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HANDLING QUALITIES EVALUATION OF A VARIABLE STABILITY NAVION AIRPLANE (N66UT) USING FREQUENCY-DOMAIN TEST TECHNIQUES

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

Randy Lee Bolding
December 1998

DEDICATION

This thesis is dedicated to my wife
Michelle, whose love, support and encouragement
give the strength for all of my accomplishments.
To my children, Joshua and Savannah, for the sacrifice of
a father's time.

ACKNOWLEGEMENTS

I would like to express my gratitude to Dr. William Lewis, Dr. Ralph Kimberlin, and Mr. Fred Stellar for their assistance in the preparation of this document. In addition, I would like to thank Charles Catterall for endless hours of computer programming and data reduction, and thanks to the airport staff in the preparation and instrumentation of the test aircraft.

ABSTRACT

General aviation small aircraft handling quality certification has predominately been accomplished using traditional time-domain test techniques. This thesis investigates the handling quality characteristics of the variable stability Navion airplane, tail number N66UT, using frequency-domain test techniques. N66UT is configured with conventional flight controls at the copilot's station and a fly-by-wire set of flight controls at the pilot's station. Time delays between the fly-by-wire and conventional flight controls were determined to be minimal. Evaluation of handling qualities of the aircraft were compared to the fixed wing flying qualities specified in MIL-HDBK-1797. Analysis of flight test was conducted using Comprehensive Identification from Frequency Responses (CIFER) program developed at NASA Ames Research Center, Moffett Field, California.

Frequency-domain system-identification methods are well suited to aircraft flight-control and handling-qualities analysis since many current design specifications, design and analysis techniques, and acceptance flight-test techniques are based in the frequency domain. The response characteristics of the Navion N66UT were stable for the longitudinal, lateral and rudder control inputs. The handling quality parameters met or exceeded proposed Level 1, Category C requirements, and Level 2, Category A requirements for fixed-wing military aircraft. The fly-by-wire system closely mirrored conventional control behavior well beyond bandwidth frequency. Time delay differences between conventional and FBW systems ranged from very near 0 to 9.5 ms. Throttle response data results were inconclusive. Further flight-testing is required.

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NOMENCLATURE

Δ	Difference (delta)
δ_r	Change in Rudder Deflection
ϕ	Roll Attitude
$L_{\delta a}$	Rolling Moment due to Change in Aileron Deflection
$M_{\delta e}$	Pitch Moment due to Change in Elevator Deflection
ms	Milliseconds
$N_{\delta r}$	Yaw Moment due to Change in Rudder Deflection
θ	Pitch Attitude
ω_{BW}	Bandwidth Frequency
ω_{135}	Frequency at 45 degree Phase Margin
ω_{180}	Frequency at 180 degree Phase
τ_p	Phase Delay
$X_{\delta t}$	x-axis Acceleration due to Change in Throttle
ψ	Yaw Attitude

Abbreviations

A/D	Analog to Digital
Accel	Acceleration
AFDD	Aeroflightdynamics Directorate

Abbreviations (continued)

ax	x-axis acceleration
ay	y-axis acceleration
az	z-axis acceleration
C.G.	Center of Gravity
C-H	Cooper-Harper
CIFER	Comprehensive Identification from Frequency Response
COMPOSITE	Multi-window Spectral Analysis
dB	Decibels
Deg	Degree
Deg/s^2	Degrees per second squared
DFC	Direct Force Control
^oC	Degrees Celsius
FBW	Fly By Wire
FCS	Flight Control System
FRESPID	Frequency Response Identification
ft	Feet
FW	Fly-by-Wire
FWD	Forward
HD	Density Altitude
Hz	Hertz

Abbreviations (continued)

IAS	Indicated Air Speed
In.	Inch
Kbytes	Kilobytes
KCAS	Knots Calibrated Air Speed
KHz	Kilo-hertz
lbs	Pounds
MAC	Mean Aerodynamic Chord
MIMO	Multi-input Multi-output
MISO	Multi-input Single output
MISOSA	Multi-input Single-output Spectral Analysis
NASA	National Aeronautics and Space Administration
OAT	Outside Air Temperature
p	Roll Rate
PC	Personal Computer
PIO	Pilot Induced Oscillation
q	Pitch Rate
r	Yaw Rate
rad/sec	Radians/second
rpm	Revolutions-Per-Minute
sec	Second

Abbreviations (continued)

SISO	Single-input Single-output
ST	CIFER Designation for Conventional Flight Controls
Std	Conventional Flight Controls
U.S.	United States
UTSI	University of Tennessee Space Institute
wt	Weight

Chapter I

INTRODUCTION

In July 1991, the University of Tennessee Space Institute (UTSI) purchased two Navion aircraft from Princeton University. These aircraft contain several modifications for conducting flight research funded by the Department of the Navy. Each aircraft is equipped with a fly-by-wire (FBW), variable control response-feedback system utilizing electronically signaled, hydraulically actuated controls located at the copilot station, and conventional flight controls at the pilot station. The FBW flight control system allows direct independent control over the aircraft six degrees of freedom through a direct force control (DFC) flight computer. The fly-by-wire system was designed and assembled using individual components purchased and integrated to form a variable stability system. Differences may exist between the fly-by-wire and conventional flight controls due to control implementation techniques.

These aircraft provide an excellent platform for performance, stability, and control test instruction for the University's Flight Research Department. For these aircraft to provide a suitable flight test platform using the variable stability systems, a thorough documentation of the basic handling characteristics of the aircraft configured with null settings for all controls is required. Data gathered from basic aircraft handling qualities tests can then be used to further investigate control responses in an attempt to model several variable settings. The responses due to various settings provide the

capability to replicate several different type aircraft flight characteristics using a single platform.

The purpose of this thesis is to evaluate the handling quality characteristics of the Ryan Navion, tail number N66UT, using frequency-domain test techniques. Short-term responses to control inputs are the primary focus for this thesis. The investigation explored differences between the two control station responses. Evaluation of aircraft handling qualities derived from the Navion are compared to the fixed wing flying qualities outlined in MIL-HDBK-1797 [1].

Analysis of frequency domain data is conducted using the Comprehensive Identification from FrEquency Response (CIFER®) [2] program developed at NASA Ames Research Center, Moffett Field, California. CIFER has been used extensively for helicopter and engine analysis, but not applied to fixed wing aircraft.

Chapter II

BACKGROUND

FLIGHT DYNAMICS AND HANDLING QUALITIES

For the majority of flight dynamics investigations time domain analysis is usually adequate, particularly when the test aircraft is a classical, unaugmented airplane. When the test airplane is an advanced modern airplane fitted with an advanced control system, flight dynamics analysis in the frequency domain can provide additional, valuable insight into its behavior. For an aircraft configured with a fly-by-wire system, the control system becomes an integral part of the primary signal flow path and the influence of its dynamic characteristics on flying and handling qualities is of critical importance. [3]

The flying and handling qualities of an airplane are those properties that describe the ease and effectiveness with which it responds to commands in the execution of some flight task. When handling qualities are described qualitatively and are formulated in terms of pilot opinion, they tend to be somewhat subjective. Flight dynamics involves the relatively short-term motion of an airplane in response to control input or to an external disturbance such as atmospheric turbulence. The motion of interest can vary from small excursions about trim to very large amplitude inputs for maneuvering when normal aerodynamic behavior may become very non-linear. The dynamic behavior of an airplane is significantly shaped by its stability and control properties, which in turn have their roots in the aerodynamics of the airframe. Modern flight dynamics is concerned not

only with the dynamics, stability and control of the basic airframe but also with interaction between airplane and flight control systems. [3]

A measure of the handling qualities of an aircraft is its stability margin when operated in a closed-loop-tracking task. The maximum frequency at which a closed-loop tracking can take place without degrading response characteristics is known as bandwidth (ω_{BW}). Aircraft capable of operating at a sufficiently large value of bandwidth will have superior performance when responding to disturbances. A bandwidth criterion is especially useful for highly augmented aircraft in which the response characteristics have a non-classical form. The concept of using bandwidth for flying qualities assessment has been proven. A 1970 utilization of bandwidth was a part of Neal-Smith [4] criterion consisting of empirical bounds on the closed-loop pitch attitude resonance $| \theta/\theta_c |_{max}$ versus pilot equalization for a piloted closure designed to achieve a specified bandwidth. Experience with this criterion has shown that the results can be sensitive to the selected value of closed-loop bandwidth. [5]

System identification is a procedure for accurately characterizing the dynamic response behavior of a complete aircraft, subsystem, or individual component from measured data. Frequency-domain identification approaches are especially well suited to the development and validation of flight-control systems. Feedback stability and noise amplification properties are determined from the open-loop frequency response, and characterized by metrics such as crossover frequency, and associated gain and phase margins. Command tracking performance is determined from the closed-loop frequency response, and characterized by metrics such as bandwidth and time-delay. Frequency-

domain identification approaches allow the direct and rapid identification of these frequency responses and metrics, without the need to first identify a parametric (state-space) model structure such as is required in applying time-domain methods.

The availability of comprehensive and reliable computational tools has substantially enhanced the acceptability of frequency-domain techniques in the flight-control and flight-test communities. Benefits from applying these techniques include the reduction of flight-test time required for control system optimization and handling-qualities evaluation, especially for complex control-law architectures, and improvements in the final system performance. Frequency-domain methods offer a transparent understanding of component and end-to-end response characteristics that can be critical in solving system integration problems. [6]

LITERATURE REVIEW (Specifications)

MIL-F-8785C [7].

MIL-F-8785C specifications contain the requirements for the flying and handling qualities, in flight and on the ground, of U.S. Military, manned, piloted airplanes. It is intended to assure flying qualities that provide adequate mission performance and flight safety regardless of design implementation or flight control system mechanization. [5] This specification defines requirements for the conventional aircraft, as tested using traditional time-domain test techniques. Testing the variable-stability Navion, an aircraft with direct force controls (DFC) which allow independent control over the six inertial degrees of freedom, exceeds the scope of MIL-F-8785C.

MIL-HDBK-1797.

MIL-STD-1797A has been redesignated as a handbook, and is to be used for guidance purposes only. The document is no longer to be cited as a requirement. The only physical change from MIL-STD-1797A was the cover page. This handbook is intended for use with fixed-wing aircraft, including a category pertaining to the Navion. The MIL Standard and Handbook are presented in terms of aircraft response or control axes. This change from MIL-F-8785C is designed to accommodate highly augmented aircraft, which is a primary objective of MIL-HDBK-1797 [1]. Within MIL-HDBK-1797, a wide variety of closed-loop tasks have been developed for the evaluation of aircraft flying qualities. Most tasks evaluated within the handbook are based on historical data and research experiments with ground-based and in-flight simulators. The handling characteristics described within the handbook are specified in terms of qualitative degrees of suitability and levels.

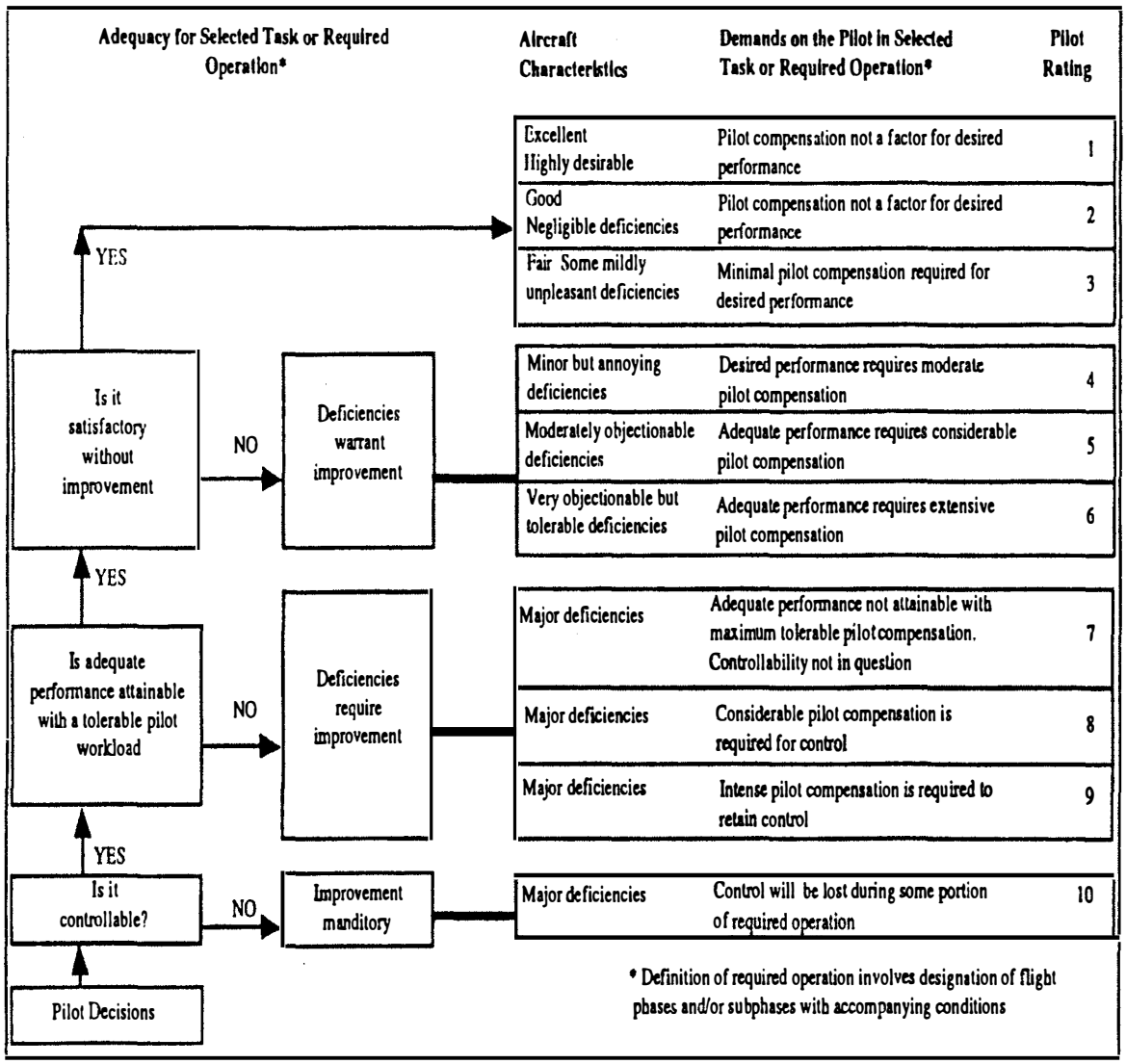


Figure 2-1. Cooper-Harper Handling Qualities Rating Scale [6]

The degrees of suitability are defined using the Cooper-Harper (C-H) rating scale (Figure 2-1) as follows:

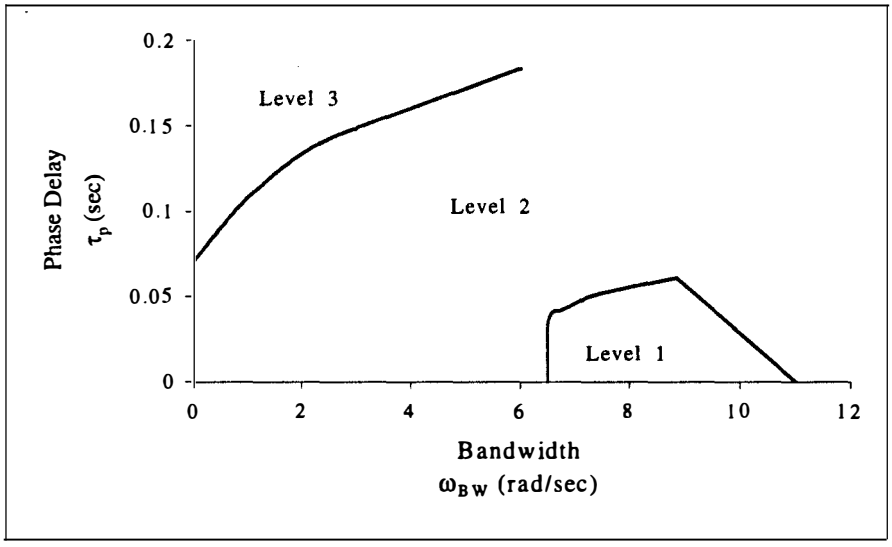
- Satisfactory: Flying qualities clearly adequate for the mission Flight Phase. Desired performance is achievable with no more than minimal pilot compensation. (C-H ratings from 1 through 3)
- Acceptable: Flying qualities adequate to accomplish the mission Flight Phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists. (C-H ratings from 4 through 6)
- Controllable: Flying qualities such that the aircraft can be controlled in the context of the mission Flight Phase, even though pilot workload is excessive or mission effectiveness is inadequate, or both. The pilot can transition from Category A Flight Phase tasks to Category B or C Flight Phases, and Category B and C Flight Phase tasks can be completed. (C-H ratings from 7 through 9)

Level 1 is Satisfactory, Level 2 is Acceptable, and Level 3 is Controllable.

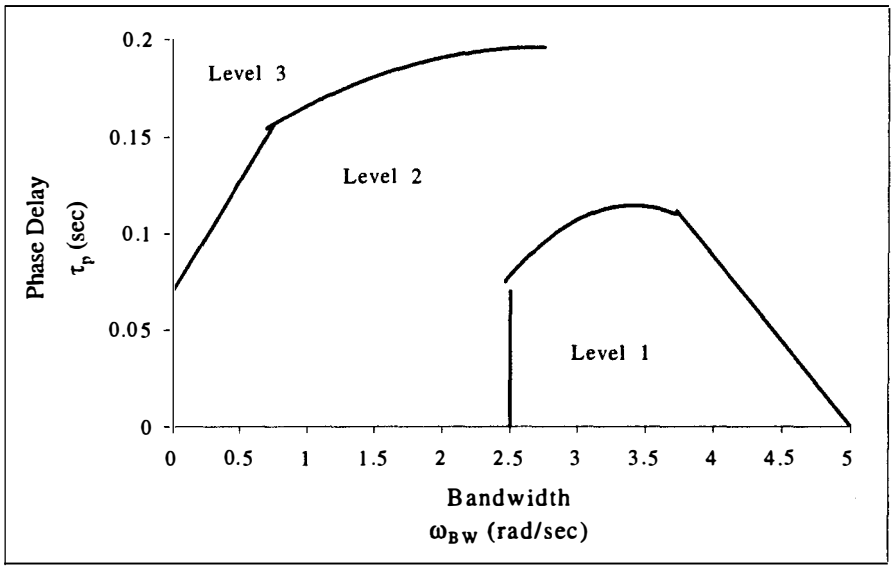
The criteria in the quantitative open-loop requirements are based on interpolation between and extrapolation beyond the configurations, flight conditions, and tasks that have been evaluated in the existing database. Pilot handling qualities ratings have been consolidated and boundaries established based on bandwidth and phase margins. It is normally recommended that more than one evaluation be conducted for each test condition, using between three to six evaluation pilots for time-domain flight testing techniques. With the use of frequency domain test techniques and specifications, one

instrumented flight has the ability to capture the required data with very little subjectivity placed on the pilot.

The bandwidth of the open-loop pitch attitude response to pilot control force is recommended to be within the bounds shown in Figure 2-2. Bandwidth (ω_{BW}) is the highest frequency at which either the response of aircraft pitch attitude to pilot control-force and control-deflection inputs have 45 degrees or more of phase margin (ω_{135}) or 6 dB of gain margin, whichever comes first. Individual specifications for uncoupled roll, yaw, and throttle axis responses are not presented in the handbook. For the purpose of this thesis, Figure 2-2 specifications will be used for all axis evaluations.



(a) Category A Requirements



(b) Category C Requirements

Figure 2-2. Bandwidth Requirements [1].

To use Figure 2-2 charts, determine whether the response being evaluated is designated a Category A or Category C maneuver. Enter the appropriate chart at the bandwidth limit (x-axis). Move up until reaching the phase delay value associated with this bandwidth limit (y-axis). Read the Level designation assigned to this combination of values.

Flight Phase Definitions:

Non-terminal Flight Phases:

Category A: Those non-terminal Flight Phases that require rapid maneuvering, precision tracking, or precise flight-path control. Included in this category are:

- Air-to-air combat (CO)
- Ground attack (GA)
- Weapon delivery/launch (WD)
- Aerial recovery (AR)
- Reconnaissance (RC)
- In-flight refueling (receiver) (RR)
- Close formation flying (FF)
- Low-altitude parachute extraction (LAPES) delivery

Category B: Those non-terminal Flight Phases that are normally accomplished using gradual maneuvers and without precision tracking, although accurate flight-path control may be required. Included in this category are:

- Climb (CL)
- Cruise (CR)
- Loiter (LO)
- In-flight refueling (tanker) (RT)
- Descent (D)
- Emergency descent (ED)
- Emergency deceleration (DE)
- Ariel delivery (AD)

Terminal Flight Phases:

Category C: Terminal Flight Phases are normally accomplished using gradual maneuvers and usually require accurate flight-path control. Included in this category are:

- Takeoff (TO)
- Approach (PA)
- Waveoff/go-around (WO)
- Landing (L)

Chapter III

FLIGHT TEST TECHNIQUE

DESCRIPTION OF TEST AIRCRAFT

General Description

The aircraft used for this flight test was a Navion, Serial Number NAV-42013, Registration Number N66UT, manufactured by Ryan Aeronautical Company, San Diego, California. (Figure 3-1) The Ryan Navion is a low-wing four-place, dual-control airplane powered by a single air-cooled engine. The fuselage is an all-metal one-piece semi-monocoque structure. The tail unit is a cantilever monoplane type with detachable tips. There is a two piece tailpane and elevators that are interchangeable left and right. Normal fuel capacity is 40 U.S. Gallons. A general description of the Navion is contained in reference 8. Aircraft specifications are listed in Table 3-1. A Navion three-view is presented in Figure 3-2. N66UT is owned and maintained by the University of Tennessee Space Institute, Tullahoma, Tennessee.

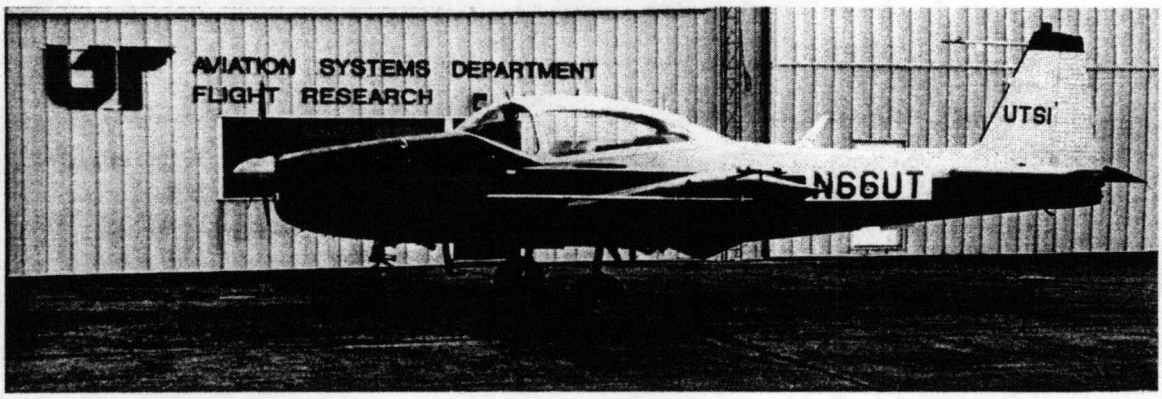


Figure 3-1. Ryan Navion N66UT

Table 3-1. Navion Specifications

Aircraft:	
Length	27 ft, 6 in. (8.38 m.)
Wing:	
Span (b)	33 ft, 5 in. (10.18 m.)
Area (S)	184 ft^2 (17.112 m^2.)
Sweep, leading edge	2.996 deg.
Aspect Ratio (A)	6.04
Taper Ratio (λ)	0.54
Mean aerodynamic chord	5.7 ft (1.74 m.)
Dihedral	7.5 deg.
Airfoil:	
Tip	NACA 6410
Root	NACA 4415
Wing loading	14.6 lb./sq. ft. (71.2 kg./$m.^2$)
Horizontal tail:	
Area	43 ft^2 (4.0 $m.^2$)
Aspect ratio	4.0
Taper ratio	0.67
Airfoil	NACA 0012
Vertical tail:	
Area above horizontal stabilizer	15.5 ft^2 (1.44 $m.^2$)
Rudder area	8.33 ft^2 (0.78 $m.^2$)

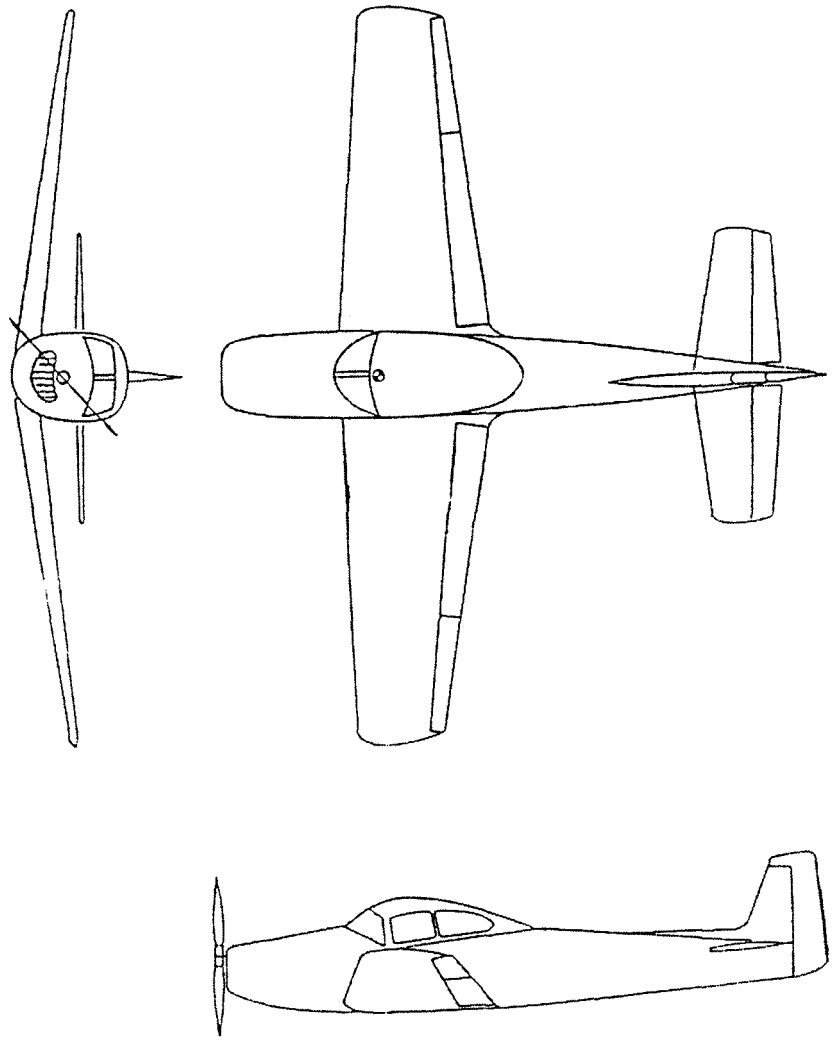


Figure 3-2. Navion Three-view

N66UT has several modifications made by Princeton University under various United States Navy Grants that contrast this aircraft from the conventional Navion. The engine is a Teledyne-Continental IO-520B producing 285 take-off horsepower at sea level, 2700 RPM. The propeller is a McCauley three-bladed constant speed propeller, part number D3663/582NC-2, serial number K103943YS. The conventional Navion main landing gear struts are replaced with those designed for the Camair twin (a Navion conversion with nearly 40 percent increase in gross weight). N66UT landing gear is permanently fixed in the down (extended) position. The system can be adjusted for landing tests to provide and allowable sink rate of 12.5 feet per second.

Conventional Flight Controls

The conventional flight control system is located at the copilot's station. This system consists of a set of rudder pedals, a wheel for elevator and aileron control, and cables and linkage connected to the respective control surfaces. The control column, to-which the control wheel shafts (through universal joints) are attached, pivots at the base to permit fore and aft movement (Figure 3-3). Sprockets on the forward end of each control wheel shaft are interconnected by a chain, the ends of which attach to cables routed through pulleys at the top and bottom of the column. The control wheel and shaft on the left side have been removed. The rudder pedal assembly is hinged to the floor (Figure 3-4). The conventional rudder pedal assembly at the left pilot's station has been removed. The rudder control system consists of two cable assemblies, connected to rudder pedal torque tube arms, and running aft to the rudder horns. Two rods, extending forward from the pedals to the bellcrank for nose wheel steering, serve as a balance cable

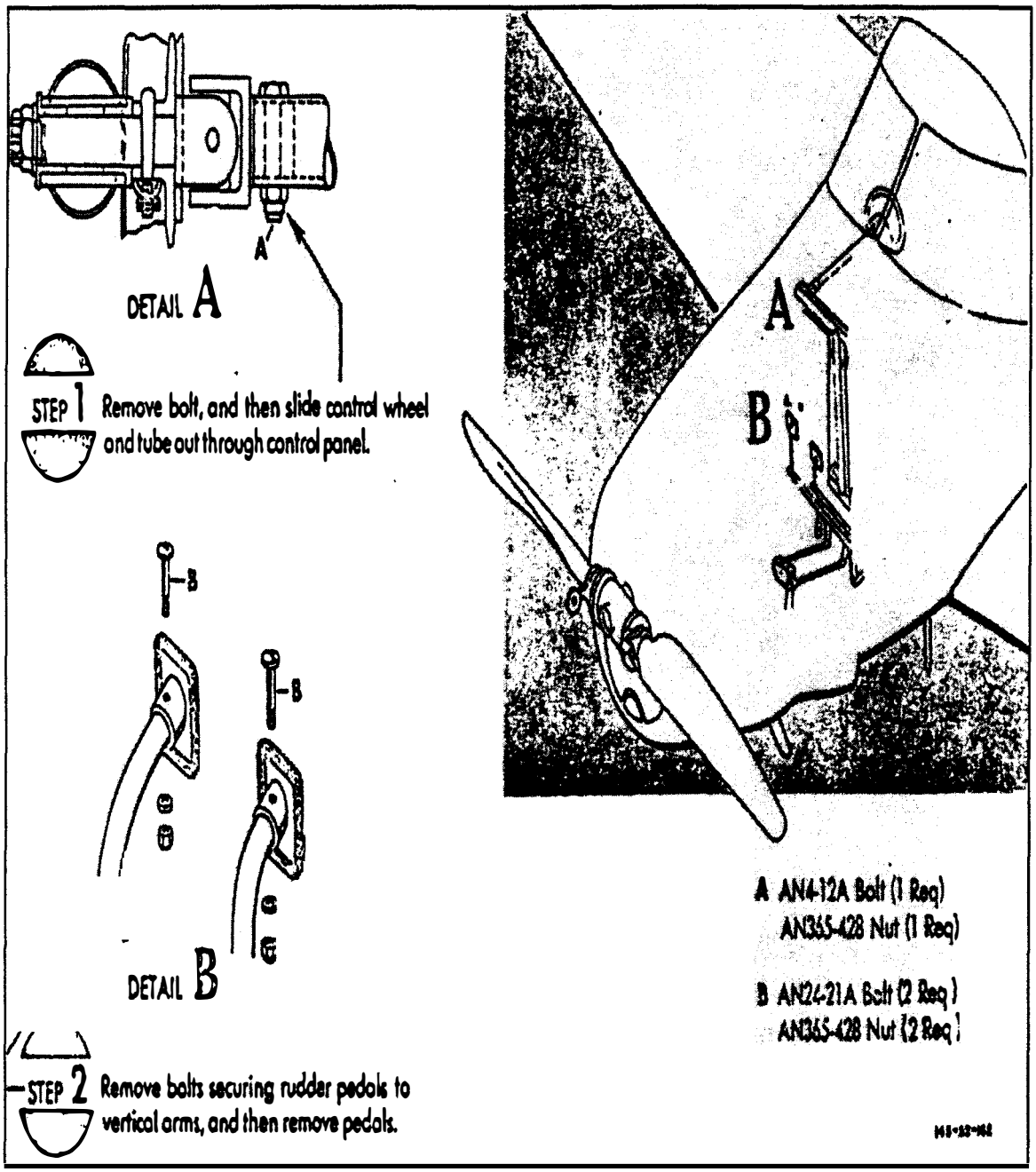


Figure 3-3. Conventional System Control Column [8]

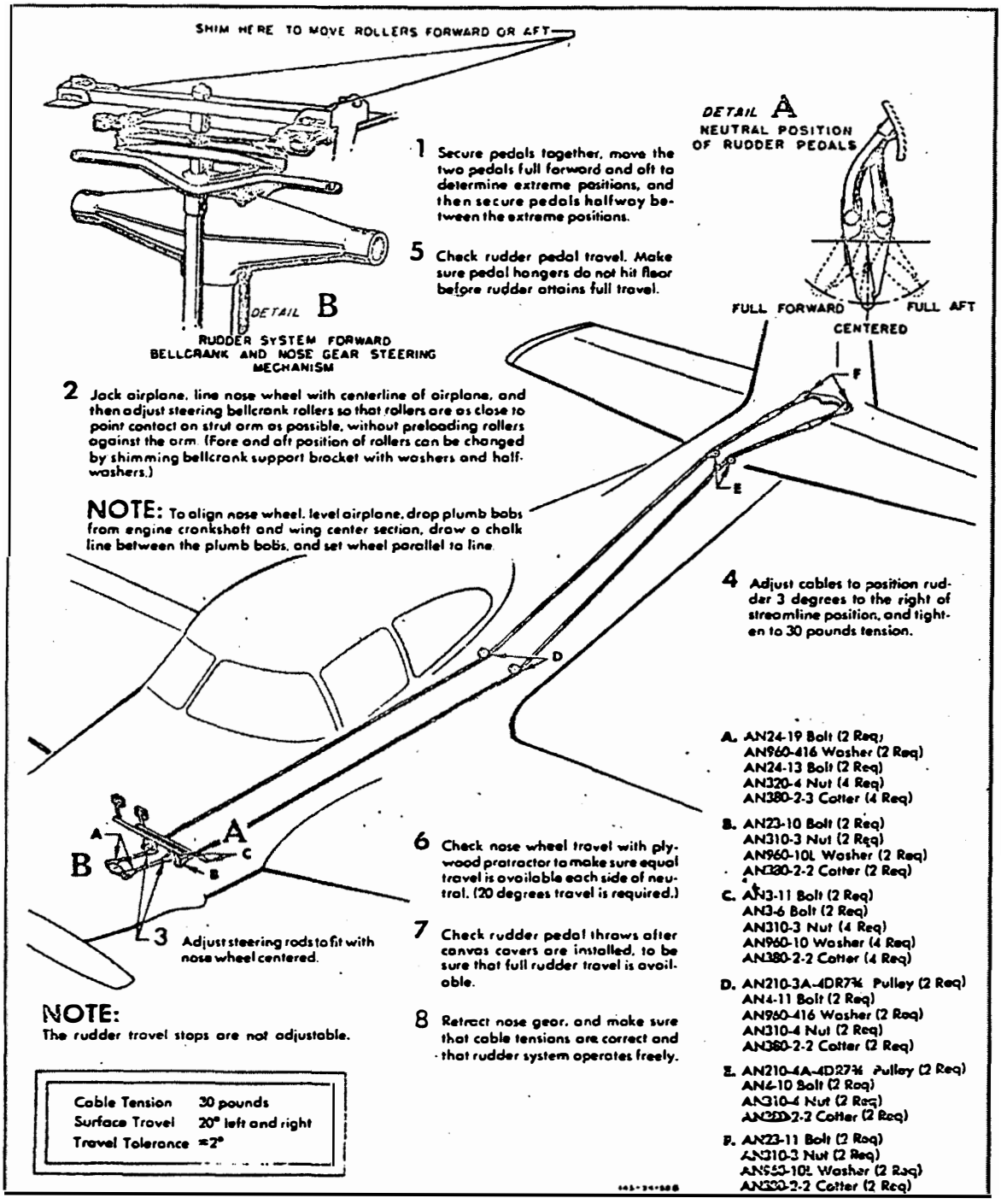


Figure 3-4. Conventional Rudder Control System [8]

for the system. The two rudder-to-aileron coordinating cables and spring assemblies have been removed from N66UT.

