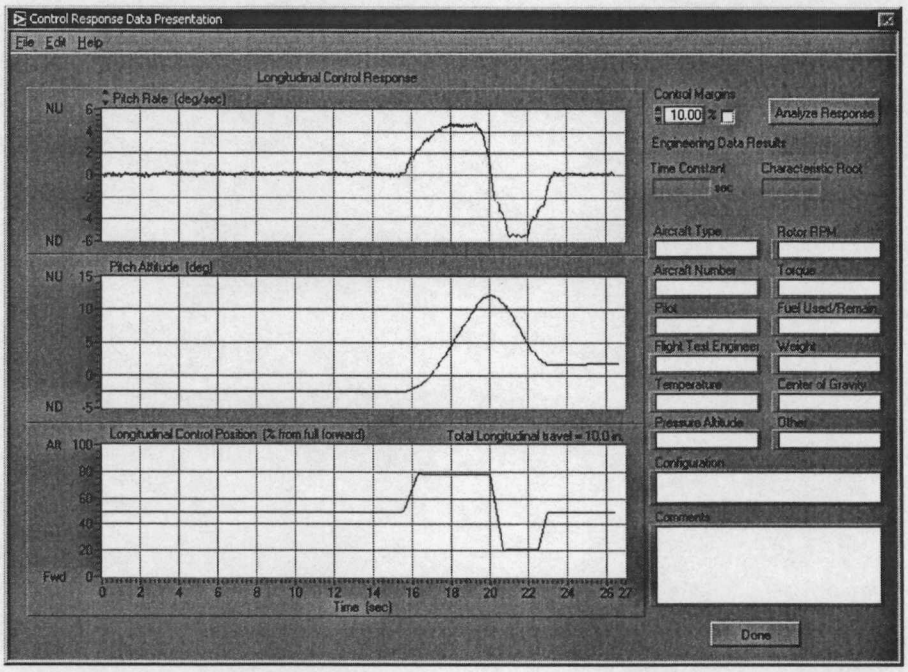


Figure 6-16 Longitudinal Control Response Data Presentation Window

automatically reentered each time a data presentation window is opened. Only the information that has changed since the last test performed needs to be updated.

Data Reduction

The engineering data results for the characteristic motion about the axis of interest can be determined by the following process:

1. Click the [Analyze Response] button.
2. Complete the procedures outlined in the First Order Dynamic Response Data Analysis section in Chapter 4.

The following characteristic parameters are returned from this analysis:

- Time Constant
- Characteristic Root

3. Once the engineering data parameters have been determined the user may opt to view the data on a complex plane by clicking the [Show Root Locus] button. For more information on the Root Locus plot see the Display of Characteristic Roots on a Complex Plane section in Chapter 4.

The data shown in the Control Response Data Presentation window can also be visually analyzed for several important characteristics. Some of the attributes that may be determined are the steady state rate, maximum rate, initial acceleration, maximum acceleration, and acceleration delay.

Analysis of control response data should also address the following questions:

1. Was the response predictable?
2. Was the response consistent?
3. Was the steady state rate adequate?
4. Are the aircraft characteristics suitable for the intended mission?

Chapter 7

FORWARD FLIGHT LATERAL-DIRECTIONAL FLIGHT TESTING

General

The forward flight lateral-directional characteristics include the roll moments about the x-axis, the yaw moments about the z-axis, and the linear force components along the y-axis. This chapter discusses the helicopter lateral-directional stability and control characteristics that are present in forward flight. As in the previous two chapters the theoretical background for lateral-directional stability and control flight testing, forward flight lateral-directional flight test techniques, and procedures for the operation of the Helicopter Stability and Control Analysis Software are presented.

Theory

As discussed in Chapter 6 the forward flight characteristics of the helicopter are governed by the static stability, dynamic stability, and control properties of the aircraft. Lateral control of a helicopter is achieved in a manner similar 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 a result lateral velocity and tail rotor thrust. The control gearing, moment of inertia, side force characteristics, and lateral speed stability together provide lateral control characteristics in forward flight that are different than for the longitudinal axis. [12]

The side force characteristics, 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 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 it's size and shape. Generally in both single main rotor and tandem rotor configurations, there are no surfaces generating roll moments equivalent to the longitudinal moments produced by the

horizontal stabilizer. Therefore, any roll moments generated by the rotor, as it is tilted to balance lateral force, results in a substantial bank angle. However, the tail rotor and/or the vertical tail provide forces stabilizing yaw motion analogous to the action of the horizontal tail in damping the pitch motion. [12]

The rotor produces many of the most dominant forces that control the helicopter dynamics, however when considering lateral-directional characteristics the effects of the tail rotor become much more evident and are included in the lateral-directional discussions. Table 7-1 presents the quasi-static rotor characteristics to indicate the rotor response to various control and external inputs.

Table 7-1 Forward Flight Lateral-Directional Quasi-Static Rotor Characteristics [12]

Perturbation	**Rotor Response**	**Forces and Moments Applied to the Aircraft**
Right Lateral Increment in Horizontal Velocity $+\Delta v$	Rotor Tilts Left (Blowside) $-\frac{\partial b_{1s}}{\partial v}$	$-\Delta L_v$, Left, Left Roll $-\Delta L_H$, Left, Left Roll $-\Delta T_{TR}$, T/R Thrust Decrease, Nose Right
Right Roll Rate Increment $+\Delta p$	Rotor Lags Shaft by Angle Proportional to Roll Rate $-\frac{\partial b_{1s}}{\partial p}$	$-\Delta L_p$ Left Roll (Damping) $-\Delta L_H$, Left Roll (Damping) $-\Delta T_{TR}$, T/R Thrust Decrease, Nose Right
Right Yaw Rate Increment $+\Delta r$	Negligible	$+\Delta T_{TR}$, T/R Thrust Increase Yaw Damping
Right Lateral Cyclic Increment $+\Delta\delta_{Lat}$	Tip Path Plane Tilts Right $+\frac{\partial b_{1s}}{\partial \delta_{Lat}}$	$+\Delta L$, Right Roll Moment
Right Tail Rotor Pedal Increment $-\Delta\delta_{Ped}$	Negligible	$+\Delta T_{TR}$, T/R Thrust Increase Nose Right

The fuselage can also have a strong influence on the lateral-directional handling qualities of the aircraft. The effect of the various contributors to the lateral-directional characteristics of a single main rotor helicopter are ranked by relative magnitude in Table 7-2. Also presented in this table are the usual sign for each the most prominent lateral-directional stability derivatives of the single main rotor helicopter.

Table 7-2 Relative Contributions to the Single Main Rotor Helicopter Stability Derivatives [12]

Derivative		Relative Contributions (1)		
Symbol	Usual Sign	Fuselage and Tail	Main Rotor	Tail Rotor
N_v	+	2 (Depends on vertical fin size)		1
N_r	–	2		1
N_p	–	2 (Depends on vertical fin size and height)		1 (Depends on tail rotor height)
$N_{\delta Ped}$	+			1
$N_{\delta Lat}$	(2)		Generally Small	
L_v	–	2	1	2 (Depends on tail rotor height)
L_r	+	2 (Depends on vertical fin size and height)	3	1 (Depends on tail rotor height)
L_p	–	3	1	2 (Depends on tail rotor height)
$L_{\delta Ped}$	+		1	
$L_{\delta Lat}$	+, –			1 (Depends on tail rotor height)
Y_v	–	1	1	1
Y_r	+	2 (Depends on vertical fin size)		1
Y_p	–	2 (Depends on vertical fin size and height)		1 (Depends on tail rotor height)
$Y_{\delta Ped}$	+		1	
$Y_{\delta Lat}$	–			1

Notes: (1) Ranking of relative contributors, 1 is largest contributor.
(2) Depends on the relative strength of the three contributors.

Lateral-Directional Static Stability

The helicopter lateral-directional static stability is evaluated by measuring the displacement of the lateral and pedal control positions when the aircraft is flown in an off trim equilibrium

flight condition. The gradient of the lateral cyclic control with sideslip indicates the sign of the lateral stability or effective dihedral. The gradient of the pedal control with sideslip indicates the sign of the directional stability. [12] The static lateral-directional characteristics of a helicopter are largely dominated by main rotor blowside, dihedral effect, and the fuselage side forces generated in sideslips. The main rotor blowside is generated in the same manner as longitudinal blowback, only blowside implies that a rolling moment rather than a pitching moment is generated. The following physical descriptions of the influencing forces and moments are presented to better understand the lateral-directional reactions of the helicopter in forward flight.

Effective Dihedral

The term *dihedral effect* is typically used in fixed wing aerodynamics to describe the rolling moment produced by placing the aircraft in a sideslip. This rolling moment is the product of an increased angle of attack on the forward wing, and a decreased angle of attack on the aft wing, in a sideslip. The angle of attack increase is caused by relative inflow angle change due to the dihedral angle of the main wing. Considering an aircraft in level, trimmed, flight; a left pedal application causes the right wing to be rotated forward of the left wing in a right sideslip. If the aircraft has positive dihedral (upward slant of the wings) the right wing will encounter an angle of attack increase and the aircraft will tend to roll left, this is a stabilizing effect. Aircraft with negative dihedral or anhedral would experience a destabilizing roll in the opposite direction of the pedal application. Dihedral was the first method used in the construction of aircraft to gain lateral stability [19]

The helicopter equivalent of dihedral is the rotor coning angle which to some degree has the same response to sideslip as an airplane with positive dihedral. Additionally, the rotor will also respond to changes in sideslip because blade flapping is produced by conditions referenced to the flight path rather than to the fuselage orientation. [5] Considering the trimmed helicopter in forward flight that is subjected to a sudden lateral gust from the right side; the rotor blade that is over the tail is affected by a relative velocity increase and the forward blade experiences a relative velocity decrease. The result is a left rolling moment or blowside. The main rotor contribution to the rolling moment due to lateral velocity is a combination of both of these positive effects.

The term *effective dihedral* refers to the total helicopter roll moment due to sideslip and is not necessarily caused entirely by dihedral. The tail rotor and the fuselage and tail of the helicopter also contribute to the effective dihedral response of the helicopter. Each of these contributions can be stabilizing or destabilizing depending upon the height of the tail rotor thrust and the height of the center of pressure with respect to the center of gravity.

Conventional helicopter notation is positive lateral velocity (v) to the right, and a positive rolling moment (L) with 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 (L_v) is negative in the case of a helicopter with positive effective dihedral. [12]

Adverse Yaw

Helicopter adverse yaw is also a result of different effects as compared to the traditional airplane aerodynamic causes. The yawing moment due to lateral control deflection in an airplane is generally due to a drag differential between the left and right wings with the ailerons deflected. In a left rolling condition the right aileron is deflected down which increases the camber of that airfoil segment. The increased camber causes an increase in lift and lift induced drag. On the left wing the opposite actions and reactions occur. The overall response to the differential lift is the desired left rolling moment; however, the differential induced drag causes a right yawing moment.

In a U.S. helicopter when the lateral control is deflected to the left in forward flight, there is an overall increase in the induced drag of the rotor system due to the increase in blade pitch angle on the advancing blades. This overall increase of the rotor system drag causes the rotor RPM to begin to decrease, which is sensed by the fuel control, which automatically increases fuel flow to increase engine power and maintain rotor RPM. This increased engine power required to maintain rotor RPM causes an imbalance of the moments about the yaw axis. Left rolling motions require an increase in power and an associated right yawing moment. Right rolling motions reduce the power required and result in a left yawing moment. Therefore the yawing moment due to lateral control deflection $\left(N_{\delta_{Lat}}\right)$ is typically negative.

The change in engine power required to maintain rotor RPM during rolling motions is known as transient torque. It is interesting to note that for U.S. single main rotor helicopters where the main rotors rotate counter-clockwise when viewed from above, there in a transient torque increase in left rolls and a transient decrease in right rolls. For single rotor helicopters where the main rotor rotates in the opposite direction, the transient torque responses will be reversed. That is, left rolls will cause torque reductions and right rolls will cause transient torque increases The interesting part is that the yawing moment produced for either direction of rotation is always opposite to the rolling direction.

