

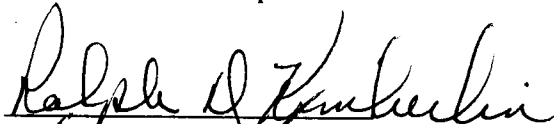
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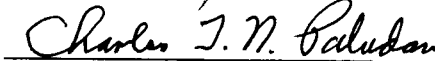
I am submitting herewith a thesis written by Eric B. Treworgy entitled "CH-46E Helicopter Automatic Flight Control System Evaluation." I have examined the final copy of this thesis for form and content and recommend that it be accepted in partial fulfillment of the requirements for the degree of Master of Aviation Systems.



Robert B. Richards, Major Professor

We have read this thesis and
recommend its acceptance:





Accepted for the Council:



Associate Vice Chancellor and
Dean of The Graduate School

**CH-46E HELICOPTER
AUTOMATIC FLIGHT CONTROL SYSTEM
EVALUATION**

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

Eric Brent Treworgy
August 1998

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DISCLAIMER

The results contained within this thesis were developed at the Naval Air Warfare Center, Aircraft Division, Patuxent River, Maryland. The discussion of the data, conclusions and recommendations presented are the opinion of the author and should not be construed as an official position of the United States Department of Defense, the Naval Air Systems Command or the Naval Air Warfare Center.

DEDICATION

To my wife, Serena,
and my two children, Tyler and Alicia,
whose love and enduring patience
have made it possible for me
to complete this program.

ACKNOWLEDGMENTS

Special thanks go to several gentlemen that have been invaluable to this undertaking. Jim Busey whose engineering talent made this testing possible and enjoyable. Frank Rudley who was never too busy to lend an ear and make the incomprehensible, comprehensible. Finally, Bob Richards who was always available for advice whenever it was needed. Again, thank you for all your help.

ABSTRACT

This thesis seeks to validate that the proposed production Automatic Flight Control System (AFCS) computers installed in the CH-46E helicopter reduce pitch extensible link actuator oscillations and provide the same aircraft handling qualities as the current AFCS computers. The testing was conducted in two phases at Naval Air Station Patuxent River, MD.

The first phase consisted of ground testing during September 1995. The ground testing included electric-hammer rap tests on the stiffened AFCS computer-mounting shelf to document the vibratory characteristics, and frequency response testing of the pitch and roll extensible link actuators to document the frequency response differences between the test aircraft and the baseline aircraft. The second phase consisted of flight tests that were conducted with the proposed production -5 AFCS computers and the current -4 AFCS computers installed. Flight tests were conducted from November 1995 to April 1996 in a fleet representative CH-46E helicopter with the exception of the installed instrumentation package. Flight-testing included classic helicopter flying qualities testing techniques to include static and dynamic control response testing and fleet representative mission maneuvers.

The stiffened AFCS shelf increased the minimum resonant frequencies to a minimum of 39 Hz; well above the aircraft's resonant frequencies of 13.2 and 26.4 Hz. During flight conditions that had been determined to cause pitch chatter and during other mission representative flight maneuvers, the -5 AFCS computers reduced pitch extensible

link actuator (PELA) oscillations and attenuated high frequency amplitudes from the levels exhibited by the test aircraft configured with -4 AFCS computers. Though not proved conclusively, the aircraft's weight appeared to have an influence on PELA response, a heavier aircraft causing less PELA frequency content. The flying qualities of the aircraft with the -5 AFCS computers installed were the same as those exhibited by the current -4 AFCS computers.

A recommendation was made to replace the -4 AFCS computers with the -5 AFCS computers. Additionally, recommended that further testing be conducted with -5 AFCS computers:

- 1) at differing aircraft gross weights to determine the extent that aircraft weight influences PELA frequency responses for all mission maneuvers, and
- 2) on an aircraft with a known history of pitch chatter while in the -4 AFCS computer configuration to confirm pitch chatter elimination.

PREFACE

The testing that is discussed in this thesis is the sum of works conducted by an integrated test team. The test team included U.S. government engineers, military officers, and industry contracted personnel. As the project officer, the author was responsible for the test planning, test execution, data reduction, data analysis, and report of test results of this evaluation. In addition, the author flew a majority of the test flights.

The original results of this evaluation were a five-page message report that did not include detailed analysis. This thesis amplifies that report and includes additional data reduction, analysis, conclusions, and recommendations.

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LIST OF SYMBOLS AND ACRONYMS

Symbol/ Acronym	Definition	Symbol/ Acronym	Definition
AFC	AirFrame Change	ND	nose down
AFCS	Automatic Flight Control System	Nf	power turbine speed
AGL	Above Ground Level	Nr	rotor revolution speed
APU	Auxiliary Power Unit	NU	nose up
ASE	Automatic Stabilization Equipment	OAT	Outside Air Temperature
avg	average	PELA	Pitch Extensible Link Actuator
BHC	Boeing Helicopter Company	q	pitch rate, vehicle referenced
Buno	aircraft serial number	Q	torque, %
C	Celsius	r	yaw rate, vehicle referenced
C.G.	Center of Gravity	rev	revolution(s)
CPT	Control Position Transducer	rpm	revolutions per minute
DASH	Dual AirSpeed Hold	RTPS	Real-time Telemetry Processing System
db	decibel(s)	sec	second(s)
deg	degree(s)	shp	shaft horsepower
ELA	Extensible Link Actuator	SR&M	Safety, Reliability, & Maintainability
ft	feet	θ	pitch angle, Earth referenced
g	normal acceleration	$\dot{\theta}$	pitch rate, Earth referenced
G.W.	gross weight	ϕ	roll angle, Earth referenced
HAC	Hover and Approach Control	τ	time constant
Hp	pressure altitude	ψ	yaw angle, Earth referenced
Hz	Hertz	$\dot{\psi}$	yaw rate, Earth referenced
in	inch(es)	$^{\circ}$	degrees
KIAS	knots, indicated airspeed		
kt	knot(s)		
lb	pound(s)		
MSL	Mean Sea Level		
LCT	Longitudinal Cyclic Trim		
NAWCAD	Naval Air Warfare Center, Aircraft Division		

CHAPTER I

INTRODUCTION

BRIEF HISTORY OF AUTOMATIC FLIGHT CONTROL SYSTEMS

We now know a method of mounting into the air, and, I think, are not likely to know more. The vehicles can serve no use till we can guide them; and they can gratify no curiosity till we mount with them to greater heights than we can reach without; till we rise above the tops of the highest mountains.