The ailerons are controlled by a combination linkage and cable system (Figure 3-5). Disconnect fittings are located within the control cable guard box on the pilot's floor, and turnbuckles are located in the right wheel well. Adjustable rods connect the bellcranks to the ailerons. The elevator control system consists of two cable assemblies, connecting the control column arm with the elevator horn (Figure 3-6). Cables pass through the empenage guided by a series of pulleys and connect to the elevator horn.

Fly-By-Wire Flight Controls

The pilot's station has been equipped with an analog fly-by-wire control system containing power-actuated control surfaces commanded by electrical signals. The signals come from the various cockpit controllers and motion sensors, and when appropriately processed and summed, provide a net signal to each servo-actuator resulting in an airplane response of a particular character and magnitude. The servos are hydraulic, supplied by an engine-driven hydraulic pump delivering approximately 9 gallons-per-minute at 750 pounds-per-square-inch pressure. Control of pitch, roll, and yaw are through conventional elevator, aileron, and rudder control surfaces. The full authority of each surface is available and the maximum deflection rate is approximately 70 degrees per second. The hydraulic servos are modified Bendix units originally designed for the Convair B-58 Hustler and incorporate built-in solenoids and pilot force-override disengage features. Servo response parameters are presented in Table 3-2.

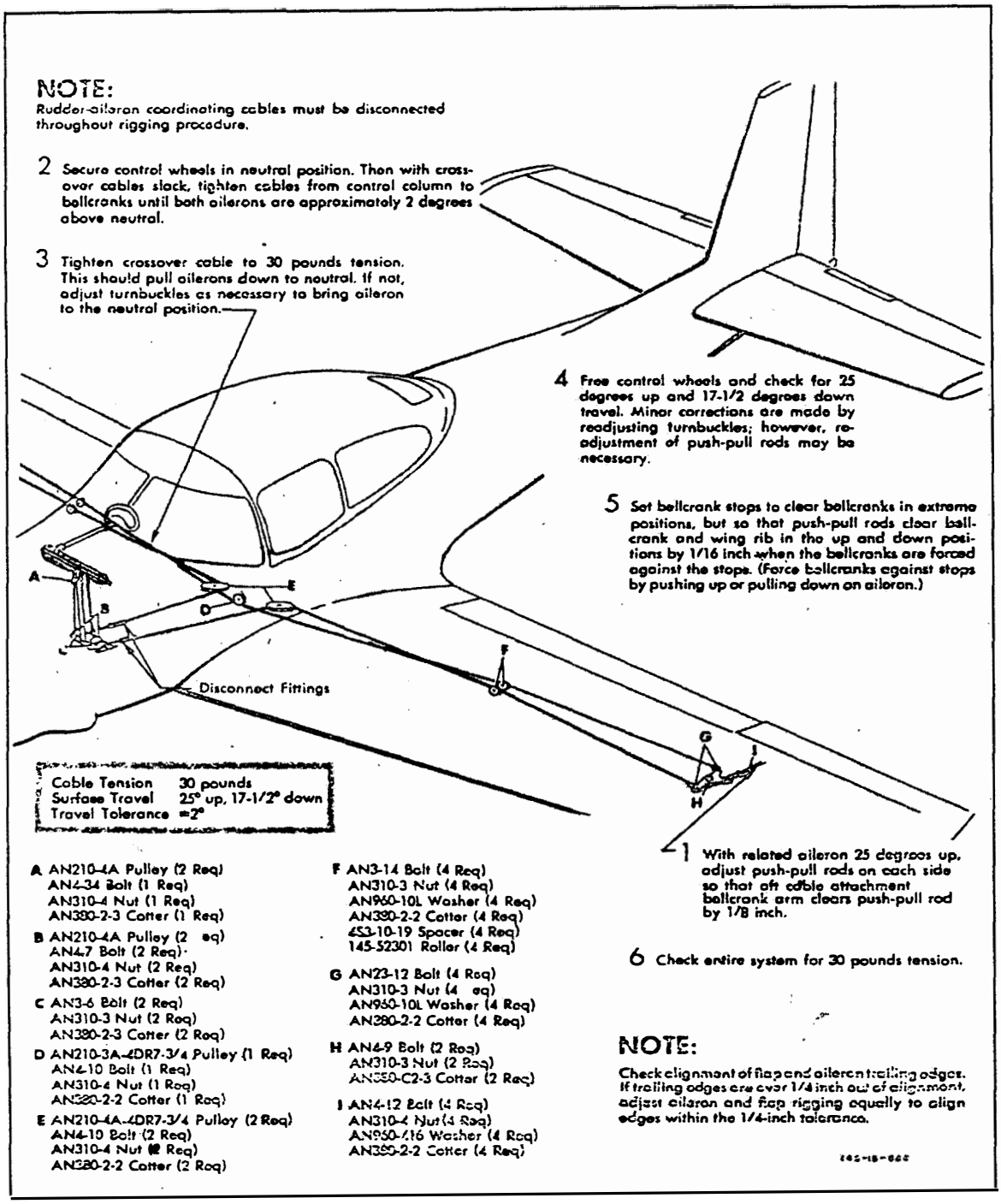


Figure 3-5. Conventional Aileron Control System [8]

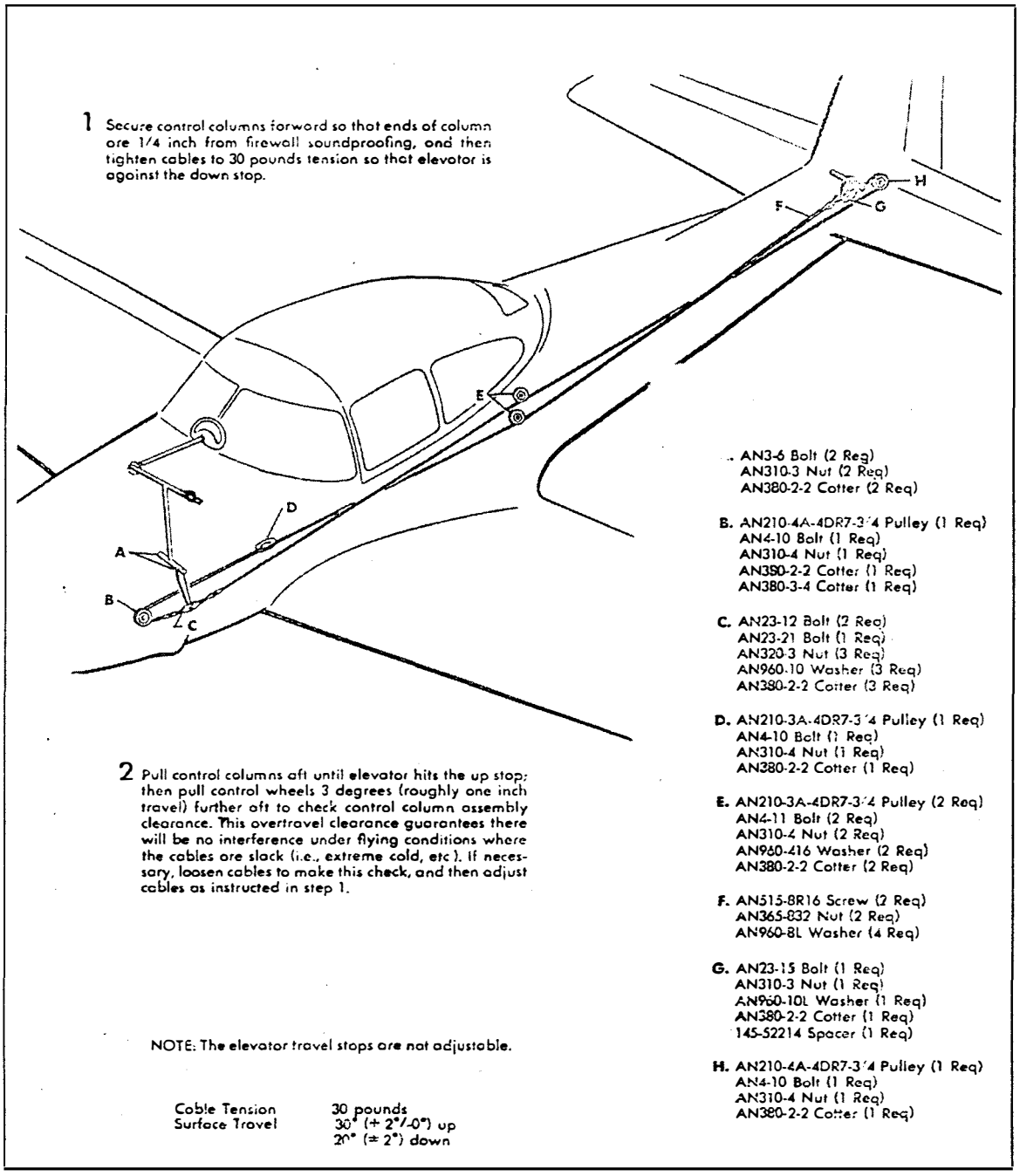


Figure 3-6. Conventional Elevator Control System [8]

Table 3-2. Servo Response [9]

Control	Displacement Limit, deg	Rate Limit, Deg/sec	Bandwidth, Flat, (6 dB Down) H_Z	Maximum Specific Force or Moment (IAS = 105)
Pitch	- 30 + 20	70	5(10)	9.9 rad/sec^2
Roll	± 20	70	5(10)	9.2 rad/sec^2
Yaw	± 20	70	5(10)	4.2 rad/sec^2
Thrust	-	-	0.6 *	.05 g
Normal	± 30	110	2(3)	1.1 g

*Limited by aircraft engine to a first order time constant of 0.25 seconds.

The FBW elevator and aileron systems incorporate redundant control channels. With the redundant channels, any substantial error between the commanded and actual control position is detected, and a switchover to a second servo is made. The fact that a channel has switched to the secondary servo is communicated to the pilot by a warning light. An "abort mode" disengagement system is installed which is activated pressing a disengagement thumb switch on the pilot's yoke. [9]

A block diagram of the FBW rudder control system is presented in Figure 3-7. Inputs to the rudder are rudder position (δ_r), lateral accelerations (ay), and yaw (ψ). Aileron, elevator, and throttle schematics are not presented in this document, but are of similar design as the rudder system. The location of the rudder hydraulic actuator is presented in Figure 3-8. One end of the actuator is connected to a bracket that is mounted to the fuselage, and the other end is connected directly to a control horn on the right side of the rudder.

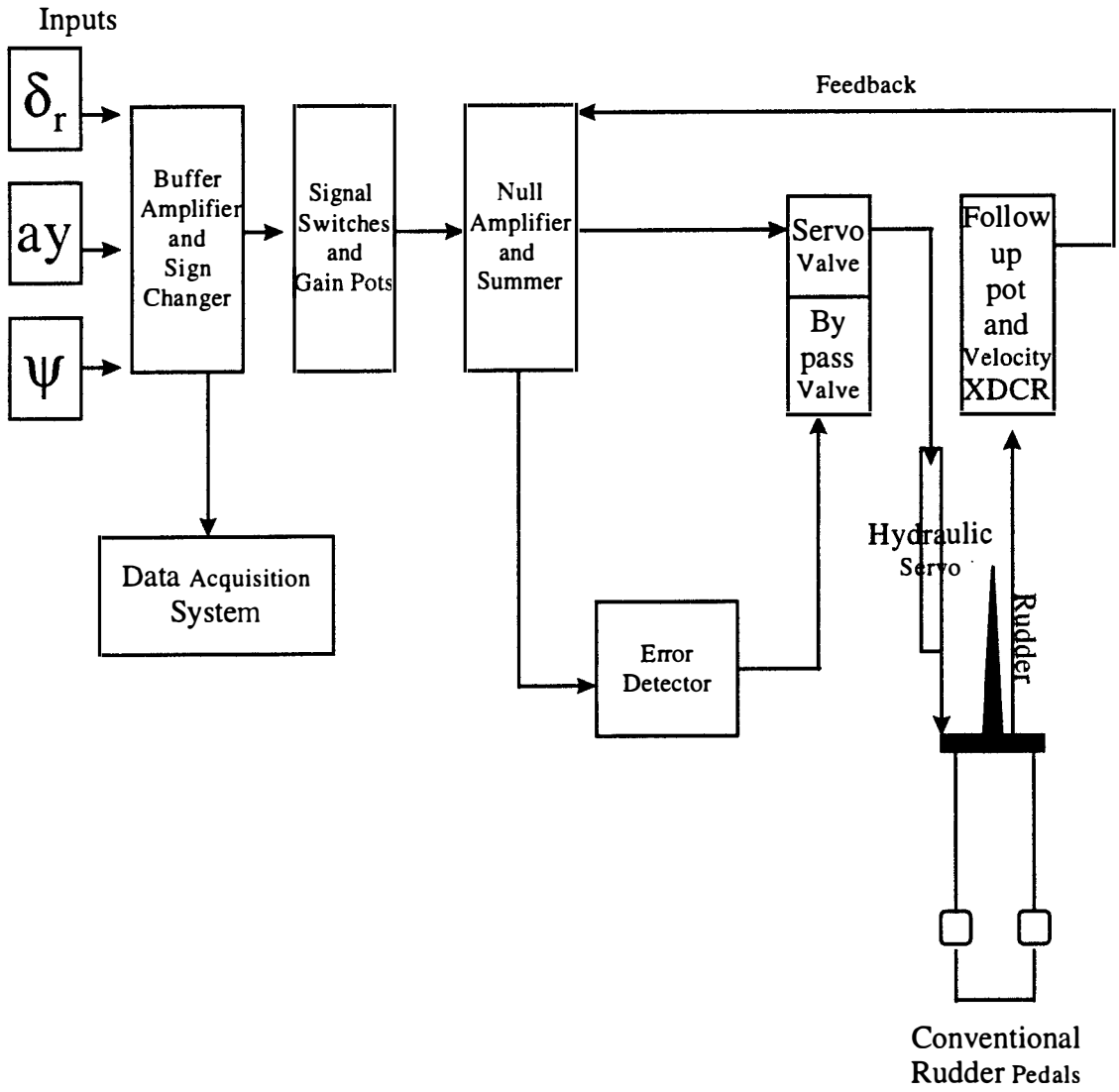


Figure 3-7. Fly-By-Wire Rudder Block Diagram

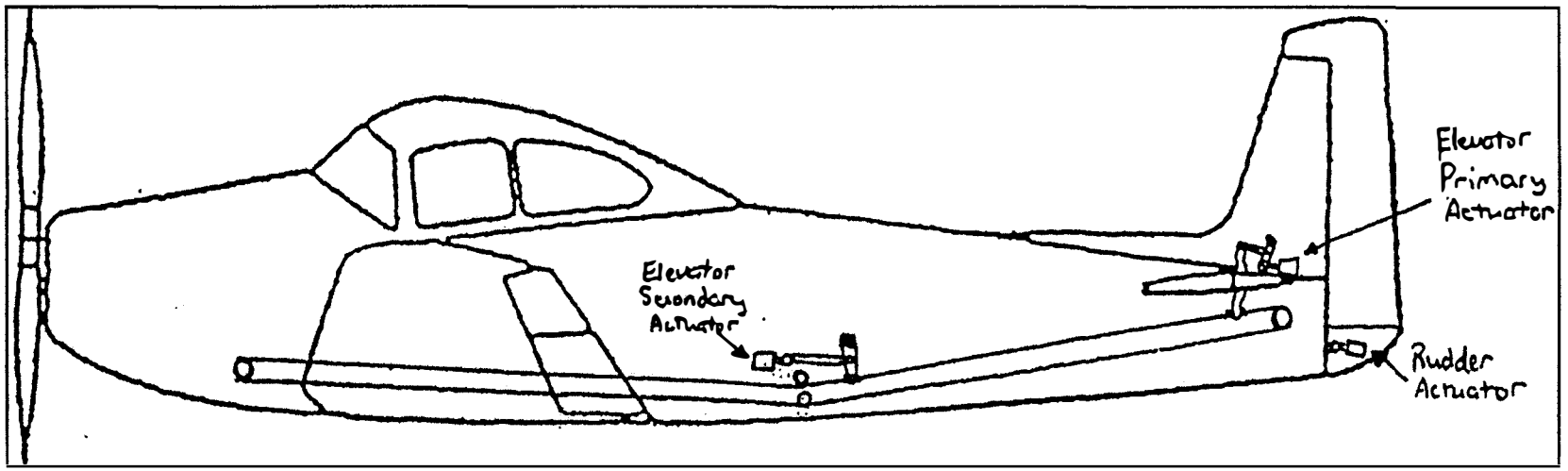


Figure 3-8. Fly-By-Wire Rudder and Elevator Diagram

The FBW elevator control system consists of two hydraulic servos, a primary and a secondary. During engagement of the system, a logic system allows for pressurization of the primary actuator for control inputs, and by-pass of the secondary actuator. If a failure of the primary actuator is sensed, the secondary actuator is pressurized and the primary actuator is placed in the by-pass mode. Should the logic system sense failure of both actuators, both are placed in the by-pass mode and the aircraft must be controlled using the conventional control system. The locations of the elevator control servos are presented in Figure 3-8. The primary actuator is located at the base of the empennage. This servo has a five-inch control horn mounted on the elevator and is connected to a bellcrank attached to the actuator. This control horn is connected to the underside of the elevator. The secondary hydraulic actuator is located just aft of the wing assembly at 173 inches aft of datum, housed within the fuselage. The secondary actuator has an eight-inch control rod connected to a control horn. This control horn is connected to the conventional control system elevator cable.

The FBW aileron control system consists of a primary actuator mounted on the underside of the right wing, and a secondary actuator mounted to the underside of the left wing (Figure 3-9). The operational priority system of the actuators is the same as stated previously for the elevator system. The actuators are connected to a bellcrank by a two-inch rod. A nine-inch rod connects the bellcrank to the aileron surface bottom side. The available inputs to each of these controls are shown in Table 3-3. Independent control over heave is exercised through the Navion flap, modified to deflect up, as well as down through a $\pm$ 30 degree range. The upward motion provides increased lift modulation

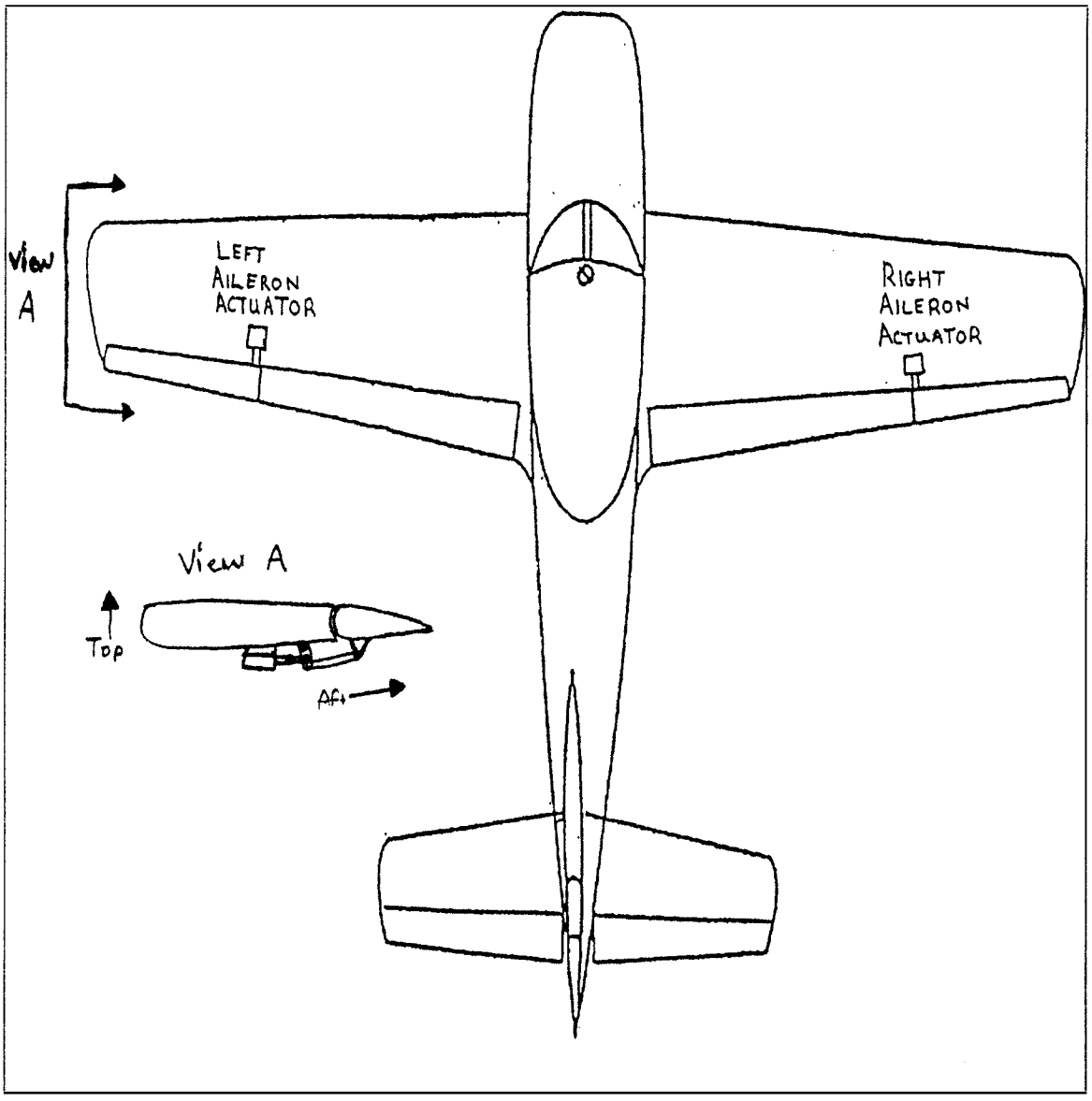


Figure 3-9. Fly-By-Wire Aileron Control System

Table 3-3. Inputs of Moment Controls [9]

CHANNEL	INPUT	FUNCTION VARIED
Pitch	Control stick displacement	Control sensitivity
	Thrust lever	Simulated moment due to thrust
	Column thumbwheel	Simulated DLC moment
	Radar altitude	Ground effect moment
	Airspeed	Speed stability
	Angle of attack	Static stability
	Pitch attitude	Attitude hold sensitivity
	Pitch rate	Pitch damping
	Flap angle	Trim change from flap
	Flap rate	Moment from flap rate
	Simulated turbulence	Turbulence response
Roll	Lateral stick displacement	Control sensitivity
	Sideslip	Dihedral effect
	Roll rate	Roll damping
	Yaw rate	Roll due to yaw rate
	Rudder pedal displacement	Roll due to rudder
	Simulated turbulence	Turbulence response
Yaw	Rudder pedal displacement	Control sensitivity
	Sideslip	Directional stability
	Yaw rate	Yaw damping
	Roll rate	Yaw due to roll rate
	Wheel displacement	Yaw due to aileron
	Simulated turbulence	Turbulence response

authority and tends to minimize the problems of drag and angle of zero lift changes. Actuation is hydraulic, with a maximum available surface rate of 110 degrees per second. Inputs to normal force control presently available are shown in Table 3-4. Thrust and drag modulation is by a hydraulic servo on the engine throttle. The engine rpm is maintained by the propeller governor, which adjusts the blade pitch for the constant speed propeller. Inputs to the thrust/drag modulation system are shown in Table 3-5.

Conventional Engine Control System

For the conventional control system, the engine is controlled from the cabin by flexible individual push-pull controls (Figure 3-10). These push-pull controls are connected to the carburetor throttle lever, manual mixture and idle cut-off lever, and the carburetor air control lever on the bottom of the air mixing chamber. Modifications have been made to incorporate two hydraulic actuators in series for operation of the FBW system. When the FBW system is disengaged, these actuators operate as fixed links within the system. Operation of the actuators during FBW engagement is explained in the subsequent FBW paragraph. A six and twelve-inch push-pull rods are used to incorporate the actuators within the throttle system. Because N66UT is equipped with a Hartzell propeller, an additional flexible control is installed for regulating the propeller governor. Control knobs normally mounted on the cabin control panel have been replaced by lever style controls mounted on the right wall next to the copilot. The conventional throttle control system is presented in Figure 3-11, labeled number 1.

Table 3-4. Inputs to Normal Force Control [9]

INPUT	FUNCTION VARIED
Control stick displacement	Lift due to control (simulated elevator lift, or direct lift control integrated with column)
Thrust lever displacement	Lift due to thrust, direct lift control integrated with throttle
Radar altitude	Ground effect/lift; wind gradients
Angle of attack	Lift response to angle of attack, lift change at stall
Simulated turbulence	Turbulence response

Table 3-5. Inputs to Thrust/Drag Modulation System [9]

INPUT	FUNCTION VARIED
Control stick displacement	Drag due to control (simulated control surface drag)
Thrust lever displacement	Thrust command/throttle sensitivity
Radar altitude	Ground effect drag changes; wind gradients
Airspeed	Drag change with speed
Angle of attack	Drag change with angle of attack
Flap displacement	Drag due to flap deflection

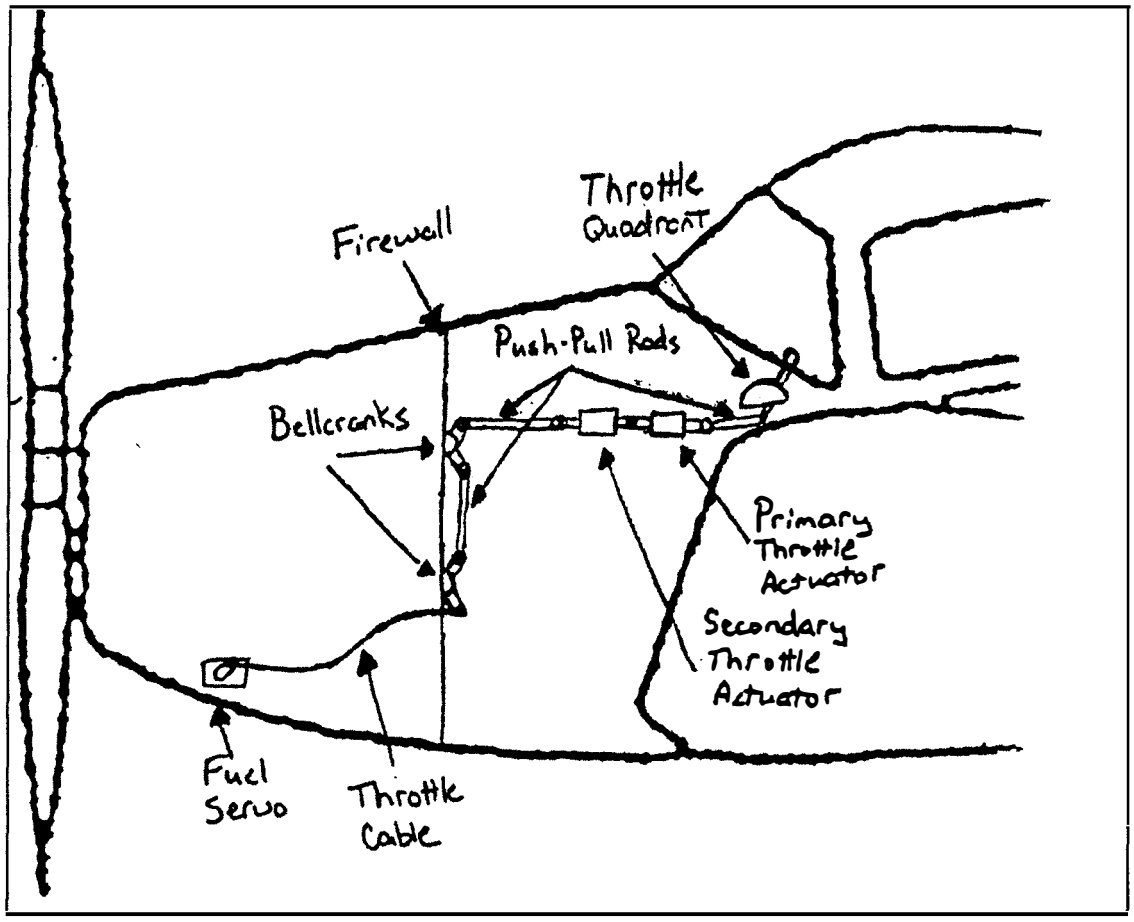


Figure 3-10. Engine Throttle Control System

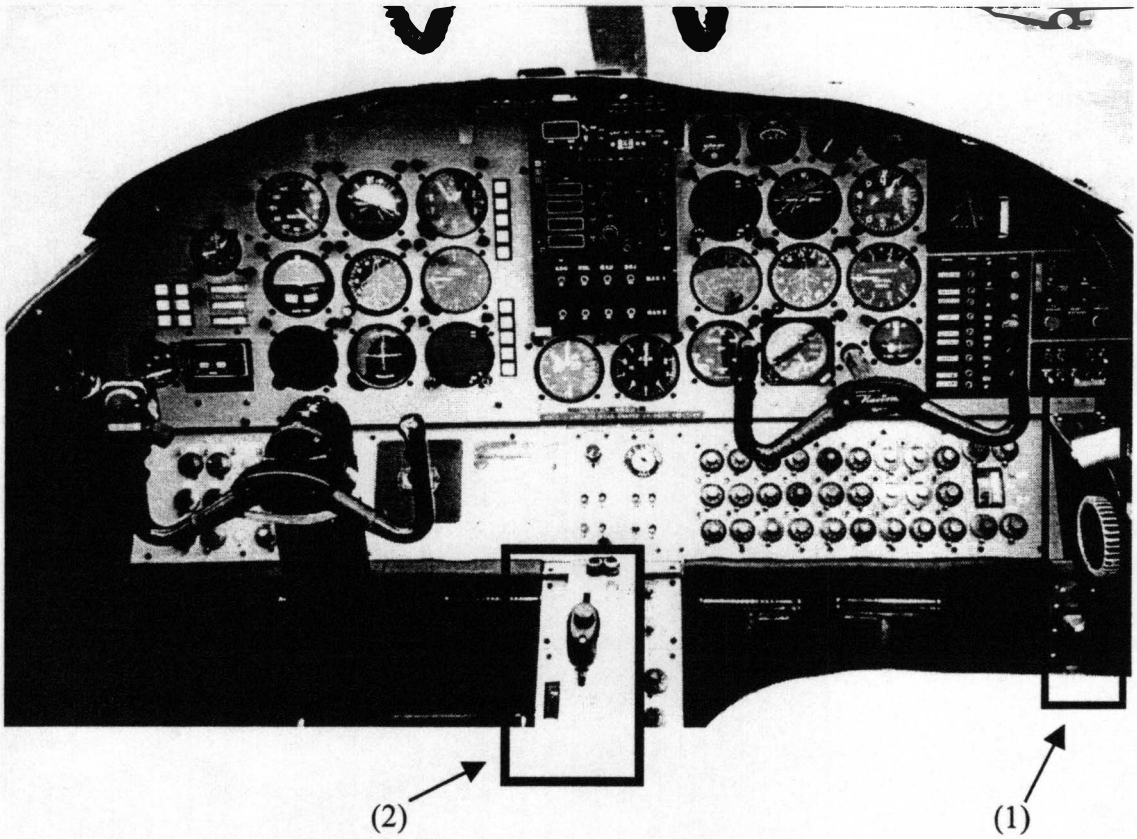


Figure 3-11. N66UT Engine Controls and Instrumentation
(1) Conventional, (2) Fly-By-Wire

Fly-by-Wire Engine Control System

For the FBW control system, the engine is controlled through using the push-pull control system described in the conventional control section (Figure 3-10). The primary actuator controls the system while the secondary actuator acts as a fixed link. Should the logic system sense a failure of the primary actuator, the secondary actuator is pressurized for control input and the primary actuator becomes a fixed link. During failure of both actuators, both act as fixed links and the throttle must be controlled using the conventional control system. The FBW control levers are centrally located on the cabin control panel. The control panel is mounted with a hinge pin assembly, allowing the controls to be moved to gain access to computer variable adjustment knobs. When the system is engaged, the hydraulic actuators receive input signals from the analog computer based on movement of the throttle control. The FBW throttle control is presented in Figure 3-11, labeled number 2.