Proverse Roll

Traditional airplane aerodynamics do not directly apply to the helicopter resultant rolling moments due to tail rotor pedal control changes. In an airplane several factors affect this derivative. First, the height of the rudder, typically above the CG. Left pedal application causes a left rudder deflection and if the rudder center of pressure is above the aircraft's CG results in a right rolling moment. Second, as the aircraft rotates about the vertical axis, the wing that is rotating forward is affected by a increased velocity due to rotation, and the other wing experiences a local velocity decrease. [19] Once a sideslip is generated the airplane responds to the dihedral effect, which generally causes a rolling moment in the direction of the yaw. The second two effects are not specifically rolling moments as a result of the rudder application $\left(L_{\delta_{Ped}}\right)$, technically they are rolling moments due to yaw rate and sideslip $\left(L_r, \& L_\beta\right)$. Although, these two effects are not strictly due to the rudder application they are often considered as a part of the discussion of proverse roll in an airplane.

For a helicopter, either at a hover or in forward flight the rolling moment due to a change in the tail rotor control position is a function of the height of the tail rotor above the vertical CG of the aircraft. Most helicopters hover with the tail rotor at a height that is at least as high as the vertical CG, and at increasing forward airspeeds the tail rotor height moves higher in relation to the CG. With the tail rotor positioned above the CG, any increment of tail rotor thrust, will cause a rolling moment about the longitudinal axis Specifically a right pedal increment will cause a decrease of tail rotor thrust that will result in the desired right yawing moment as well as a left

rolling moment, or adverse roll. By sign convention a right pedal increment is positive and a left rolling moment is negative; therefore, $L_{\delta_{Ped}}$ is usually negative in a helicopter

Inherent Sideslip

A typical U.S. single rotor helicopter (counter-clockwise main rotor rotation) in wings level, coordinated, equilibrium flight has a tail rotor and/or vertical stabilizer that is producing a left yawing moment to balance the torque on the main rotor. Consequently, the resultant aerodynamic force has a lateral component to the right along the aircraft body y-axis. This lateral force component requires an equal and opposite main rotor force component to maintain equilibrium The lateral main rotor force is generated by a left tilt of the tip path plane. Thus, the aircraft is normally flying in a right sideslip This sideslip is required to achieve a balance between the sideforce contribution of the tail rotor thrust and the fuselage-main rotor side forces. The sideslip angle required to achieve the balance in side forces is the inherent sideslip angle. The inherent sideslip angle increases when main rotor torque required is increased by either increasing gross weight or climbing at the same forward speed. In addition, a decrease in airspeed at a 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. [12] Several design factors play a role in the amount of inherent sideslip present for a given helicopter, including: main rotor hub offset, rotor height above the center of gravity, and the height of the anti-torque force relative to (above or below) the center of gravity.

Some 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 the right and left disturbances from trimmed flight [12]

Lateral-Directional Dynamic Stability

Typically, three distinct dynamic modes of motion exist in the lateral-directional sense. These modes include the roll mode, the spiral mode and the lateral-directional oscillation (LDO) or dutch roll mode. The first two are non-oscillatory and the third is oscillatory.

Roll Mode

Consider the roll rate response to a step lateral cyclic step input No rudder pedal input is present For a hypothetical helicopter in which such an input did not excite the spiral or the LDO modes the roll response is shown in figure 7-1 A steady roll rate would be achieved in some period of time Two main effects are at work The cyclic input creates a rolling moment, which is opposed by a rolling moment due to roll rate (roll damping) When the roll rate builds up to the point where the damping moment equals the control moment, no further increase in roll rate is achieved The less damping, the higher the steady state roll rate is achieved for a given cyclic input, and the longer the time will be to achieve steady state A well damped roll mode, with no other modes interfering, would be ideal [18] Unfortunately, the damping contribution from the rotor is usually quite low and is not augmented by any significant fuselage contributions, even at high forward speed Consequently, the very long response times make it appear to the pilot like an acceleration control in roll An initial cyclic input must be reduced when the desired roll rate response is achieved [18]

Spiral Mode

The spiral mode is best described by considering a helicopter placed in steady flight with zero angular velocity, but with one wing low The flight controls of the aircraft are in the proper positions for wings level flight. If the aircraft rights itself to a wings level attitude, the spiral

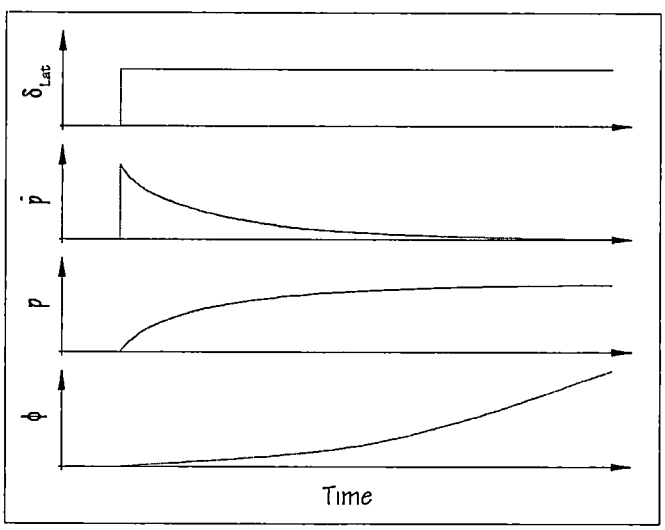


Figure 7-1 Idealized Time History Plot of Roll Responses to a Lateral Cyclic Step Input

mode is convergent. A roll toward the existing bank angle indicates spiral divergence and no roll motion in either direction indicates a neutral spiral mode. The dynamic stability of the spiral mode depends on the relative magnitude of the dihedral effect, static directional stability, yaw damping, and roll due to yaw rate. When the directional static stability is high and the stable dihedral effect is low, the roll due to yaw rate rolling moment can cause the aircraft to roll into a larger and larger bank angle. The continued increase in bank angle results in an increase in the sideslip, which in turn causes an increase in yaw rate and classic spiral divergence. Helicopters that do exhibit stable spiral modes (often with the aid of stability augmentation devices) usually take a long time to converge and do not produce a significant effect on the short term maneuvering of the aircraft. [18]

Lateral-Directional Oscillation

The lateral-directional oscillation (LDO) is an oscillatory mode coupling roll and sideslip excursions. It is usually lightly damped and relatively low frequency. It is a "nuisance" mode, requiring continual pilot compensation in precise lateral maneuvers. The LDO is characterized by a combination of sideslip and roll attitude excursions at the same frequency but not in phase. In aircraft where the dihedral effect is large in comparison to the directional stability the roll-to-sideslip (ϕ/β) ratio is large, that is the LDO is mostly rolling with very little sideslip. A low roll-to-sideslip ratio indicates a "flat" LDO, being mostly sideslip with very little roll.

Lateral-Directional Equations of Motion for a Helicopter in Forward Flight

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. The equations of motion presented below assume that the initial flight path is near level ($\cos\gamma_0 \approx 1$) and that there is minimal coupling with the longitudinal motion of the aircraft. Also, inertial and cross-axis couplings in roll and yaw are not considered. [12]

$$\dot{v} - Y_v\Delta v - Y_r\Delta r + u_0\Delta r - Y_p\Delta p - g\phi = Y_{\delta_{Lat}}\Delta\delta_{Lat} + Y_{\delta_{Ped}}\Delta\delta_{Ped} \qquad \textit{eq. 7-1}$$

$$\dot{p} - L_v\Delta v - L_r\Delta r - L_p\Delta p = L_{\delta_{Lat}}\Delta\delta_{Lat} + L_{\delta_{Ped}}\Delta\delta_{Ped} \qquad \textit{eq. 7-2}$$

$$\dot{r} - N_v\Delta v - N_r\Delta r - N_p\Delta p = N_{\delta_{Lat}}\Delta\delta_{Lat} + N_{\delta_{Ped}}\Delta\delta_{Ped} \qquad \textit{eq. 7-3}$$

Where:

$\dot{v}$	=	*Linear acceleration along the y-axis.*
$\dot{p}$	=	*Angular acceleration about the x-axis.*
$\dot{r}$	=	*Angular acceleration about the z-axis.*
Δv	=	*Change in linear velocity along the y-axis.*
Δr	=	*Change in angular velocity about the z-axis (change in yaw rate).*
Δp	=	*Change in angular velocity about the x-axis (change in roll rate).*
Y_v	=	*Lateral side force due to lateral velocity.*
Y_p	=	*Lateral side force due to roll rate.*
Y_r	=	*Lateral side force due to yaw rate.*
L_v	=	*Roll moment due to lateral velocity.*
L_p	=	*Roll moment due to roll rate.*
L_r	=	*Roll moment due to yaw rate.*
N_v	=	*Yaw 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 component along the x-axis.*
g	=	*Gravity.*
ϕ	=	*Roll angle.*
$Y_{\delta_{Lat}}$	=	*Lateral side force due to lateral control.*
$Y_{\delta_{Ped}}$	=	*Lateral side force due to pedal control.*
$L_{\delta_{Lat}}$	=	*Roll moment due to lateral control.*
$L_{\delta_{Ped}}$	=	*Roll moment due to pedal control.*
$N_{\delta_{Lat}}$	=	*Yaw moment due to lateral control.*
$N_{\delta_{Ped}}$	=	*Yaw moment due to pedal control*
$\Delta\delta_{Lat}$	=	*Change in lateral control.*
$\Delta\delta_{Ped}$	=	*Change in pedal control.*

Lateral-Directional Stability Derivatives

The lateral-directional stability derivatives give the change in the total aerodynamic side force, rolling moment, and yawing moment resulting from a change in one of the lateral-directional flight or control variables (v, r, p, δ_{Lat}, and δ_{Ped}). [12] The general forms of the

equations for the lateral-directional force and moment stability derivatives are presented as equations 7-4 through 7-6. These general equations are used for each pertinent flight or control variable while the holding the remaining variables constant. The six terms of these equations can be examined individually and relative magnitudes can be assessed for each term, then the relative magnitudes can be summed to gauge the overall impact of the examined variable.

$$Y_{(\ )} = \frac{1}{m}\left[\overbrace{T\frac{\partial b_{1s}}{\partial(\ )}}^{\langle 1\rangle} + \overbrace{b_{1s}\frac{\partial T}{\partial(\ )}}^{\langle 2\rangle} + \overbrace{\frac{\partial Y_R}{\partial(\ )}}^{\langle 3\rangle} + \overbrace{\frac{\partial T_{TR}}{\partial(\ )}}^{\langle 4\rangle} + \overbrace{\frac{\partial Y_{f\&t}}{\partial(\ )}}^{\langle 5\rangle}\right] \qquad \textit{eq. 7-4}$$

$$L_{(\ )} = \frac{1}{I_{xx}}\left[\overbrace{\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial b_{1s}}{\partial(\ )}}^{\langle 1\rangle} + \overbrace{hb_{1s}\frac{\partial T}{\partial(\ )}}^{\langle 2\rangle} + \overbrace{h\frac{\partial Y_R}{\partial(\ )}}^{\langle 3\rangle} - \overbrace{h_{TR}\frac{\partial T_{TR}}{\partial(\ )}}^{\langle 4\rangle} + \overbrace{\frac{\partial L_{f\&t}}{\partial(\ )}}^{\langle 5\rangle}\right] \qquad \textit{eq. 7-5}$$

$$N_{(\ )} = \frac{1}{I_{zz}}\left[\overbrace{T\ell_m\frac{\partial b_{1s}}{\partial(\ )}}^{\langle 1\rangle} + \overbrace{\ell_m b_{1s}\frac{\partial T}{\partial(\ )}}^{\langle 2\rangle} + \overbrace{\ell_m\frac{\partial Y_R}{\partial(\ )}}^{\langle 3\rangle} - \overbrace{\ell_t\frac{\partial T_{TR}}{\partial(\ )}}^{\langle 4\rangle} + \overbrace{\frac{\partial N_{f\&t}}{\partial(\ )}}^{\langle 5\rangle} + \overbrace{\frac{\partial Q_{MR}}{\partial(\ )}}^{\langle 6\rangle}\right] \qquad \textit{eq. 7-6}$$