Samuel Johnson
(Johnson, 1759)

The value of an aircraft depends on its ability to fly a controllable flight path between its departure point and its destination or target. If the pilot cannot accurately control the aircraft in flight, then the aircraft serves no useful purpose. The degree of aircraft stability will determine how controllable an aircraft is in flight.

Aircraft are usually of two types: stable or unstable. A stable aircraft is one in which returns to equilibrium when disturbed. An aircraft that is stable in the pitch axis will return to its original pitch attitude position once a disturbance causes a nose pitch deviation from its equilibrium condition. An unstable aircraft is one that does not return to equilibrium when disturbed, but continues to increase its deviation from equilibrium. An aircraft that is unstable in the pitch axis will continue to increase nose pitch deviation once a disturbance causes nose pitch from its equilibrium condition.

Stability in aircraft can be either inherent or synthetic. Inherent stability is the tendency of an aircraft that is disturbed from equilibrium to return to equilibrium by its

own accord. Synthetic stability is stability provided by an automatic flight control system by controlling the aircraft to return it to equilibrium. If the automatic flight control system is removed, the synthetic stability is also removed (Etkin and Reid 1996). Automatic flight control systems can provide stability to inherently unstable aircraft allowing the aircraft to fly a controllable flight path between departure point and destination.

A decade before the Wright Brothers first flew their historic flight at Kittyhawk in 1903, Sir Hiram Maxim was developing an automatic flight control for a heavier-than-air vehicle in which he was trying to improve the longitudinal stability. A steam-driven vertical gyroscope operated a valve that ported steam to a servo cylinder. The elevators were then moved by the motion of the servo cylinder's piston. A feedback link from the piston repositioned the gyro-operated valve body so as to close the valve. (McRuer, Ashkenas, and Graham 1968). Though Sir Hiram's automatic flight control system was quite simple and only controlled the pitch axis, it encompassed the major elements of modern automatic flight controls: a sensor (steam driven-gyroscope), an amplifier (steam valve), and the control surface positioning servo (piston).

As the early airplanes progressed, so did automatic flight control systems (AFCS). The first AFCSs were called gyrostabilizers, as their purpose was to maintain an aircraft in level flight. AFCSs were deemed necessary, as the early airplanes were marginally stable requiring continuous pilot vigilance. Between 1910 and 1912, Dr. Elmer Sperry and his son, Lawrence, developed the first true automatic flight control system. The system was a two-axis gyrostabilizer. Pitch and roll angles were measured by roller

contacts on a gyro platform. The roller contacts actuated magnetic clutches that connected the elevator and ailerons to the propeller for propulsion (Figure I- 1); the propeller being driven by the engine. In 1914, Lawrence Sperry won a French safety prize by demonstrating the stability of a Curtiss flying boat incorporating his gyrostabilizer. The flight was conducted at low altitude over the Seine River with a mechanic walking on top of the wing while Sperry stood up in the cockpit with his hands held over his head.

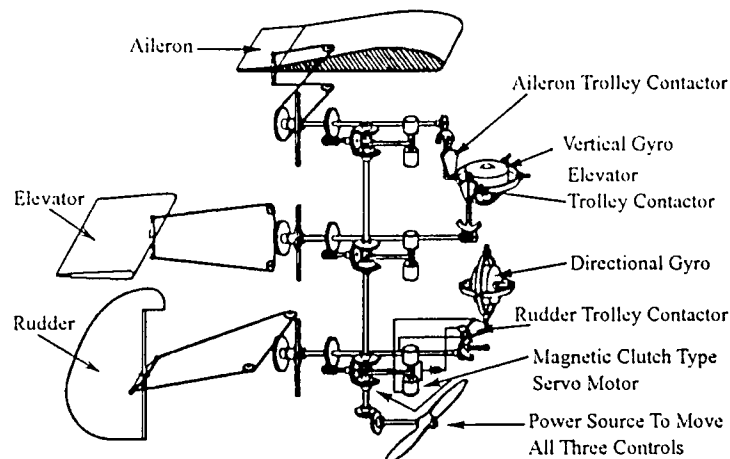


Figure I- 1
Layout of Early Sperry Automatic Pilot

Source: McRuer, Ashkenas, and Graham, 1968.

As airplanes were developed, they became more inherently stable to the point that automatic flight control systems were not needed to make them stable. However, during the late 1920's, the range and endurance of the aircraft became so great that pilot fatigue became a factor; therefore, AFCSS were needed for pilot relief (Roskam 1979). In 1933,

Wiley Post set a solo round-the-world record of 7 days, 18 hours while utilizing a Sperry autopilot that he had installed in his Lockheed Vega 5-C. The autopilot greatly reduced pilot fatigue and even allowed Post to nap in flight. The Sperry autopilot was composed of air-driven gyros with the gyro gimbals operating air valves. An example of such an autopilot is shown in Figure I- 2. Commercial aviation became particularly interested in Sperry's autopilot as the U.S. Commerce Department allowed the airlines to fly passengers during instrument conditions.