INSTRUMENTATION, CALIBRATIONS AND CORRECTIONS

The Navion is equipped with a standard array of aircraft instruments (Figure 3-11). These include the manifold pressure, engine RPM, airspeed indicators, and altimeters. The manifold pressure and engine RPM gauges are connected to sensors in the engine. The airspeed and altitude indicators are connected to the ship's pitot-static pressure systems. Instrument calibration information was obtained from previous flight tests conducted at UTSI and is included in Appendix A, Figures 1-3. In addition to the standard instruments, the aircraft is equipped with a junction box (Figure 3-12) for access

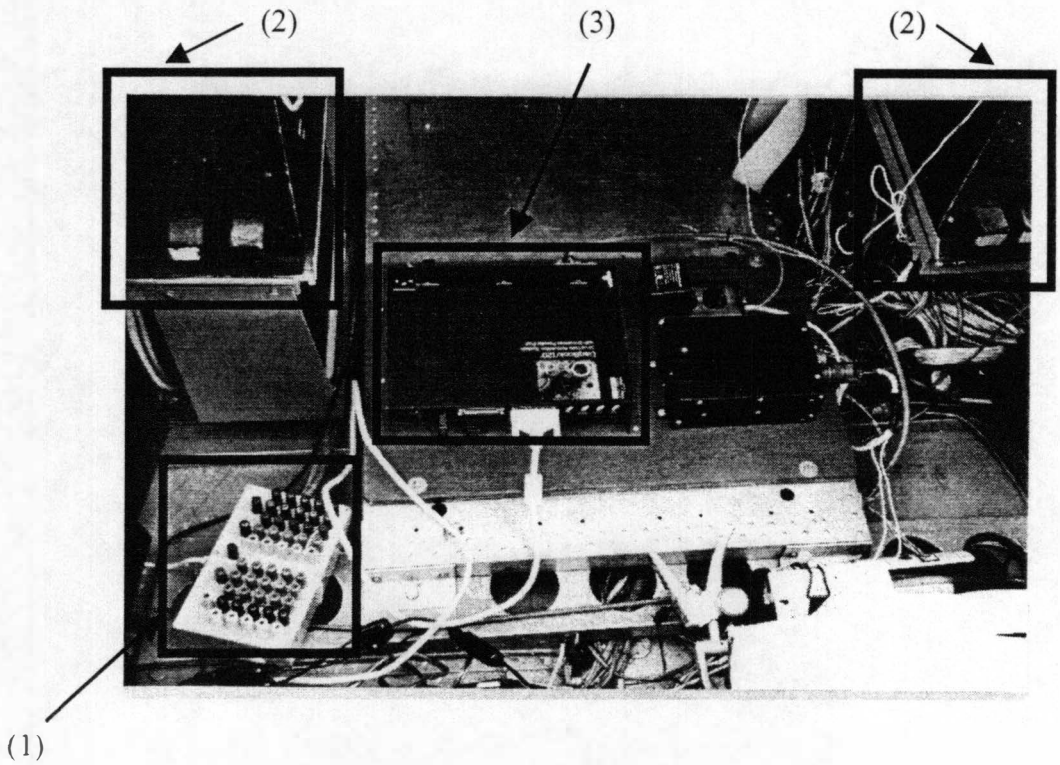


Figure 3-12. Fly-By-Wire Computer Interface

to the variable-stability flight data computer described in the previous section. Data extraction instrumentation are labeled as follows: "1" is the junction box, "2" is the analog computer system, and "3" is the DaqBook.A static on-ground calibration was conducted to determine potentiometer dial settings that would provide equal control surface movement for equal control displacement between conventional and fly-by-wire system. Control movement distance measurements were taken from the conventional control system and correlated to the actual control surface response. The same measurements were made of the fly-by-wire control system, and potentiometers were

to match the conventional system deflection. This calibration resulted in the following dial settings for the axes of interest:

- Aileron $L_{\delta a} = 58$ Pedals $N_{\delta r} = 50$
- Throttle $X_{\delta t} = 57$ Elevator $M_{\delta e} = 68$

Calibration data are included as Appendix A, Figures A-4 through A-19. Measurements required for handling quality analysis consisted of angular rates and attitudes, linear velocities and accelerations, and control positions. Accelerometers mounted on the test aircraft were located 98 inches aft of datum, 2.35 inches forward of the 100.35 inches aircraft center of gravity during the test. Vertically, the accelerometers were located at 109.0 inches above datum placing them 1.7 inches below the vertical C.G., located at 110.7 inches above datum.

TEST METHOD AND PROCEDURES

The short-term frequency response flight-tests for the Ryan Navion consisted of three flights of 1.5, 1.9, and 0.7 hours respectively. The flight tests were flown under daylight visual conditions at Tullahoma Municipal Airport, Tullahoma, Tennessee, on the 9th and 10th of October 1998 for data recording, and a flight on the 11th of November 1998, for validation of data. The essential test elements included 26 frequency sweeps of 120-second duration each and doublets at the frequency sweep condition. A summary of tests and test conditions are presented in Table 3-6. The flight tests were conducted at 90 knots calibrated airspeed (KCAS), and a density altitude (H_D) of 5000 feet for comparison by UTSI, to wind tunnel data. The tests were flown within the existing envelope of the aircraft. The flight crew consisted of two crewmembers. Average gross

weight and CG for the first flight were 3,177 pounds and 100.37 inches aft of datum, with an average M.A.C. value of 24.45 %. Average gross weight and CG for the second flight were 3,199 pounds and 100.32 inches aft of datum, with an average M.A.C. value of 24.37 %. Fuel at engine start was full at 39.5 gallons. Weight and Center of Gravity calculations are presented in Figure 3-13. The tests were conducted with the canopy closed, heater and carburetor heat off.

Table 3-6. Tests and Test Conditions

Test	**Axis**	**Control Station**	**Airspeed KCAS**	**Altitude Ft. H_D**	**Average OAT °C**	**Average Gross Wt. (lbs)**	**Average C.G. (Inches)**
Forward Flight Frequency Sweep	Pitch, Roll, Yaw, Throttle	P, C/P	90	5000	4, 10 *	3188	100.35
Doublet	Pitch, Roll, Yaw, Throttle	P, C/P	90	5000	4, 10 *	3188	100.35

* Indicates average temperature during the fly-by-wire flight test

Navion Weight and Balance

N66UT

Flight Identification: Conventional Frequency Sweep Crew Pilot: Dr. William Lewis

Date: 9 Oct 98 Copilot: Charles Catterall

Takeoff Conditions	**Weight (lbs)**	**Arm (in)**	**Moment (in-lbs)**
Basic Aircraft	2546	100.79	256,605.69
Pilot/Copilot*	430	96.0	41,280
Laptop Computer	10	96.0	960
DaqBook Shelf	20	116.0	2,320
Fuel (39.5 gallons)	237	103.0	24,411
Total	3242		325,576.69

CG = Moment/weight = 100.39 in

CG Location 100.39 in. – 83.66 in = 16.73 in/68.35 in x 100 = 24.5% M.A.C

Landing Conditions	**Weight (lbs)**	**Arm (in)**	**Moment (in-lbs)**
Fuel Burned (21.7gal)	-130.2	103.0	-13,410.6
Total	3111.8		312166.09

CG = Moment/weight = 100.32 in

CG Location 100.32 in. – 83.66 in = 16.66 in/68.35 in x 100 = 24.4% M.A.C

Flight Identification: FBW Frequency Sweep Crew Pilot: Dr. William Lewis

Date: 10 Oct 98 Copilot: Randy L. Bolding

Takeoff Conditions	**Weight (lbs)**	**Arm (in)**	**Moment (in-lbs)**
Basic Aircraft	2546	100.79	256,605.69
Pilot/Copilot*	460	96.0	44,160
Laptop Computer	10	96.0	960
DaqBook Shelf	20	116.0	2,320
Fuel (39.5 gallons)	237	103.0	24,411
Total	3272		328,456.69

CG = Moment/weight = 100.38 in

CG Location 100.38 in. – 83.66 in = 16.72 in/68.35 in x 100 = 24.46% M.A.C

Landing Conditions	**Weight (lbs)**	**Arm (in)**	**Moment (in-lbs)**
Fuel Burned (24.5gal)	-147	103.0	-15,141
Total	3125		313,315.69

CG = Moment/weight = 100.26 in

CG Location 100.26 in. – 83.66 in = 16.6 in/68.35 in x 100 = 24.29% M.A.C

Figure 3-13. Flight Test Weight and Balance Information

The flight test technique to acquire data was to make a "frequency sweep" using the desired control. The sweep was performed by making control displacements to either side of trim at frequencies from 0.1 Hz (10 sec period) to 2.0 Hz (0.5 sec period). The control displacements were large enough to effect a noticeable aircraft response, but not so large as to generate large airspeed changes (+/- 10 knots) or translations. A desirable sweep contains at least three seconds of trim followed by gradually increasing frequencies throughout the range of interest, and ending in three seconds of trim. The frequency range of interest was between 0.1 ω_{BW} and $2\omega_{180}$. Uniformity in the magnitude of control input throughout the frequencies is of no particular concern. It is both natural and acceptable for the magnitude at low frequencies to be quite small so that the attitude change will not be excessive, become larger in the mid-frequency range, then become smaller at high frequencies because of the physiological constraints of making rapid hand movements. Care must be taken to prevent unwanted movement of another control at the same frequency during the frequency sweep. While exact frequency inputs are not required, it is necessary to sweep the entire range of frequencies. This is not an intuitive process and requires practice to preclude wasted flight test time. The ideal method is to practice in the aircraft and record the data. Data analysis should be used to identify any unexcited frequencies.

Input and response data were recorded over a 120-second period (minimum to maximum frequency). Three frequency sweeps per axis were recorded for data reduction. Each frequency sweep ended with three seconds at trim condition followed by a doublet control input to be used for time domain verification of results. The pilot not on

the controls held the laptop computer, activated the recording process, and counted seconds for control input. There is a high potential for structural damage to the aircraft during bandwidth testing, particularly if inputs are made with higher magnitudes or frequencies than required. Safety precautions were taken to start frequency sweeps with small magnitude inputs at lower frequency. Magnitude of input was increased through the middle frequency ranges and decreased again approaching higher frequencies. While performing the frequency sweep tests, data were collected for handling quality analysis, for comparison of conventional and fly-by-wire control systems, and for a math model verification effort.

SPECIAL PRECAUTIONS

The following special precautions were observed:

- All normal limits and emergency procedures contained in the aircraft operator's manual were reviewed prior to flight and observed during flight.
- Crew coordination, test techniques, data recording and lookout responsibilities were reviewed prior to flight.
- Incremental build-up techniques were utilized.

DATA COLLECTION

A personal computer based data acquisition system designed and marketed by IoTech Corporation, Cleveland, Ohio, was used for data collection (Figure 3-14). The IoTech DaqBook® 120 is a portable data acquisition system designed to interface with a

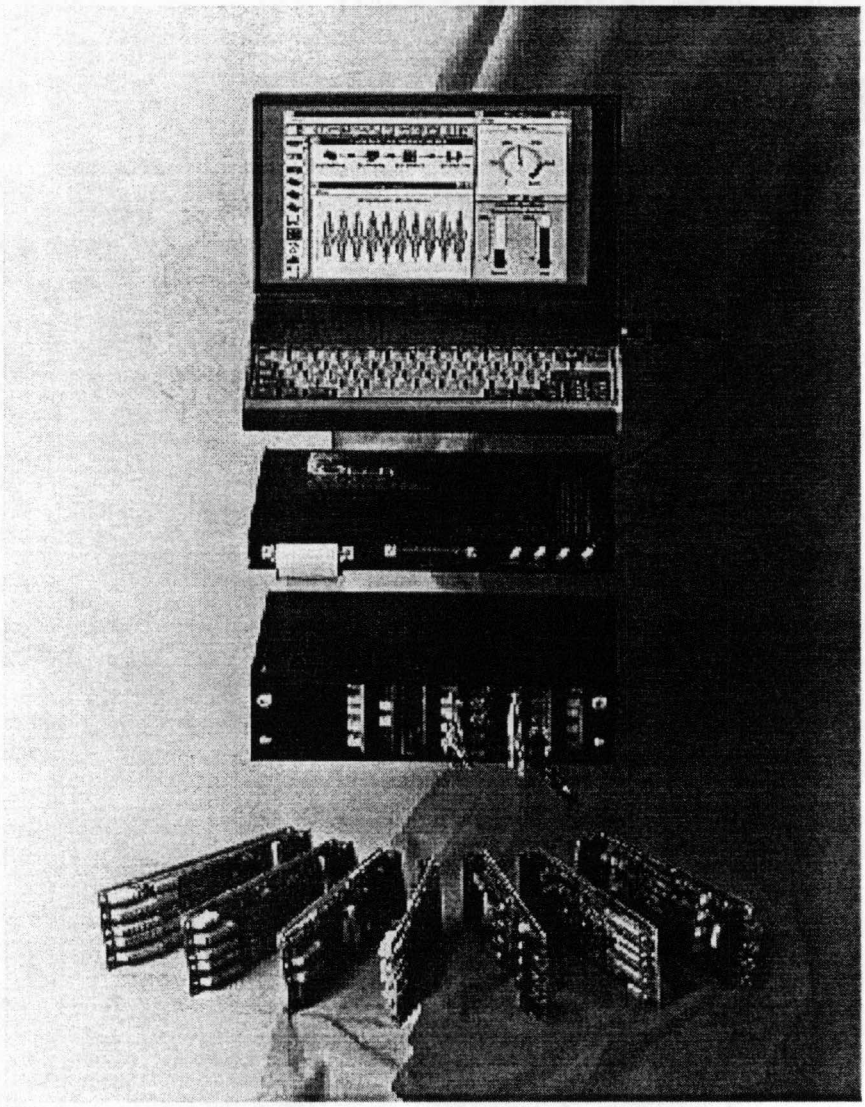


Figure 3-14. IoTech Data Acquisition System [10]

notebook computer via the standard parallel port. DaqView®, a spreadsheet style software provided with the system was used for data acquisition. System characteristics include 12-bit, 100-kHz A/D conversion, 100 K reading/sec sampling and real-time storage to disk. Data acquisition for system identification required a 100 Hz sampling rate. The system allows 8 differential- or 16 single-ended inputs, expandable to 256 channels. Channel/gain sequencing is at 10 millisecond intervals. Programmable gain adjustments are available. Gain adjustments by a factor of one, two, four, or eight are included as selection options. DaqBook can support up to 800 Kbytes/sec total data transfer to a standard or enhanced parallel port interface or PC-Card link [10].

The conventional flight controls were not instrumented to provide data for control positions. Calibration data relating conventional control inputs to control surface deflection were used, and therefore an assumption of a rigid flight control system was made. The aircraft was not instrumented for continuous measurement of real time structural loads. The following data parameters were monitored and recorded for system identification:

Control Positions

- Pilot's yoke wheel (aileron) position (FBW) Channel 0
- Pilot's yoke (elevator) position (FBW) Channel 1
- Pilot's Pedal (rudder) Position (FBW) Channel 2
- Pilot's Throttle Position (FBW) Channel 3

Body Rates

- Roll Rate (p) Channel 4
- Pitch Rate (q) Channel 5
- Yaw Rate (r) Channel 6

Accelerations

- ax — Channel 7
- ay — Channel 8
- az — Channel 9

Actuators

- Aileron Position — Channel 10
- Elevator position — Channel 11
- Rudder position — Channel 12
- Throttle position — Channel 13

Attitudes

- Roll — Channel 14
- Pitch — Channel 15

Chapter IV

DATA REDUCTION TECHNIQUE

DaqView files were saved as binary files saved during the flight test. Conversion of binary to ASCII character format is completed using DaqView software. Flight data files were imported into a Microsoft Excel spreadsheet to convert data for each channel from volt readings to the respective engineering units using appropriate calibrations. Calibration data for recorded channels are presented as Appendix A, Figures 4 – 20. Files were saved as tab delimited ASCII files and transferred via network transfer (File Transfer Protocol) to a Silicon Graphics Incorporated Octane II parallel processing computer, using an IRIX 6.4 operating system. Time history data were processed into a CIFER usable format by a compiling program called Convert. The Convert program separated the file containing all sixteen channels into sixteen individual program files.

The analysis of the flight-test data was conducted by first identifying frequency-response from time histories and second by determining bandwidth and phase-delay from frequency responses. Requirements for bandwidth (ω_{BW}) and phase delay (τ_p) are as depicted in Figure 2-2. Data were analyzed using the CIFER® 3.0 (Comprehensive Identification from FrEquency Responses) program for aircraft system identification. CIFER® was developed at the NASA Ames Research Center by Mark Tischler and Mavis Cauffman to provide a tool for system identification. System identification is the determination of a mathematical description of aircraft dynamic behavior from measured aircraft motion. This process creates transfer functions from flight test data. The AFDD/NASA frequency-domain system identification procedure is shown in Figure 4-1.

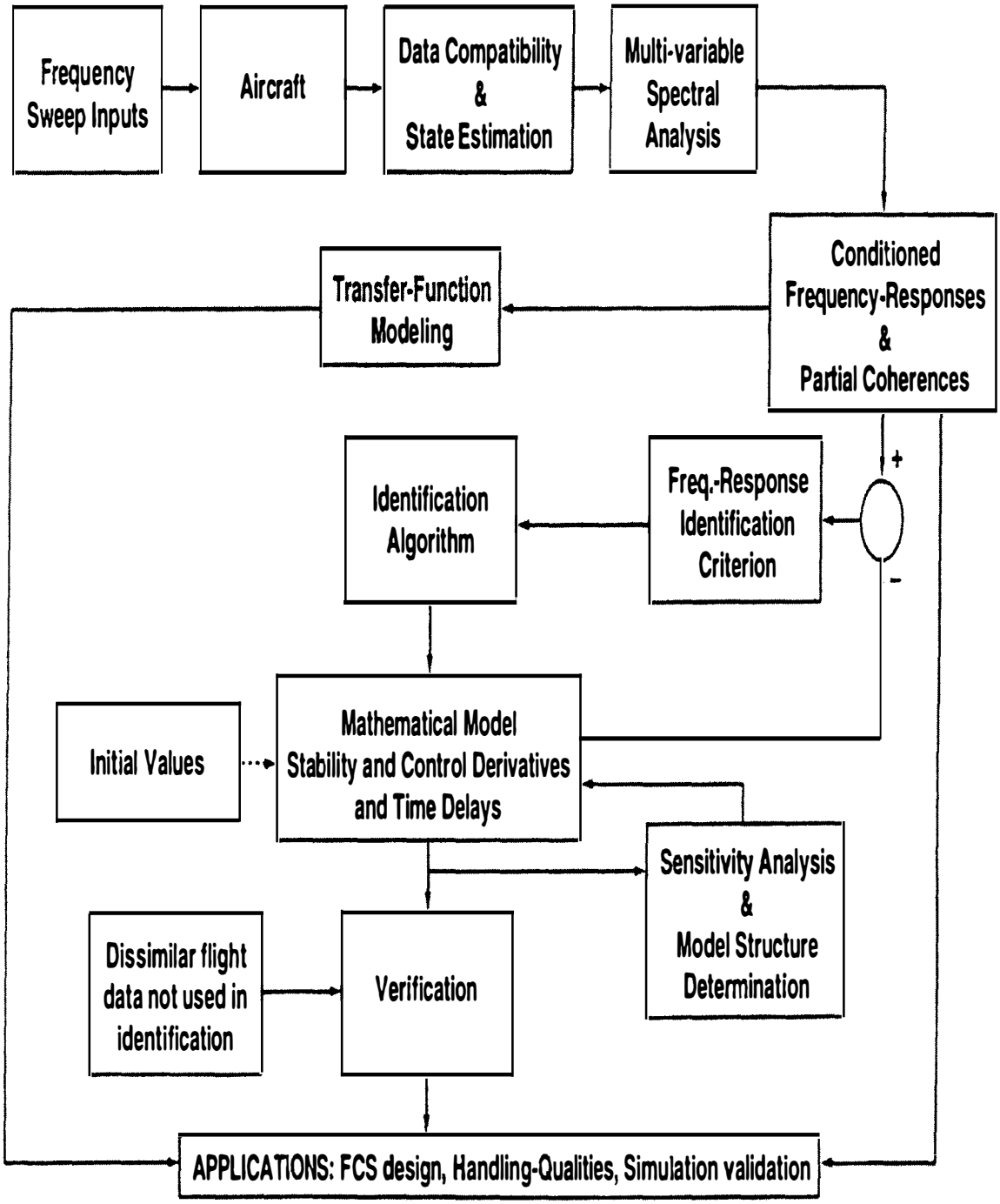


Figure 4-1. Frequency Response Method for System Identification [9]

The frequency domain analysis is based on bandwidth and phase delay criteria (Bode plot). The bandwidth criteria are established using adequate phase and gain margin to insure piloted loop closure does not threaten stability. The foundation of the AFDD/NASA approach to data analysis is the extraction of a complete multi-input/multi-output (MIMO) frequency-response database using an advanced multivariable spectral analysis with the Chirp-Z (advanced Fast Fourier) transform and composite optimal window technique [6]. The contaminating effects of any off axis control are removed by an inversion of the frequency response for all inputs to a single output. [11]

Three steps were used to generate the frequency response database. The first step was to produce the single-input/single-output (SISO) frequency response from the time histories using the Chirp-Z transform. The second step was to condition the responses to account for the effect of secondary inputs. These conditioned multi-input/single-output (MISO) responses were the same as the SISO frequency responses that would have been obtained had no correlated controls been present during the frequency sweep of a single control. Step three was to combine multiple window lengths into a composite response. The overall result of these three steps (FRESPID, MISOSA, and COMPOSITE) was the rapid identification of a set of broadband frequency responses for all input/output pairs [5].

Data were analyzed first using the Frequency Response Identification Program (FRESPID), an integrated part of the CIFER. The four plots from FRESPID that are of interest are the time history, auto spectrum, coherence, and transfer function (Bode plot). It has been determined that better results can be obtained by applying Laplacian

integration to rate data rather than by using attitude data directly. The time history used two separate frequency-sweeps concatenated to make a single data sampling. As an example, the longitudinal stick inputs for two concatenated frequency sweeps, each lasting about 90 seconds is shown in Figure 4-2. No other stick input time histories are included. The inputs do not exceed 10 % and are smaller for low frequencies. Output data presented are pitch rates (q). A desirable sweep contained several seconds of trim followed by gradually increasing frequencies throughout the range of interest. The frequency range of interest was between 0.1 ω_{BW} and 2.0 ω_{180}. Uniformity in the magnitude of control input throughout the frequencies has no effect on the outcome. Irregularities in the shape of the input are usually interpreted as high-frequency data by the FRESPID analysis, and are acceptable.

The input and output auto spectrum show the frequency distribution of the sweeps. As an example, Figure 4-3 presents the composite input auto spectrum for the elevator-surface input. This data indicates that good excitation was achieved over the range of 0.1 to 10.0 rad/sec. A drop-off in the auto spectrum occurs for input frequencies outside this range. No other auto spectrum data are included in this document. Relatively constant input power is desired throughout the frequency range of interest. A drop-off at higher frequencies is to be expected in the output spectrum.

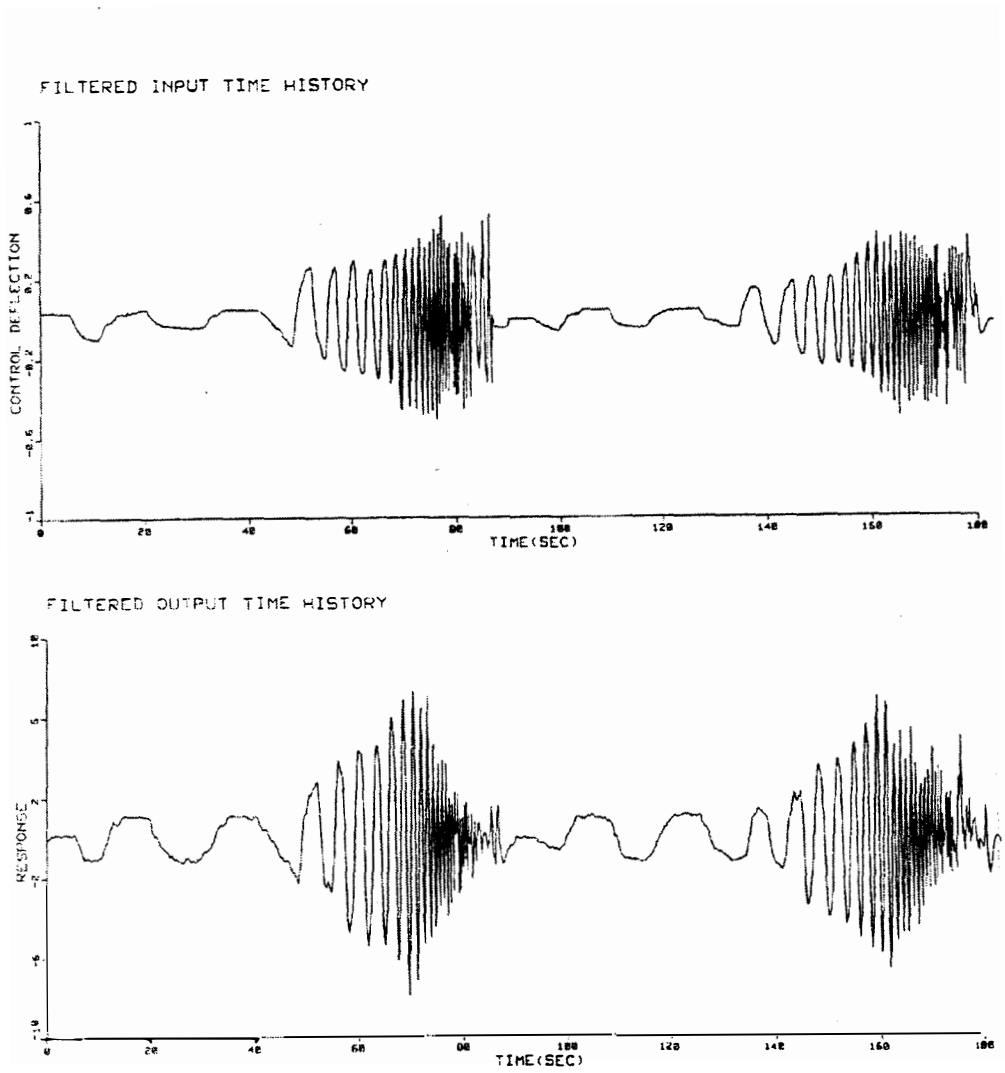


Figure 4-2. Time History: Longitudinal Stick Input
to Pitch Rate Response

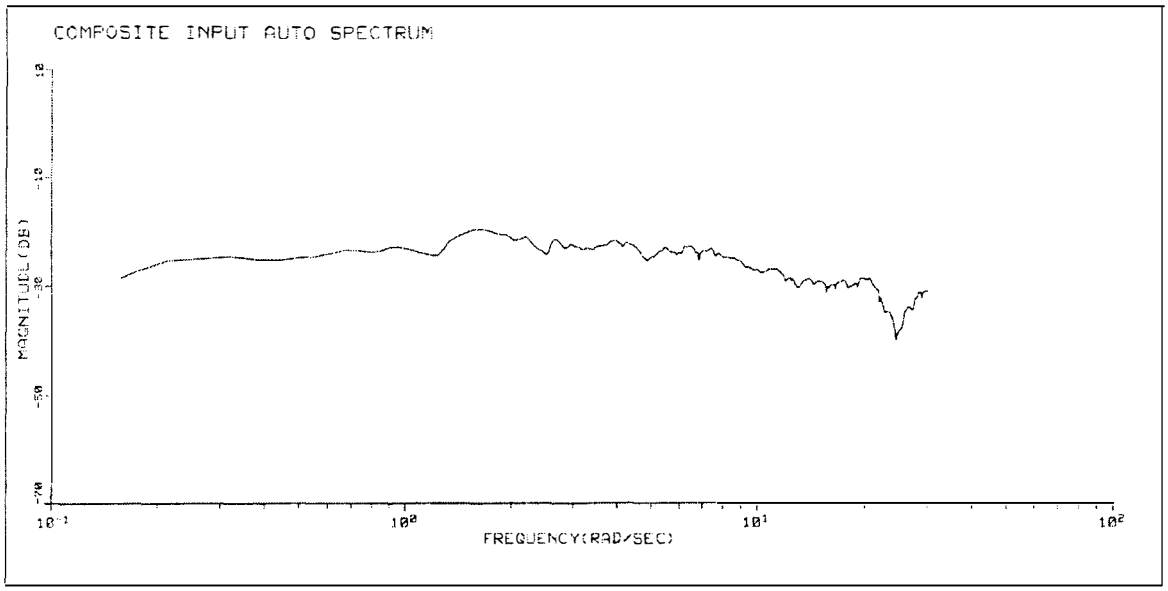


Figure 4-3. Conventional Controls Longitudinal Input
to Pitch Attitude Response Auto Spectrum

The coherence function is a measure of the extent to which the output appears to be linearly related to the input. A value of one is ideal. For the purpose of MIL-HDBK-1797 data interpretation, a coherence function greater than 0.8 that does not oscillate is acceptable. Common sources of reduced coherence are: atmospheric turbulence, off-axis inputs at or near the evaluated frequency, sensor noise, insufficient excitation of the aircraft, or significant non-linearity in the relationship between control input and aircraft response.

The Bode plot presents the attitude response to the control input in terms of magnitude and phase versus frequency. An example plot is presented in Figure 4-4. The Bode plot from each axis was used to determine MIL-HDBK-1797 compliance in terms of bandwidth and phase delay. Bandwidth is generally a measure of the ability of a system output to satisfactorily follow input over a range of frequencies; good systems having high bandwidth and poor systems having low bandwidth. The system bandwidth (ω_{BW}) is defined as the lesser of two frequencies from the Bode plot: ω_{GM}, the frequency for 6dB of gain margin before neutral stability (180 degree phase), ω_{180}; and ω_{135}, the frequency for 45 degrees phase margin before the same point. Techniques used for calculating the gain margins are depicted in Figure 4-5.

The phase delay (τ_p) is a measure of the steepness with which the phase drops off after –180 degrees and indicates the behavior of the aircraft as the pilot attempts to control the aircraft with inputs at frequencies higher than the bandwidth frequency. Aircraft with large phase delays tend to be prone to pilot induced oscillation (PIO). If the phase between ω_{180} and 2 ω_{180} is linear, two points may be used to define τ_p. If it is non-linear, a linear curve fit must be applied. The recommended procedure for performing a curve fit of the data is to use a linear, least squares fit. This will eliminate any uncertainty in the determination of linearity, and assure that specification compliance numbers will be similar regardless of individual judgement [12].

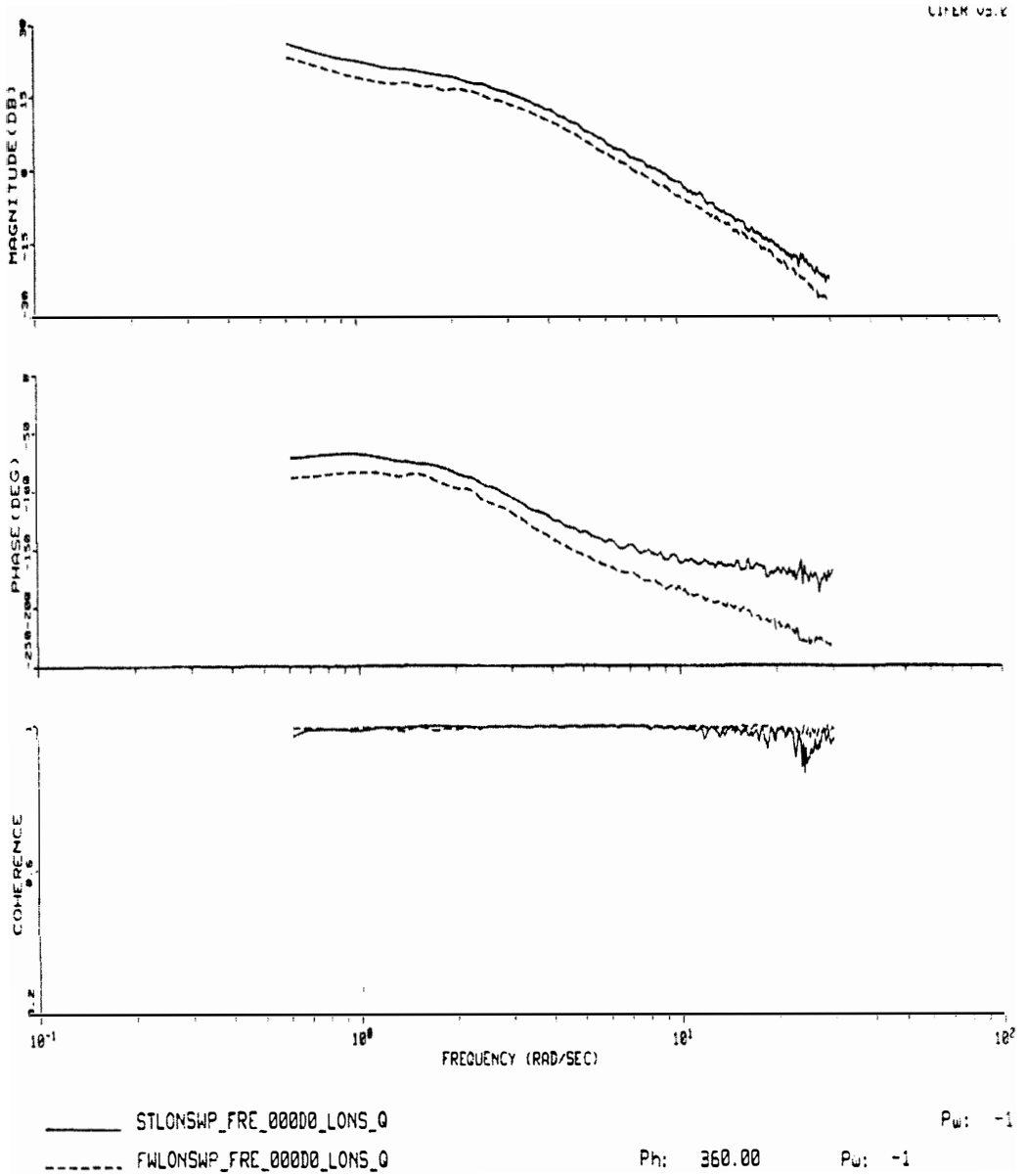


Figure 4-4. Typical CIFER Bode Plot; A Comparison of the Fly-by-wire and Conventional Control Systems

Figure 4-4 contains CIFER data plot coding which can be located at the top or bottom of the data plot, depending on which utility was used to process the Bode Plot. The following summation is presented to explain what each part of the coding system means to the reader.

For the Heading : **STLONSWP_FRE_000D0_LONG_Q**

The first two letters in the label "ST" indicate the control system. In this case it is the standard (conventional) flight controls. The second part of the label "LONSWP" is the case name assigned for CIFER identification. The third section of the label "FRE" is the CIFER program used for data analysis, in this example, FRESPID. "000D0" was assigned to indicate a 40-second window. Each position has a window size assigned for the program to process. During use of COMPOSITE, several windows could be used simultaneously for better identification. Within a window identifier of ABCD0, A = 40 sec, B = 30 sec, C = 20 sec, and D = 10 sec window lengths. "LONG" indicates the input for the data. "Q" (pitch rate) indicates the channel from which the data were recorded. A shortcoming in the utility used for individual control system data plots neglects to indicate when a laplacian integration has been used. No provisions are available to change source information labels. In the case above, an appropriate symbol in place of "Q" would have been "θ" for pitch attitude. This problem is apparent with many Bode plots included within Appendix B. Particular attention must be paid to the figure title to insure an understanding of data being presented. "Ph: 360" indicates any phase shift required for data analysis. "Pw: -1" is the power of a Laplacian integration

(s) multiplied into the transfer function. Above, an s^{-1} has been multiplied into the transfer function for pitch attitude acceleration to obtain pitch attitude change.