Where:

$Y_{(\ )}$	=	*Side force due to examined variable.*
$L_{(\ )}$	=	*Roll moment due to examined variable.*
$N_{(\ )}$	=	*Yaw moment due to examined variable.*
$(\ )$	=	*Examined flight or control variable* (v, p, r, δ_{Lat}, *or* δ_{Ped})
m	=	*Mass of the aircraft.*
I_{xx}	=	*Moment of inertia about the x-axis.*
I_{zz}	=	*Moment of inertia about the z-axis.*
T	=	*Main rotor thrust.*
T_{TR}	=	*Tail rotor thrust.*
h	=	*Height of the rotor hub above the center of gravity.*
h_{TR}	=	*Height of the tail rotor above the CG.*
b_{1s}	=	*Lateral flapping angle (shaft referenced).*
e	=	*Flapping hinge offset.*

M_S = *Blade mass moment.*

Ω = *Rotor angular velocity.*

Y_R = *Rotor side force.*

$Y_{f\&t}$ = *Side force due to aerodynamic forces of fuselage and tail.*

$N_{f\&t}$ = *Yaw moment due to aerodynamic forces of fuselage and tail.*

ℓ_m = *Longitudinal distance between the mast and the CG.*

ℓ_t = *Length of the tail.*

Q_{MR} = *Main rotor torque.*

The terms of these equations represent the major contributors to the lateral-directional stability of the helicopter. The physical explanations for the six terms are as follows:

1. The terms proportional to the change in lateral tilt of the tip path plane (Δb_{1s}):
   a. $T\Delta b_{1s}$, increment in lateral component of rotor thrust due to a change in lateral tip path plane tilt.
   b. $Th\Delta b_{1s}$, roll moment produced by side force increment due to a change in lateral tip path plane tilt.
   c. $T\ell_m\Delta b_{1s}$, yaw moment produced by side force increment due to a change in lateral tip path plane tilt.
   d. $\frac{ebM_S\Omega^2\Delta b_{1s}}{2}$, roll moment produced hub moments due to Δb_{1s}.
2. The terms proportional to the change in the thrust increment (ΔT).
   a. $b_{1s}\Delta T$, increment in lateral component of rotor thrust due to a thrust increment for a trim condition with a constant lateral tip path plane tilt (b_{1s}).
   b. $hb_{1s}\Delta T$, roll moment produced by side force increment due to a change in main rotor thrust.
   c. $\ell_m b_{1s}\Delta T$, yaw moment produced by side force increment due to a change in main rotor thrust.
3. The terms proportional to the increment in the rotor side force (ΔY_R):
   a $h\Delta Y_R$, roll moment produced by rotor side force increment.
   b. $\ell_m\Delta Y_R$, yaw moment produced by rotor side force increment.

4. The terms proportional to the tail rotor thrust increment (ΔT_{TR}):
   a. $h_{TR}\Delta T_{TR}$, roll moment produced by tail rotor thrust increment.
   b. $\ell_T\Delta T_{TR}$, yaw moment produced by tail rotor thrust increment.
5. The terms proportional to the increments in the aerodynamic forces acting on the fuselage and tail ($_{f\&t}$).
6. Increment in yaw moment due to a change in main rotor torque (ΔQ_{MR}).

Test Procedures

Lateral-Directional Static Stability

The purpose of this test is to evaluate the static lateral-directional stability or the change in the total lateral-directional restoring moment generated by off-trim sideslip conditions. Since the aircraft's forces/moments can not be directly measured in flight, the intent of this test is to measure the control displacements necessary to balance the lateral-directional moments caused by sideslip variations from trim. [12] Positive static lateral stability is indicated by increasing right lateral cyclic stick displacement with increasing right sideslip. Positive static directional stability is indicated by increasing left pedal control displacement with increasing right sideslip.

Although the direction of the control displacements are an indicator of the sign of the static stability, neither the linear displacement nor the percentage of total travel, for a given off-trim condition, is a measure of the strength of the stability. The amount of control displacement is a function of the flight control linkage gearing.

Test Method

This test is conducted by establishing a wings level, steady, unaccelerated, trimmed flight condition. Then without varying the collective control position or rotor speed the sideslip angle is varied incrementally left and right of the trim attitude. At each stabilized off-trim flight condition, the lateral and directional controls are used to maintain steady non-turning flight. The longitudinal control is used to maintain a constant airspeed. If the aircraft is equipped with an airspeed indicating system that is not effected by sideslip (i.e. a swiveling boom mounted pitot probe) then this system is used to maintain airspeed. Otherwise, as the sideslip angle is increased, the airspeed indicator may become inaccurate. To confirm the airspeed at large

sideslips, the pedals can be quickly returned to trim to verify the airspeed. Maintaining airspeed at larger sideslip angles will normally require the aircraft be in a slight descent. During this test an altitude band should be maintained; typically ±1000 feet is used. The first sideslip increment left and right of trim should be approximately 2 degrees with subsequent increments of 5 degrees.

Pilot Procedures

1. Establish stabilized, wings level, unaccelerated, level flight with the control forces trimmed to zero Record the initial trim conditions.
2. Stabilize on each sideslip increment with the collective fixed at the trim airspeed position. The pedals are used to establish the sideslip angle while the lateral cyclic control is simultaneously applied to prevent the helicopter from turning Record data. Longitudinal control is applied as necessary to maintain the initial trim airspeed.
3. Repeat step 2 until the desired sideslip range is covered.

Program Operation

The Lateral-Directional Static Stability Data Collection window (figure 7-2) is used to accumulate the necessary flight test data. To access this window select Acquire Data | Forward Flight Lateral-Directional Tests | Steady Heading Sideslips from the menu on the main window. The top half of this window displays a 30 second time history of the position of each flight control. The lower half of the window is used to collect data for each desired off-trim condition

Data Acquisition Procedures

1. Enter the length of time that the control positions will be monitored on each data point in the 'Sample Length' field. The 'Sample Length' setting can be set from 0 seconds (instant sample) to 10 seconds, the default is 3 seconds. This setting can be changed for subsequent data points as desired.
2 With the aircraft stabilized on the trim speed and control forces trimmed to zero, select the [Mark] button associated with the 'Initial Trim Condition'. After [Mark] is pressed, the pilot should keep the aircraft stabilized for the duration of the Sample Length setting as the software will monitor the control positions for movement during this period.

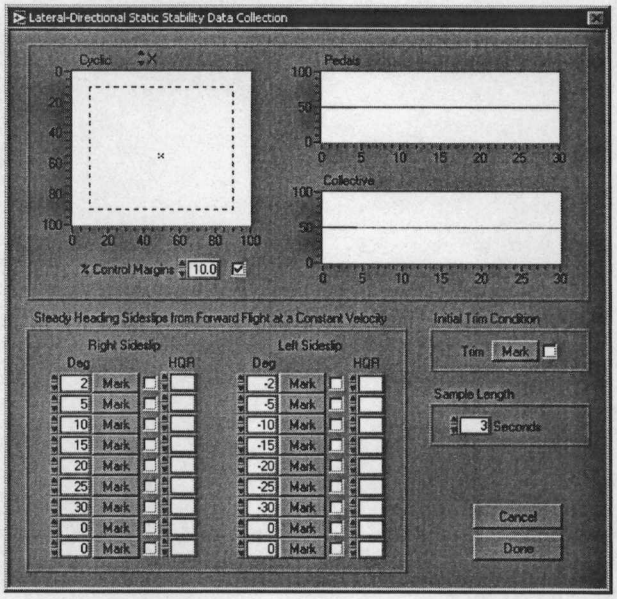


Figure 7-2 Lateral-Directional Static Stability Data Collection Window

3. Restabilize the aircraft on the off-trim sideslip angle without retrimming the control forces.
4. With the aircraft stabilized on the off-trim sideslip and the 'Degrees' set to the sideslip angle being flown, select the corresponding [Mark] button, to record data. The data will be sampled for the duration set in the 'Sample Length' setting.
5. Repeat steps 3 – 4 for each desired sideslip angle data point.
6. When the all desired data points have been collected, select the [Done] button to close the Lateral-Directional Static Stability Data Collection window and open the Lateral-Directional Static Stability Data Presentation window.

Program Notes:

The default values for the off-trim sideslip angles can be changed to any values of interest. If these values are changed from the default values, they must be set before the [Mark] button is pressed for that data point.

After a data point is marked, the [Mark] button mnemonic changes to [Clear]. If the user wishes to discard the data collected for that data point the [Clear] button can be pressed. The data for that speed will be discarded, the button mnemonic changes back to [Mark], and new data can be acquired.

Selecting the [Cancel] button discards all collected data and closes the Lateral-Directional Static Stability Data Collection window and returns function to the main window.

Selecting the [Done] button closes the Lateral-Directional Static Stability Data Collection window and opens the Lateral-Directional Static Stability Data Presentation window.

Data Reduction and Presentation

An example of the Lateral-Directional Static Stability Data Presentation window is presented as figure 7-3. Analysis of this information is primarily based on the slopes of the lateral cyclic control position and the pedal control position plots versus sideslip angle. A lateral cyclic stick position plot that indicates increasing right stick displacement with increasing right sideslip or increasing left stick displacement with increasing left sideslip is indicative of positive dihedral effect (positive lateral static stability). Graphically this would be plotted as a positive sloping line for positive static stability. A pedal position plot that indicates increasing left displacement with increasing right sideslip or increasing right displacement with increasing left sideslip is indicative of positive directional static stability. Graphically this would be plotted as a negative sloping line for positive static stability. A summary of the lateral-directional static stability graphical indications are presented in table 7-3.

The information on the lower right corner of the data presentation window can be filled in by the user to save relevant test condition information with the data. Once this information has been entered by the user it will be automatically reentered each time a data presentation window is opened. Only the information that has changed since the last test performed needs to be updated.

Analysis of lateral-directional static stability data should also consider the following questions:

1. Were control margins approached or exceeded?
2. Is there a speed band about trim within which there are no displacement (or force) cues to indicate the off-trim condition?
3. Were there any pitot static problems in sideslips?
4. Are the aircraft characteristics suitable for the intended mission?