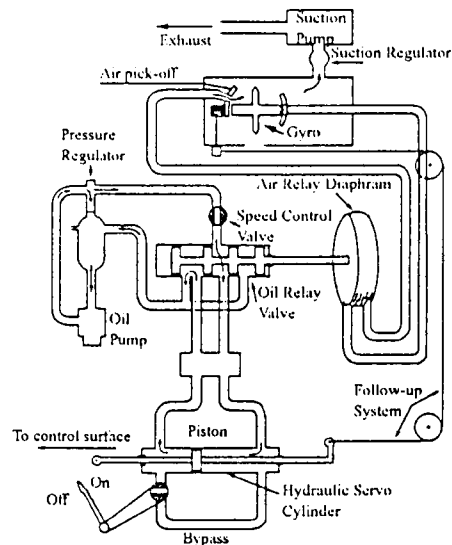


Figure I- 2
Layout of Improved Sperry Autopilot

Source: Roskam, 1979.

World War II brought about major advances in automatic flight control systems. The effective use of feedback signals was refined during this period. The first all-electric autopilot was developed in 1941 that was a 3-axis controller. A free gyro was developed to obtain a heading reference; though frequent resetting was necessary to compensate for

gyro drift. Manual synchronization of gyro pick-offs was replaced by automatic synchronization. Also introduced was altitude control. In the late 1940's, the first automatic-landing approach-equipment was successfully flown. With the exception of updated components, the basic AFCS of today is composed of essentially the same mechanization as those of the 1940's.

GYROSCOPES

A gyroscope is any rotating body that exhibits gyroscopic inertia and precession. Gyroscopic inertia means that a flywheel that is rotating (rotor) tends to continue to rotate in the same plane about the same axis in space. Precession is the tilting of the rotor about an axis that is at right angles to both the spin axis and the axis about which the torque is applied. A conventional gyroscope consists of a rotor (gyro) spinning at high speed and mounted in set of rings (gimbals) that allow one or two degrees of angular freedom. Figure I- 3 is an example of a two-degree of freedom gyro. This type of arrangement allows the airframe to rotate while the rotor remains fixed in space.

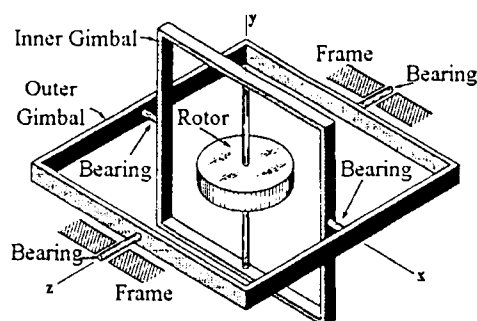


Figure I- 3
Two-Degree of Freedom Gyroscope

Source: Raven, 1961.

A vertical gyroscope is one in which the rotor spin axis is aligned with the local vertical, (Figure I- 4). The vertical gyro is a two-degree of freedom type gyro in that it is free to rotate about two axes. With motion of the gimbals, the rotor spin axis tends to precess due to torques caused by friction in the gimbals. To maintain the rotor spin axis aligned with the local vertical, erection mechanisms of various types are used. The erection mechanism applies a correction signal to torque motors mounted on the gimbal axes that align the spin axis with the local vertical.

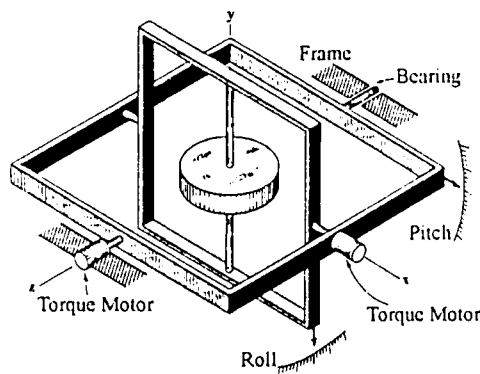


Figure I- 4
Two-Degree of Freedom Vertical Gyro

Source: Raven, 1961.

A directional gyroscope is one in which the rotor spin axis is horizontal and remains aligned with a fixed horizontal direction. A directional gyro is a vertical gyro rotated 90 degrees. The rotor spin axis is maintained in the horizontal plane by one of the torque motors while the other torque motor aligns the rotor spin axis to north.

A rate gyro is one that has only one degree of freedom and its gimbal rotation is restrained by means of springs (Figure I- 5). With this arrangement, the precession

torque causes deflection of the restraining springs. The amount of deflection is proportional to angular rate.

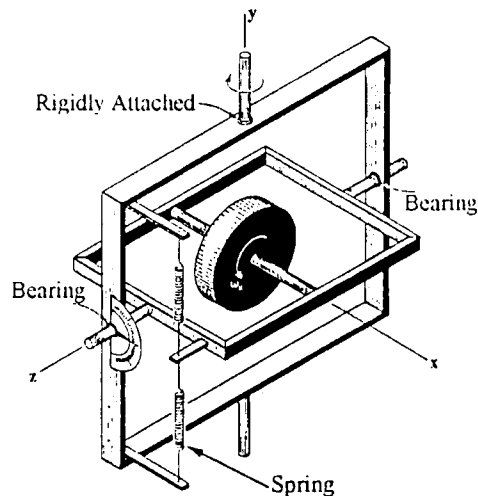


Figure I- 5
Rate Gyro

Source: Raven, 1961.

For the vertical and rate gyros, an electrical pick-off is attached to the gyro to measure the angular rotation of the gimbal. The electrical signal is then forwarded for processing.

SIGNAL PROCESSING

The type of signal processing performed on a gyro signal depends on the intended use of the signal. For example, the gyro signal can be sent to either an attitude indicator or an AFCS computer for processing. We will limit the discussion to pitch attitude and pitch rate signal processing for use in an AFCS. A pitch attitude signal, θ (Earth

referenced), obtained from a displacement gyro must first be differentiated to obtain pitch rate, $\dot{\theta}$ (Earth referenced).

Another method for obtaining pitch rate is by the use of rate gyros. Using a rate gyro negates the necessity of differentiating pitch attitude to obtain pitch rate. The differentiating process introduces errors into the calculated pitch rate.