Phase Delay:

$$\tau_p = \frac{\Delta\Phi 2\omega_{180}}{57.3(2\omega_{180})}$$

Note: if phase is nonlinear between ω_{180} and $2\omega_{180}$, τ_p shall be determined from a linear least squares fit to phase curve between ω_{180} and $2\omega_{180}$

Rate Response-Types:

ω_{BW} is lesser of $\omega_{BW_{gain}}$ and $\omega_{BW_{phase}}$

Attitude Command/Attitude Hold Response-Types (ACAH):

$\omega_{BW} \equiv \omega_{BW_{phase}}$

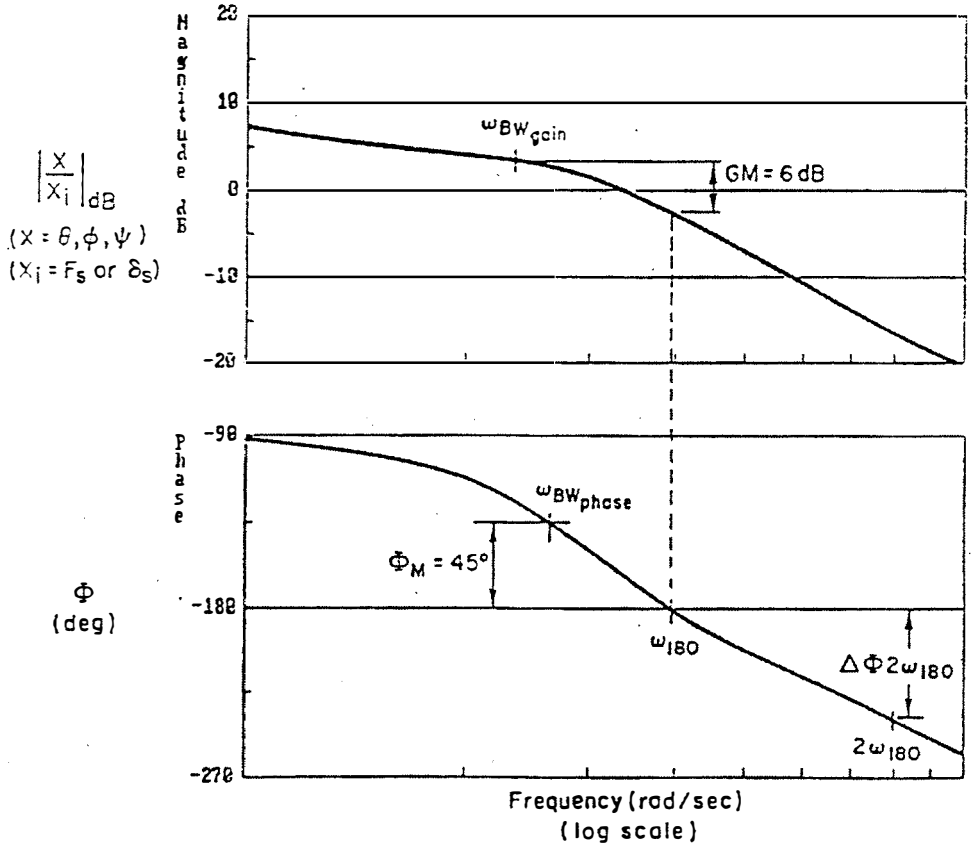


Figure 4-5. Definitions of Bandwidth and Phase Delay [6]

Chapter V

DISCUSSION OF TEST RESULTS

DATA ANALYSIS

CIFER programs FRESPID, MISOSA, and COMPOSITE were used to evaluate frequency-response data. Corrections to FRESPID data for off-axis response were minimal. Final COMPOSITE response data plots are presented in Appendix B. Frequency response identification was completed using the angular rate variables (p, q, r) rather than the angular attitude variables (ϕ, θ, ψ) because the middle and high frequency content of the rate variables is greater. The middle to high frequency signals are better suited for identification of the bandwidth and phase-delay parameters. When the identification of the low-frequency characteristics is more important, the attitude response variables are better suited for the analysis. The COMPOSITE program determines the required attitude responses from the rate responses by applying a Laplacian iteration to the magnitude and phase curves. The bandwidth and phase-delay parameters needed for demonstration of specification compliance are defined in terms of the attitude frequency response in Figure 2-2. As stated previously, the bandwidth, ω_{BW}, for a rate-response type system, like the Navion, is the lower of two frequencies: one, ω_{GM}, based on a gain margin of 6 dB; and the other, ω_{135}, based on a phase margin of 45 degrees. Bandwidth calculations for all data plots within this document resulted in a phase margin of 45 degrees (ω_{135}) determining bandwidth limits. No minus 6 dB gain margins were reached during this test.

GENERAL

Frequency sweep data are presented from a frequency of 0.6 rad/sec and truncated with a 5 Hz filter. Pitch, roll, and yaw control on axis responses had Level 1; Category C ratings, and Level 2; Category A ratings. Figure B-2 was used to demonstrate calculation techniques used in determining time delays from doublet time histories. Considering the high rate at which the initial doublet input is made, calculation of time delays from frequency-response data were taken at 20 rad/sec for comparison. Doublet control input information for all FBW systems were contaminated by interference, making this data unusable. Time delay differences between the conventional and FBW systems were calculated at FBW bandwidths ω_{135}.

During the handling qualities data validation flight, a technique was used to identify the maximum amplitude response by a control sweep of increasing frequency. Once identified, this amplitude was maintained for ten seconds and the number of cycles noted. An approximate frequency was calculated. Validation flight damping estimates were determined by displacing the aircraft away from trim and by observing the number of overshoots until returning to trim. Approximation of damping was accomplished using the following formula:

$$\frac{7-(\text{number of overshoots})}{10}$$

Flight check of spiral mode during the validation flight indicated positive stability with an 11-second time-to-half amplitude in the left and right roll directions.

LONGITUDINAL CONTROL INPUT

Pitch Mode, Conventional Control System

The difficulty in generating large low-frequency input signals in the pitch axis resulted in poor pitch response identification at the lowest input frequencies. However, the coherence function indicates good identification in the range of 0.6 to 10.0 rad/sec, which is satisfactory for specification compliance. Figure B-1 indicates a heavily damped 1st order pitch mode at 3.2 rad/sec. The resultant time constant is 313 milliseconds (ms). A heavily damped 1st order system was evident to the pilot during the flight. During the verification flight 6 pitch attitude cycles were experienced in 10 seconds, equating to a period of approximately 3.78 rad/sec. No overshoot of pitch attitude was noticed during the flight. Bandwidth from frequency data was at 4.84 rad/sec. The effective time delay is 487 ms measured at bandwidth (ω_{135}). Time delays of 487 ms at bandwidth will be very noticeable to the pilot as sluggishness in the control response. During the flight, attitude changes to control inputs decreased as frequency of movement increased. Since the phase curve is nearly flat at ω_{180}, the phase-delay for the conventional control pitch response is negligible. Figure 5-1 presents time history data for the conventional control system at bandwidth. Control deflection and pitch attitude change are presented. A line is drawn at the maximum control displacement, and another line is drawn at the point of maximum nose down deflection. Time delay indicated by the time history was determined to be 480 ms, only an 8 ms difference from the frequency determined time delay, and thus verifying the calculation.

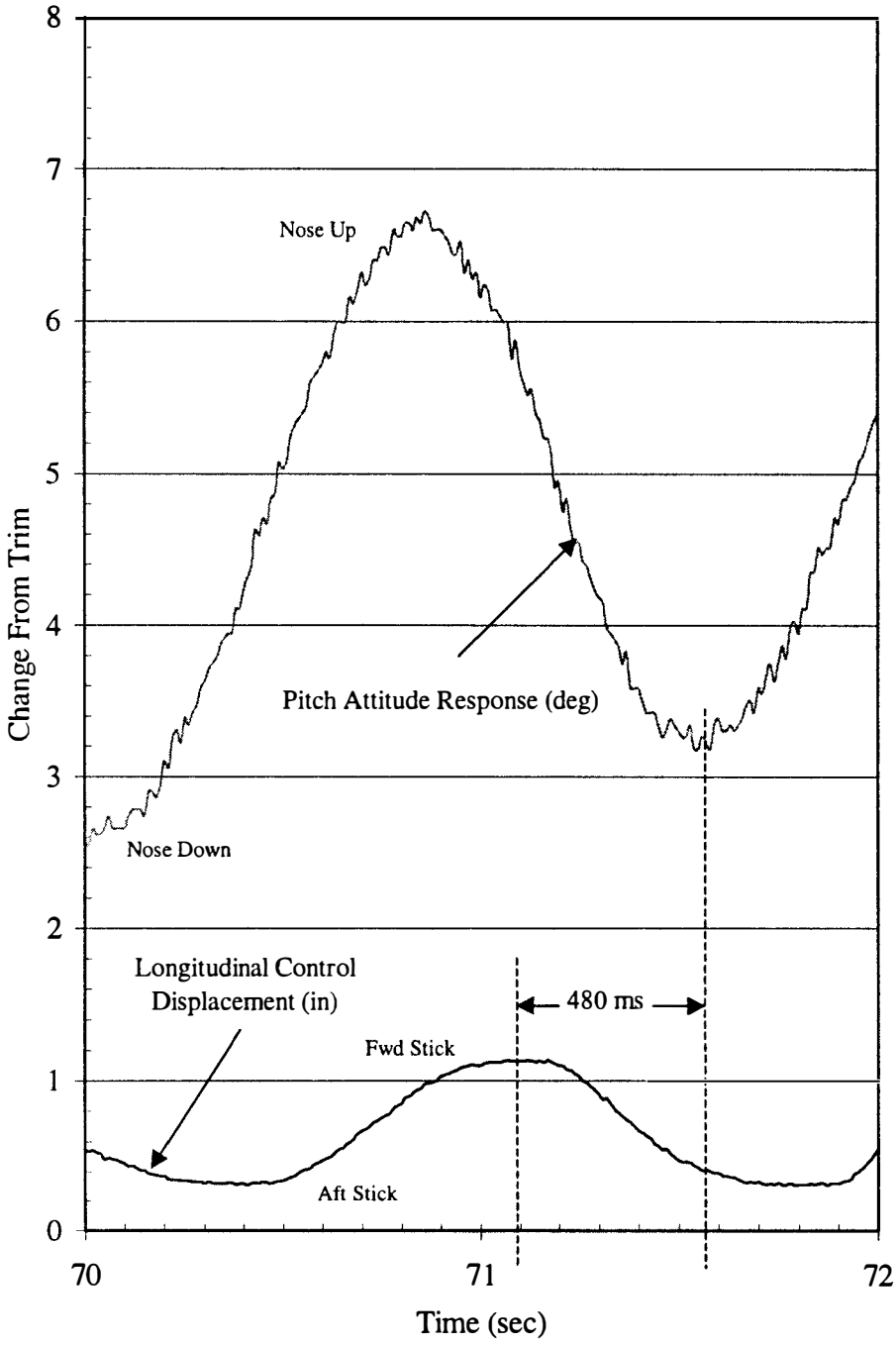


Figure 5-1. Longitudinal Time History (Conventional Controls); Longitudinal Input to Response

Pitch Mode, Conventional Control System Doublet

Figure B-2 presents the doublet input used in validating test results for the conventional control system. The time delay indicated for the doublet initial input was approximately 172 ms. The frequency derived time delay was 161 ms, only 12 ms difference from the doublet time delay, validating the test data. Handling-qualities experience indicates that the pilot may notice an equivalent time delay in excess of 180 ms. Initial aircraft response to step-type input should be within 172 ms. No delay was noticeable to the pilot during the flight.

Pitch Mode, FBW Control System

Figure B-3 data indicated a heavily damped 1st order pitch mode at 3.68 rad/sec for the FBW control system. Bandwidth was at 3.68 rad/sec. The time constant calculation resulted in a value of 278 ms. Since the phase curve is almost flat near ω_{180}, the phase-delay calculation for the FBW control pitch response was calculated using a least-squares curve fit routine, resulting in τ_p = 27 ms. The effective time delay measured at ω_{135} was 655 ms. A time delay of 655 ms at bandwidth will be very noticeable to the pilot. During the flight, sluggishness in control response was noticeable as frequency of control movement increased. At high frequency control inputs, a state was reached of no attitude change with control input. The time delay indicated by the frequency data is approximately 192 ms. Initial aircraft response to step-type input should be within 192 ms. Any delay in aircraft response was unnoticeable to the pilot.

Pitch Mode, Conventional and FBW Control Systems

Figure B-4 presents both conventional and FBW system responses relating longitudinal stick to pitch attitude. The FBW control system closely models conventional control system responses. A time delay difference of 9.5 ms between FBW and conventional control systems was measured at FBW bandwidth of 3.68 rad/sec. This small difference in time delay is not noticeable to pilots. FBW system electrical component delays and actuator dynamics did not affect phase delays until beyond bandwidth at approximately 10 rad/sec. This value correlates directly with the 9.9 deg/s^2 actuator maximum specific force response limit specified in Table 2-2. For input frequencies below 0.6 rad/sec and above 10 rad/sec, the coherence function becomes erratic and the transfer-function identification is not reliable.

Airspeed variation during longitudinal testing was $\pm$ 5 knots. Close monitoring was required to prevent control migration forward from the original trim point. Any deviation from the original trim point would result in a change to trim airspeed.

Pitch Mode to Normal Acceleration

The Bode plot of normal acceleration response to longitudinal control input for both control systems is shown in Figure B-5. The magnitude curve is flat between 0.6 and 3 rad/sec, indicating a normal acceleration response to control input. The pilot would perceive an upward force of the seat as the nose pitches up, and a downward drop in the seat as the nose is pitched down. The phase plot indicated approximately a 90-degree phase shift in the response. The pilot would perceive a reduction in the normal acceleration with an increase in the frequency of control movement. These flight

characteristics were verified by the pilot qualitative assessment. The coherence function for the normal acceleration response is strong over the frequency range of 0.6 to 3 rad/sec, with the drop-off above 4.5 rad/sec.

Pitch Mode, Cross-coupling

Figure B-6 presents an example of the cross-coupling off-axis responses related to longitudinal inputs. Off-axis coherence plots indicated no linear correlation between inputs and outputs. To the pilot, this equates to minimal unwanted roll or yaw coupled response being generated with a commanded pitch attitude change. Aircraft behavior during the flight test indicated no unwanted off axis responses associated with longitudinal control inputs.

LATERAL CONTROL INPUT TO ROLL ATTITUDE CHANGE

Roll Mode, Conventional Controls

The Bode plot for roll attitude response to control input for the conventional control system is presented in Figure B-7. The response is characterized by a heavily damped 1st order roll mode at 2.5 rad/sec. The time constant calculation resulted in a value of 400 ms. Bandwidth was determined to be 5.41 rad/sec. Pilot assessment verified the data. During the data verification flight, a maximum output to input response frequency was visually established using the conventional control system. Results were 5.5 roll attitude cycles in 10 seconds, equating to a period of approximately 3.45 rad/sec. No overshoot was experienced with roll mode responses. The effective time delay was 436 μs measured at ω_{135}. A time delay of 436 ms at bandwidth will be noticeable to the pilot. During the frequency sweep, sluggishness in control response was noticeable as

frequency of control movement increased. A state of no roll attitude change with control input resulted at a higher frequency control movement. Since the phase curve is nearly flat near ω_{180}, the phase-delay calculation for the conventional control roll response is negligible.

Roll Mode, Conventional Control Doublet

Figure B-8 presents the doublet input used to validate test results for the conventional control system. The time delay indicated for the doublet initial input is approximately 154 ms. Calculation from frequency-response data at 20 rad/sec resulted in a delay of 147 ms, a 7 ms difference from the doublet time delay, validating the test data. Initial aircraft response to a step input should be within 154 ms. This delay was unnoticeable during the flight. No roll attitude overshoot or oscillations were experienced during the test.

Roll Mode, FBW Control System

The Bode plot for roll attitude response to control input for the FBW control system is presented in Figure B-9. The response is characterized by a heavily damped 1st order roll mode at 2.5 rad/sec. The time constant calculation resulted in a value of 400 ms. Bandwidth frequency was 3.85 rad/sec. The effective time delay was 612 ms measured at ω_{135}. A time delay of 612 ms at bandwidth will be very noticeable to the pilot. During the flight, delay in control response was noticeable as frequency of control movement increased. A state of no roll attitude change with control input resulted at highest control movement frequencies tested. Using a least-squares curve fit routine, the phase-delay calculation for the FBW control roll response was $\tau_p = 30$ ms.

The time delay indicated by the frequency data at 20 rad/sec was approximately 146 ms. Any delay in control response was unnoticed by the pilot during the flight.

Roll Mode, Conventional and FBW Control Systems

Figure B-10 presents both conventional and FBW system responses relating lateral stick input to roll attitude response. The coherence function for the yaw response in both systems was unreliable below 0.9 rad/sec, and dropped below a value of 0.8 at frequencies above 10.8 rad/sec. The FBW control system closely models conventional control system responses, with a time delay difference of 8.9 ms, measured FBW roll bandwidth 3.85 rad/sec. This small difference in time delay will not be noticeable to the pilot. FBW system electrical component delays and actuator dynamics do not begin to affect phase delay until approximately 6 rad/sec, which is beyond bandwidth for this system. Table 3-2 indicates a roll actuator limit for maximum specific force or moment of 9.2 rad/sec^2. For input frequencies below 0.6 rad/sec and above 17 rad/sec, the coherence function becomes erratic and the transfer-function identification is not reliable.

Adverse Yaw

The Bode plot for yaw attitude response to roll control input for both the conventional and FBW control systems is presented in Figure B-11. This adverse yaw response is characterized by a moderately damped 2nd order mode with a natural frequency of about 2.4 rad/sec, and a damping ratio of approximately 0.5. This mode manifests itself at 180 degrees out of phase response to control input, and creates an adverse yaw effect. As the aircraft is rolled right, the nose of the aircraft yaws left. When rolled left, the aircraft yaws to the right. During roll axis frequency sweeps, a

continuous yawing motion was noted. This was apparent to the pilot during flight-testing. All other cross-coupling responses showed no linear correlation with roll inputs.

Roll Mode Lateral Acceleration (AY)

The Bode plot of the Y-axis acceleration response to lateral control input for both control systems is shown in Figure B-12. The magnitude curve rises slightly between 0.6 and 2.3 rad/sec and drops off rapidly above that frequency. The characteristic of the magnitude data indicated an increasing lateral acceleration response to control input, with a drop-off in response for frequencies above 2.3 rad/sec. The phase plot indicated a negative Y-axis acceleration with positive roll input. This was perceived as a sideward force of the aircraft away from the direction of roll as the pilot maneuvers in the roll axis. This is opposite of what was experienced by the pilot during this flight. This was due to the accelerometers being located below the vertical CG of the aircraft. The phase plot also indicates approximately a 90-degree phase shift in the response before coherence drops off. This data indicates the pilot would experience a reduction in the sideward force with an increase in the frequency of control movement. This was noticed during the flight. The coherence function for the y-axis acceleration response within acceptable range over the frequency range of 0.6 to 2.3 rad/sec, with a drop-off above 2.3 rad/sec.

PEDAL CONTROL INPUT TO YAW ATTITUDE CHANGE

Yaw Mode, Conventional Controls

The Bode plot for yaw attitude response to control input for the conventional control system is presented in Figure B-13. The response is characterized by a lightly damped 2nd order mode with a natural frequency of approximately 2.5 rad/sec, and a small damping ratio. Flight observations indicated a lightly damped oscillatory response. Bandwidth at ω_{135} was 2.9 rad/sec. During the verification flight 4.5 yaw attitude cycles were experienced in 10 seconds, equating to a period of approximately 2.8 rad/sec. The yaw response had 5 overshoots of trim before recovery, indicating a damping ratio of approximately 0.2. The effective time delay was 812 ms measured at ω_{135}. A time delay of 812 ms will be very noticeable to the pilot in all tasks associated with yaw axis control near 2.9 rad/sec. During the flight, yaw response became sluggish as frequency was increased until a frequency of no response to control input at approximately 23 rad/sec. The pilot experienced slight overshoot and oscillation (about the desired attitude). Small in-flight oscillations in the yaw axis were noticed with no associated control inputs, indicating weak directional stability about the trim point. Since the phase curve is nearly flat at ω_{180}, the phase-delay for the conventional control yaw response is negligible. Data for sideslip (ψ) were not recorded, therefore no doublet information is available for the yaw axis. The time delay indicated by the frequency data at 20 rad/sec was approximately 157 ms. Delay in initial aircraft response to control input was not noticeable. Figure 5-2 presents time history data for the conventional yaw control system at bandwidth. Due to lack of instrumentation for the yaw attitude, control deflections are

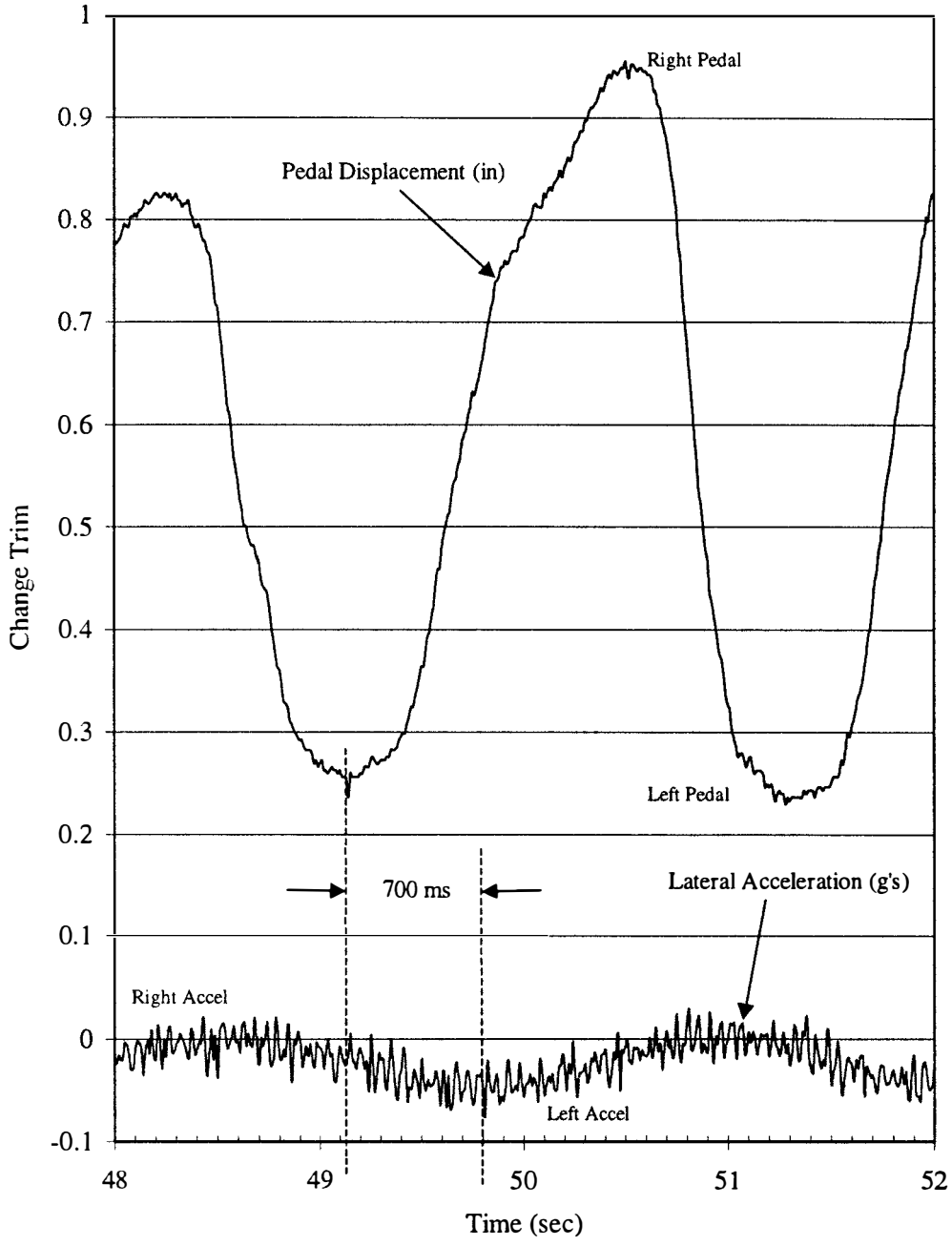


Figure 5-2. Conventional Control Pedal Input to Lateral Acceleration Time History

plotted against lateral accelerations, which closely represent a sideslip attitude. Time delay indicated by the time history was determined to be approximately 700 ms, 112 ms difference from the frequency determined time delay. Phase delay differences between sideslip and lateral acceleration probably accounts for some of the difference.

Yaw Mode, FBW Control System

The Bode plot for yaw attitude response to control input for the FBW control system is presented in Figure B-14. The response is characterized by a lightly damped 2^{nd} order mode with a natural frequency of about 2.4 rad/sec, and a small damping ratio. Bandwidth at ω_{135} was 2.83 rad/sec. The effective time delay was 833 ms measured at ω_{135}. A time delay of 833 ms at bandwidth will be very noticeable to the pilot in all tasks associated with yaw axis control. During the flight, yaw response became sluggish as frequency was increased until a point of no response to input at 18 rad/sec. The phase-delay calculation for the FBW control yaw response was τ_p = 29 ms. Data for sideslip (ψ) were not recorded, therefore no doublet information is available for the yaw axis. The time delay indicated by the frequency data at 20 rad/sec was approximately 196 ms. The pilot noticed no delay in aircraft yaw to pedal input.

Small cockpit design combined with pilot size resulted in limited space for control movement. During the pedal sweep of the FBW control system, the copilot was inadvertently making aileron inputs at the same frequency by the contact of his knees with the yoke assembly. An additional data run was required because of problems encountered with frequency-sweep techniques. This problem can be avoided by not resting your hands on your knees during input. Alternatively, practicing the input with

smaller magnitude may solve the coupled input problem. In worst case, a smaller pilot may be required.

Yaw Mode, Conventional and FBW Control Systems

Figure B-17 presents both conventional and FBW system responses for pedal control input-to-yaw attitude response. The FBW control system closely models conventional control system responses. There was no measurable time delay between the systems at FBW bandwidth, 2.4 rad/sec. FBW system electrical component delays and actuator dynamics do not begin to affect phase delays until a frequency of approximately 5 rad/sec, beyond bandwidth for this system. Table 3-2 indicates a yaw actuator limit for maximum specific force or moment of 4.2 rad/sec^2. For input frequencies below 0.9 rad/sec and above 17 rad/sec, the coherence function becomes erratic and the transfer-function identification is not reliable.

Dihedral Effect

The Bode plot for roll attitude response to pedal control input for both the conventional and FBW control systems is presented in Figure B-16. The dihedral response is characterized by a moderately damped 1st order mode at 2.7 rad/sec. The resultant time constant is 370 ms. This response manifests itself at 90 degrees out of phase with the control input, and creates a positive dihedral response. During flight positive sideslip resulted in a negative roll. As the aircraft was yawed nose right (left sideslip), the aircraft pitched slightly nose down and rolled to the right, away from the sideslip. Nose down pitch rate was slightly faster with a right sideslip.

Yaw Mode Lateral Acceleration (AY)

The Bode plot of the Y-axis acceleration response to pedal control input for both control systems is shown in Figure B-17. The magnitude curve rises between 0.6 to 2.5 rad/sec and drops off rapidly at higher frequencies. This indicates an increasing acceleration response to control input, with a drop-off in response for frequencies beyond 2.5 rad/sec. The phase plot indicates an in-phase Y-axis acceleration up through 2.0 rad/sec, with a 180-degree phase shift for frequencies above 4.0 rad/sec. This is perceived as a sideward force of the aircraft into the direction of yaw. This result was verified by the pilot during flight. All other cross-coupling responses indicated no linear correlation with yaw inputs.

THROTTLE CONTROL INPUT TEST RESULTS

The Bode plot of the X-axis acceleration response to throttle control input for both control systems is shown in Figure B-18. Test result coherence for all channels recorded indicated poor results. Transfer-function identification and handling qualities evaluation for all throttle inputs were inconclusive. Table 3-2 indicated a throttle servo response bandwidth of only 0.6 rad/sec. Inputs of 0.6 rad/sec were at the lower bound of the test data interval. Throttle sensitivity was a concern during testing. The FBW throttle was more sensitive and less predictable in response than the conventional control. Frequency inputs to both systems were made to an estimated 5 percent either side of trim, however throttle response was inconsistent. Engine accelerations were large and non-linear, with a

sluggish response after initial movement followed by rapid acceleration. Responsiveness of the propeller was not sufficiently adequate to maintain consistent RPM. During flight 70-100 RPM engine fluctuations were experienced. Control response was jerky, and the engine neared stall during one iteration indicating a need for a control fixture for this test.

Chapter VI

CONCLUSIONS

The purpose of this thesis was to evaluate the handling quality characteristics of the Ryan Navion (N66UT) using frequency-domain test techniques. Conventional and Fly-by-Wire flight control systems were compared to determine any time differences in response between the two control systems. Evaluation of aircraft handling qualities were compared to fixed wing flying quality specifications outlined in MIL-HDBK-1797. Due to the lack of specifications regarding low gain roll and yaw maneuvers, specification compliance in these axes was assumed to follow pitch attitude specification data.

The flight test developed and conducted by the University of Tennessee Space Institute resulted in the following conclusions:

1. Frequency-domain system-identification methods are well suited to aircraft flight-control and handling-qualities analysis since many current design specifications, design and analysis techniques, and acceptance flight-test techniques are based in the frequency domain. Frequency-domain methods extract large amounts of broad frequency flight-test data using a minimal amount of flight-time.

2. The response characteristics of the Navion N66UT are stable for the longitudinal, lateral and rudder control inputs. All bandwidth frequencies were determined by 45 degree phase margins (ω_{135}). The handling quality parameters meet or exceed proposed Level 1, Category C requirements, and Level 2, Category A requirements for fixed-wing military aircraft. A summary table of test results is presented in Table 6-1.

3. The fly-by-wire system closely mirrored conventional control behavior well beyond bandwidth frequency. Results are presented in Table 6-2. Time history data from flight test were used to verify frequency time delay calculations at bandwidth for the conventional controls, longitudinal control input to pitch attitude and pedal control input to lateral acceleration.

4. Time delays in excess of 400 ms at bandwidth were experienced in pitch, roll and yaw control axis. This would prohibit use of the variable stability airplane, N66UT; to simulate any high-bandwidth, minimal control delay designed aircraft.

5. Activation of the FBW system caused interference with attitude response data channels 14 (roll attitude) and 15 (pitch attitude) signals, and prevented the use of doublet information for FBW verification.

Table 6-1. Summary of Test Results

Control System	Input	Output	Bandwidth ω_{135} (rad/sec)	Phase Delay τ_p (ms)	Time Delay at ω_{135} (ms)	Time Constant (ms)
Conventional (ST)	Long stick	Pitch(θ)	4.84	Negligible	487	313
	Lat stick	Roll(ϕ)	5.41	Negligible	436	400
	Pedal	Yaw(ψ)	2.9	Negligible	812	2nd order
Fly-by-wire (FW)	Long stick	Pitch(θ)	3.68	27	655	278
	Lat stick	Roll(ϕ)	3.85	30	612	400
	Pedal	Yaw(ψ)	2.83	29	833	2nd order

Table 6-2. Time Delay in the Fly-by-Wire Control System with Respect to Conventional Controls

	Longitudinal	Lateral	Directional (Pedal)
Time Delay Difference*	9.5 ms	8.9 ms	negligable

* calculated at FBW control system bandwidth

6. The Navion demonstrated the following flight characteristics:
    - A positive dihedral effect with sideslip input.
    - Adverse yaw response to roll attitude changes were noticed during flight and verified in the frequency domain data.

7. Control delays due to hydraulic actuator dynamics did not affect pitch, roll, or yaw FBW control response until beyond bandwidth frequencies.

8. Throttle response data was below acceptable levels of coherence (0.8). No conclusions could be made regarding specification compliance. Table 3-2 indicates a servo response bandwidth of only 0.6 rad/sec. Inputs of 0.6 rad/sec were at the lower bound of the test data interval. Control sensitivity was a concern during testing of the conventional and FBW throttle systems. The FBW throttle system was more sensitive to control changes and less predictable in response than the conventional system. Frequency inputs were made to an established 5 percent either side of trim, however throttle response was inconsistent. Engine accelerations were large and non-linear, with an initially sluggish response to control movement followed by rapid acceleration.

9. Lateral acceleration data indicated the opposite of observed aircraft accelerations during roll inputs. This false lateral acceleration data is the result of accelerometer package location below the vertical CG.

10. An additional frequency sweep of the FBW roll control and pedals were required due to improper frequency sweep techniques.

11. Inputs were inadvertently made to roll controls during the pedal sweep of the FBW control system. This input was due to the pilot's knees contacting the yoke assembly and making roll inputs at the same frequency as the pedal inputs. An additional roll frequency sweep was made due to delays in input and excessive control displacement.

Chapter VII

RECOMENDATIONS

1. Review the instrumentation and data collection equipment prior to further testing. Determine the cause for the loss of pitch (ch 15) and roll (ch 14) attitude data during use of the Fly-by-Wire control systems.

2. Conventional control systems need to be instrumented for accurate position data. Using an assumption of a rigid control system does not allow for determination of any control hysteresis.

3. Fixtures should be considered to control range of motion during frequency-domain testing to increase input consistency. At a minimum, a control fixture should be designed for throttle sweeps of both conventional and Fly-by-Wire control systems.

4. Insure that the pilot performing the frequency sweep has adequate space to avoid unwanted control movements. If the cockpit area is not large enough to facilitate the pilot and all of his equipment, consideration should be made to using smaller pilot or different flight equipment.

5. Increase pilot training in frequency-domain testing techniques to avoid inadvertent off-axis response inputs while sweeping frequencies in the tested axis. Practice of frequency-sweep techniques using the control system to be tested should be arranged.

6. Locate the accelerometer package on the vertical and longitudinal center of gravity. If this is not possible, corrections should be made to accelerometer data to account for distance between sensor and center of gravity.