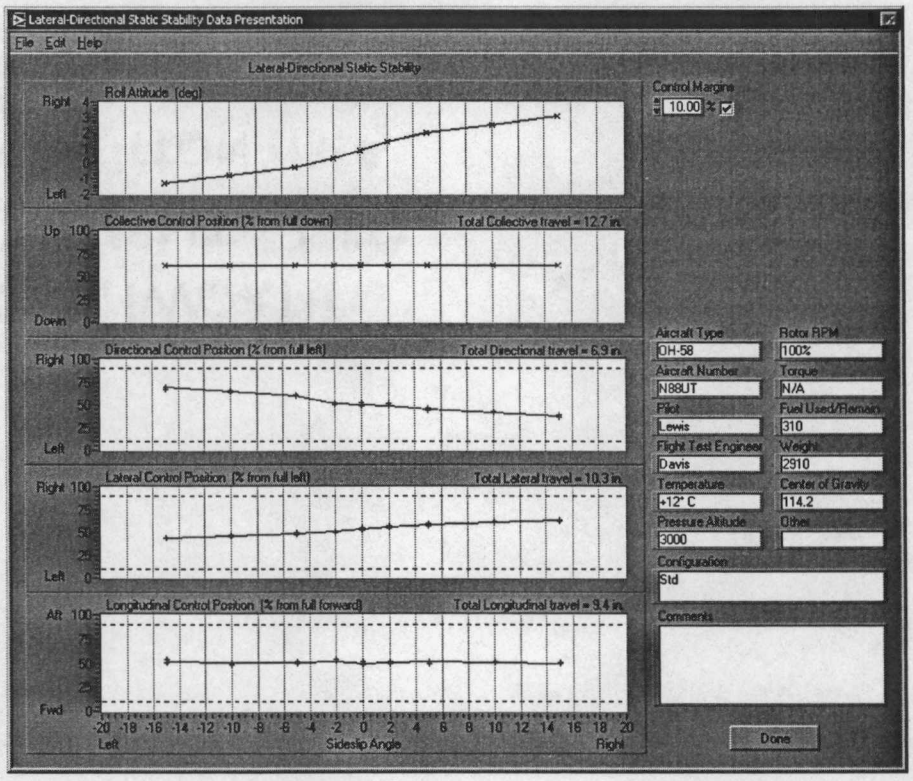


Figure 7-3 Lateral-Directional Static Stability Data Presentation Window

Table 7-3 Summary of Lateral-Directional Static Stability Graphing Indications

Aircraft Motion	What control input was required?	Slope of the control position curve	Static Stability
Increasing Right Sideslip	Increasing Right Cyclic	Positive	Positive
	Neutral	Zero	Neutral
	Increasing Left Cyclic	Negative	Negative
	Increasing Right Pedal	Positive	Negative
	Neutral	Zero	Neutral
	Increasing Left Pedal	Negative	Positive
Increasing Left Sideslip	Increasing Right Cyclic	Negative	Negative
	Neutral	Zero	Neutral
	Increasing Left Cyclic	Positive	Positive
	Increasing Right Pedal	Negative	Positive
	Neutral	Zero	Neutral
	Increasing Left Pedal	Positive	Negative

Spiral Stability

The purpose of this test is to determine the helicopter response to bank angle deviations from trim. A preliminary assessment of the spiral mode can be evaluated during turns on one control testing. For cyclic only turns, the position of the lateral cyclic control when the bank is steady is an indication of the spiral mode. Likewise, the pedal position during pedal only turns when the bank angle is steady, also provides an indication of the spiral mode. The spiral stability test will quantify the spiral mode. [12]

Test Method

This test is conducted by establishing wings level, steady, unaccelerated, level flight with the control forces trimmed to zero. While maintaining the other controls fixed, lateral cyclic is applied to establish the desired test bank angle. Then the lateral cyclic must be smoothly returned to the trim position. Since the spiral mode is usually weak, the response is easily contaminated if the controls are not returned precisely to trim. With the controls at the trim positions a time history of the aircraft response is recorded. A bank angle which converges back toward the level attitude indicates positive spiral stability. A bank angle which diverges from trim corresponds to negative spiral stability.

Pilot Procedures

1. Establish a stabilized, wings level, unaccelerated, level flight with the control forces trimmed to zero.
2. Note the control positions (primarily the lateral cyclic control position).
3. Use lateral cyclic control to establish the desired bank angle, then smoothly return the cyclic to the trim position. Precise positioning of the lateral control to the trim position is important to prevent contaminating the spiral response.
4. Record a time history of the roll attitude response.

Program Operation

The Spiral Stability Data Collection window (figure 7-4) is used to accumulate the necessary flight test data. To access this window select Acquire Data | Forward Flight Lateral-Directional Tests | Spiral Stability from the menu on the main window. This window displays

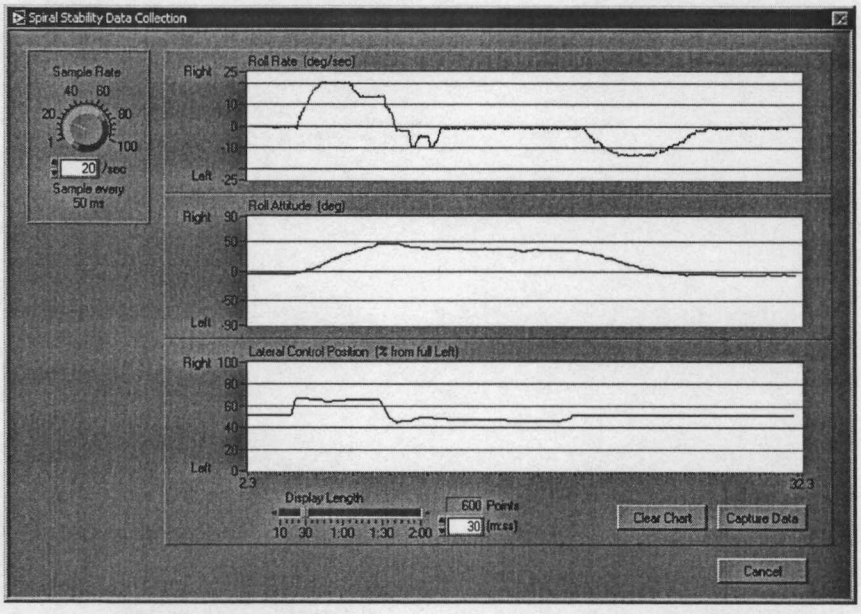


Figure 7-4 Spiral Stability Data Collection Window

time history plots of the roll rate, roll attitude and the lateral flight control position. Since the data collected for this test is sampled over a period of time the user can select the rate of acquisition of the data samples. This is done with the 'Sample Rate' knob at the top left of the window. This setting adjusts the number of samples per second that will be acquired in the range of 1 sample per second to 100 samples per second. The time interval between each data sample is displayed below the 'Sample Rate' knob. The default setting for the sample rate is 20 samples per second and is generally suitable for this test.

The length of time that is displayed on these graphs can be adjusted with the 'Display Length' slider control at the bottom of the graphs. The valid range for this control, on this test, is from ten seconds to two minutes. The default length for this test is 30 seconds. Also displayed with the 'Display Length' slider control is the 'Points' indicator which displays the number of data samples that are held in memory. This value is calculated by multiplying the sample rate (in samples per second) times the display length (in seconds). Data that is older than the display length is 'expired' and can not be recovered. The display length must be long enough to capture the characteristic motion of the aircraft without relevant data expiring.

The middle graph displays the roll attitude. The data presented on this graph are the basis for determining the engineering data results. The intent is to measure the time required for the

perturbed roll attitude to subside to one-half (or double) the initial perturbation. The spiral stability test procedures are conducted while this window is displayed. Before the data on the time history plots expire, the [Capture Data] button must be pressed. When the [Capture Data] button is pressed the time history data from the aircraft body attitude, body rates, accelerations, and stick positions are sent to the Spiral Stability Data Presentation window.

Data Acquisition Procedures

1. Set the 'Sample Rate' as desired.
2. Set the 'Display Length' as desired.
3. Perform the Pilot Procedures above to test the spiral stability.
4. When the desired response data is shown in the time history graph select the [Capture Data] button to close the Spiral Stability Data Collection window and open the Spiral Stability Data Presentation window.

Program Notes:

If the data on the time history graph scrolls off the window or 'expires', the data is lost forever. To prevent this from happening increase the 'Display Length' setting. Setting the 'Display Length' to an unnecessarily long setting will cause the data presentation window to display unneeded data. The default settings for the sample rate and the display length are normally sufficient to properly document most typical helicopter responses but some experimentation with these settings may be necessary.

The 'Sample Rate' setting should *not* be changed during a flight test. All data for a specific test must be acquired at a constant sampling rate or the results will be inaccurate.

The [Clear Data] button clears all three time history graphs. It is not necessary to clear data before beginning an attempt to acquire flight test data, but it may be desirable to clear extraneous data before a "for-record test" is performed. This is especially recommended if the display length is set to a long length and there are several minutes of unnecessary data preceding the test.

Data Reduction and Presentation

An example of the Spiral Stability Data Presentation window is presented as figure 7-5. The left side of this window presents the relevant time histories, while the right side contains the

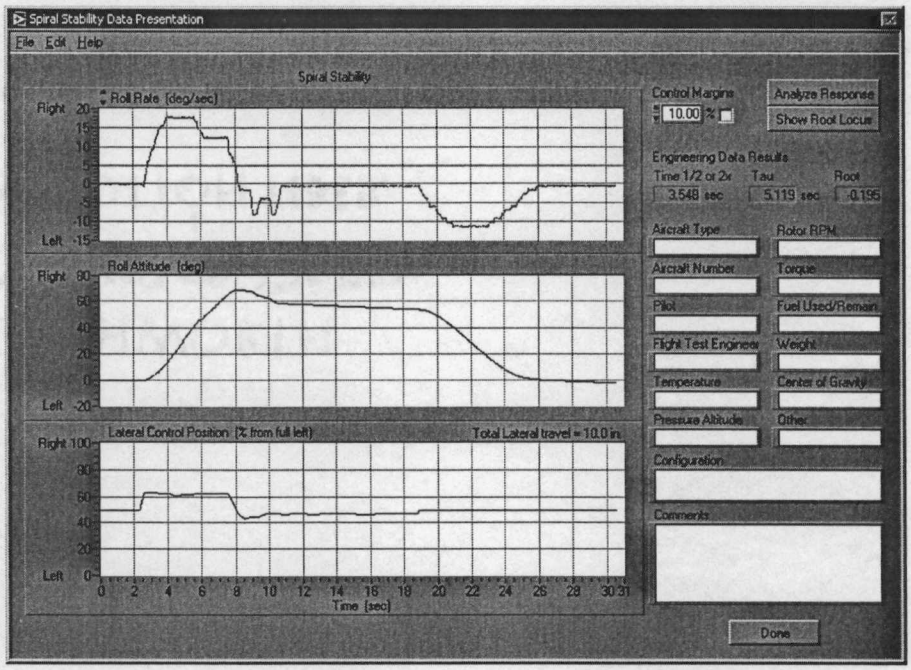


Figure 7-5 Spiral Stability Data Presentation Window

analysis tools, engineering data results, and general test condition information. The time histories display the scaled raw data from which the engineering data results are calculated.

When this window is initially opened, the 'Engineering Data Results' section is blank. These data are determined by means of a process that is initiated by clicking the [Analyze Response] button. This analysis can be done when the data is collected or the data can be saved and the analysis can be performed at a later time. The results of the analysis are returned and filled into the 'Engineering Data Results' section of the window. For more information on the response analysis process see the First Order Dynamic Response Data Analysis section in Chapter 4.

The information on the lower right can be filled in by the user to save relevant test condition information with the data. Once this information has been entered by the user it will be automatically reentered each time a data presentation window is opened. Then only the information that has changed since the last test performed needs to be updated.

Data Reduction

The engineering data results for the characteristic motion about the axis of interest can be determined by the following process:

1. Click the [Analyze Response] button.
2. Complete the procedures outlined in the First Order Dynamic Response Data Analysis section in Chapter 4.

The following characteristic parameters are returned from this analysis:

- Time to half or double
- Time Constant
- Characteristic Root

3 Once the engineering data parameters have been determined the user may view the data on a complex plane by clicking the [Show Root Locus] button. For more information on the Root Locus plot see the Display of Characteristic Roots on a Complex Plane section in Chapter 4.

Analysis of spiral stability response data should also address the following questions:

1. Was the spiral stability response different to the left and right?
2. Did the spiral response vary with airspeed or power changes?
3. Are the spiral stability characteristics suitable for the intended mission?

Lateral-Directional Oscillation Dynamic Stability

The purpose of this test is to evaluate the long term lateral-directional response of the helicopter as a function of time following a disturbance This long term dynamic response is generally a low frequency lightly damped oscillation that is typically damped by the pilot or by the AFCS. The intent of this test is to determine the lateral-directional response; therefore, small off-axis control inputs may be applied to maintain the initial trim pitch attitude. If the off-axis response is large the controls should be fixed and the full response documented.

Test Method

This test is conducted by establishing steady, unaccelerated, level flight with all control forces trimmed to zero. The lateral-directional oscillation is excited with one of the methods described below, and the aircraft response is recorded If the response is moderately divergent the artificial stimulus must be small enough that the response can be adequately documented without exceeding a test limit.

Excitation methods

Release from steady heading sideslips. This technique requires that the cyclic and/or directional controls are quickly and simultaneously returned to trim from the steady sideslip. The response is measured from the time when the controls are returned to trim.