Most helicopters are subjected to various vibrations; the most common being vibrations due to the rotor system. The rotor system imparts oscillatory vibrations at various frequencies. The frequencies range from a one per revolution (rev) corresponding to a single rotor blade rotation to frequencies corresponding to multiples of the number of total rotor blades in the system (i.e., a three bladed system would have vibrations at 3 per rev, 6 per rev, 9 per rev, etc.). For a helicopter operating at 260 rpm, a 3 per rev vibration occurs at 13 Hz, a 6 per rev vibration occurs at 26 Hz, and a 9 per rev occurs at 39 Hz.

The vibrations can introduce noise (errors) into the pitch attitude signal. The errors are then amplified during the differentiating process, creating large cyclic errors in pitch rate. If the pitch attitude is imparted a cyclic noise in the form of a sine wave as vertical vibrations in a helicopter would, then the change in pitch attitude would be:

$$\Delta\theta = A\sin(\omega t)$$

Where A is the amplitude of the noise, ω is the frequency of the noise in Hz, and t is time.

Differentiating with respect to time yields a change in pitch rate of:

$$\dot{\Delta\theta} = A\omega \cos(\omega t)$$

Therefore, the amplitude of the errors in pitch rate is the amplitude of the pitch attitude error multiplied by the frequency of the noise signal. Consequently, small errors in pitch attitude create large errors in pitch rate.

Once the pitch rate is obtained, either from differentiation or directly from a pitch rate gyro, Euler angles are then used to convert an Earth referenced pitch rate, $\dot{\theta}$, to a vehicle referenced pitch rate, q . During wings-level flight, the vehicle referenced pitch rate is equal to the Earth referenced pitch rate. Once the aircraft rolls to an attitude, ϕ , other than wings-level, the amount of vehicle referenced pitch rate contributing to Earth referenced pitch rate becomes a function of cosine of the roll angle; such that at 90 degrees of roll, no vehicle referenced pitch rate contributes to Earth referenced pitch rate.

During wings-level flight, vehicle referenced yaw rate, r , does not contribute to the Earth referenced pitch rate. Once the aircraft rolls to an attitude other than wings-level, the amount of vehicle referenced yaw rate contributing to the Earth referenced pitch rate becomes a function of sine of the roll angle; such that at 90 degrees of roll, all vehicle referenced yaw rate is contributing to Earth referenced pitch rate. Therefore, derived Earth referenced pitch rate is given by the following equation:

$$\dot{\theta} = q \cos \phi - r \sin \phi$$

Whereas, a positive pitch rate is nose up, a positive yaw rate is to the right, and a positive roll angle is to the right. Rearranging the equation to solve for the vehicle referenced pitch rate yields:

$$q = \frac{r \sin \phi + \dot{\theta}}{\cos \phi}$$

Therefore, the large errors incurred by differentiating pitch attitude to obtain pitch rate introduce large errors into the vehicle referenced pitch rate. These large errors then introduce excessive errors into the aircraft's AFCS.

BACKGROUND ON PITCH CHATTER PHENOMENA

Since the incorporation of the current AFCS into the CH-46E Safety, Reliability, & Maintainability (SR&M), aircraft have experienced high frequency AFCS pitch extensible link actuator (PELA) oscillations, otherwise known as pitch link "chatter". The problem is manifested to the crew as a chattering noise from the pitch extensible link actuators and by increased airframe vibration; however, pitch stability is not compromised. Though not fully known nor tested, the effect of the pitch chatter is believed to be detrimental to the aircraft's dynamic flight control components. The pitch chatter seems not to have any effect on the aircraft's ride quality with the exception of the increased vibrations.

The chatter and vibrations appeared to be caused by gyro interface problems as the AFCS computers in pre-SR&M aircraft incorporated internal rate gyros and were chatter free. The current AFCS computers (-4 computers) obtained the pitch and roll attitude signals from the vertical gyros (Earth referenced) and in conjunction with yaw rate synthesized pitch and roll rates (vehicle referenced).

One of the earlier unsuccessful solutions was to stiffen the attitude gyros mounting-shelves. In 1987, a program was initiated to reduce the pitch chatter by using a variety of newer attitude gyros. The program also examined the results of moving the gyros to different aircraft locations. Neither the various attitude gyros nor various gyro locations were able to reduce the pitch chatter (DON 1987).

Since the pre-SR&M aircraft that had rate gyros incorporated into the AFCS computers were pitch chatter free, the next solution was to redesign the AFCS computers to include rate gyros. This would alleviate the problems of differentiating the attitude signals to obtain pitch and roll rates. In 1992, testing was conducted on -4 AFCS computers modified with internal pitch rate and roll rate gyros, 3 per revolution and 6 per revolution pitch rate notch filters, stiffened AFCS computer shelf, and variable pitch rate lag, τ_3 . Though pitch chatter was not encountered during this testing, the optimized version of the modified -4 computers (rate gyros incorporated) did reduce the high frequency PELA oscillations during both dynamic and steady state flight conditions. The recommended AFCS configuration was to incorporate internal pitch rate and roll rate gyros, 3 and 6 per revolution pitch rate notch filters, a pitch rate lag time of 0.008 seconds, and stiffening of the AFCS computer shelf (DON 1994).

CHAPTER II

DESCRIPTION OF THE CH-46E (SR&M)

The CH-46E (SR&M) Sea Knight helicopter, pictured in Figure II-1, is a dual-piloted, all-weather, tandem-rotor, medium-lift, combat assault helicopter manufactured by the Boeing Helicopter Company. The aircraft's principle dimensions and general components are depicted in Figure A- 1 and Figure A- 2. It is currently in active service with the US Marine Corps. An earlier version, CH-46D, is in active service with the US Navy. The primary mission of the CH-46E is to transport 25 combat loaded Marines, support equipment, and supplies from naval shipping to unprepared landing zones under combat conditions.