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APPENDICES

APPENDIX A

NAVION CALIBRATION CHARTS

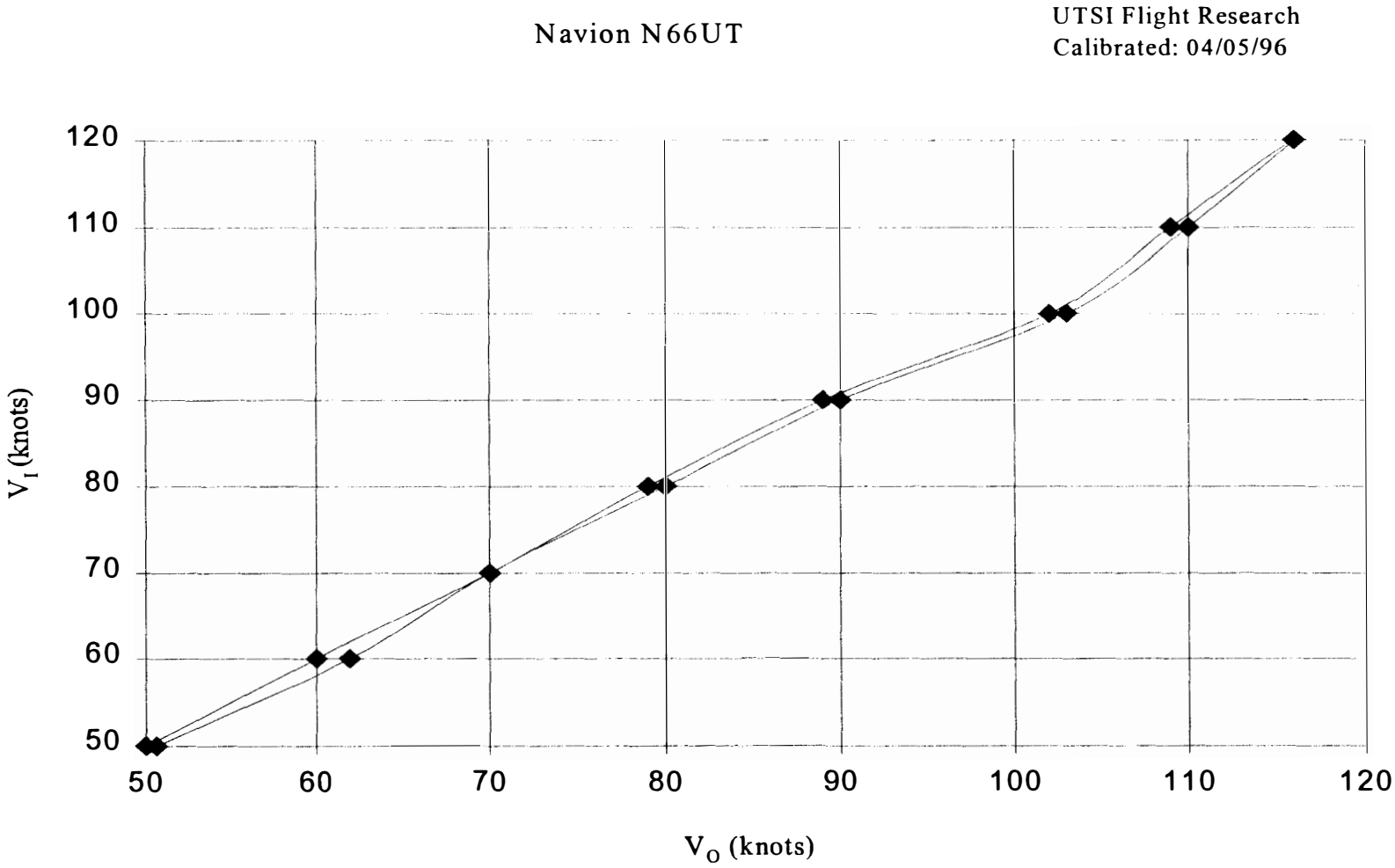


Figure A-1. Copilot's Airspeed Instrument Error Correction

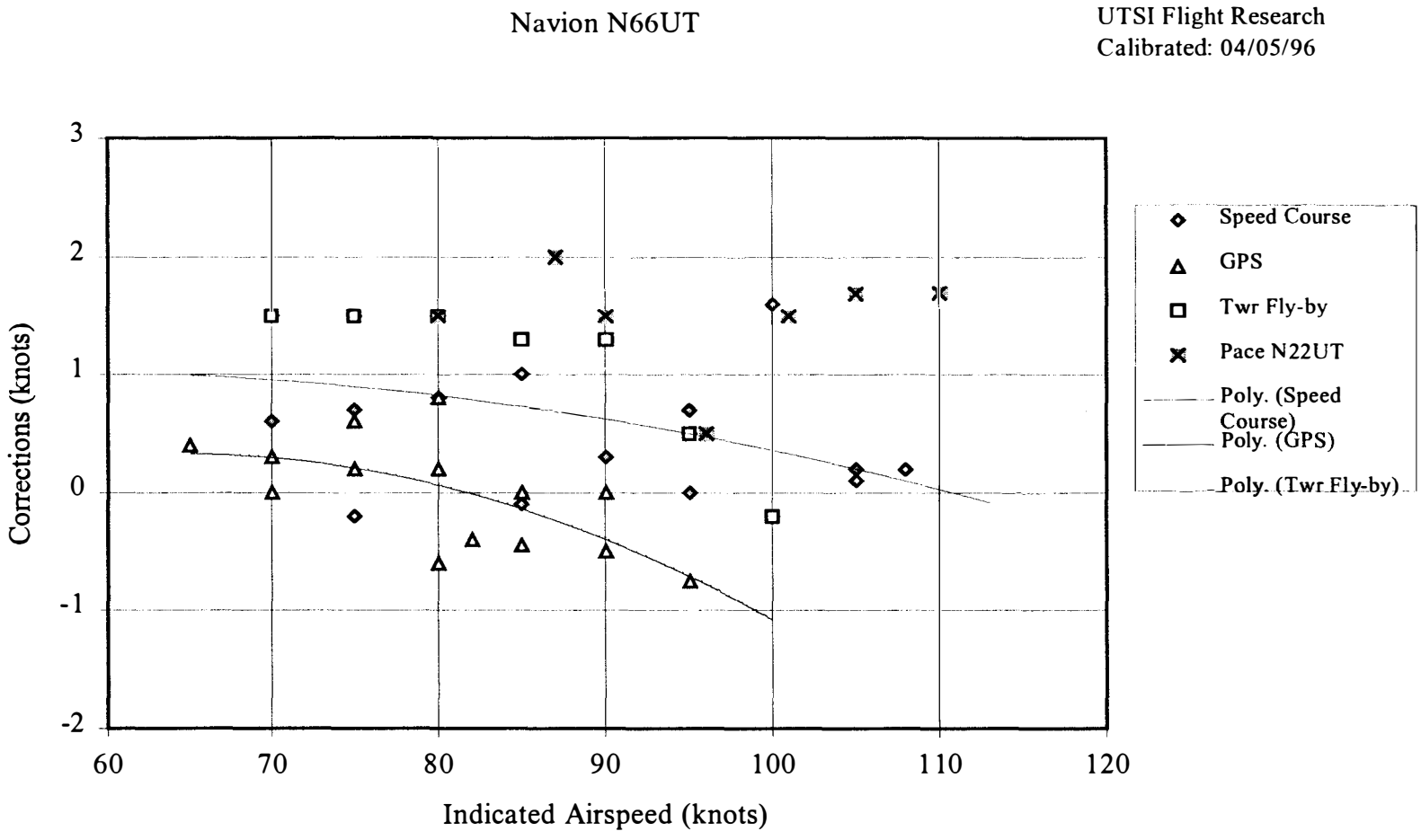


Figure A-2. Airspeed Position Error Correction

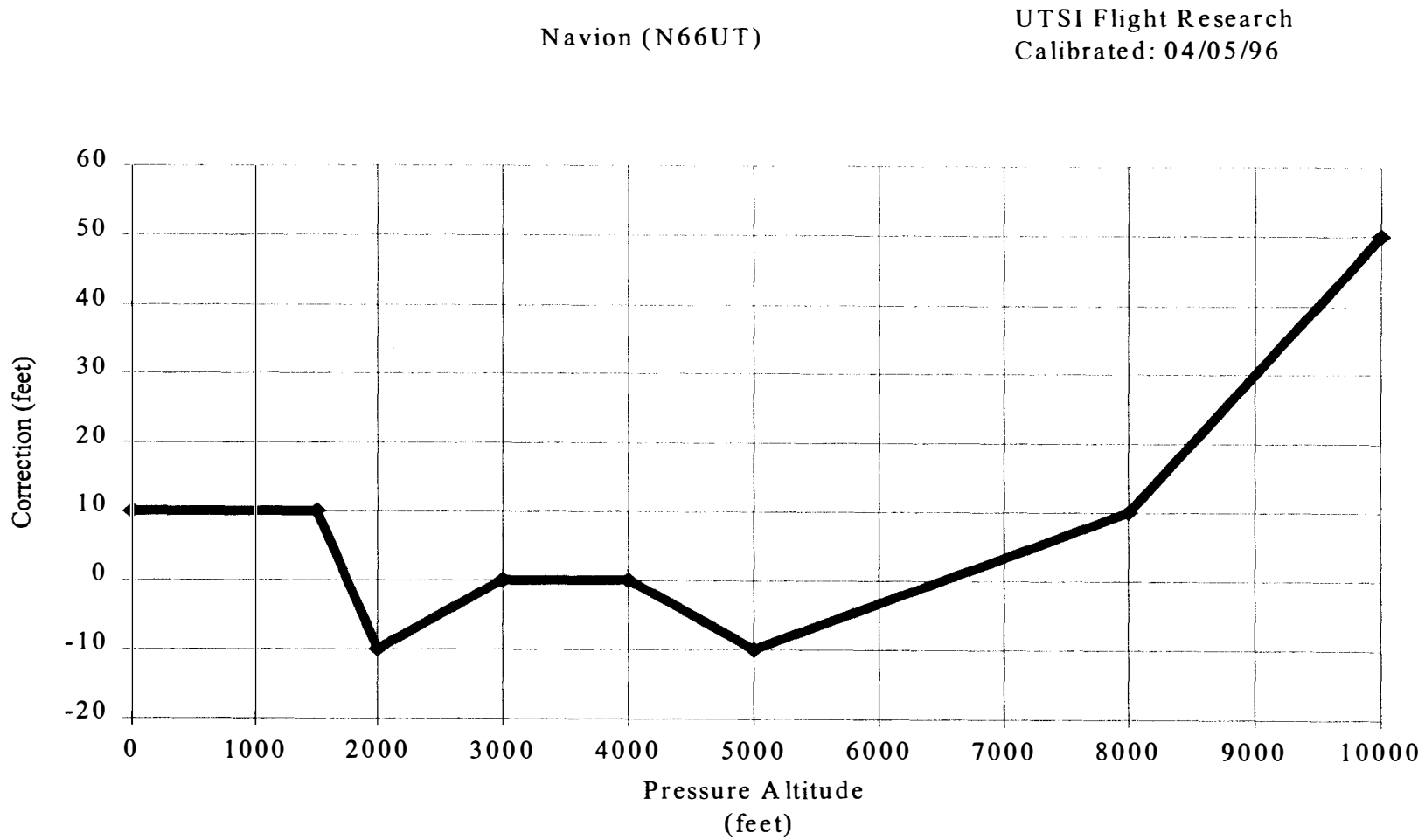


Figure A-3. Copilot's Altimeter Instrument Error Correction

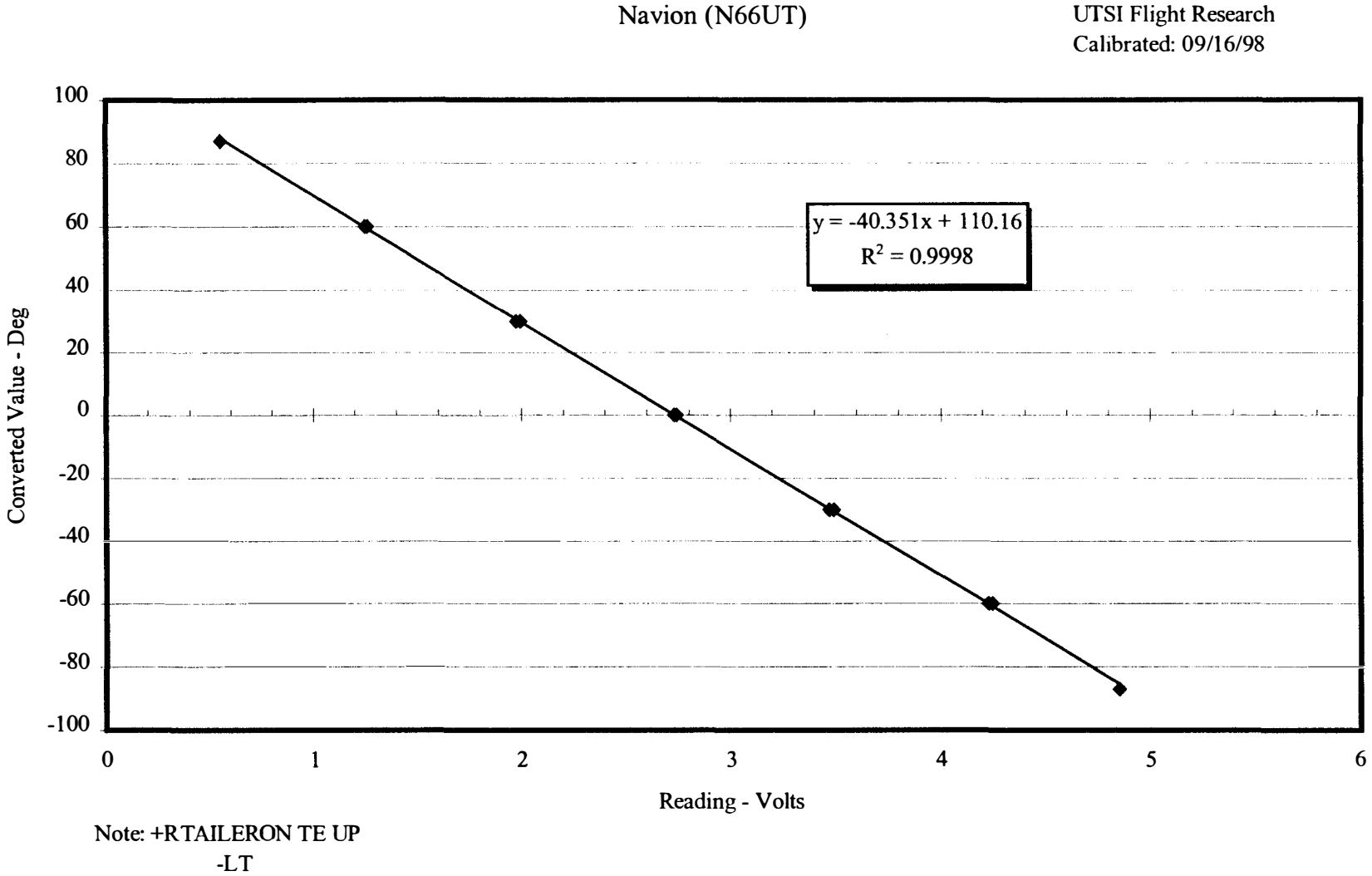


Figure A-4. FBW Aileron Stick Position (CH00)

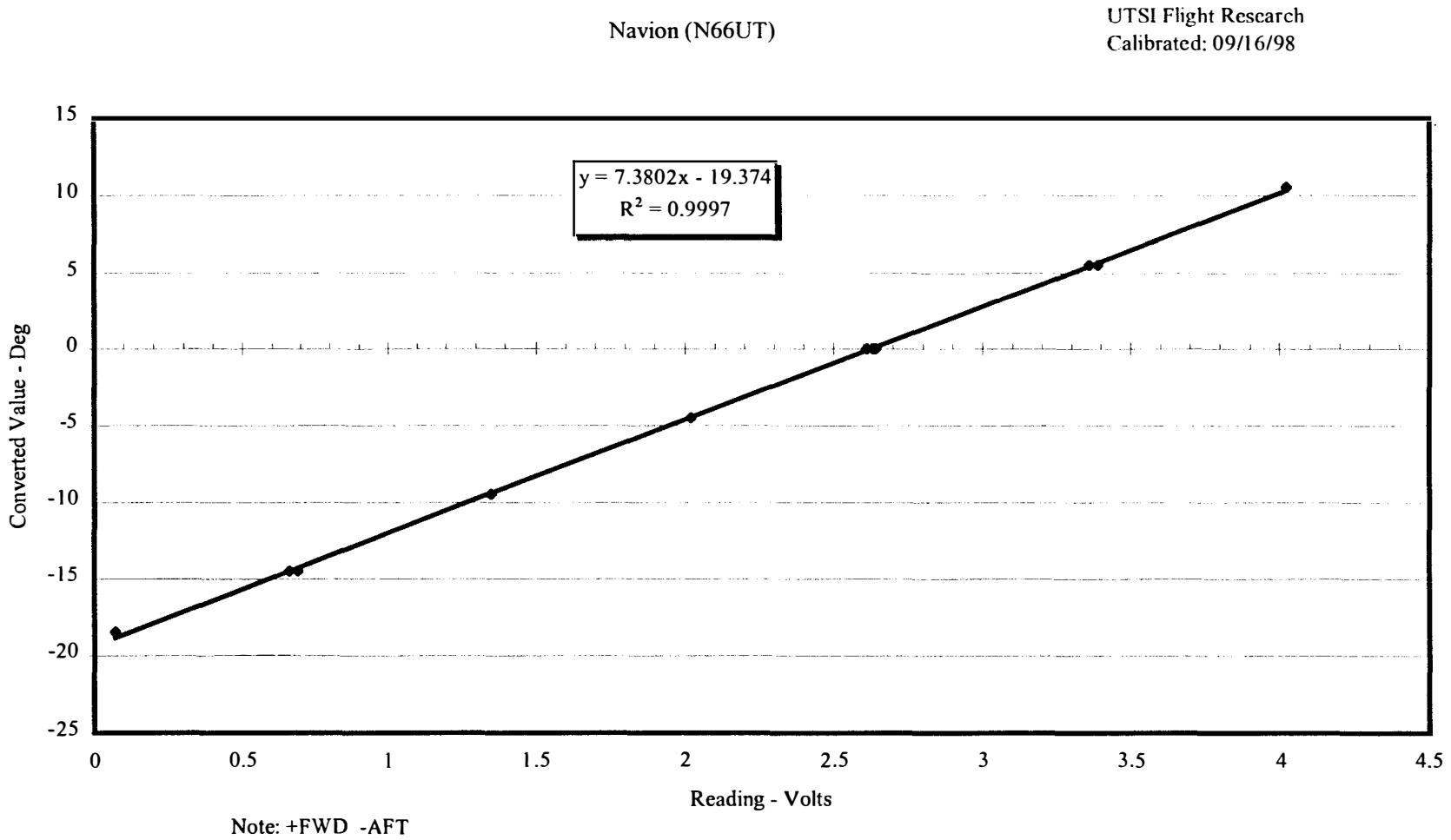
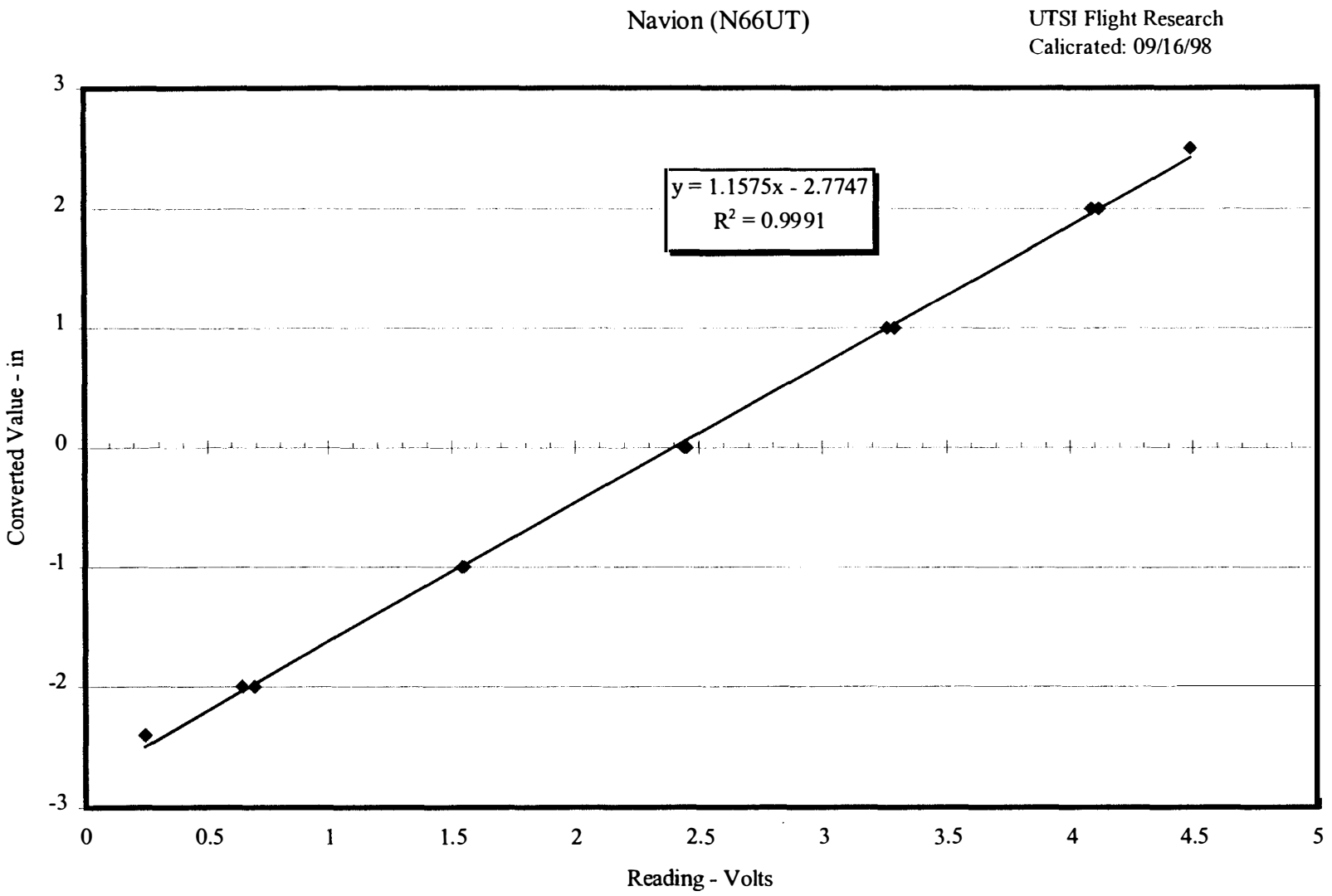


Figure A-5. FBW Elevator Stick Position (CH01)



Note: + RT PEDAL
- LT PEDAL

Figure A-6. FBW Rudder Pedal Position (CH02)

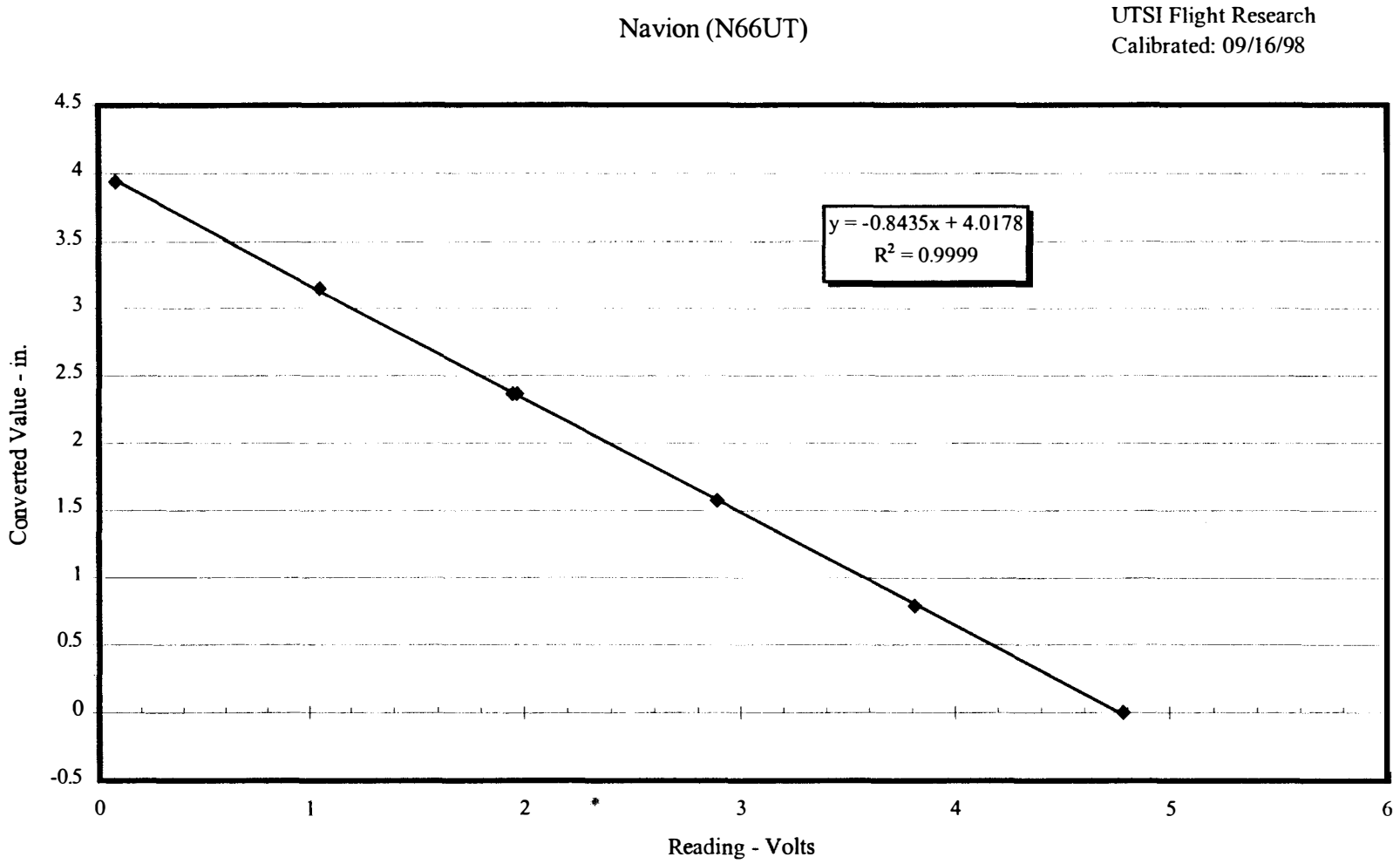


Figure A-7. FBW Throttle Command Position (CH03)

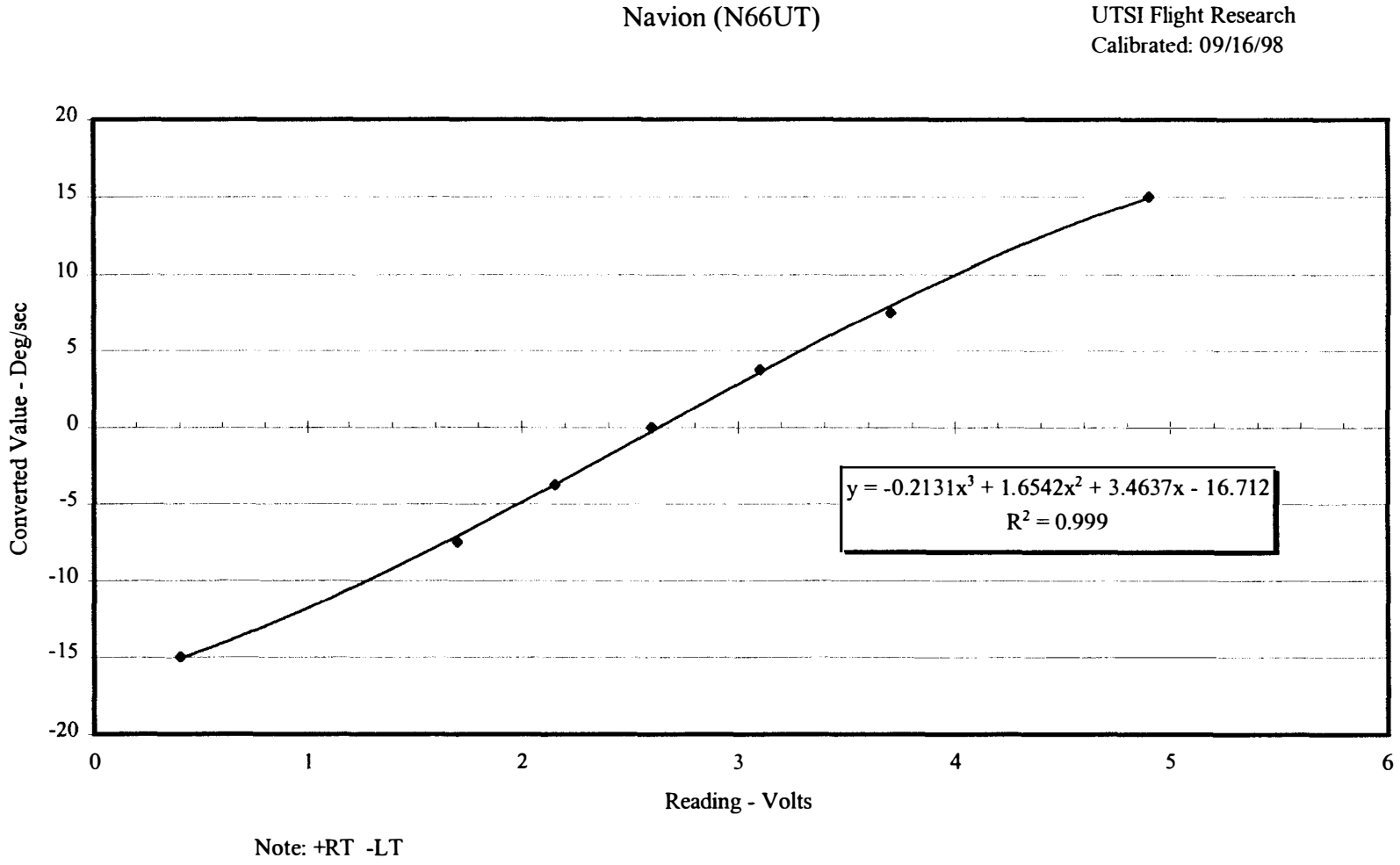


Figure A-8. Roll Rate (p)
(CH04)

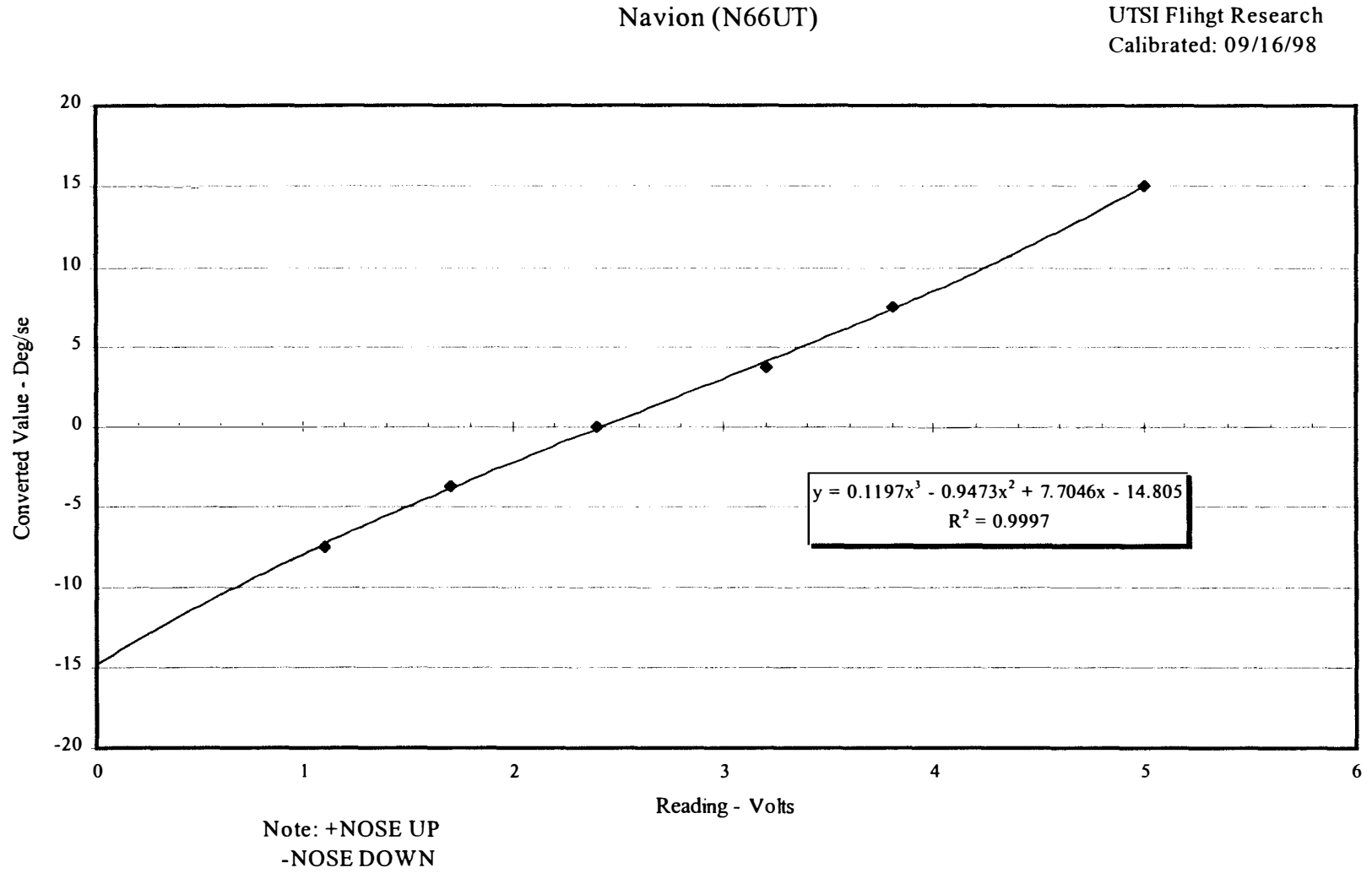
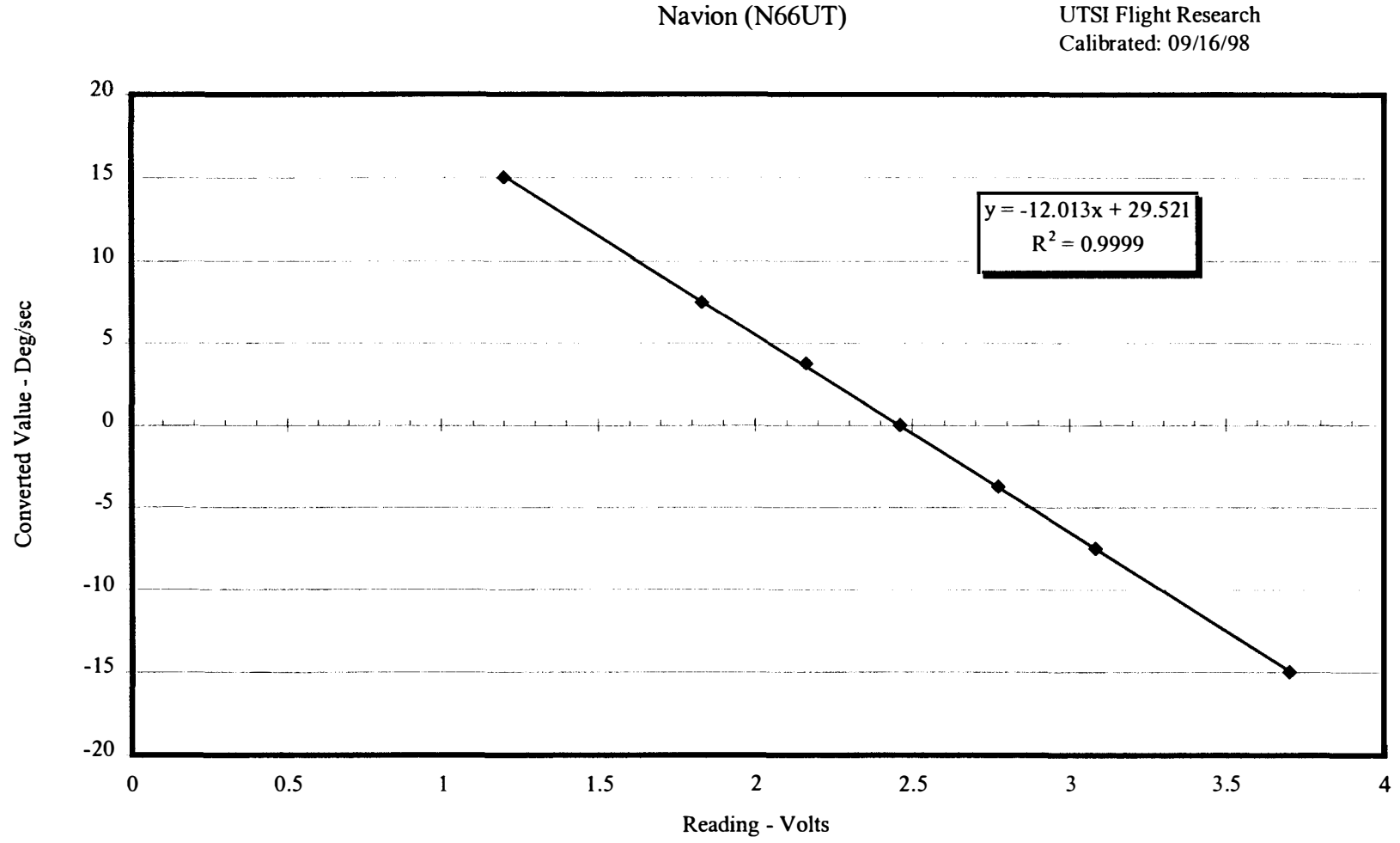