Pedal Doublet. To determine the frequency of the doublet application a frequency sweep of the directional control should be conducted. This is accomplished by applying sinusoidal inputs of constant amplitude at a series of frequencies and observing the yaw response. The LDO frequency is found by starting with a low frequency input and increasing the input frequency while comparing the yaw attitude response. The desired frequency is indicated by a maximum amplitude of the yaw response to the pedal inputs. The LDO is excited by applying a pedal doublet at the noted frequency with sufficient amplitude to disturb the yaw attitude.

Release from pedal driven LDO. This technique also utilizes a frequency sweep of the pedal controls to determine the LDO frequency as described above. Once the LDO frequency is determined, a sinusoidal pedal input is applied at the determined frequency and at a sufficient amplitude to excite the LDO. Then the pedal controls are quickly and precisely returned to trim and the aircraft response is recorded.

Pilot Procedures

1. Establish steady, unaccelerated, level flight with all control forces trimmed to zero.
2. Excite the LDO by one of the methods described above and quickly return the flight controls to the trim positions as accurately as possible.
3. Record the aircraft response.

Program Operation

The Lateral-Directional Oscillation Data Collection window (figure 7-6) is displayed during the lateral-directional oscillation flight testing. To access this window select Acquire Data | Forward Flight Lateral-Directional Tests | Lateral-Directional Oscillation from the menu on the main window. Since the data collected for this test is sampled over a period of time the user can select the rate of acquisition of the data samples. This is done with the 'Sample Rate' knob at the top left of the window. This setting adjusts the number of samples per second that will be

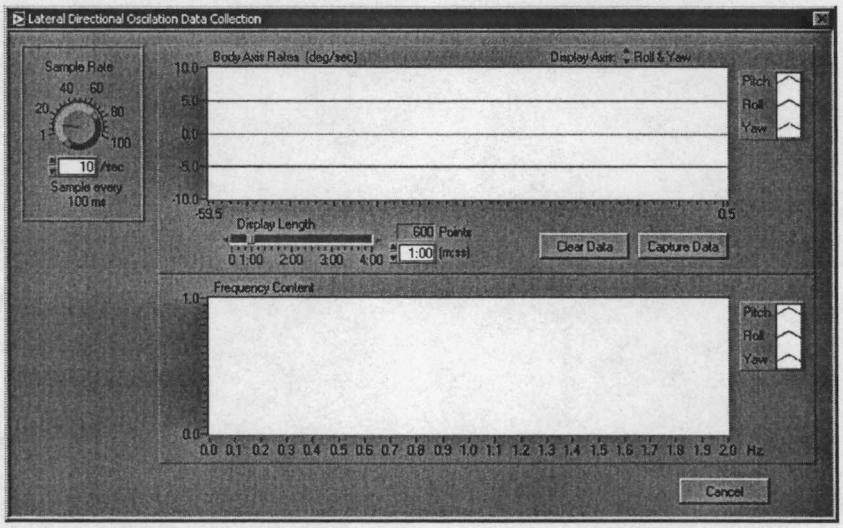


Figure 7-6 Lateral Directional Oscillation Data Collection Window

acquired in the range of 1 sample per second to 100 samples per second. The time interval between each data sample is displayed below the 'Sample Rate' knob. In the example shown in figure 7-6 the sample interval is 100 milliseconds. The default setting for the sample rate is 10 samples per second and is generally suitable for this test.

Two continuously updating graphs are displayed in this window. The top graph is a time history of the roll and yaw rates in degrees per second, the bottom graph displays the frequency content present for the roll and yaw body axis rates. The user may select specific axes to monitor by selecting the desired axes with the 'Display Axis' selection control. Since this test is primarily concerned with roll and yaw rates, it is recommended that this selection remain on 'Roll & Yaw' for this test. The user can also set the length of time for the time history display, this is done by adjusting the 'Display Length' slider control. The valid range for this test is from thirty seconds to four minutes. The default length for this test is 60 seconds. The number of data points held in memory can be calculated by multiplying the sample rate times the display length (for example, in figure 7-6, $10 \frac{\text{samples}}{\text{sec}} \times 60 \text{ sec} = 600$ samples). Data that exceeds the display length is 'expired' and can not be recovered. The display length must be long enough to capture the characteristic motion of the aircraft without the relevant data expiring.

The bottom graph displays the frequency information for the axes displayed in the top graph. The frequency information is determined by a Fourier transformation of the time domain data.

The range of frequencies that can be determined are directly related to the sample rate. The highest frequency that can be determined is one-half of sample rate in accordance with the Nyquist Theorem.

The LDO test procedures are conducted while this window is displayed. Before the data on the time history plot expires, the [Capture Data] button must be pressed. When the [Capture Data] button is pressed the time history data from the aircraft body attitude, body rates, accelerations, and stick positions are sent to the data presentation window.

Data Acquisition Procedures

1. Set the 'Sample Rate' as desired.
2. Set the 'Display Length' as desired.
3. Perform the Pilot Procedures above to stimulate the LDO response.
4. When the desired response data is shown in the time history graph select the [Capture Data] button to close the Lateral-Directional Oscillation Data Collection window and open the Lateral-Directional Oscillation Data Presentation window.

Program Notes:

If the data on the time history graph is allowed to scroll off the window or 'expire', the data is lost forever. To prevent this occurrence increase the 'Display Length' setting. Setting the 'Display Length' to an unnecessarily long setting will cause the data presentation window to display unnecessary data. The default settings for the sample rate and the display length are normally sufficient to properly document most typical helicopter responses but some experimentation with these settings may be necessary.

The 'Sample Rate' setting should *not* be changed during a flight test. All data for a specific test must be acquired at a constant sampling rate or the results will be inaccurate.

The [Clear Data] button clears both the time history and the frequency content graphs. It is not necessary to clear data before beginning an attempt to acquire flight test data, but it may be desirable to clear extraneous data before a "for-record test" is performed. This is especially recommended if the display length is set to a long length and there are several minutes of unnecessary data preceding the test.

Data Reduction and Presentation

The time history data of the aircraft body attitudes, body rates, accelerations, and stick positions are available for analysis and presentation in the Lateral-Directional Oscillation Data Presentation window (figure 7-7). The left side of this window presents the relevant time histories, while the right side contains the analysis tools, engineering data results, and general test information. The time histories display the scaled raw data from which the engineering data results are calculated. The time histories that are displayed for this test are roll rate, roll attitude, yaw rate, yaw attitude, and directional control position. The user may select other plots as desired.

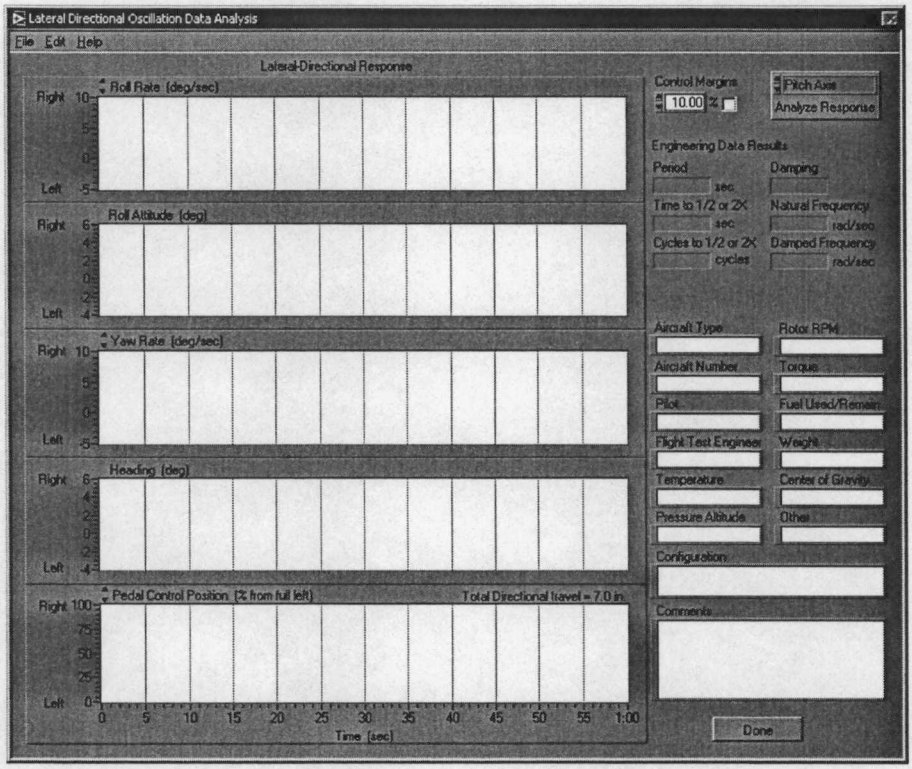


Figure 7-7 Lateral-Directional Oscillation Data Presentation Window

When this window is initially opened, the 'Engineering Data Results' section is blank. This analysis can be done when the data is collected or the data can be saved and the analysis can be performed at a later time. The data analysis is done by clicking the [Analyze Response] button. The process used to determine the engineering data is detailed in the Second Order

Dynamic Response Data Analysis section in Chapter 4. The results of the analysis are returned and filled into the 'Engineering Data Results' section of the window.

The information on the lower right can be filled in by the user to save relevant test condition information with the data. Once this information has been entered by the user it will be automatically reentered each time a data presentation window is opened. Only the information that has changed since the last test performed needs to be updated.

Data Reduction

The engineering data results for the characteristic motion about the axis of interest can be determined by the following process:

1. Select the 'Roll Axis' or 'Yaw Axis' from the selection control at the top of the [Analyze Response] button.
2. Click the [Analyze Response] button.
3. Complete the procedures outlined in the Second Order Dynamic Response Data Analysis section in Chapter 4.

The following characteristic parameters are returned from this analysis:

- Period of the Motion
- Time to half or double Amplitude
- Cycles to half or double Amplitude
- Damping Ratio
- Natural Frequency
- Damped Natural Frequency

4. Once the engineering data parameters have been determined the user may view the data on a complex plane by clicking the [Show Root Locus] button. For more information on the Root Locus plot see the Display of Characteristic Roots on a Complex Plane section in Chapter 4.

Analysis of lateral-directional oscillation data should also address the following questions:

1. How easy was it to excite the LDO response?
2. How easy was it for the pilot to suppress the LDO response?
3. Are the aircraft characteristics suitable for the intended mission?

Control Response

The purpose of this test is to evaluate the aircraft response to a flight control input. The aircraft response is determined by the control sensitivity, damping, and response lags in the examined axis. Control sensitivity is the acceleration per unit control input. Damping is the acceleration per unit of velocity. The response lags are the time from the control input to the aircraft response. The response lags are a consequence of the flight control system characteristics and rotor system type.

As noted in previous chapters, several characteristics contributing to desirable response qualities are listed below. [12]

1. Pilots want a vehicle responsive enough to achieve some level of attitude change within a certain time after a control input (without being overly responsive).
2. Pilots want a reasonably steady state rate for a given input size.
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.

Each of these qualities are directly related to control sensitivity, damping, or both. The resulting motion due to a given control input can be characterized as a first order system.

Test Method

This test is conducted by establishing steady, unaccelerated, level flight with all control forces trimmed to zero. A control step input is applied, and the aircraft response is monitored. Control fixtures are generally used to restrain control inputs and an incremental buildup technique should be used to build up to the desired control input magnitude. The control input should be quick and precise. If a fixture is used, the control should be firmly held against the fixture but not forcefully. Recovery is initiated when a steady state roll rate is achieved or a predetermined test flight limit is reached.

Test limits on bank angle can prevent achieving a steady state roll rate. If the test helicopter can achieve high roll rates and/or the test program is restricted to low roll attitudes, the recovery may occur prior to achieving a steady state roll rate. In this case a smaller control input may be

evaluated that allows the aircraft to achieve a steady state rate, or a 'false start' technique may be used. In this technique, the test pilot establishes a bank attitude in the direction opposite to the test input. Then the control step input is made from the banked control position. The use of the false start technique permits a greater roll attitude change and a longer time to elapse prior to recovery at the test attitude limit. This technique may permit the helicopter to achieve the steady state rate for the desired control input.