Figure II-1
CH-46E (SR&M) Sea Knight Helicopter

The CH-46E is equipped with two main rotors in a tandem configuration and tricycle type landing gear. Two General Electric T58-GE-16 free turbine engines rated at 1,870-shp power the helicopter. The drive train consists of high-speed shafts from the engines being combined in a mix box. Output from the mix box is directed forward and aft. The aft output is directed to the aft rotor head via the aft transmission. The forward output is directed to the forward rotor head via a synchronization shaft and forward transmission.

The rotor system consists of two rotor heads with three fiberglass rotor blades each, rotating at 264 rpm. Each rotor head is fully articulated and incorporates lead-lag and flapping hinges. The rotor heads also incorporate lead-lag dampers. The forward rotor head rotates counterclockwise while the aft rotor head rotates clockwise, when viewed from above. Each rotor system has a diameter of 51-ft. Each blade has a metal spar with a fiberglass shell and a blade chord of 18 3/4 inches. The effective offset for the rotors is 1.8%.

The aircraft is controlled by conventional helicopter controls: cyclic, collective, and directional pedals. The controls are irreversible and hydraulically powered. Artificial feel devices in the control system provide off-trim force cues and control centering. Pitch control utilizing the cyclic stick is provided by differential collective on the forward and aft rotor heads. A forward cyclic input increases collective pitch (i.e., lift) on the aft rotor head while decreasing the collective pitch on the forward rotor head. The change in forward and aft lift results in a nose down pitching moment.

Roll control utilizing the cyclic stick is provided by swash plate tilting (i.e., rotor plane tilting) in the direction of roll on both rotor heads. A right cyclic input tilts both rotor systems to the right causing a right rolling moment. Altitude control utilizing the collective is provided by collectively increasing or decreasing pitch on both rotor systems in the same amount. Directional control utilizing the directional pedals is provided by opposite swash plate tilting of the rotor heads. A right pedal input tilts the forward rotor head to the right and tilts the aft rotor head to the left creating a right yawing moment.

A dual automatic flight control system stabilizes the aircraft about the pitch, roll, and yaw axes and also provides airspeed, heading, bank angle, and altitude hold. A detailed description of the CH-46E automatic flight control system is presented in Chapter III.

With the exception of the installed instrumentation and recording packages, the aircraft was representative of production aircraft that included enlarged stubwings for additional fuel (AFC 379). A detailed description of the CH-46E helicopter is contained in DON 1992.

CHAPTER III

AUTOMATIC FLIGHT CONTROL DESCRIPTION

GENERAL

The following section describes the operation of the CH-46E's Automatic Flight Control System (AFCS) with A02VS100-5 computers installed. The original AFCS utilizes AFCS A02VS100-4 computers, which are designated -4 computers, while the improved AFCS utilizes the AFCS A02VS100-5 computers, which are designated -5 computers. All functions and components of the two AFCSs are exactly identical except where noted. The pitch axis is the focus of this description.

The AFCS consists of two electrically and mechanically independent systems, that operate simultaneously to stabilize the aircraft about the pitch, roll, and yaw axes. The AFCS also automatically maintains selected airspeed, heading, altitude, and bank angle. AFCS control of the aircraft is through movement of electro-hydraulic and electro-mechanical actuators located in series within the flight control system. The actuators receive manual inputs from the pilot operated cockpit controls and electrical inputs from the AFCS computers. Actuator output motion is transferred through mechanical flight controls into swashplate motion that controls the pitch of the rotor blades, thereby controlling the attitude, airspeed, and altitude of the helicopter. Attitude or altitude deviations are sensed and compensated for, resulting in the aircraft's return to the selected trim condition. An AFCS interface block diagram is depicted in Figure A- 3.

FUNCTIONS

The AFCS performs the following functions:

1. Stabilizes the helicopter about the pitch, roll, and yaw axes by providing rate damping of transient changes in attitude and providing corrective control.
2. Maintains a constant pitch attitude and airspeed for airspeeds above 40 kt. It also provides for a positive stick force gradient.
3. Provides turn coordination for airspeeds above 40 kt.
4. Maintains a selected heading, bank angle, and barometric or radar altitude.
5. Reduces rotor shaft stress by reducing fuselage drag and rotor blade flapping.
6. Provides a semi-automatic approach to a hover at a pre-selected radar altitude and holds the hover (for aircraft with a doppler radar installed).

COMPONENTS

The AFCS consists of the following components:

1. Two AFCS computers mounted on the third electronics equipment shelf (Figure A- 4, Sheets 1 and 2).
2. Four ASE actuators, one each for the collective, pitch, roll, and yaw axes, located in the flight control closet (Figure A- 4, Sheets 1 and 3).
3. Three electro-hydraulic dual extensible links, one each for the pitch, roll, and yaw axes located in the flight control closet (Figure A- 4, Sheets 1 and 3) and are powered by the aircraft's two hydraulic flight control boost systems

(Figure A- 5). Each extensible link actuator has a control movement of ± 0.2 inches.

4. Two separate pitot-static systems (Figure A- 4, Sheets 1 and 2). One system is connected to the copilot's instruments and to the No. 1 AFCS computer. The other system is connected to the pilot's instruments and to the No. 2 AFCS computer.
5. Four sideslip ports, two on either side of the aircraft's nose, one on each side connected to the No. 1 AFCS computer and the second one on each side connected to the No. 2 AFCS computer (Figure A- 4, Sheet 1).
6. A dual airspeed hold (DASH) actuator in the longitudinal flight controls located in the flight control closet (Figure A- 4, Sheet 1).
7. Longitudinal, lateral, and directional control position transducers (CPT) attached to the longitudinal, lateral, and directional lower idler bellcranks, respectively (Figure A- 4, Sheets 1 and 3).
8. Forward and aft longitudinal cyclic trim actuators (LCT) in the forward and aft upper flight controls (Figure A- 4, Sheet 1).
9. Forward and aft LCT indicators on the center instrument panel.
10. An AFCS control panel on the center console (Figure A- 4, Sheet 2; Figure A- 6; and Figure A- 7).