Figure A-9. Pitch Rate (q)
(CH05)



Note: + NOSE RT
- NOSE LT

Figure A-10. Yaw Rate (r)
(CH06)

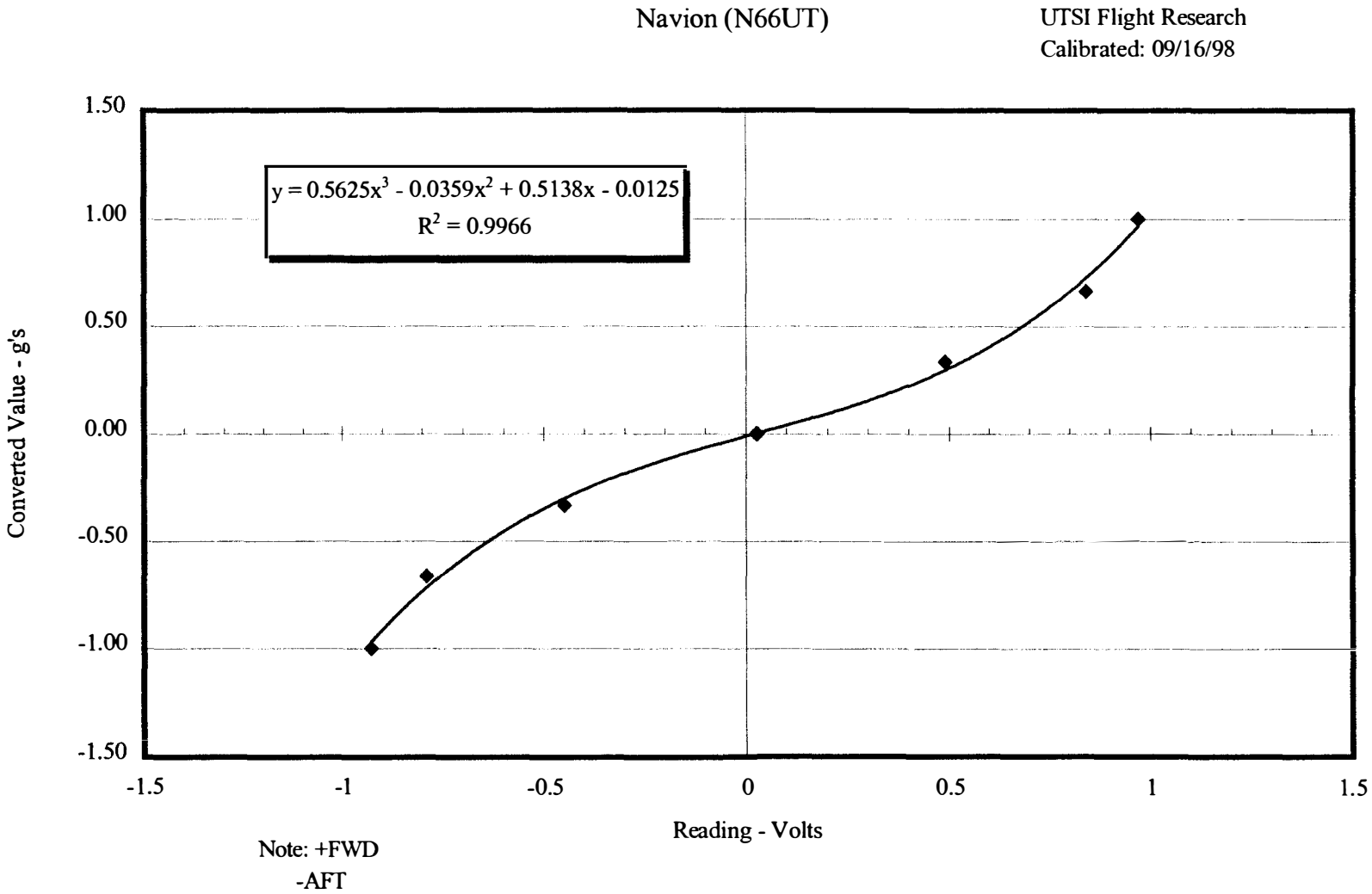


Figure A-11. Longitudinal Acceleration (ax)
(CH07)

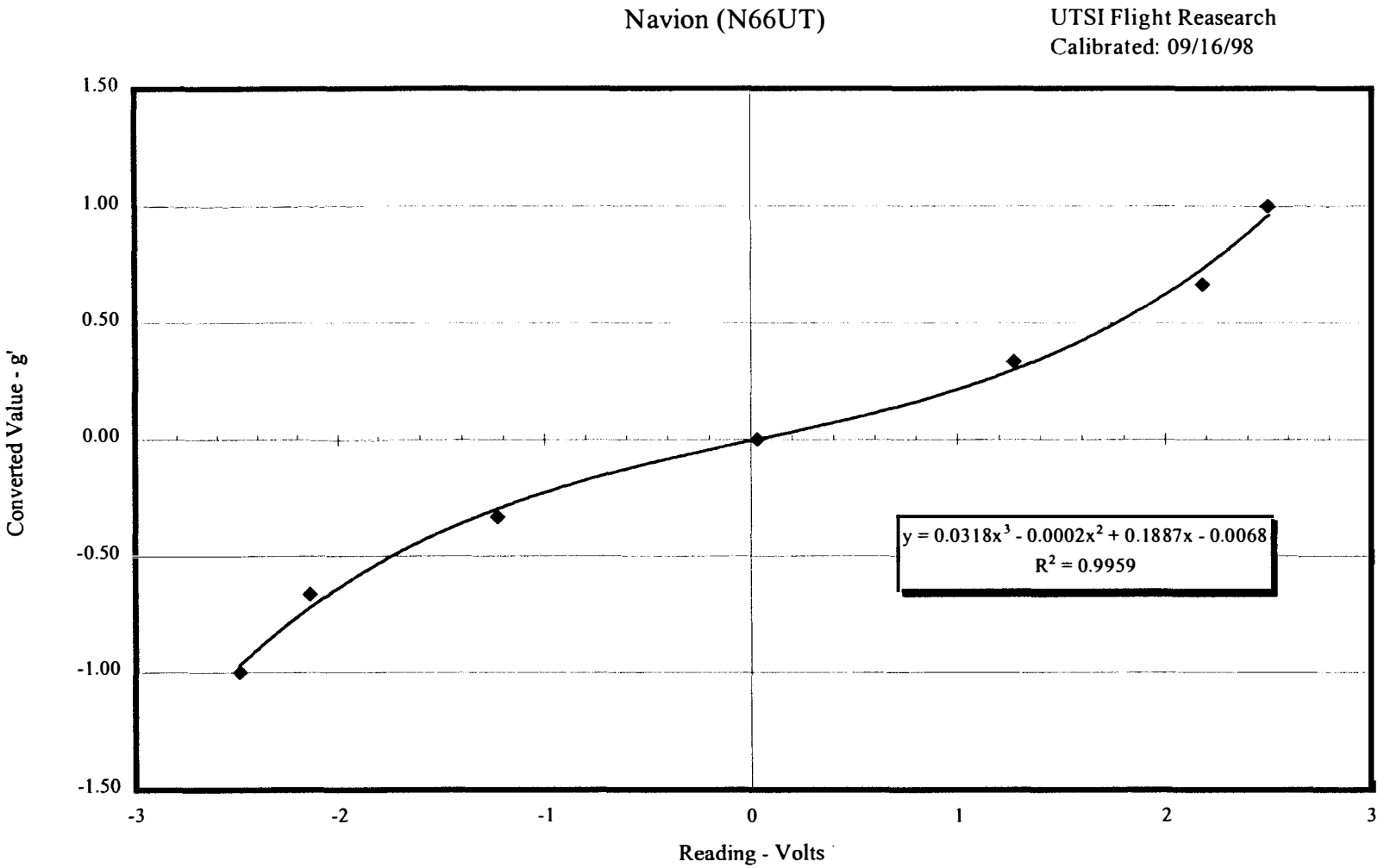


Figure A-12. Lateral Acceleration (ay)
(CH08)

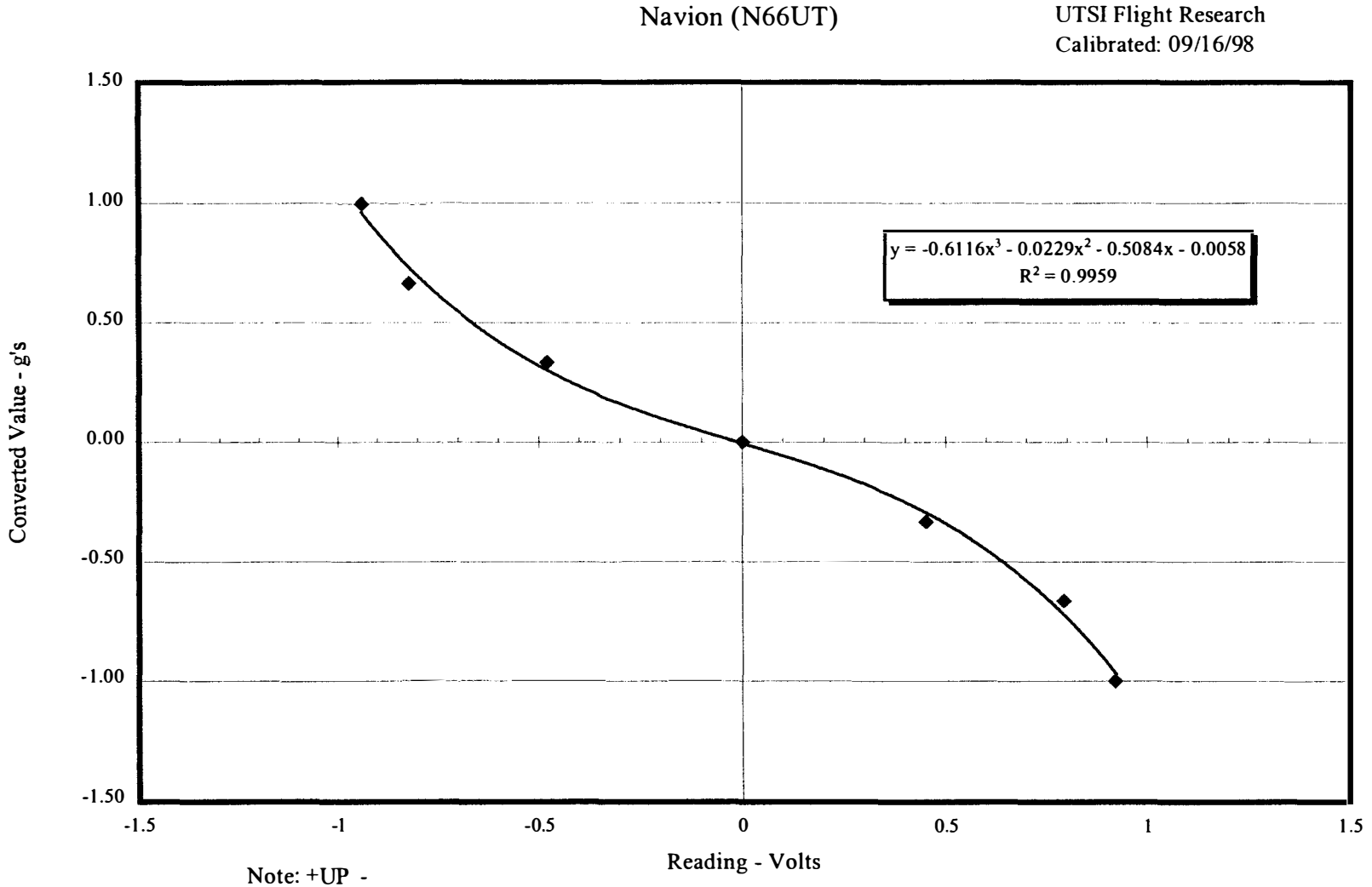


Figure A-13. Normal Acceleration (az) (CH09)

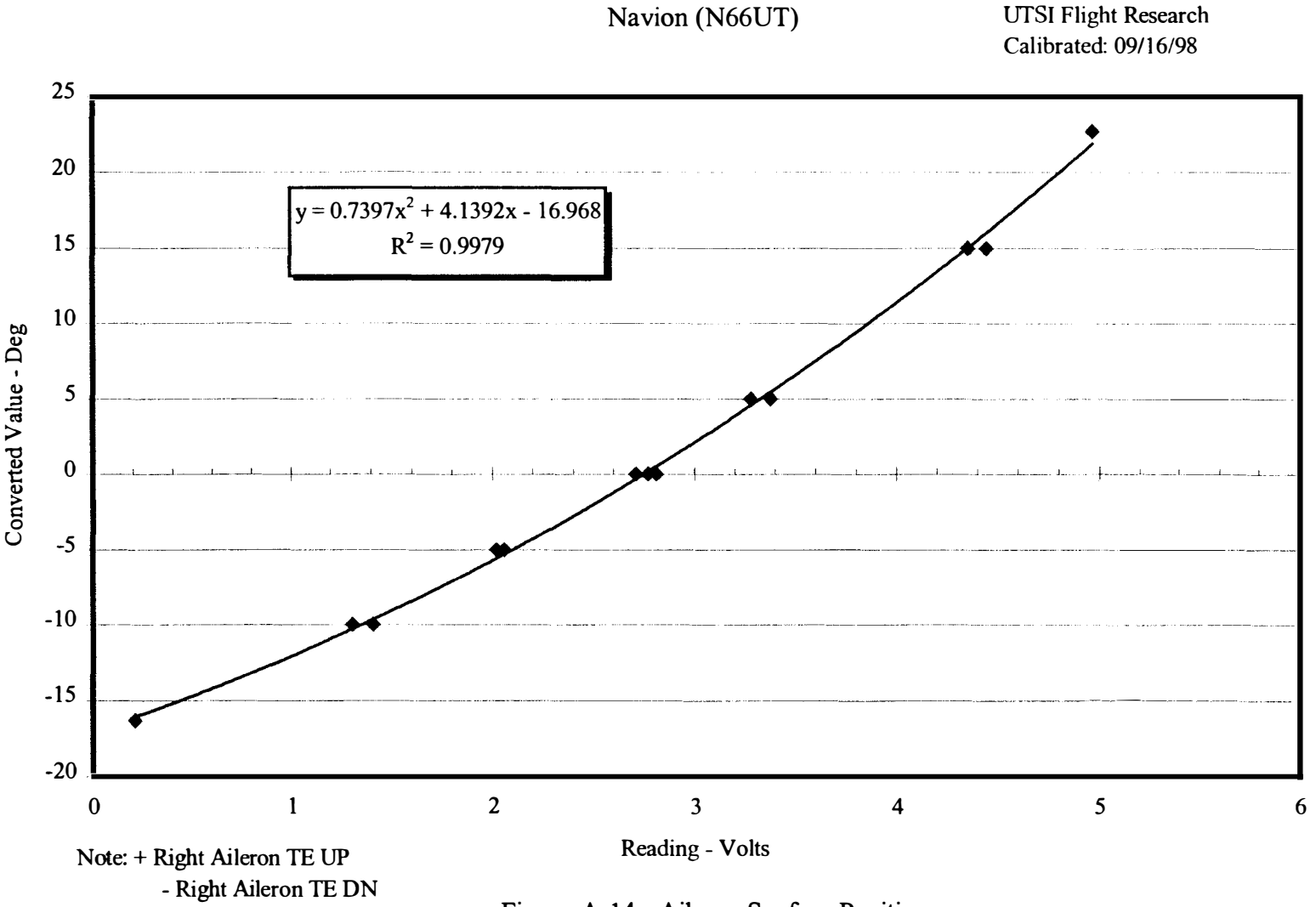
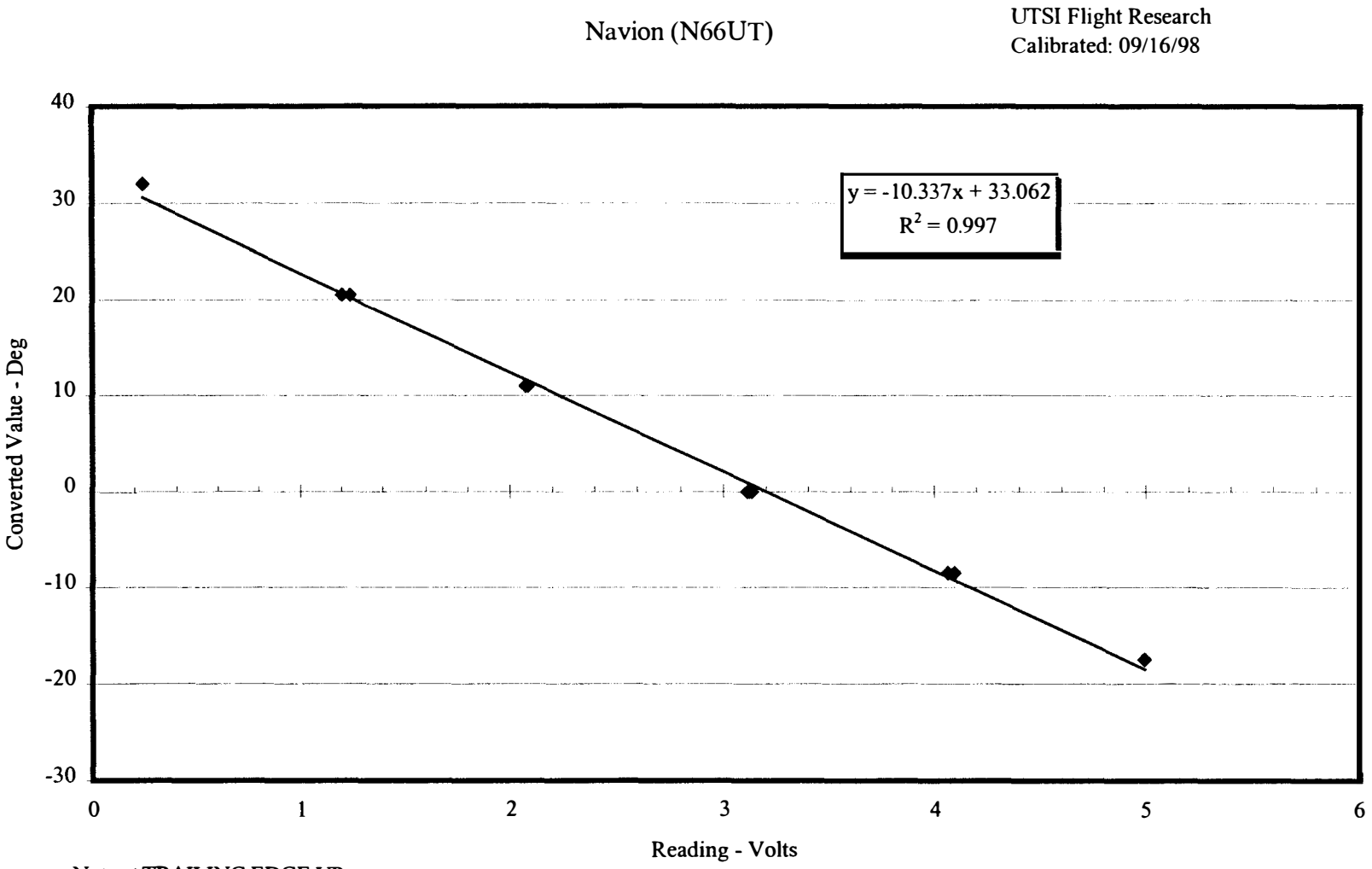
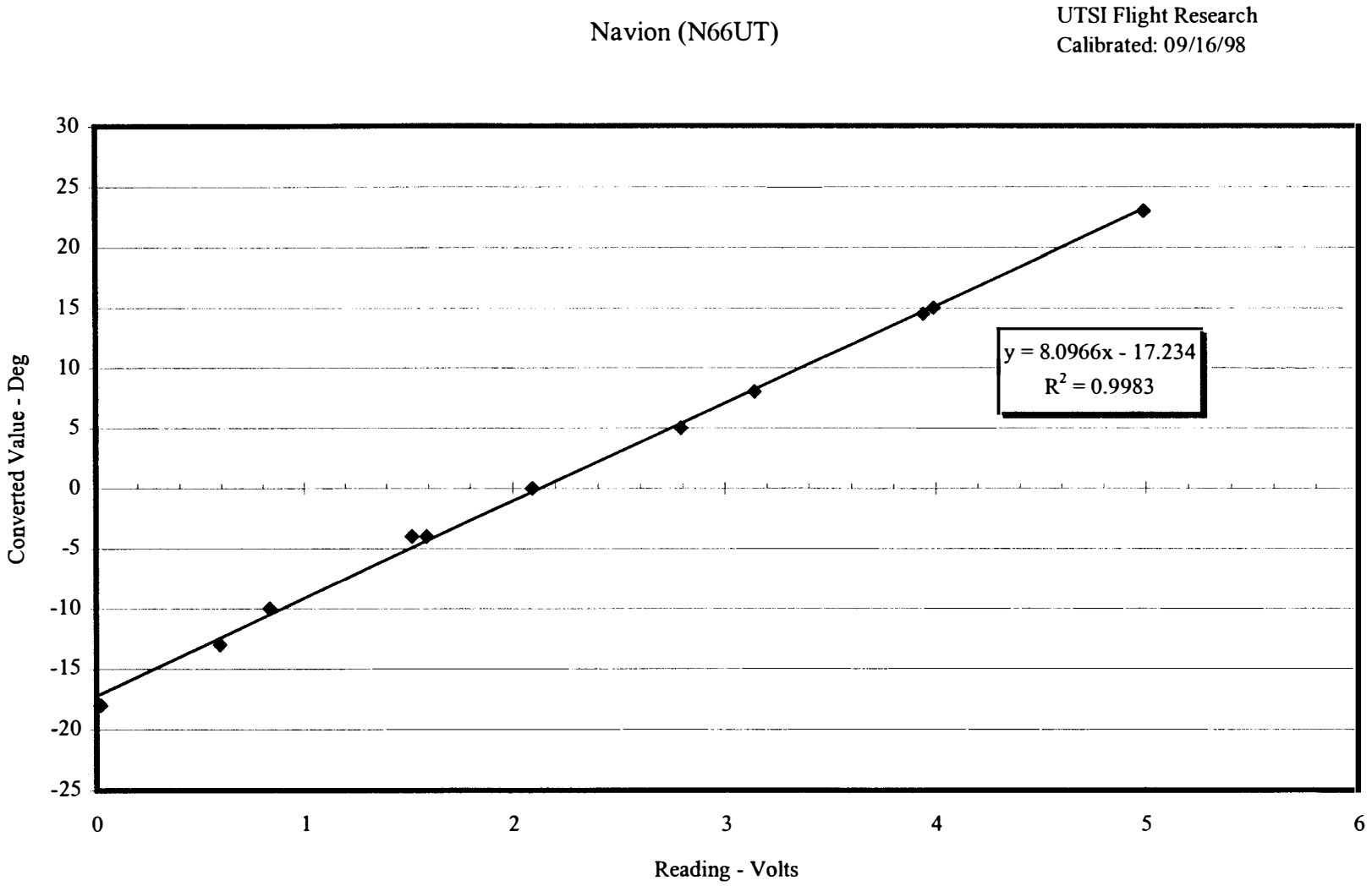


Figure A-14. Aileron Surface Position (CH10)



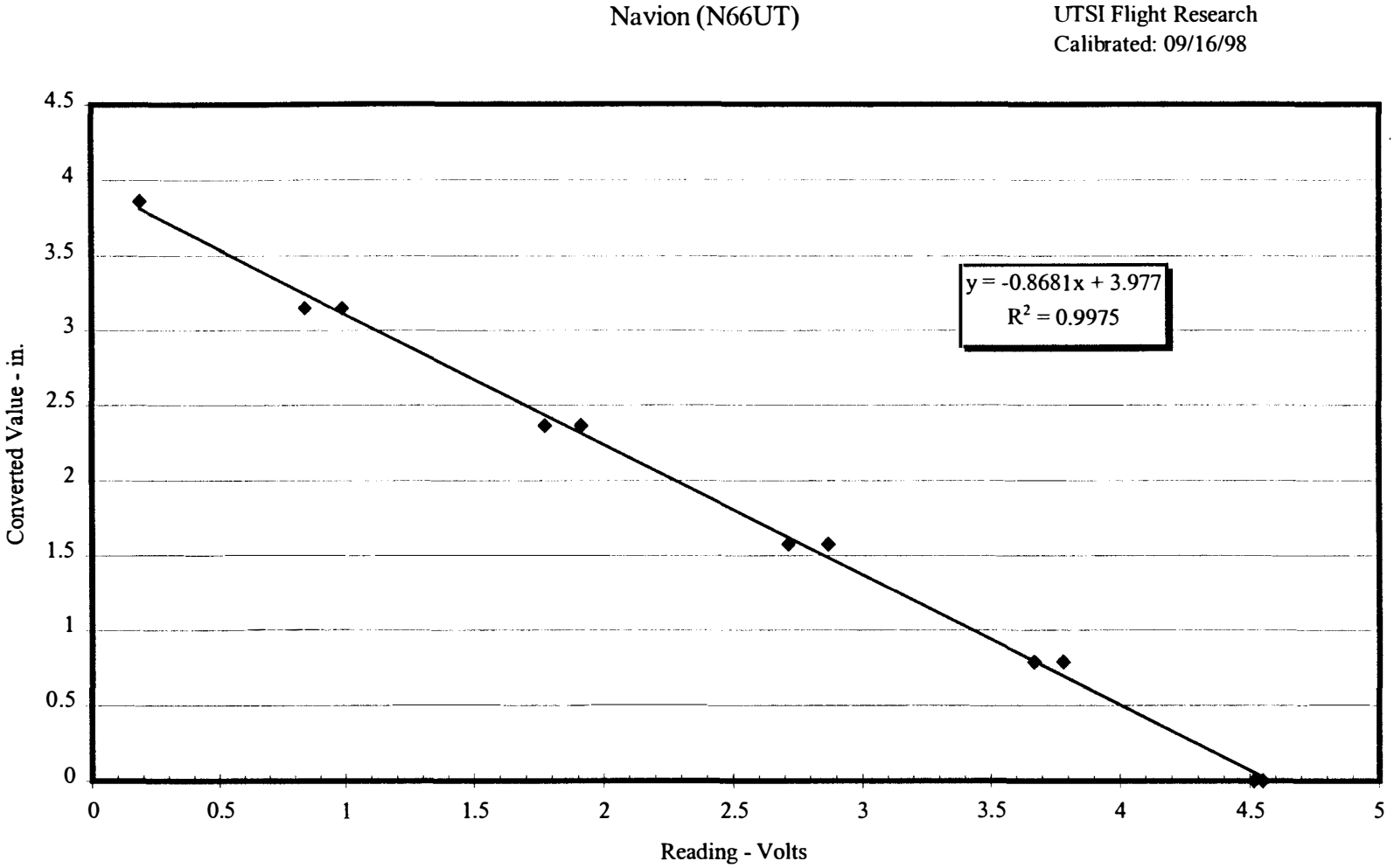
Note: +TRAILING EDGE UP
- TRAILING EDGE DN

Figure A-15. Elevator Surface Position (CH11)



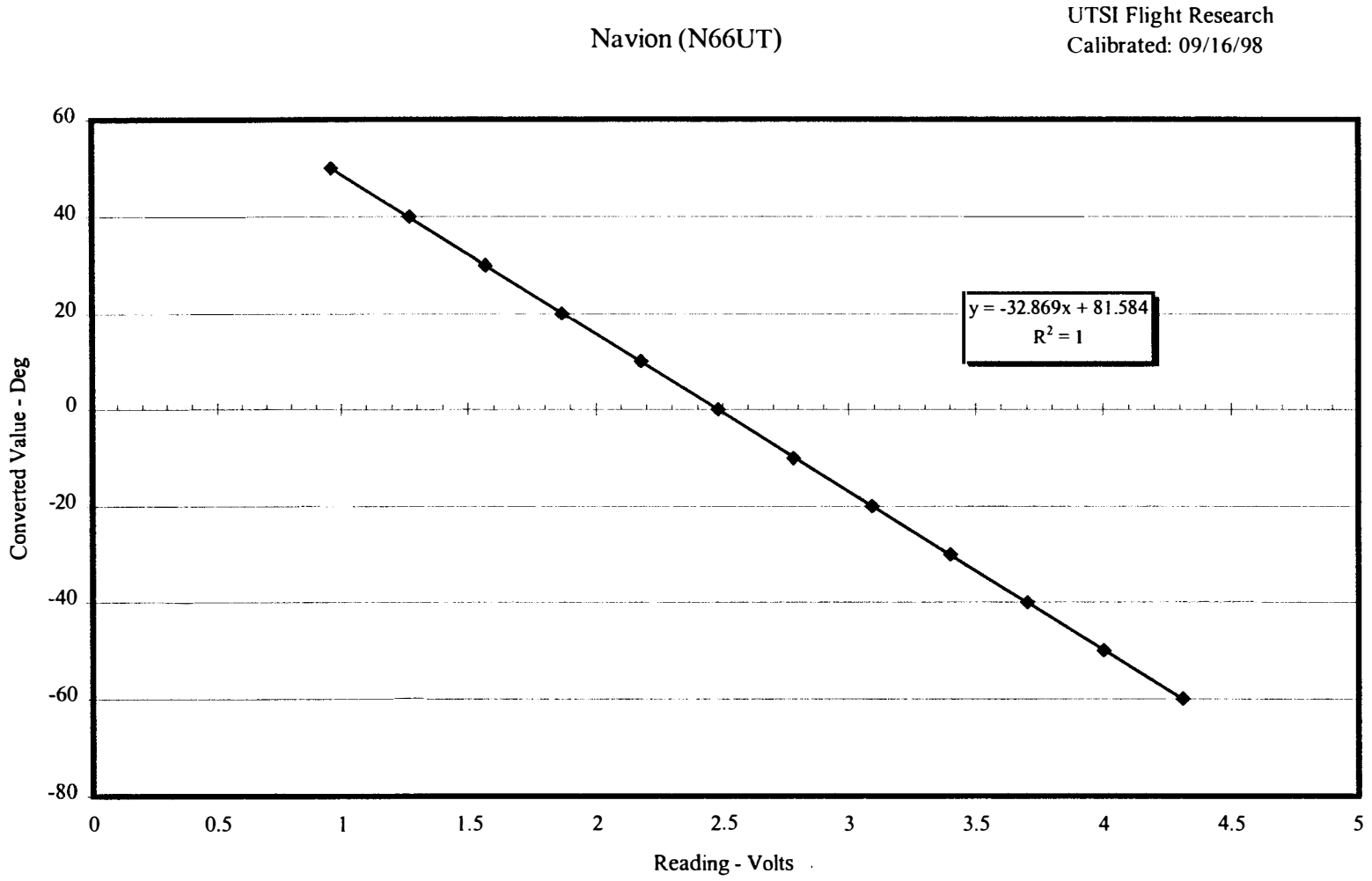
Note: +RT -LT

Figure A-16. Rudder Surface Position (CH12)



Note: + FWD
- AFT

Figure A-17. Std. Throttle Control Position (CH13)



Note: +RT -LT

Figure A-18. Roll Attitude (ϕ)
(CH14)

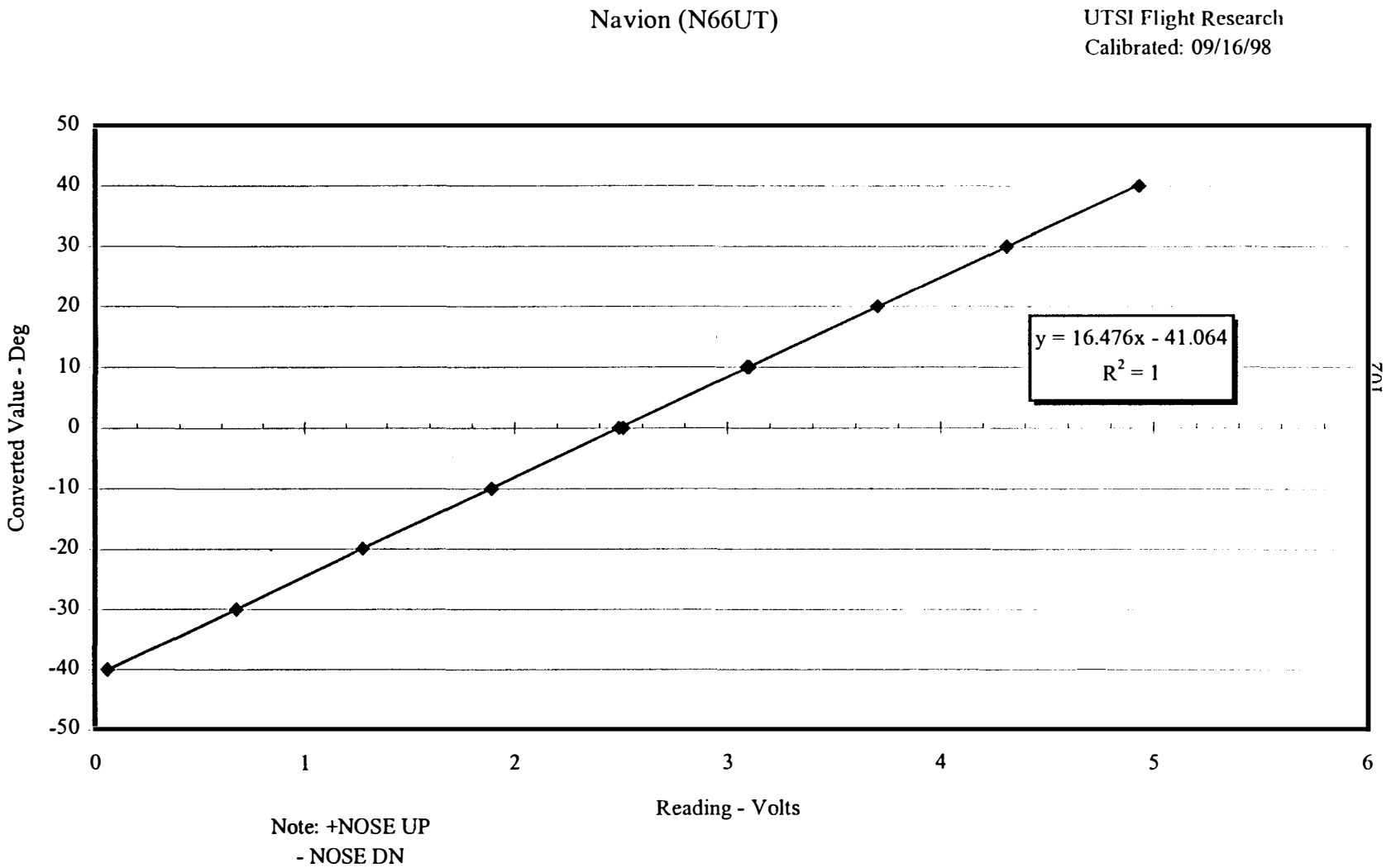


Figure A-19. Pitch Attitude (θ)
(CH15)