The data reduction process uses the steady state rate to determine aircraft characteristics, but does *not* critically depend on discerning a steady state rate. There are alternate analysis methods included in the program that do not require a steady state to determine the desired aircraft characteristics.

Pilot Procedures

1. Establish steady, unaccelerated, level flight with all control forces trimmed to zero.
2. Set the control fixture for the desired displacement and position it for the desired input direction. The control input size and direction are verbally and visually verified between the pilot and copilot/flight test engineer Start with small inputs and increase in size incrementally until desired input is achieved.
3. Apply the lateral cyclic control input against the fixture and hold the controls fixed. Recover when a steady state roll rate is obtained or a predetermined test flight limit is reached.
4. Record the aircraft response.
5. When the maneuver is complete, announce the recovery so the control fixture can be quickly removed.
6. Repeat steps 1 through 5 for increasing lateral cyclic control displacements until the desired deflection is attained.
7. Qualitatively assess the predictability of the control response.

Program Operation

The Control Response Data Collection window (figure 7-8) is displayed during forward flight control response flight testing. To access this window select Acquire Data | Forward Flight

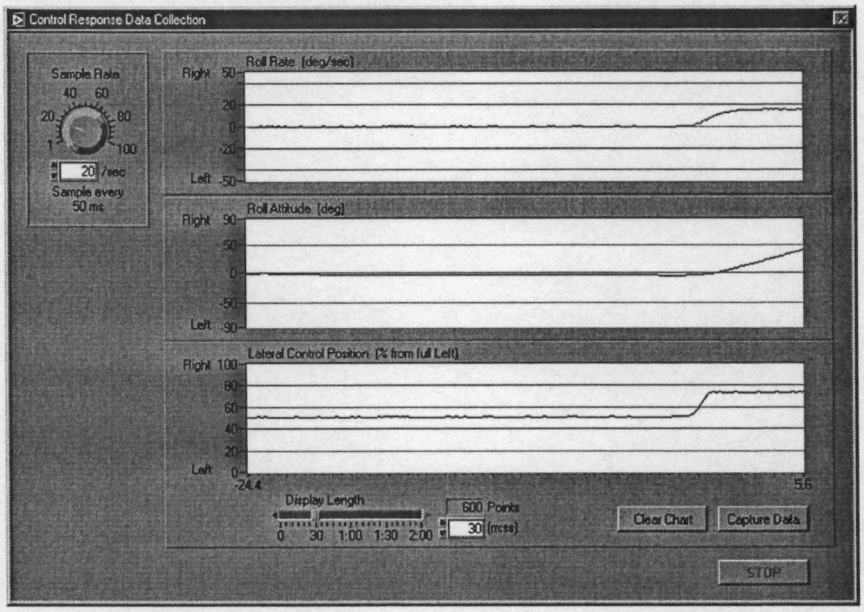


Figure 7-8 Lateral Control Response Data Collection Window

Lateral-Directional Tests | Roll Control Response from the menu on the main window. Since the data collected for this test is sampled over a period of time the user can select the rate of acquisition of the data samples. This is done with the 'Sample Rate' knob at the top left of the window. This setting adjusts the number of samples per second that will be acquired in the range of 1 sample per second to 100 samples per second. The time interval between each data sample is displayed below the 'Sample Rate' knob. The default setting for the sample rate is 20 samples per second and is generally suitable for this test.

Three continuously updating time history plots are displayed in this window. The length of time that is displayed on these graphs can be adjusted with the 'Display Length' slider control at the bottom of the graphs. The valid range for this test is from ten seconds to two minutes. The default length for this test is 30 seconds. Also displayed with the 'Display Length' slider control is the 'Points' indicator which displays the number of data samples that are held in memory. This value is calculated by multiplying the sample rate (in samples per second) times the display length (in seconds). Data that exceeds the display length is 'expired' and can not be recovered. The display length must be long enough to capture the characteristic motion of the aircraft without relevant data expiring.

The top graph is a time history of the roll rate in degrees per second. The data presented on this graph are the basis for determining the engineering data results. Ideally the data presented on this graph will attain a steady state rate during the performance of the particular flight test. The middle graph displays the roll attitude. The bottom graph shows the lateral cyclic control position versus time plot.

The control response test procedures are conducted while this window is displayed, then before the data on the time history plots expire, the [Capture Data] button must be pressed. When the [Capture Data] button is pressed the time history data from the aircraft body attitude, body rates, accelerations, and stick positions are sent to the Control Response Data Presentation window.

Data Acquisition Procedures

1. Set the 'Sample Rate' as desired.
2. Set the 'Display Length' as desired.
3. Perform the Pilot Procedures above to test the control response.
4. When the desired response data is shown in the time history graph select the [Capture Data] button to close the Control Response Data Collection window and open the Control Response Data Presentation window.

Program Notes:

If the data on the time history graph is allowed to scroll off the window or 'expire', the data is lost forever. To prevent losing relevant data, increase the 'Display Length' setting. Setting the 'Display Length' to an unnecessarily long setting will cause the data presentation window to display unnecessary data. The default settings for the sample rate and the display length are normally sufficient to properly document most typical helicopter responses but some experimentation with these settings may be necessary.

The 'Sample Rate' setting should *not* be changed during a flight test. All data for a specific test must be acquired at a constant sampling rate or the results will be inaccurate.

The [Clear Data] button clears all three time history graphs. It is not necessary to clear data before beginning an attempt to acquire flight test data, but it may be desirable to clear irrelevant

data before a "for-record test" is performed. This is especially recommended if the display length is set to a long length and there are several minutes of unnecessary data preceding the test.

Data Reduction and Presentation

The time history data of the aircraft body attitudes, body rates, accelerations, and stick positions are available for analysis and presentation in the Control Response Data Presentation window (figure 7-9). The left side of this window presents the relevant time histories, while the right side contains the analysis tools, engineering data results, and general test information. The time histories display the scaled raw data from which the engineering data results are calculated.

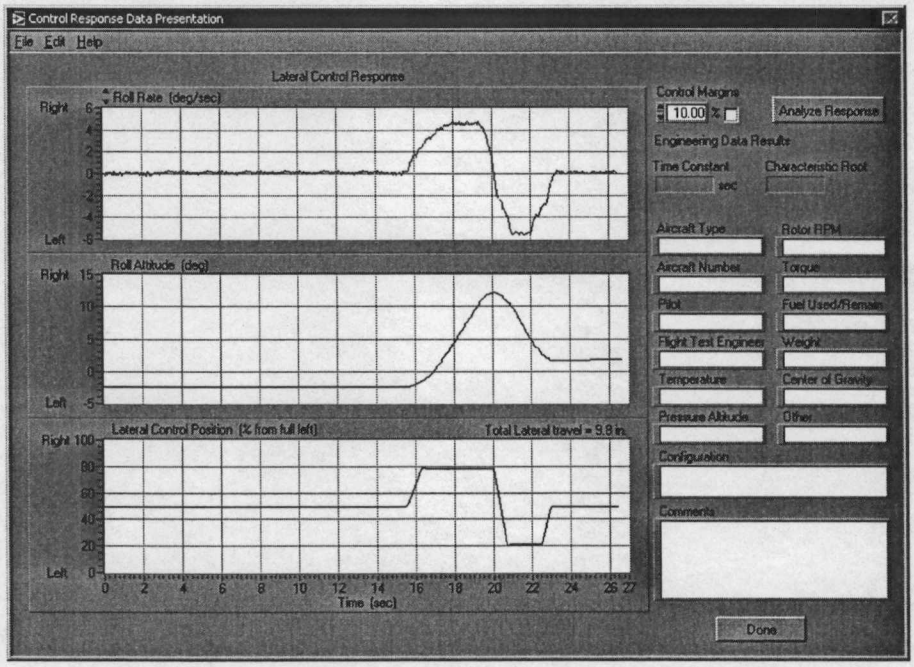


Figure 7-9 Lateral Control Response Data Presentation Window

When this window is initially opened, the 'Engineering Data Results' section is blank. These data are determined by means of a process that is initiated by clicking the [Analyze Response] button. This analysis can be done when the data are collected or the data can be saved and the analysis can be performed at a later time. The results of the analysis are returned and filled into the 'Engineering Data Results' section of the window. For more information on the response analysis process see the First Order Dynamic Response Data Analysis section in Chapter 4.

The information on the lower right can be filled in by the user to save relevant test condition information with the data. Once this information has been entered by the user it will be automatically reentered each time a data presentation window is opened. Then only the information that has changed since the last test performed needs to be updated.

Data Reduction

The engineering data results for the characteristic motion about the axis of interest can be determined by the following process:

1. Click the [Analyze Response] button.
2. Complete the procedures outlined in the First Order Dynamic Response Data Analysis section in Chapter 4.

The following characteristic parameters are returned from this analysis:

- Time Constant
- Characteristic Root

3. Once the engineering data parameters have been determined the user may view the data on a complex plane by clicking the [Show Root Locus] button. For more information on the Root Locus plot see the Display of Characteristic Roots on a Complex Plane section in Chapter 4.

The data shown in the Control Response Data Presentation window can also be visually analyzed for several important characteristics. Some of the attributes that may be determined are the steady state rate, maximum rate, initial acceleration, maximum acceleration, and acceleration delay.

Analysis of forward flight control response data should also address the following questions:

1. Was the response predictable?
2. Was the response consistent?
3. Was the steady state rate adequate?
4. Are the aircraft characteristics suitable for the intended mission?

Chapter 8

CONCLUSIONS AND RECOMMENDATIONS

Conclusions

The objective of this effort was to develop an automated software system for recording, monitoring, and analyzing helicopter stability and control flight test data. The primary design goals for this project were successfully accomplished. During the development process, several difficulties were encountered. The primary obstacles included:

1. Design of the user interface and data flow model.
2. Determining the lateral and longitudinal cyclic control position.
3. Determining which data parameters to sample and the rate at which to sample to properly document each flight test.
4. Selecting the data from the appropriate time period when collecting from a streaming data source.
5. Saving the acquired data to disk, and reading data from the disk for later analysis and presentation.
6. Presenting the acquired data in a format consistent with the format commonly utilized by experimental test pilots and flight test engineers.
7. Designing the analysis tools necessary to quantify the engineering parameters which characterize the aircraft responses.

Each of the obstacles encountered during the development process were adequately surmounted; however, the last one deserves some discussion. The analysis tools were necessary to quantify the aircraft dynamic responses. These dynamic responses were classified into low order equivalent systems. The first order systems analysis did not present significant challenges except to mention that a high signal to noise ratio can easily contaminate the engineering data results. Analysis of the second order systems did however present a major difficulty. A heavily damped oscillation proved to be very difficult to characterize in terms of frequency and damping. The method chosen in this work depends a graphical analysis technique. This graphical

technique works extremely well for a pure and perfect first order response; however, a noisy signal or a response that is contaminated with other aircraft characteristics is likely to be inaccurately characterized. Although this analysis tool is sensitive to noise or contaminated aircraft responses, a heavily damped response is generally considered a desirable flight characteristic. Classifying the response as being heavily damped is often sufficient.

Testing the software also presented some challenges. Specifically, since the hardware was not available to acquire real data from the aircraft, artificial data inputs had to be used to simulate the aircraft responses. The software testing indicated that the user interface is functional and the data handling routines are relatively reliable. This testing also indicated that some data filtering may be necessary to minimize the noise on some channels.

The final product of this effort is a software system capable of collecting, presenting and analyzing helicopter stability and control flight test data near real time inflight The software system is designed to be relatively simple to operate.

Recommendations

With most engineering design projects, hindsight allows the engineer to recommend approaches that might produce a better final product if the project were to be done over again or is revised in the future. The following recommendations should be considered for inclusion in future developments of this software.