11. AFCS with -4 computers utilized MD-1 and AHRS displacement gyros.

MD-1 gyro provides inputs to the No. 1 AFCS computer while the AHRS gyro provides inputs to the No. 2 AFCS computer.

SYSTEM OPERATION

The -5 AFCS computers are pilot controllable through the AFCS panel which provides for mode selection and system engagement. Each system can be engaged by itself or both can be engaged together (normally both are engaged). The analog AFCS computer accepts signals from its internal pitch, roll, and rate yaw gyros and air data sensors as well as from external sensors such as the directional gyro and the three control position transducers. These analog signals are then processed into control commands for the pitch, roll, and yaw extensible link actuators. The extensible link actuators then make direct inputs into each control axis. These actuators provide three-axis damping of the helicopter motions and attitude control of the helicopter in roll and yaw.

The AFCS computers also provide signal processing for control of three additional actuators:

1. Differential Airspeed Hold Actuator (DASH) which provides airspeed hold capability plus a positive stick gradient.
2. Automatic Stabilization Equipment (ASE) actuators which provide radar and barometric altitude hold in collective axis and pitch and roll response to Hover and Approach control system (HAC) signals in pitch and roll axis.

3. Longitudinal Cyclic Trim Actuators (LCTA), one for each forward and aft rotors, which provide pitch trim capability, either automatically as a function of airspeed and altitude, or manually at the discretion of the pilot.

AFCS COMPUTERS

The two interchangeable -5 AFCS computers are mounted on the third shelf of avionics compartment. Each analog computer contains twelve plug-in, printed-wiring, circuit cards that receive inputs from various sensors. The computers then compute and control the required actuator response. The cards are numbered from A1 through A12, Figure A- 8. Pitch axis computations occur on card A-9.

Each computer also contains an airspeed transducer, a sideslip transducer, an altitude transducer, a pitch rate gyro, a roll rate gyro, a yaw rate gyro, and an accelerometer, Figure A- 9. These sensors convert helicopter airspeed, sideslip, barometric altitude, pitch rate, roll rate, yaw rate, and normal acceleration parameters to analog signals.

The focus of the remaining description will concentrate on the pitch axis, as this is where the problems reside. A more detailed description of the roll and yaw axes and other AFCS features can be found in DON 1992 and DON 1995.

PITCH STABILIZATION

Deviations from the aircraft's trimmed pitch attitude are sensed by each AFCS computer's internal pitch rate gyro. The deviation signals are amplified and sent to the

corresponding system's pitch extensible link actuator's torque motor. The torque motor servo valve directs hydraulic fluid to the extensible link, either extending or retracting the link. Each link has a total travel of 0.4 inch. The pitch extensible link extends to raise the aircraft's nose if a pitching down moment is sensed or retracts to lower the aircraft's nose if a pitching up moment is sensed. Link position transducers apply a feedback signal to the summing amplifier when the link is properly positioned to stop the aircraft pitching. As the aircraft returns to the proper attitude, the gyro's signal is nulled, the link centers, and the link torque motor shuts off hydraulic fluid to the link.

Each dual pitch extensible link incorporates two identical extensible link actuators, one for each AFCS computer. Each half of the dual extensible link incorporates a triple link position transducer that extends and retracts with the extensible link. Two of the position transducer sections feedback link positions to its associated AFCS computer. These signals cancel the gyro correction signals as the link reacts to stabilize the aircraft. The third position transducer section is cross-coupled to the AFCS computer of the opposite system. Should a link section react to a hard-over position because of a hydraulic or electronic failure, the cross coupling will cause the opposite link section to react in the opposite direction and reduce the hard-over effect.

A simplified pitch channel interface diagram is presented in Figure A- 10. A pitch axis stabilization block diagram is presented in Figure A- 11. The pitch axis control laws diagram is presented in Figure A- 12.

COMPUTER MODIFICATIONS

As stated earlier, all the functions and components of the AFCS systems are identical. The differences between the systems lie within the AFCS computers themselves. The -4 computers derive pitch and roll rates from attitude signals supplied by the MD-1 and AHRS gyros. The -5 computers incorporate internal pitch rate and roll rate gyros; negating the need to derive pitch and roll rates from pitch and roll attitudes.

The major modifications that are incorporated into the -4 AFCS computers to convert them into -5 AFCS computers are as follows:

1. Addition of one pitch rate gyro and one roll rate gyro and associated circuitry.
2. Deletion of the derived pitch rate and roll rate circuitry.
3. Addition of a 6 per revolution notch filter to the pitch rate signal to attenuate the signal at the aircraft's forcing frequency of 26.4 Hz.
4. Addition of a 3 per revolution notch filter to the roll rate signal to attenuate the signal at the aircraft's forcing frequency of 13.2 Hz.
5. Increased pitch low pass filter time constant, τ_3 , from 0.004 to 0.008 seconds to provide increased high frequency noise filtering of the pitch rate gyro signal.
6. Increased roll low pass filter time constant, τ_{12} , from 0.0034 to 0.125 seconds to provide increased high frequency noise filtering of the roll rate gyro signal.
7. Changes in control laws, built-in test equipment, and internal wiring to account for the installed gyros and notch filters.

8. The AFCS computer-mounting shelf was stiffened to detune the shelf from the resonant 3 per revolution and 6 per revolution frequencies.

AFCS COMPUTER SHELF STIFFENING MODIFICATION

The AFCS computer shelf stiffening modification was designed to detune the AFCS computers from the aircraft natural frequencies of 13.2 Hz (3 per rev) and 26.4 Hz (6 per rev) by increasing the minimum resonant frequency to 40 Hz. The modification consisted of formed angle and shims of 2024 T-3 aluminum alloy mounted to the AFCS computer shelf.