APPENDIX B

CIFER BODE PLOTS

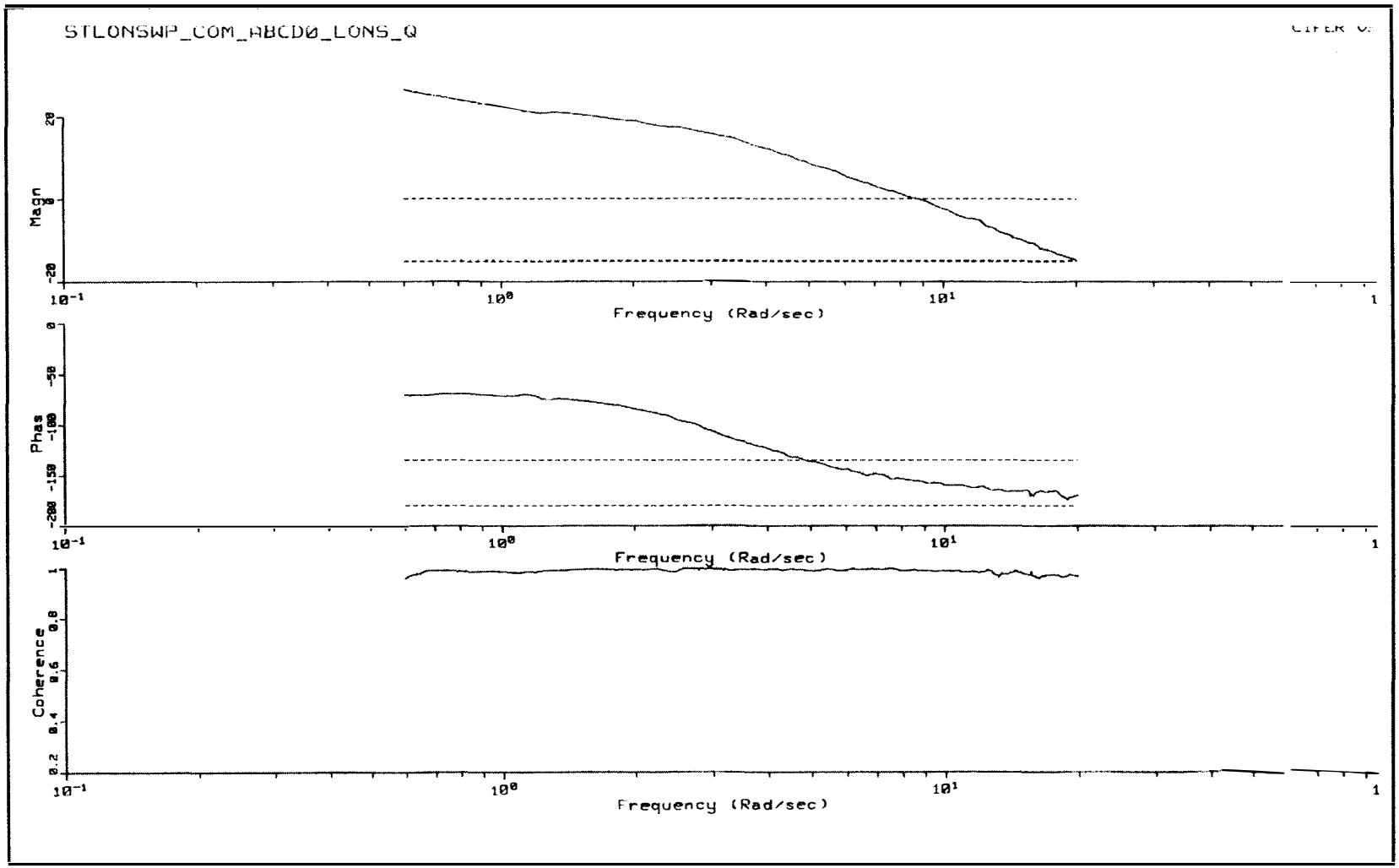


Figure B-1. Conventional Control Longitudinal Input To Pitch Attitude Response

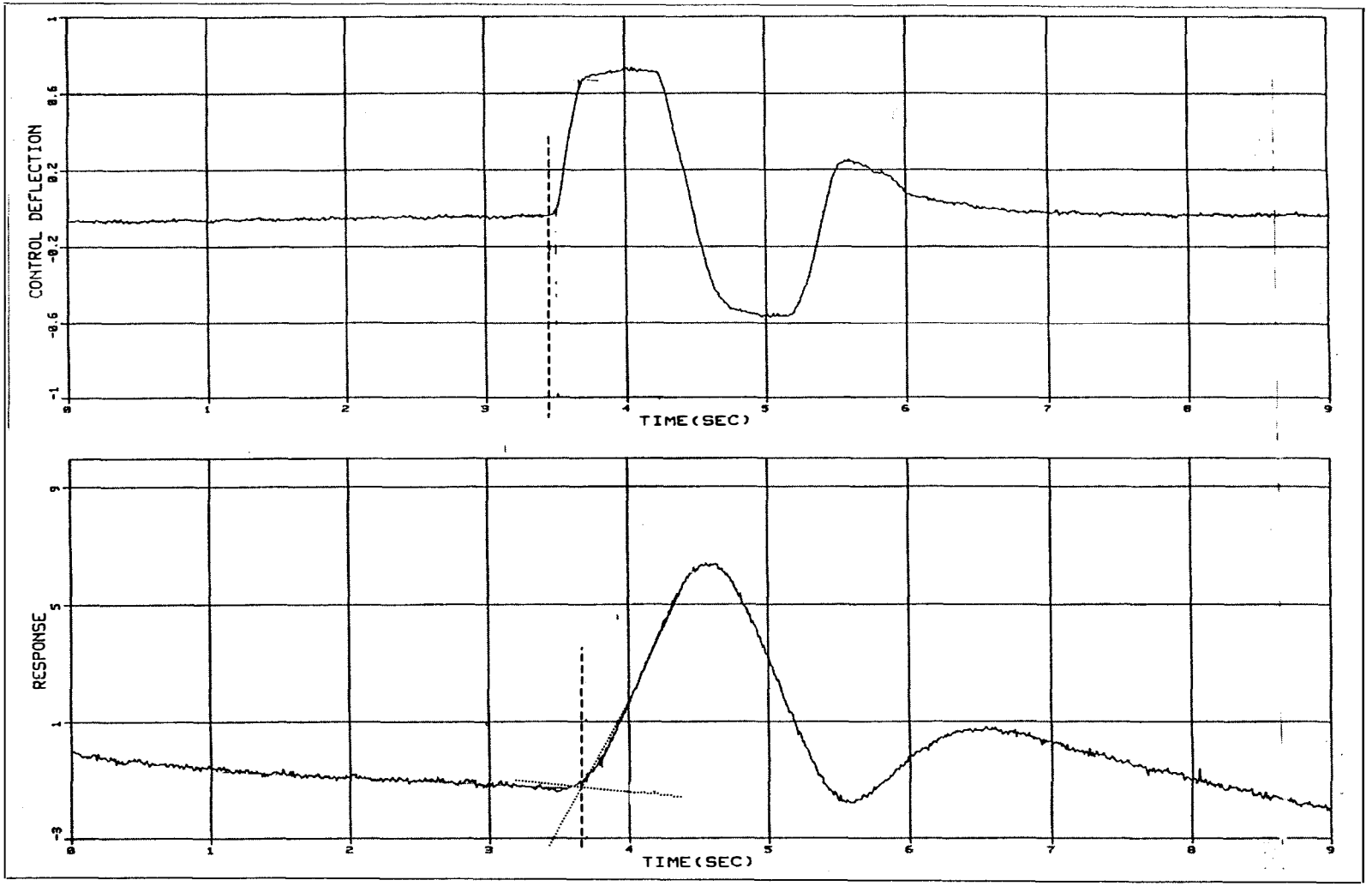


Figure B-2. Conventional Control Longitudinal Doublet

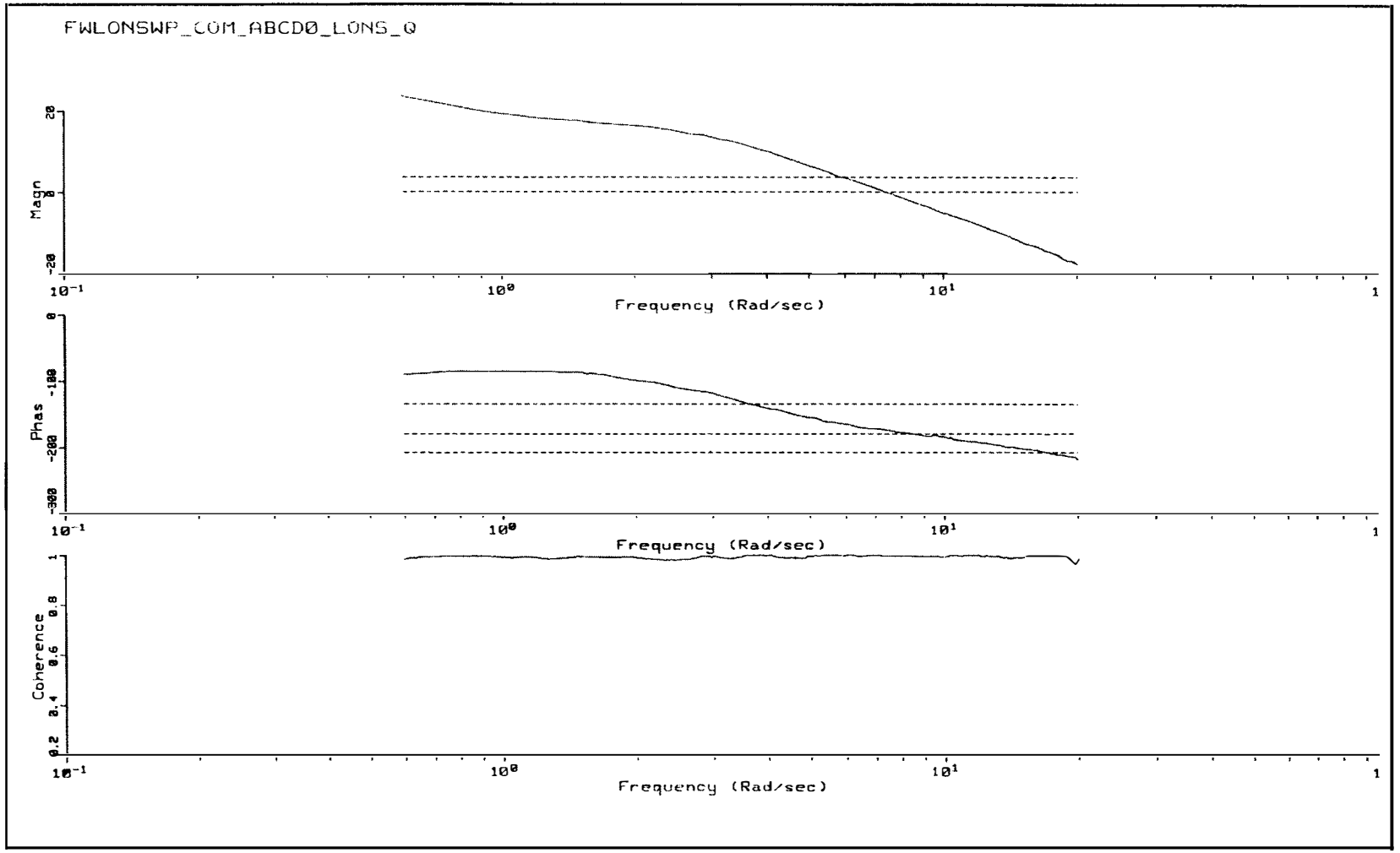


Figure B-3. FBW Control Longitudinal Input To Pitch Attitude Response

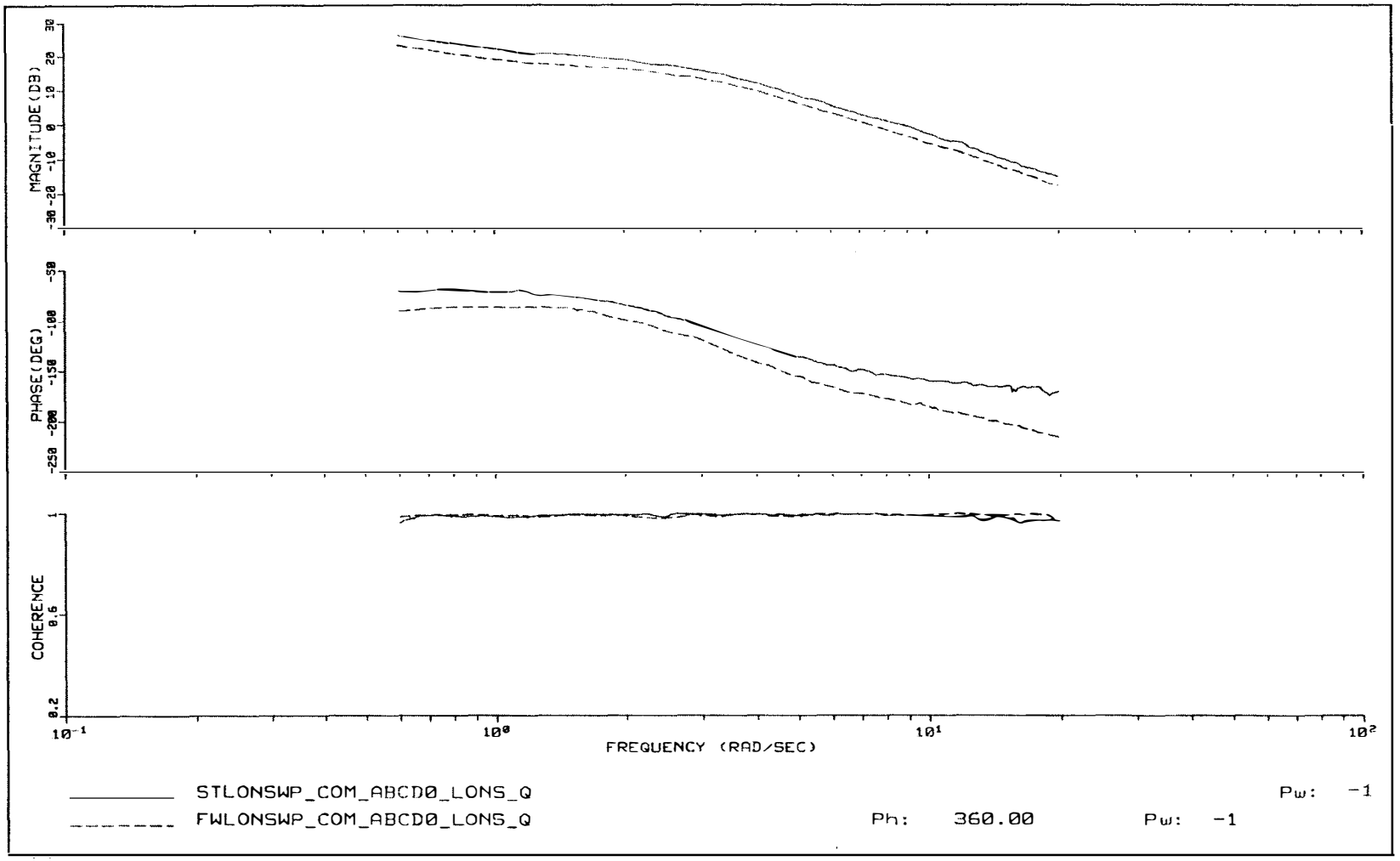


Figure B-4. Conventional and FBW Longitudinal Control Input to Pitch Attitude Response

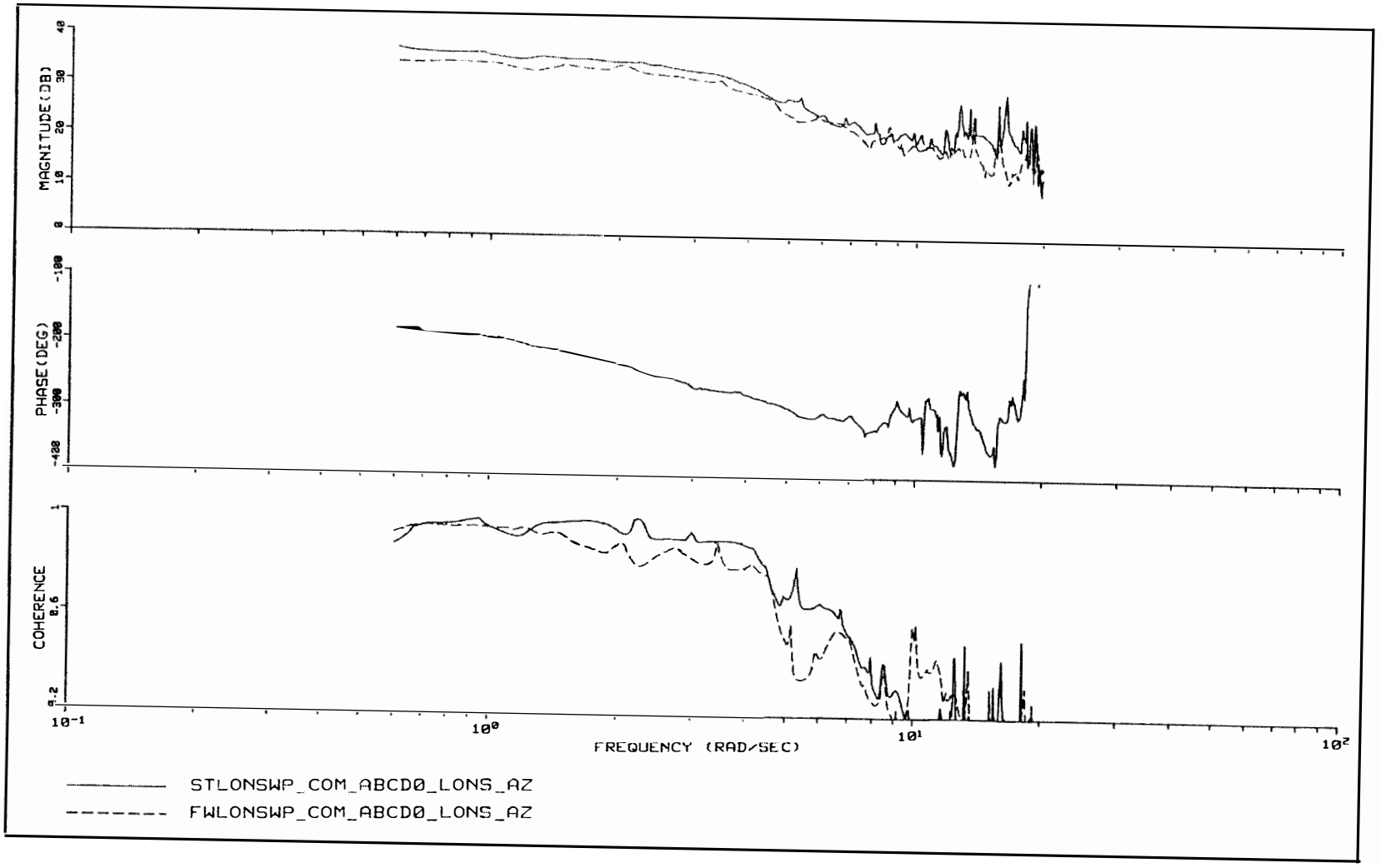


Figure B-5. Conventional and FBW Longitudinal Control Input to Heave

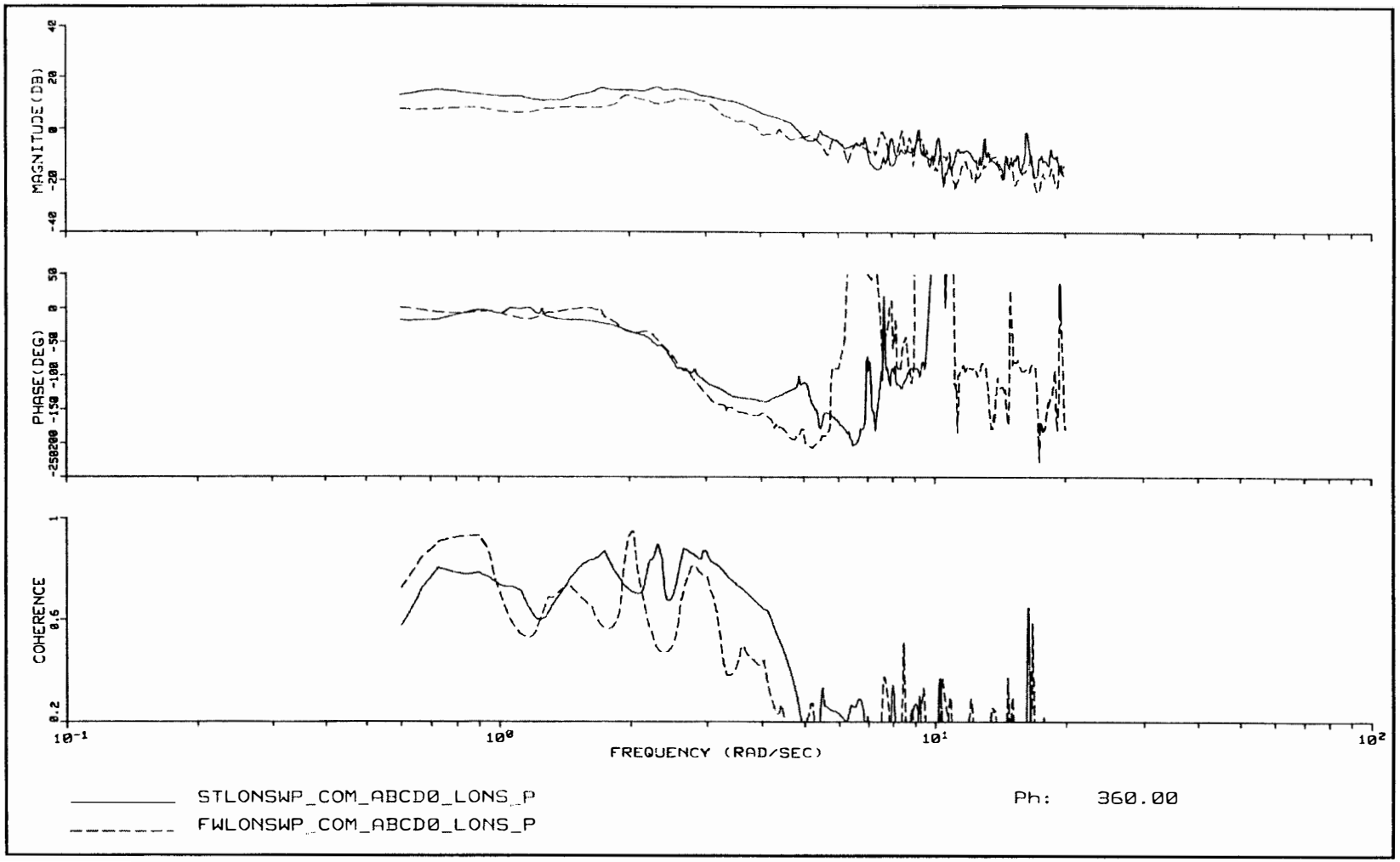


Figure B-6. Conventional and FBW Longitudinal Control Input to Roll Rate Response

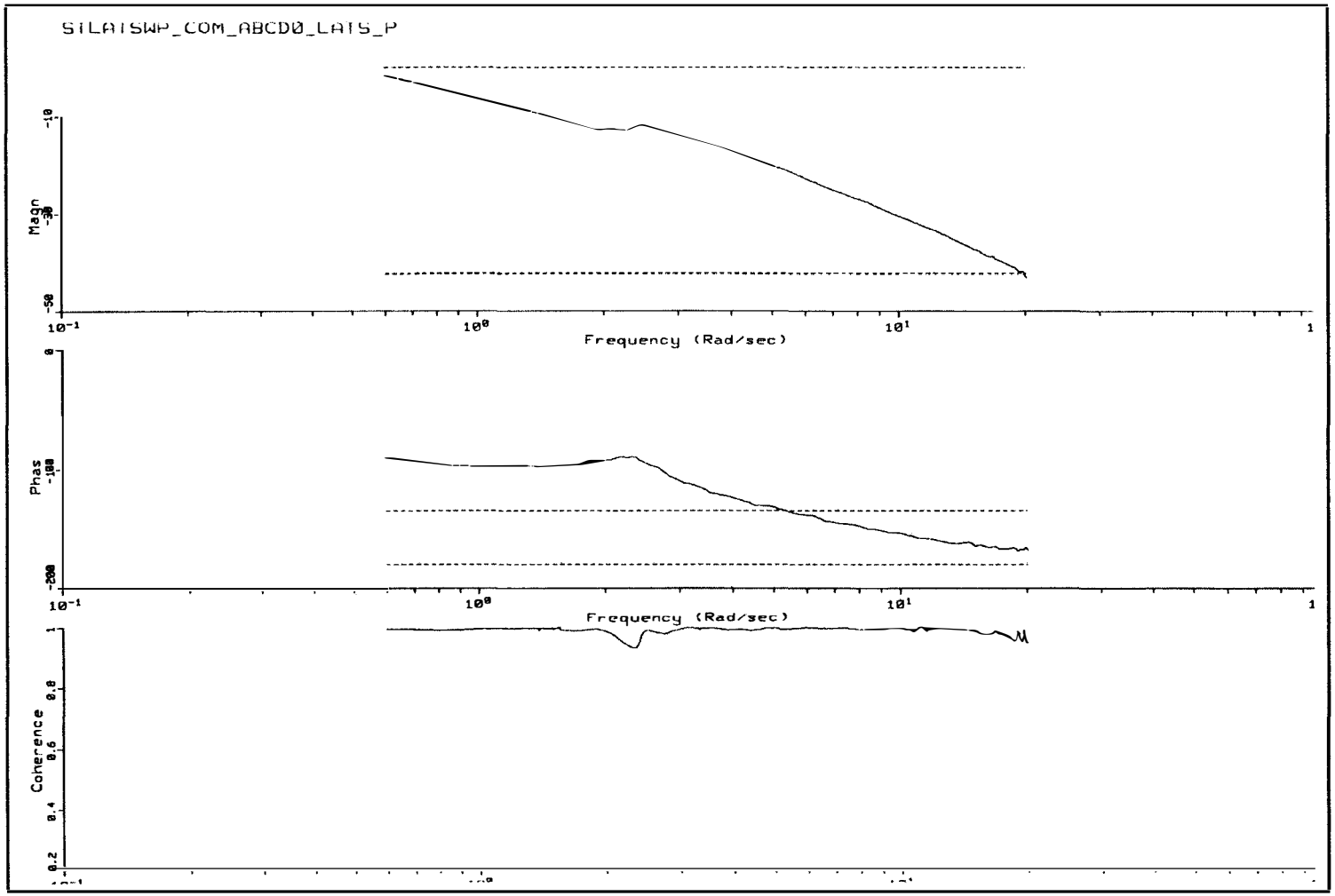


Figure B-7. Conventional Control Lateral Input To Roll Attitude Response

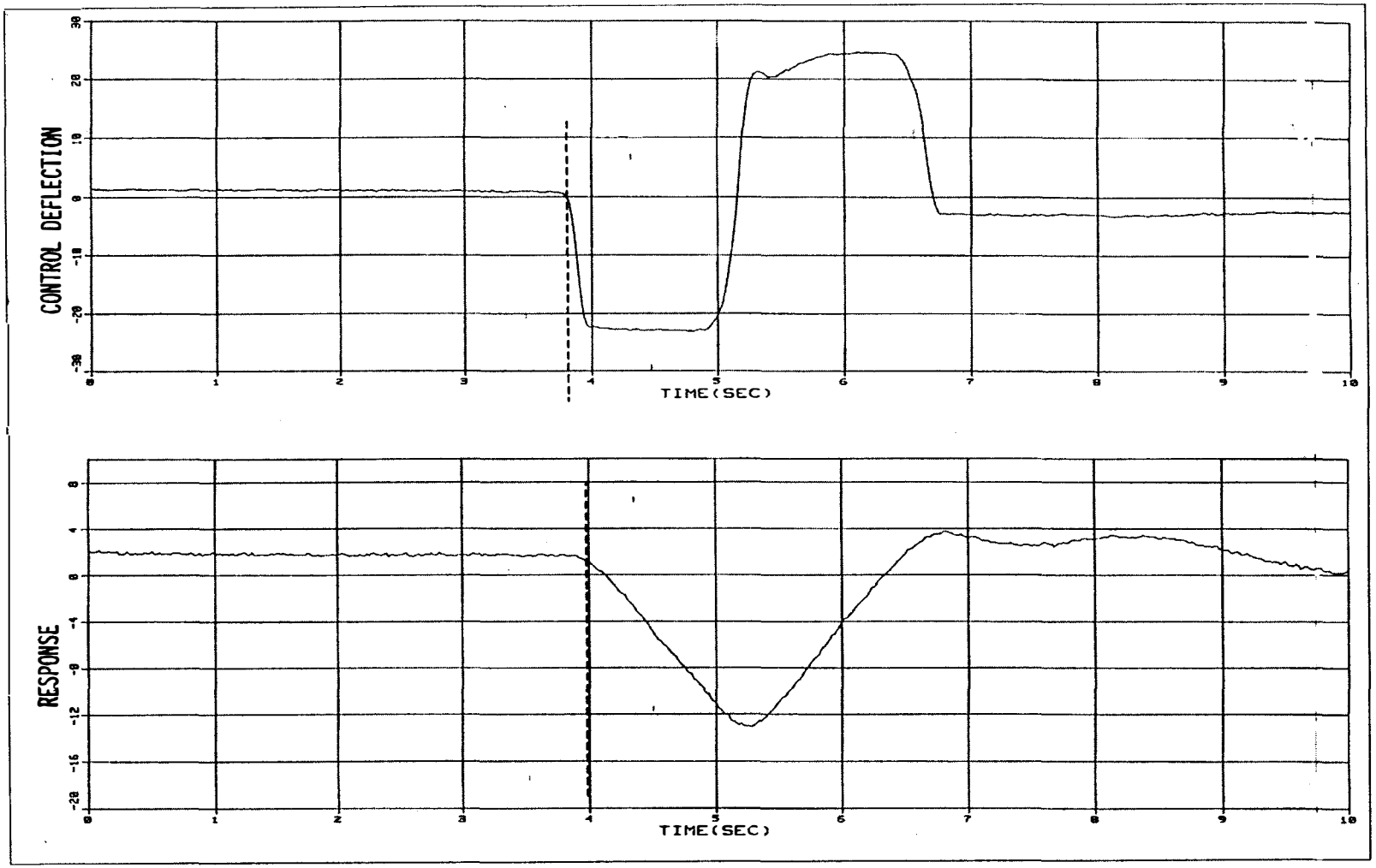


Figure B-8. Conventional Control Lateral Input to Roll Attitude Response Doublet

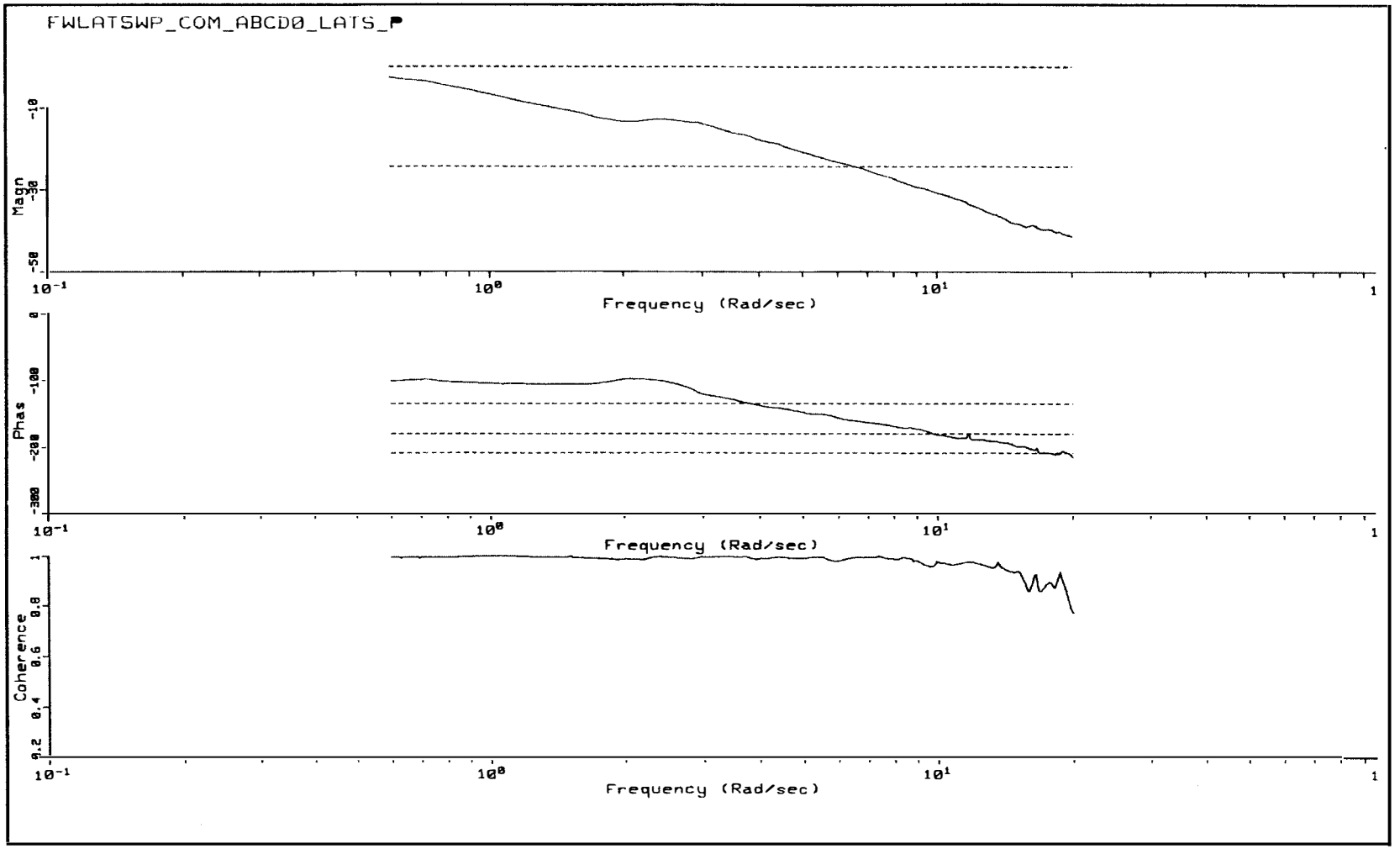


Figure B-9. FBW Control Lateral Input To Roll Attitude Response

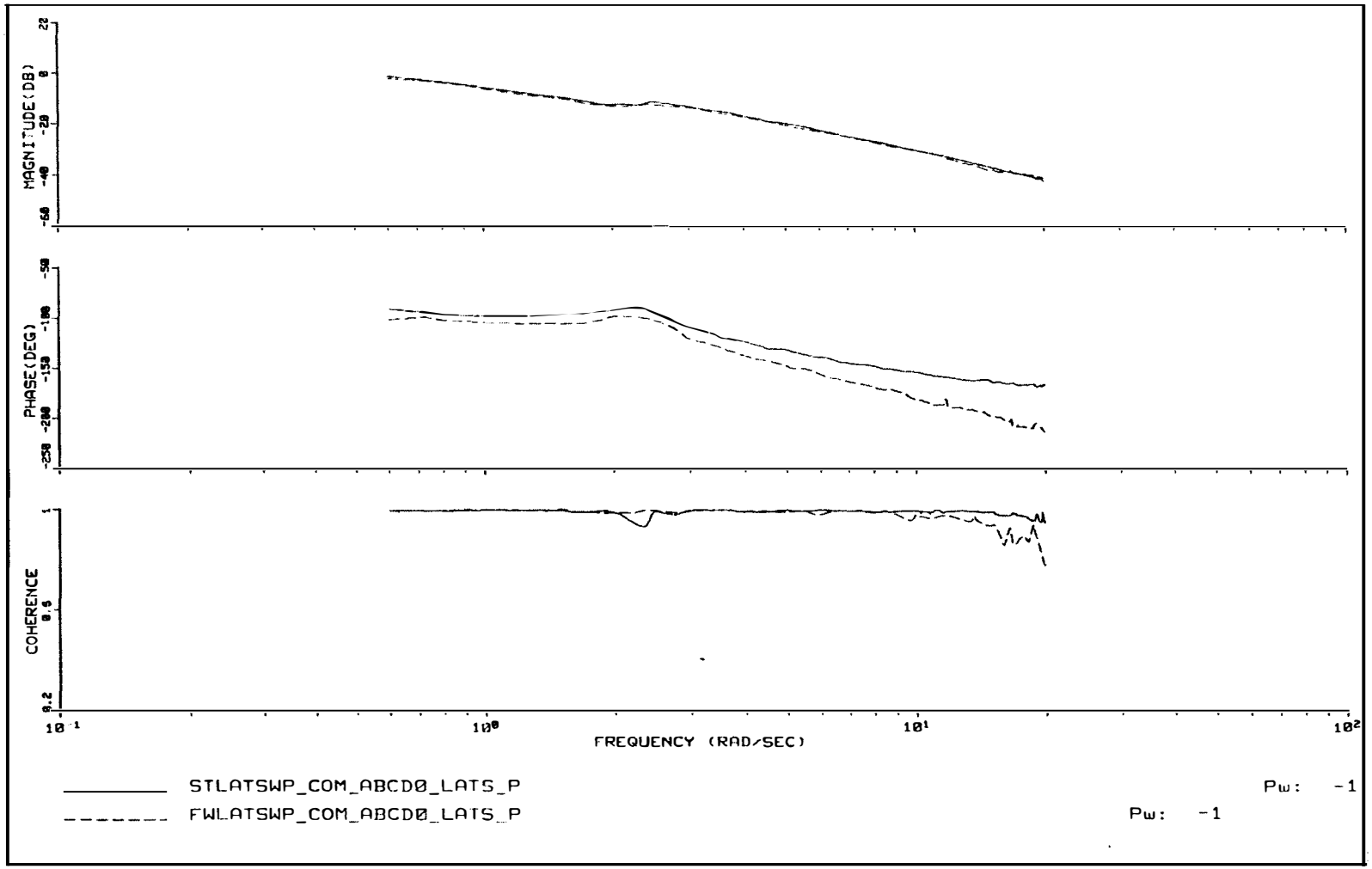


Figure B-10. Conventional and FBW Lateral Control Input to Roll Attitude Response