Although the analysis tools developed in this effort are functional, they are relatively simplistic. More robust analysis tools are available commercially. Specifically, National Instruments' Advanced Analysis Toolkit has professionally developed functions that are capable of more accurately characterizing the aircraft responses of interest. The functions in this commercial toolkit are modular and could be easily incorporated into the Helicopter Stability and Control application developed in this effort.

The data that is acquired is not currently filtered or processed in any way It may be advantageous to allow the user to selectively apply signal processing to the data channels. For example, the normal acceleration (N_z) may need to be filtered to remove the vertical vibration caused by the main rotor. The implementation of this signal processing is very simple; however a

user interface change would be required to allow the user to select the desired signal processing for each channel.

This program currently has the capability to overlay some of the requirements for military rotorcraft handling qualities (as specified in ADS-33D) over the data results determined from flight testing. This permits a quick assessment of the compliance with the requirements of the specification. The current capability is limited to the comparison of some of the oscillatory response requirements. Future developments of this software should consider expanding the capabilities of the software to assess additional requirements of the specification, for example, bandwidth and attitude quickness.

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APPENDICES

APPENDIX A

EXAMPLE PRINTED DATA

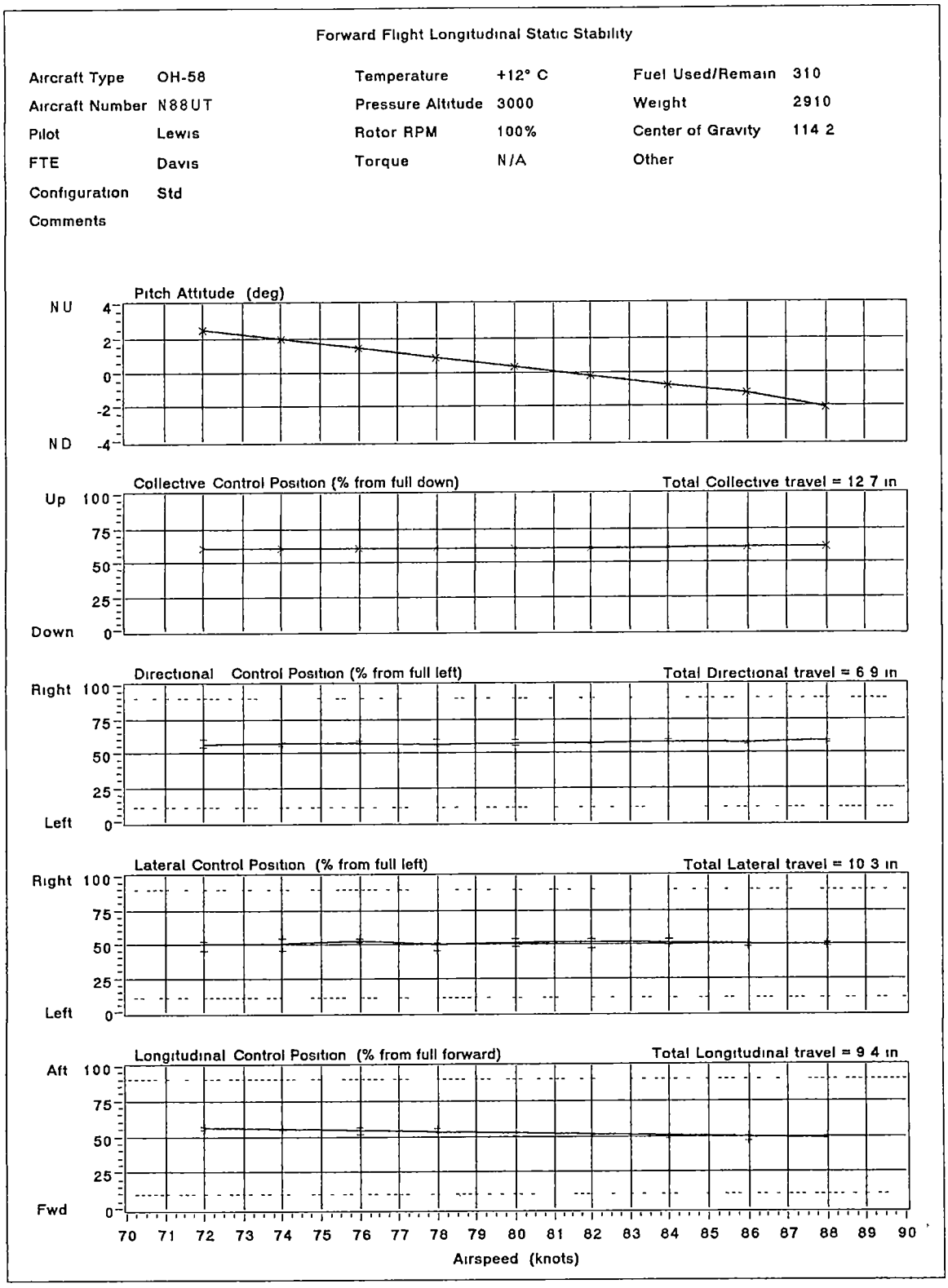


Figure A-1 Example Printed Data

APPENDIX B

EXAMPLE RAW DATA

0
Hover Long Term Dynamic Stability Response
OH-58
N 88UT
Lewis
Davis
+12° C
2010
100%
75%
310
2910
114 2

Standard
No Comments

2 447
1 769
0 723
0 151
0 413
0 409
*****Begin Data*****

Time	Pitch	Roll	Heading	Pitch Rate	Roll Rate	Yaw Rate	Lateral	Longitudina	Collective	Pedals	Nx	Ny	Nz
sec	deg	deg	deg	deg/sec	deg/sec	deg/sec	%	%	%	%	g	g	g
0 000	0 005	0 005	359 998	0 052	0 048	-0 021	50 30	46 12	41 10	52 21	0 06	0 22	1 29
0 100	0 004	0 011	359 990	-0 015	0 062	-0 077	50 21	46 22	41 15	52 26	0 06	0 28	1 06
0 200	-0 002	-0 005	359 975	-0 059	-0 156	-0 157	50 29	46 10	41 07	52 03	0 02	0 05	1 26
0 300	-0 019	-0 011	359 975	-0 167	-0 066	0 002	50 19	46 20	41 26	52 07	0 09	0 15	1 13
0 400	-0 026	-0 022	359 988	-0 073	-0 106	0 135	50 19	46 25	41 13	52 04	0 06	0 27	1 14
0 500	-0 029	-0 022	359 997	-0 026	-0 005	0 086	50 22	46 09	41 28	52 26	0 09	0 26	1 21
0 600	-0 044	-0 026	0 000	-0 155	-0 040	0 036	50 05	46 11	41 01	52 17	0 08	0 20	1 21
0 700	-0 050	-0 022	359 997	-0 051	0 041	-0 032	50 02	46 03	41 18	52 23	0 01	0 04	1 20
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0 900	-0 065	0 000	359 989	-0 116	0 064	0 045	50 00	46 25	41 29	52 02	0 04	0 25	1 26
1 000	-0 072	0 014	359 983	-0 063	0 138	-0 054	50 11	46 06	41 18	52 11	0 05	0 01	1 06
1 100	-0 090	0 021	359 984	-0 188	0 075	0 008	50 15	46 26	41 04	52 02	0 07	0 24	1 14
1 200	-0 116	0 032	359 976	-0 261	0 105	-0 079	50 03	46 03	41 04	52 22	0 02	0 27	1 05
1 300	-0 112	0 023	359 970	0 046	-0 087	-0 062	50 22	46 26	41 27	52 26	0 10	0 04	1 16
1 400	-0 120	0 013	359 974	-0 084	-0 101	0 044	50 08	46 09	41 03	52 11	0 04	0 14	1 06
1 500	-0 141	0 028	359 979	-0 210	0 149	0 047	50 09	46 24	41 28	52 28	0 06	0 01	1 24
1 600	-0 160	0 021	359 985	-0 190	-0 070	0 064	50 00	46 29	41 20	52 27	0 01	0 27	1 18
1 700	-0 161	0 029	359 988	-0 007	0 082	0 031	50 19	46 22	41 19	52 14	0 09	0 01	1 11
1 800	-0 158	0 023	359 979	0 033	-0 062	-0 092	50 14	46 26	41 15	52 08	0 07	0 02	1 11
1 900	-0 161	0 009	359 986	-0 037	-0 144	0 063	50 12	46 25	41 05	52 03	0 04	0 24	1 29
2 000	-0 161	0 007	359 980	0 000	-0 014	-0 054	50 11	46 29	41 23	52 08	0 03	0 21	1 25
2 100	-0 165	0 009	359 970	-0 037	0 014	-0 102	50 24	46 25	41 25	52 04	0 03	0 14	1 25
2 200	-0 183	-0 002	359 980	-0 181	-0 110	0 101	50 29	46.23	41 05	52 09	0 02	0 18	1 12
2 300	-0 208	0 004	359 988	-0 249	0 064	0 082	50 10	46 12	41 27	52 03	0 09	0 22	1 13
2 400	-0 230	0 014	359 982	-0 225	0 094	-0 059	50 27	46 17	41 28	52 03	0 07	0 14	1 09
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2 600	-0 279	0 001	359 988	-0 216	0 021	0 061	50 19	46 16	41 04	52 05	0 07	0 29	1 25
2 700	-0 277	-0 014	0 004	0 017	-0 149	0 160	50 06	46 16	41 11	52 07	0 06	0 12	1 22
2 800	-0 295	-0 002	0 013	-0 180	0 117	0 084	50 15	46 05	41 08	52 03	0 08	0 03	1 22
2 900	-0 295	-0 011	0 011	-0 004	-0 088	-0 015	50 18	46 05	41 14	52 09	0 03	0 06	1 07
3 000	-0 314	-0 018	0 026	-0 186	-0 071	0 144	50 06	46 05	41 02	52 05	0 04	0 18	1 06
3 100	-0 315	-0 013	0 038	-0 011	0 049	0 127	50 14	46 25	41 01	52 06	0 06	0 09	1 02
3 200	-0 342	-0 020	0 053	-0 268	-0 071	0 146	50 22	46 22	41 22	52 26	0 06	0 23	1 04
3 300	-0 365	-0 030	0 038	-0 228	-0 097	-0 147	50 12	46 27	41 09	52 22	0 09	0 17	1 29
3 400	-0 381	-0 020	0 036	-0 159	0 101	-0 019	50 01	46 05	41 15	52 27	0 06	0 22	1 08
3 500	-0 388	-0 020	0 032	-0 077	0 003	-0 041	50 04	46 10	41 06	52 02	0 08	0 07	1 26
3 600	-0 401	-0 019	0 017	-0 127	0 012	-0 150	50 19	46 26	41 07	52 20	0 03	0 02	1 24
3 700	-0 396	-0 022	0 008	0 049	-0 036	0 090	50 13	46 19	41 05	52 12	0 10	0 15	1 02
3 800	-0 411	-0 012	359 999	-0 146	0 100	0 096	50 17	46 07	41 07	52 18	0 06	0 07	1 14
3 900	-0 435	-0 028	359 990	-0 248	-0 155	-0 087	50 28	46 28	41 00	52 25	0 08	0 28	1 05
4 000	-0 439	-0 028	359 986	-0 033	-0 008	-0 040	50 05	46 02	41 09	52 04	0 01	0 06	1 10
4 100	-0 439	-0 017	359 982	-0 002	0 117	-0 036	50 18	46 25	41 04	52 22	0 10	0 13	1 10
4 200	-0 442	-0 025	359 980	-0 036	-0 084	-0 023	50 05	46 06	41 16	52 08	0 03	0 22	1 06
4 300	-0 439	-0 033	359 989	0 031	-0 079	0 085	50 16	46 27	41 15	52 23	0 05	0 24	1 08
4 400	-0 438	-0 043	359 990	0 014	-0 102	0 011	50 03	46 24	41 17	52 07	0 03	0 22	1 21
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4 700	-0 466	-0 019	359 988	-0 106	-0 022	0 109	50 10	46 03	41 27	52 01	0 08	0 06	1 29
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4 900	-0 464	-0 031	359 970	-0 029	-0 139	-0 058	50 26	46 01	41 11	52 23	0 07	0 15	1 29
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5 400	-0 408	-0 009	359 985	0 015	0 040	0 098	50 08	46 25	41 08	52 26	0 01	0 05	1 23
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Figure B-1 Example Raw Data