CHAPTER IV

CONDUCT OF TESTING

SCOPE OF TESTS

Ground Tests

An evaluation of the CH-46E (SR&M) frequency response of the pitch and roll extensible link actuators and structural resonance of the AFCS mounting shelf were conducted in 16 hours. The extensible link actuator response testing was conducted by Boeing Helicopter Company and the aircraft structural resonance testing was performed by the Maritime Vertical Lift Reliability and Maintainability Section at the Naval Air Warfare Center, Aircraft Division (NAWCAD), Patuxent River, MD between 14 Sep 95 to 18 Sep 95.

Flight Tests

An evaluation of the CH-46E (SR&M) handling qualities and frequency response of the pitch extensible link actuators was conducted in nine flights for a total of 12.5 flight hours. Of the nine data gathering flights, five were flown with -5 computers installed and four were flown with -4 computers installed. Although the original -4 computer configuration did not include a stiffened AFCS computer shelf, the -4 computer configuration was flown with the stiffened AFCS shelf to maintain the test program schedule and budget. The aircraft testing was performed at the Naval Air Warfare Center,

Aircraft Division (NAWCAD), Patuxent River, MD between 29 Nov 95 to 9 Apr 96. The tests were conducted during daylight visual meteorological conditions.

The test program was conducted within the operator's flight envelope (DON 1992). Testing was conducted with a crew of four, full internal fuel capacity, and doors and windows closed (cabin door upper hatch was open). An external store was attached to the cargo hook during the external load flight tests. A full auxiliary internal fuel tank was utilized for the heavy gross weight flight. The gross weight varied from 18,630 lb to 23,080 lb and the center gravity ranged from 292.8 to 310.4 inches aft of datum (mid C.G.). Test conditions were selected to replicate as closely as possible those conditions flown during the baseline evaluation (DON 1994). Tests and test conditions are presented in Table B- 1. Specific test limitations are presented in Table IV-1.

Table IV-1
Test Limitations

Maneuver	Limitation
Minimum Normal Acceleration	+0.5 g
Maximum Normal Acceleration	+2.5 g
Maximum Pitch Attitude	±45°
Maximum Roll Attitude	±45°
Maximum Yaw Rate	24°/sec
Maximum Landing Rate of Descent	8 ft/sec
Maximum Sideslip at 100 KIAS	1/2 Turn Coordination Ball Deflection
Maximum Airspeed	145 KIAS
Maximum Gross Weight	24,300 lb

Naval Rotary Wing Aircraft Test Squadron provided the aircraft, pilots and engineering support for the testing. Dyncorp provided contract maintenance for the

project aircraft. Test Article Preparation, NAWCAD prepared and maintained the aircraft test instrumentation.

METHOD OF TEST

General

An integrated project team conducted the testing. Flight-testing was conducted by the H-46 Platform team of the Naval Rotary Wing Aircraft Test Squadron. Maritime Vertical Lift Reliability and Maintainability Section, NAWCAD conducted the AFCS shelf rap tests and Boeing Helicopter Company conducted the extensible link actuator response testing.

Ground Tests

General

The ground testing was conducted inside the Naval Rotary Wing Aircraft Test Squadron's maintenance hangar. External hydraulic and electrical power was applied. The engines and rotors were static. Detailed test and test conditions are presented in Table B- 1.

AFCS Shelf Resonant Frequencies

Rap tests were performed on the stiffened AFCS computer-mounting shelf to document the vibratory characteristics and determine its resonant frequencies. The testing was conducted with the -5 AFCS computers installed. The accelerometer and

force application locations are presented in Figure A- 13. Testing was conducted using a PCB electronic type instrumentation hammer, four PCB 308 accelerometers, and a Zonic WCA analyzer. The force was applied in the vertical direction at frequencies from 0 to 400 Hz. All data of interest were digitally stored in the analyzer during the testing and were plotted after test completion. Data were plotted as magnitude and normal acceleration per pound of force for frequencies between 0 and 400 Hz (DON 1995). Resonant frequencies were identified where local maxima in the magnitude corresponded to a 180-degree phase change.

PELA Frequency Response

The PELA frequency response was evaluated by introducing a command frequency from 0.10 to 100.0 Hz to each PELA. The amplitudes of the command frequency were 0.01 and 0.025 inches. The PELA gain and phase changes were then plotted as a function of the command frequency (BHC 1995).

Flight Tests

General

Flight-testing was performed in two AFCS configurations. The first AFCS configuration was with the -5 AFCS computers installed and the stiffened shelf incorporated in the test aircraft, Buno 157692. The results of this configuration were then compared to previous testing conducted with -4 AFCS computers and the optimized

version of the modified -4 AFCS computers (rate gyros incorporated) installed in Buno 153355, as reported in DON 1994.

The second AFCS configuration was with -4 AFCS computers installed in the same aircraft as the -5 AFCS computer testing and utilizing the same instrumentation package. Basically, the -5 AFCS computers were removed and replaced with the -4 AFCS computers without changing the aircraft configuration or instrumentation. The second AFCS configuration was required as the results from the -5 configuration did not match the results of the optimized -4 computers (rate gyros incorporated) in all flight regimes due to differences in aircraft PELA response characteristics, rotor system track and balance, and aircraft structure. The original aircraft from the optimized -4 AFCS computer testing was not employed for the -5 AFCS computer testing due to aircraft availability. In addition, the -4 AFCS computer testing in the current aircraft was performed with the stiffened AFCS shelf installed due to time and fiscal considerations.

Each AFCS configuration included flying qualities testing and mission maneuvers. Flying qualities testing was conducted in accordance with the USNTPS Rotary Wing Stability and Control Manual (DON 1995). Specific mission maneuvers flown were selected as they were maneuvers that had been previously identified as having the highest probability of inducing pitch chatter. Some mission tasks were given specific tolerances and the aircraft's handling qualities were rated according to the Cooper-Harper Handling Qualities Rating Scale (Figure A- 14).