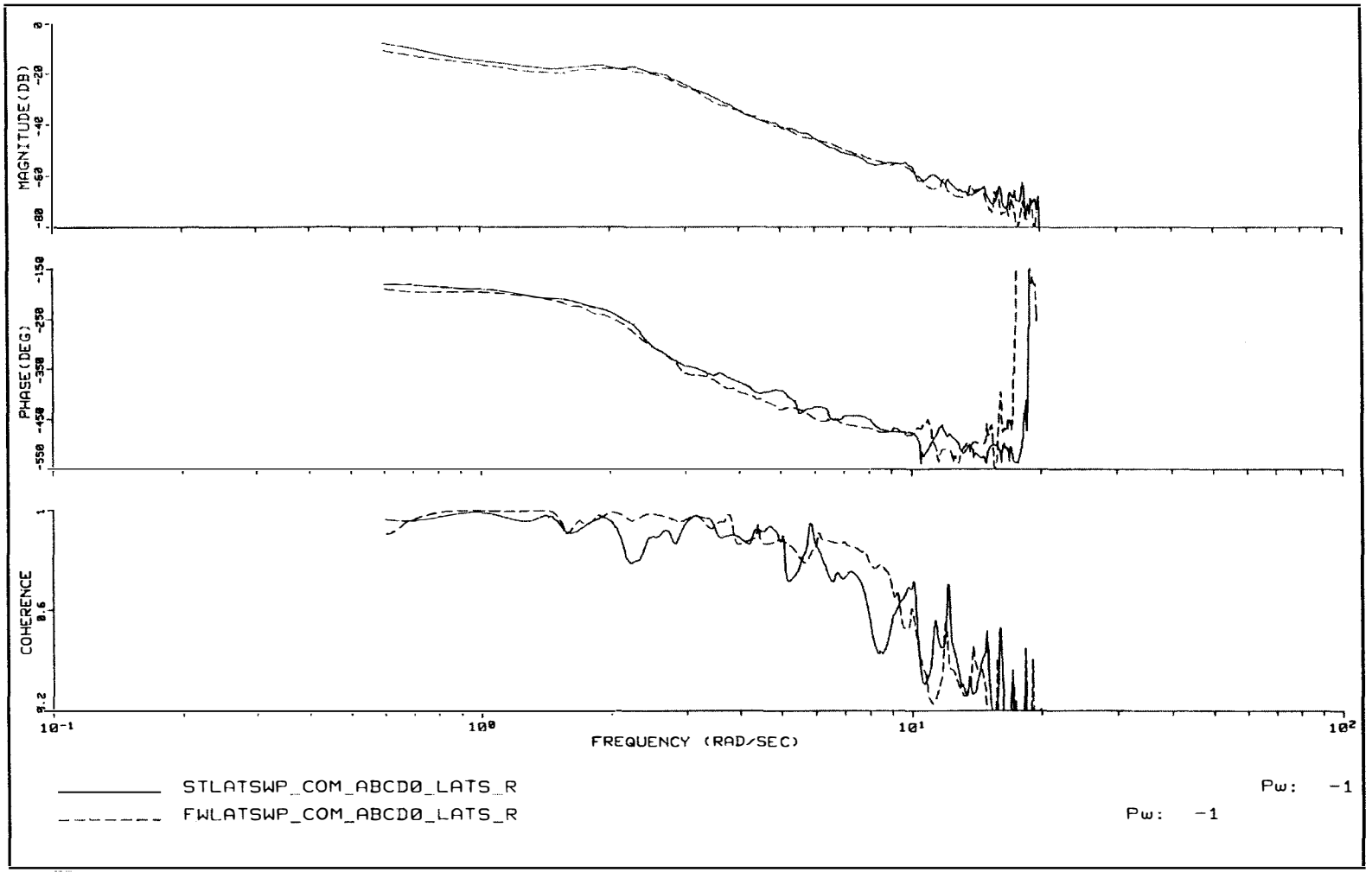


Figure B-11. Conventional and FBW Lateral Control Input to Yaw Attitude Response

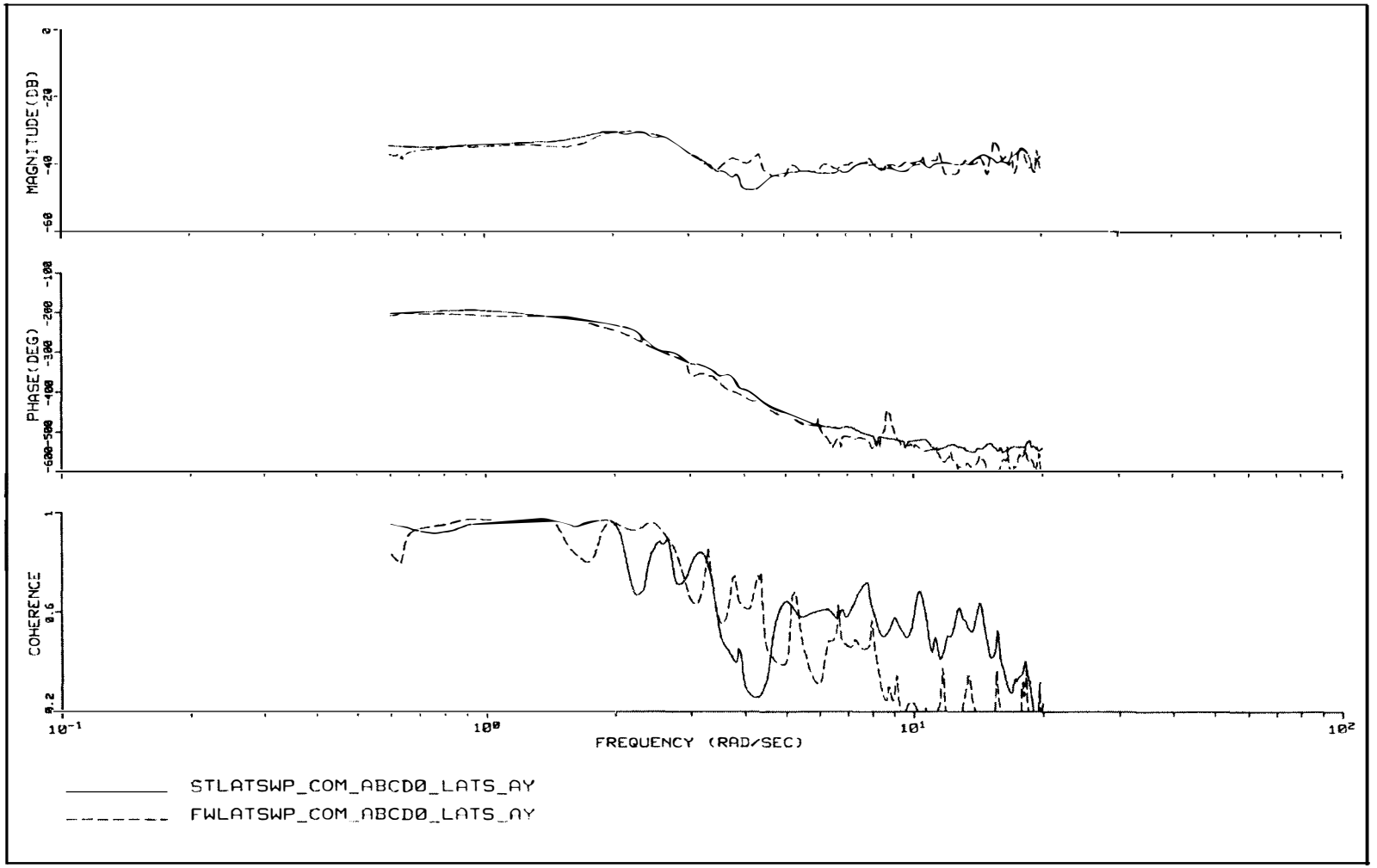


Figure B-12. Conventional and FBW Lateral Control Input To Lateral Acceleration

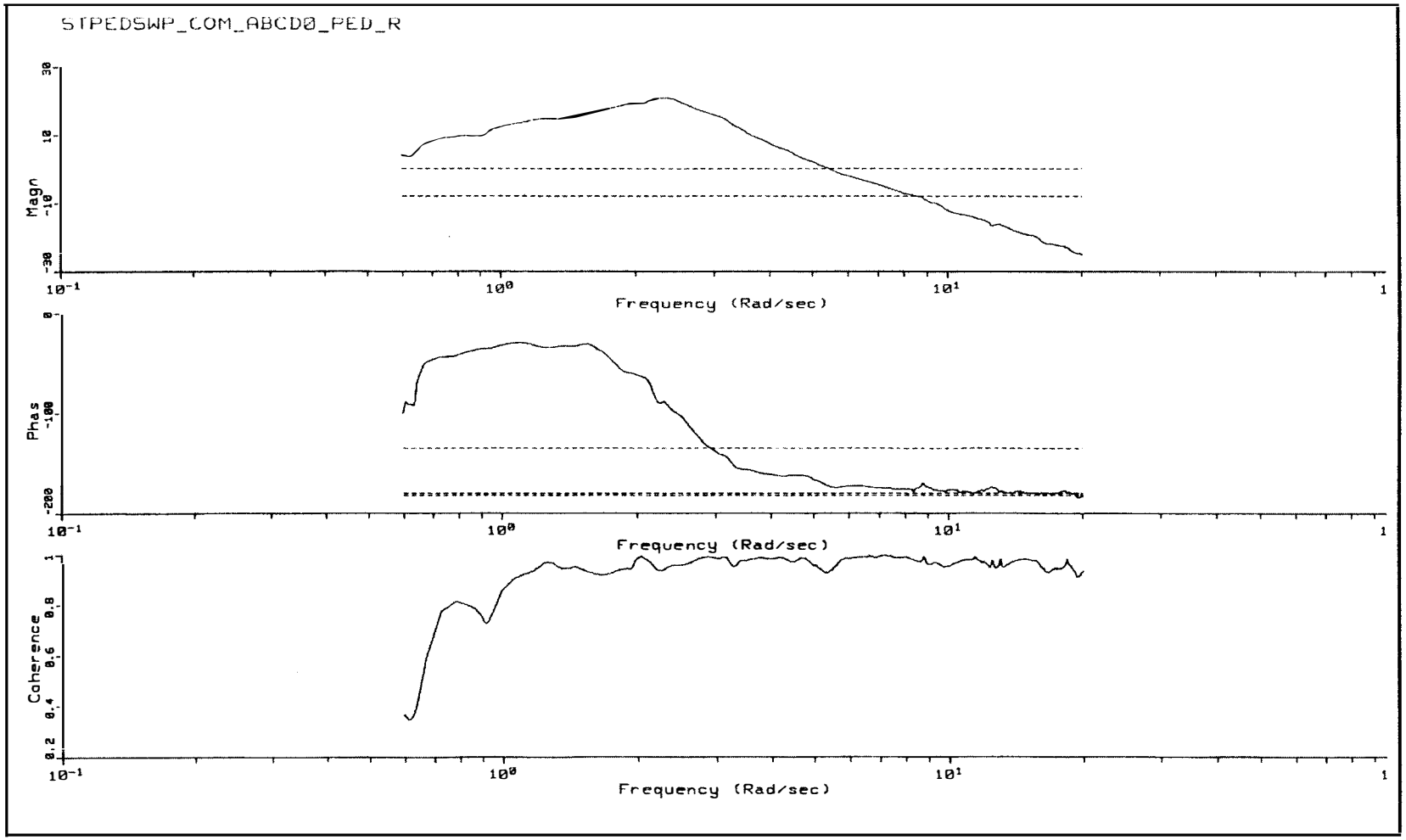


Figure B-13. Conventional Control Pedal Input to Yaw Attitude Response

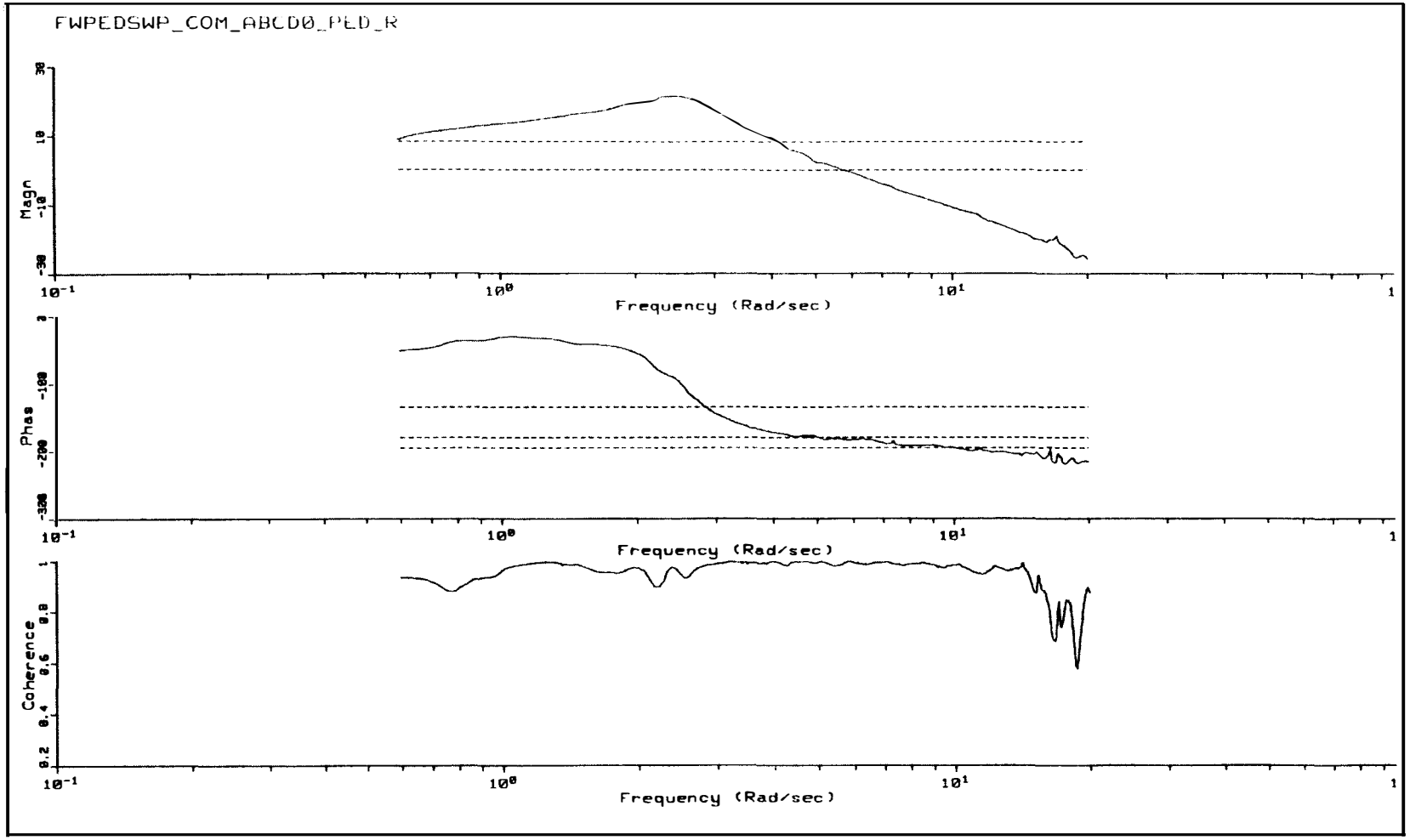


Figure B-14. FBW Control Pedal Input to Yaw Attitude Response

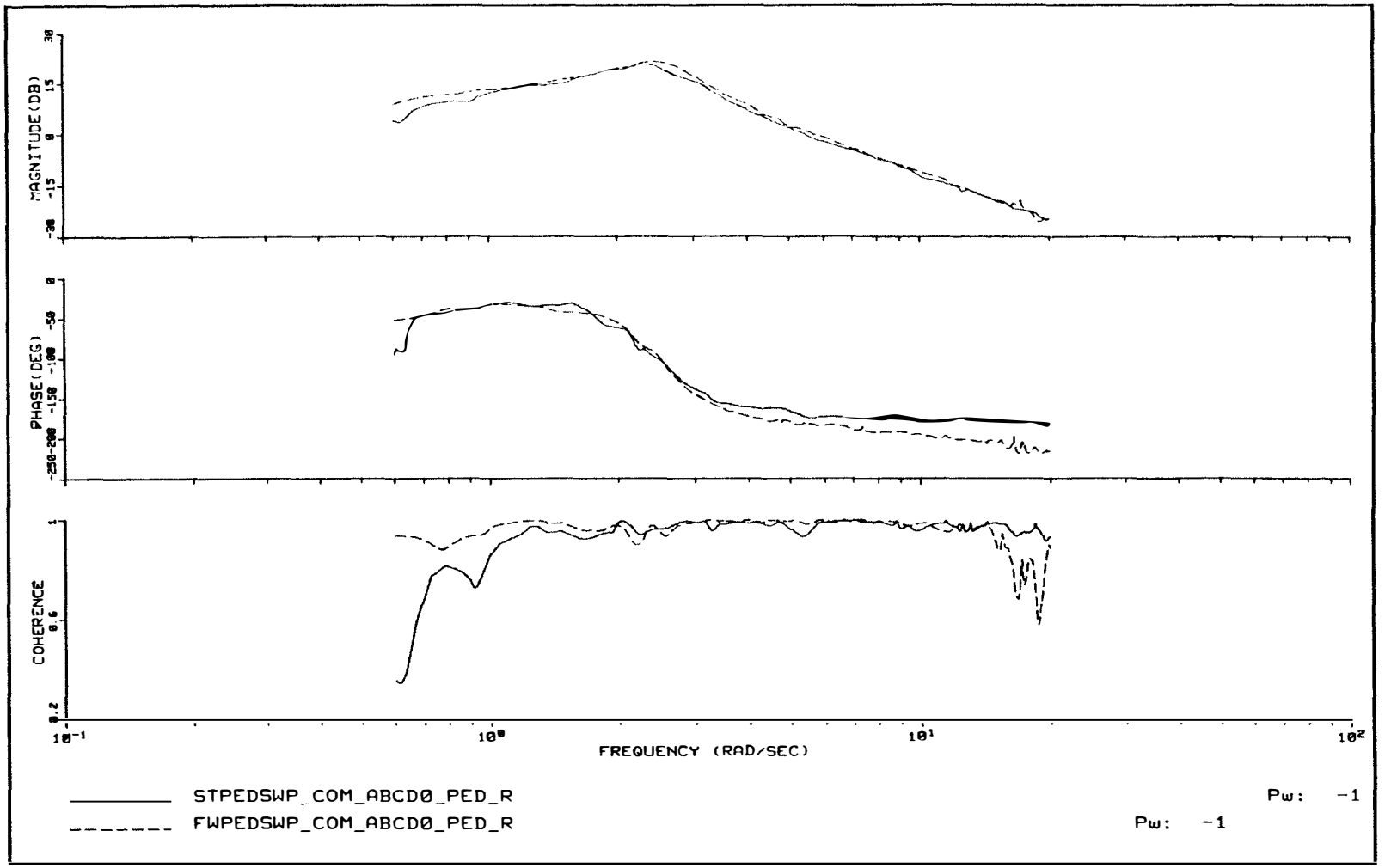


Figure B-15. Conventional and FBW Pedal Control Input to Yaw Attitude Response

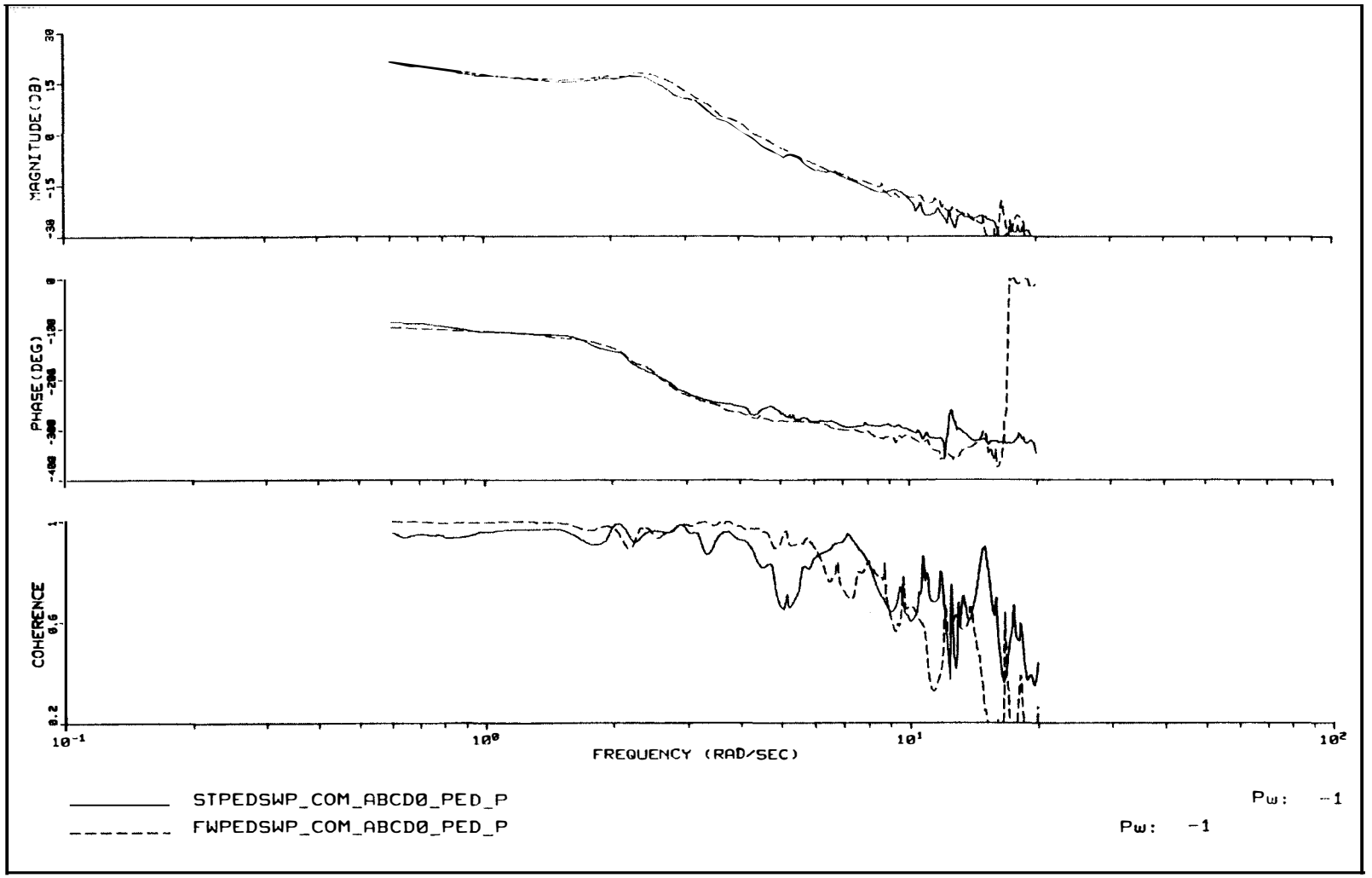


Figure B-16. Conventional and FBW Pedal Control Input to Roll Attitude Response

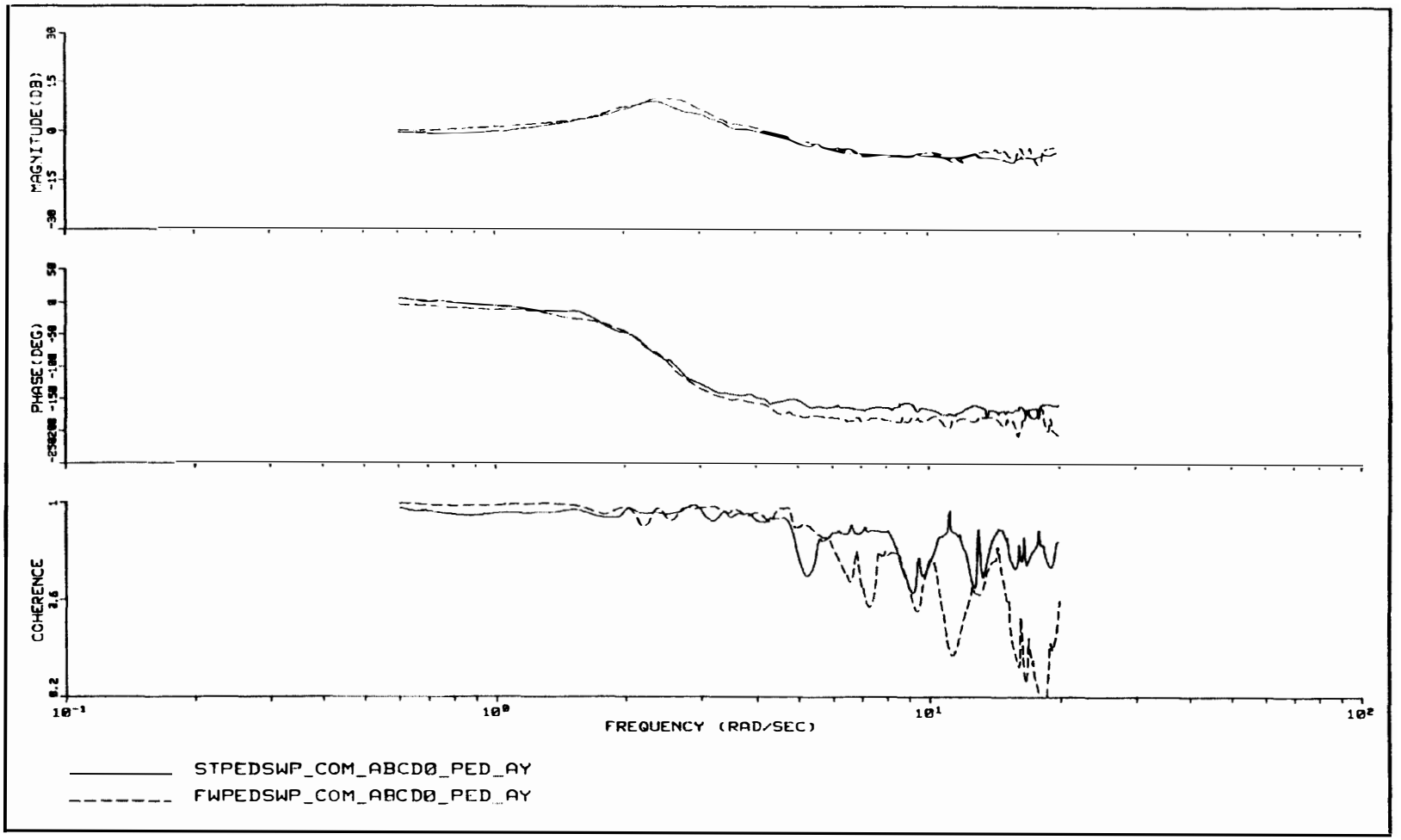


Figure B-17. Conventional and FBW Pedal Control Input to Lateral Acceleration

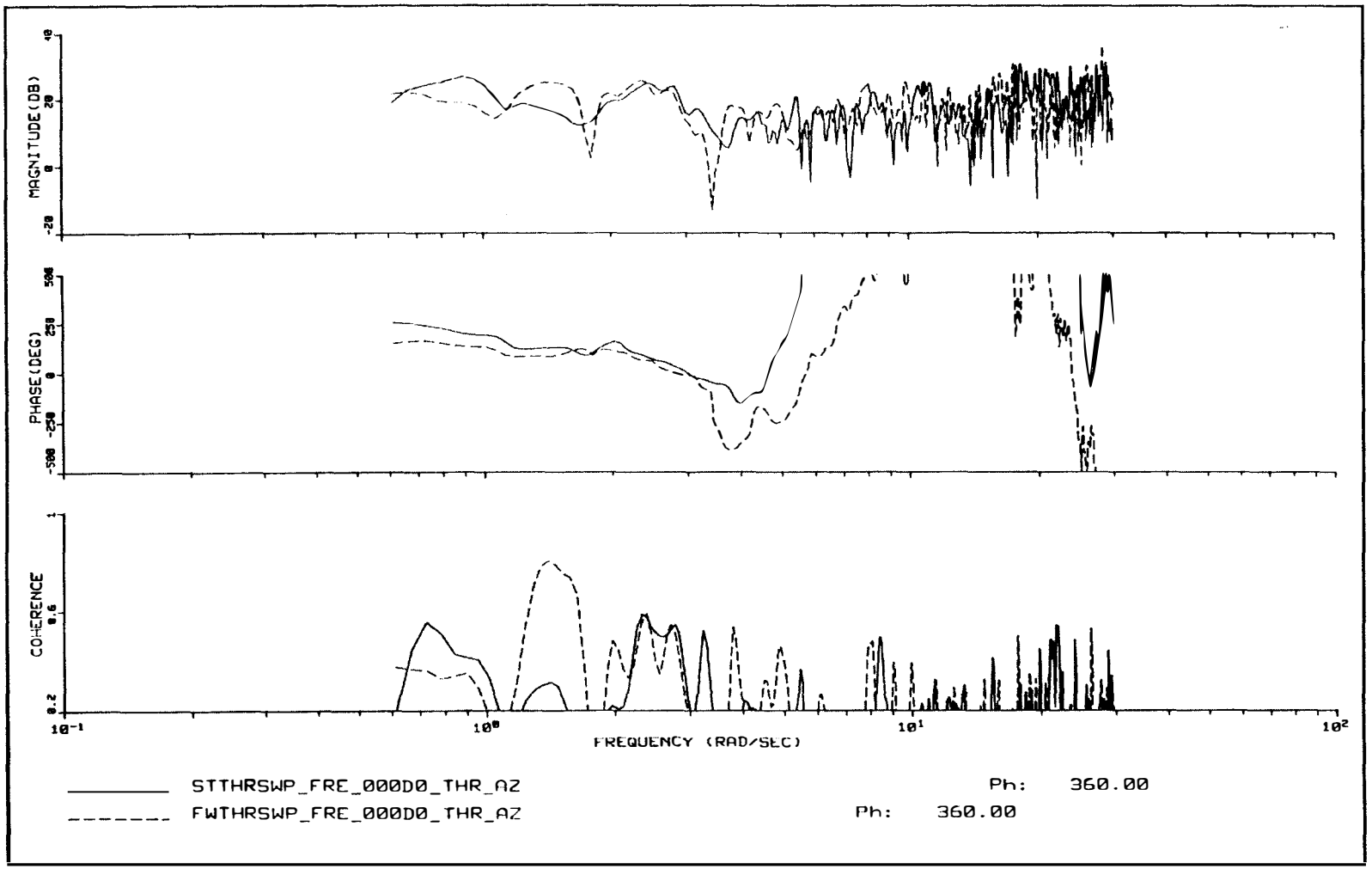


Figure B-18. Throttle Control Input to Longitudinal Acceleration

VITA

Randy Bolding was born in Mason City, Iowa on August 23, 1957. He attended schools in the public systems of Oklahoma City, Oklahoma, and Manly, Iowa, where he graduated from North Central High School in May 1975. He entered the United States Army in February of 1977 as a Military Policeman. In September 1981, he completed Rotary Wing Flight Training at Ft. Rucker, Alabama and was commissioned as a Warrant Officer. He completed an Associate of Science Degree in Pre-Engineering at Methodist College, Fayetteville, North Carolina, in 1986. He is a veteran of the Operation Urgent Fury/Grenada and Operation Desert Shield/Storm. He served a total of fourteen months working drug interdiction in the Bahamas for the State Department. Through use of continuing education programs, he was able to complete a Bachelor of Science degree in Business from the University of Maryland in 1994 and a Master of Science Degree in Engineering Technology from Murray State University, Murray, Kentucky in 1995. In August of 1997, he attended The University of Tennessee Space Institute, graduating in December 1998 with a Master of Science degree in Aviation Systems with a follow on assignment to attend the U.S. Naval Engineering Test Pilot School at Patuxent River, Maryland.