5 600	-0 362	-0 016	359 997	0 239	-0 080	0 138	50 16	46 27	41 20	52 04	0 06	0 06	1 18
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5 800	-0 102	-0 025	0 002	2 052	-0 099	-0 068	50 10	46 01	41 29	52 25	0 03	0 22	1 01
5 900	0 185	-0 029	359 999	2 864	-0 046	-0 034	50 16	46 15	41 15	52 11	0 07	0 21	1 26
6 000	0 546	-0 021	0 004	3 613	0 081	0 053	50 05	46 28	41 04	52 24	0 05	0 16	1 26
6 100	1 024	-0 035	359 994	4 774	-0 142	-0 102	50 07	46 01	41 13	52 28	0 10	0 24	1 10
6 200	1 557	-0 046	359 981	5 334	-0 105	-0 131	50 26	46 25	41 04	52 05	0 05	0 18	1 28
6 300	2 204	-0 048	359 974	6 467	-0 019	-0 073	50 28	46 28	41 25	52 08	0 05	0 19	1 22
6 400	2 867	-0 039	359 967	6 629	0 089	-0 064	50 02	46 03	41 05	52 11	0 09	0 23	1 24
6 500	3 420	-0 034	359 969	5 532	0 045	0 020	50 25	46 18	41 19	52 28	0 06	0 04	1 19
6 600	3 875	-0 029	359 969	4 554	0 049	-0 001	50 27	46 03	41 22	52 18	0 10	0 27	1 26
6 700	4 172	-0 040	359 971	2 970	-0 110	0 020	50 28	46 19	41 13	52 26	0 01	0 22	1 10
6 800	4 328	-0 040	359 973	1 556	0 004	0 016	50 18	46 12	41 17	52 04	0 02	0 29	1 03
6 900	4 352	-0 048	359 959	0 245	-0 076	-0 135	50 21	46 16	41 30	52 22	0 09	0 08	1 16
7 000	4 244	-0 044	359 956	-1 080	0 039	-0 032	50 24	46 22	41 14	52 27	0 03	0 28	1 16
7 100	4 019	-0 032	359 956	-2 252	0 122	-0 004	50 18	46 17	41 17	52 13	0 07	0 12	1 18
7 200	3 647	-0 047	359 970	-3 724	-0 152	0 146	50 01	46 04	41 06	52 22	0 10	0 04	1 18
7 300	3 240	-0 050	359 967	-4 066	-0 037	-0 034	50 12	46 27	41 05	52 30	0 02	0 07	1 06
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8 300	-0 013	-0 033	359 926	-0 127	-0 027	0 069	50 22	46 29	41 07	52 26	0 05	0 20	1 27
8 400	0 055	-0 044	359 937	0 676	-0 110	0 115	50 17	46 17	41 05	52 14	0 02	0 04	1 20
8 500	0 226	-0 045	359 935	1 710	-0 012	-0 023	50 18	46 01	41 03	52 08	0 08	0 16	1 03
8 600	0 406	-0 046	359 945	1 804	-0 011	0 097	50 14	46 02	41 19	52 18	0 08	0 06	1 28
8 700	0 614	-0 057	359 948	2 081	-0 111	0 034	50 16	46 17	41 17	52 28	0 09	0 02	1 18
8 800	0 823	-0 067	359 952	2 091	-0 094	0 043	50 23	46 15	41 23	52 22	0 04	0 13	1 02
8 900	1 039	-0 056	359 959	2 154	0 112	0 067	50 26	46 16	41 14	52 07	0 08	0 07	1 13
9 000	1 191	-0 040	359 947	1 518	0 151	-0 120	50 03	46 25	41 11	52 15	0 08	0 15	1 22
9 100	1 290	-0 033	359 954	0 990	0 073	0 065	50 02	46 21	41 11	52 22	0 02	0 27	1 19
9 200	1 355	-0 037	359 959	0 654	-0 042	0 051	50 28	46 15	41 24	52 25	0 09	0 28	1 12
9 300	1 380	-0 051	359 957	0 247	-0 142	-0 014	50 29	46 18	41 05	52 18	0 03	0 05	1 19
9 400	1 327	-0 055	359 945	-0 527	-0 037	-0 128	50 06	46 17	41 00	52 05	0 04	0 18	1 12
9 500	1 228	-0 047	359 951	-0 994	0 087	0 064	50 23	46 06	41 15	52 17	0 02	0 21	1 19
9 600	1 075	-0 061	359 964	-1 530	-0 140	0 129	50 02	46 28	41 20	52 29	0 02	0 15	1 25
9 700	0 902	-0 067	359 957	-1 725	-0 062	-0 066	50 24	46 23	41 04	52 17	0 05	0 11	1 04
9 800	0 720	-0 069	359 954	-1 820	-0 019	-0 036	50 10	46 29	41 02	52 23	0 07	0 27	1 17
9 900	0 534	-0 058	359 940	-1 859	0 102	-0 137	50 12	46 07	41 05	52 11	0 07	0 29	1 09
10 000	0 351	-0 060	359 947	-1 831	-0 018	0 071	50 01	46 27	41 28	52 22	0 00	0 16	1 11
10 100	0 183	-0 054	359 952	-1 686	0 060	0 050	50 14	46 27	41 28	52 02	0 03	0 21	1 01
10 200	0 079	-0 040	359 956	-1 038	0 143	0 042	50 25	46 09	41 18	52 20	0 02	0 01	1 14
10 300	0 015	-0 024	359 960	-0 642	0 161	0 037	50 01	46 20	41 03	52 05	0 07	0 18	1 03
10 400	0 019	-0 038	359 962	0 042	-0 142	0 021	50 14	46 14	41 01	52 14	0 04	0 06	1 04
10 500	0 016	-0 036	359 948	-0 025	0 016	-0 140	50 30	46 08	41 06	52 14	0 01	0 16	1 04
10 600	0 015	-0 050	359 939	-0 011	-0 133	-0 089	50 27	46 08	41 16	52 02	0 08	0 30	1 17
10 700	0 047	-0 050	359 925	0 315	0 000	-0 140	50 25	46 16	41 06	52 02	0 09	0 29	1 14
10 800	0 082	-0 061	359 917	0 353	-0 108	-0 082	50 14	46 26	41 18	52 04	0 08	0 21	1 08
10 900	0 098	-0 054	359 921	0 162	0 067	0 039	50 06	46 19	41 28	52 25	0 05	0 25	1 11
11 000	0 117	-0 061	359 932	0 192	-0 072	0 114	50 10	46 29	41 06	52 10	0 01	0 14	1 24
11 100	0 094	-0 071	359 920	-0 232	-0 096	-0 119	50 17	46 05	41 03	52 06	0 10	0 16	1 18
11 200	0 089	-0 068	359 935	-0 056	0 025	0 147	50 25	46 05	41 17	52 16	0 04	0 29	1 14
11 300	0 065	-0 068	359 949	-0 240	0 003	0 145	50 01	46 16	41 13	52 09	0 07	0 22	1 00
11 400	0 024	-0 076	359 949	-0 408	-0 082	-0 004	50 22	46 05	41 11	52 27	0 02	0 28	1 21
11 500	0 000	-0 088	359 960	-0 239	-0 121	0 114	50 24	46 08	41 24	52 29	0 04	0 01	1 04
11 600	-0 030	-0 081	359 958	-0 297	0 072	-0 025	50 28	46 18	41 23	52 25	0 10	0 18	1 14
11 700	-0 076	-0 071	359 970	-0 461	0 098	0 120	50 16	46 29	41 11	52 20	0 05	0 22	1 14
11 800	-0 109	-0 073	359 960	-0 327	-0 021	-0 099	50 07	46 16	41 26	52 06	0 08	0 27	1 13
11 900	-0 155	-0 062	359 950	-0 463	0 114	-0 102	50 11	46 29	41 18	52 15	0 01	0 20	1 29
12 000	-0 189	-0 060	359 947	-0 345	0 019	-0 025	50 24	46 14	41 11	52 03	0 09	0 07	1 05
12 100	-0 220	-0 058	359 945	-0 309	0 022	-0 030	50 22	46 11	41 28	52 22	0 07	0 28	1 15
12 200	-0 257	-0 047	359 956	-0 364	0 109	0 119	50 25	46 05	41 15	52 24	0 02	0 27	1 29
12 300	-0 299	-0 044	359 959	-0 427	0 030	0 030	50 02	46 06	41 28	52 27	0 08	0 29	1 07
12 400	-0 342	-0 058	359 956	-0 422	-0 146	-0 038	50 24	46 07	41 29	52 21	0 05	0 19	1 15
12 500	-0 361	-0 061	359 959	-0 197	-0 025	0 035	50 29	46 02	41 03	52 04	0 04	0 29	1 15
12 600	-0 396	-0 064	359 949	-0 350	-0 035	-0 103	50 20	46 15	41 20	52 19	0 08	0 15	1 15
12 700	-0 434	-0 068	359 949	-0 376	-0 038	0 000	50 25	46 29	41 28	52 05	0 01	0 17	1 16
12 800	-0 461	-0 054	359 945	-0 268	0 144	-0 041	50 19	46 19	41 16	52 06	0 01	0 09	1 23
12 900	-0 480	-0 066	359 956	-0 198	-0 121	0 114	50 01	46 20	41 09	52 02	0 06	0 05	1 07
13 000	-0 504	-0 058	359 943	-0 241	0 081	-0 130	50 01	46 02	41 16	52 26	0 05	0 01	1 18
13 100	-0 525	-0 044	359 944	-0 206	0 135	0 006	50 05	46 16	41 20	52 25	0 01	0 24	1 01
13 200	-0 548	-0 032	359 932	-0 229	0 119	-0 121	50 25	46 14	41 22	52 01	0 06	0 07	1 10
13 300	-0 564	-0 037	359 946	-0 162	-0 043	0 140	50 16	46 16	41 26	52 04	0 07	0 01	1 30
13 400	-0 587	-0 034	359 945	-0 231	0 029	-0 008	50 22	46 08	41 16	52 18	0 01	0 19	1 06
13 500	-0 629	-0 034	359 938	-0 419	-0 008	-0 064	50 19	46 28	41 21	52 14	0 05	0 11	1 16
13 600	-0 667	-0 030	359 951	-0 379	0 046	0 126	50 14	46 26	41 23	52 17	0 00	0 03	1 09
13 700	-0 708	-0 028	359 955	-0 405	0 016	0 034	50 14	46 19	41 28	52 26	0 08	0 17	1 01

Figure B-1 (cont) Example Raw Data

APPENDIX C

COOPER-HARPER AIRCRAFT HANDLING QUALITY PILOT RATING SCALE

Figure C-1 The Cooper-Harper Aircraft Handling Quality Pilot Rating Scale

VITA

Chief Warrant Officer Four, Alan Davis, was born on Long Island, New York on October 12, 1962 He graduated from Sachem High School in Lake Ronkonkoma, New York in 1980 He enlisted in the United States Army in July 1980 and attended the Army's Initial Entry Rotary Wing Course at Fort Rucker, Alabama During his 20 year career as a Warrant Officer, he served as an Army level Standardization Instructor Pilot, accumulated 4000 flight hours and is qualified as a Master Army Aviator He is also a combat veteran of operations Earnest Will, Prime Chance, Desert Shield, and Desert Storm Alan is a graduate of Embry Riddle Aeronautical University with a Bachelor of Science degree in Professional Aeronautics Currently, he is pursuing a Master of Science degree within the Aviation Systems program at the University of Tennessee Space Institute (UTSI) with a specialization in Research and Development After completion of his degree at UTSI, he will attend the U S Navy Test Pilot School in Patuxent River, Maryland, after which he will continue to serve the U S Army as an experimental test pilot