Flying Qualities

The following are descriptions of the flying qualities tests that were conducted. The term "trim condition" indicates that the aircraft was stabilized on airspeed and in wings-level, coordinated flight with the control forces trimmed to zero.

1. Longitudinal Static Stability: The purpose of this test was to evaluate the aircraft's initial tendency to return to or depart from equilibrium once disturbed. Longitudinal static stability is indicated by the variation of the longitudinal control position with airspeed; therefore, increasing forward longitudinal cyclic displacement with increasing airspeed indicates positive static stability. From the trimmed condition with the collective constant, the aircraft was stabilized at airspeeds within ± 15 kt of trim while not retrimming the control forces.

2. Maneuvering Stability: The purpose of this test was to determine the aircraft's tendency to return to one g flight following a disturbance. Maneuvering stability is indicated by the variation of the longitudinal control position with changes in normal acceleration; therefore, increasing aft longitudinal cyclic displacement with increasing normal acceleration indicates positive maneuvering stability. Maneuvering stability was determined using the windup turn method. From the trim condition, the aircraft was stabilized at increasing angles of bank in order to increase normal acceleration while the airspeed and collective were held constant. Control forces were not retrimmed during the windup turn.

3. Dynamic Longitudinal Stability: The purpose of this test was to evaluate the aircraft's long term tendency to return to equilibrium following a disturbance. Dynamic longitudinal stability, a nuisance mode, is indicated by the aircraft's airspeed and altitude variations over time following a disturbance. Positive dynamic longitudinal stability is indicated by the aircraft returning to initial airspeed and altitude after a period of time. From the trim condition with all other controls held fixed, the longitudinal cyclic was used to slow the aircraft 10 kt without retrimming control forces and then the cyclic was returned to the initial trimmed position.

4. Short Term Dynamic Stability: The purpose of this test was to determine the characteristics of the short term response due to longitudinal cyclic inputs and to evaluate the pilot's ability to make small attitude changes. The short term dynamic stability characteristics were determined by applying a one-inch longitudinal doublet at 1/2 Hz. The one-inch doublet is a NAWCAD standard in lieu of frequency sweeps. From the trim condition, a longitudinal cyclic doublet was applied while holding all other controls fixed.

5. Static Lateral-Directional Stability: The purpose of this test was to evaluate the aircraft's directional static stability and lateral static stability (effective dihedral). Directional static stability is the tendency of the aircraft to return to the inherent sideslip once disturbed. Directional stability is indicated by the variation of pedal control with change in sideslip; therefore, increasing left sideslip with increasing right pedal control displacement indicates positive static directional stability. Effective dihedral is the tendency of the aircraft to roll away from increasing sideslip. Effective dihedral is

indicated by the variation of lateral cyclic control with change in sideslip; therefore, increasing right cyclic for increasing right sideslip indicates positive effective dihedral.

The method used to determine the static lateral-directional characteristics was steady heading sideslips. From the trim condition with the collective held constant, pedal control was applied to create sideslip while lateral cyclic was used to maintain ground track and longitudinal cyclic was used to maintain airspeed. Control forces were not retrimmed. Pedal control was increase until a test limit was reached.

6. Dynamic Lateral-Directional Stability: The purpose of this test was to evaluate the aircraft's tendency to return to equilibrium following a lateral-directional disturbance. Dynamic lateral-directional stability also known as lateral-directional oscillation (LDO) is a nuisance mode and is indicated by the aircraft's angle of bank and sideslip variations over time following a disturbance. The aircraft returning to initial angle of bank and sideslip after a period of time indicates positive dynamic lateral-directional stability. The dynamic lateral-directional stability was evaluated using an artificial excitation method. From the trim condition, the aircraft was placed in a steady heading sideslip and then the controls were returned to the initial trimmed positions. All controls where then held fixed as the aircraft responded to the lateral-directional disturbance.

7. Control Response: The purpose of this test was to evaluate the aircraft's response to longitudinal and lateral cyclic control input due to the rotor system moments generated from the cyclic inputs. The control response characteristics were determined during forward flight and during out-of-ground effect hover. From the trim condition, a

longitudinal or lateral cyclic step input was crisply made against a control fixture while holding all other controls fixed. Step inputs were increased until a test limit was reached. The process was the same for out-of-ground effect hover where the aircraft was stabilized over a hover point.

Mission Maneuvers

The following are the mission maneuvers that were conducted and the method of each maneuver. The term "hover" denotes the aircraft stabilized over a hover point, at desired altitude, and the aircraft's nose oriented into the prevailing wind with the control forces trimmed to zero. The term "trim condition" indicates that the aircraft was stabilized on airspeed and in wings-level, coordinated flight with the control forces trimmed to zero.

1. In-Ground Effect Hover (HIGE): From an in-ground effect hover, the aircraft was maintained over the hover point and at the desired altitude. In addition, while maintaining the hover point, pedal turns were conducted up to 360° of heading change at yaw rates up to 24 deg/sec.

2. Out-of-Ground Effect Hover (HOGE): From an out-of-ground effect hover, the aircraft was maintained over the hover point and at the desired altitude.

3. Maximum Power Climb with Turns: The aircraft was stabilized on desired airspeed with maximum continuous power applied (100% Q). The angle of bank was then varied during the climb.

4. Minimum Power Descent with Turns: The aircraft was stabilized on desired airspeed with the power turbine just powering the rotor system as indicated by the gas generator speed beginning to increase when collective was applied. The angle of bank was then varied during the descent.

5. Constant Altitude Turns: From the trim condition, the aircraft was rolled into a desired angle of bank while applying power to maintain altitude. Power was applied until maximum continuous power was reached, at which point airspeed was bled off in order to maintain the angle of bank.

6. Angle of Bank Capture: From the trim condition, the aircraft was rolled into a 45 degree left or right angle of bank at approximately 30 deg/sec roll rate while maintaining fixed power. The objective was to capture the angle of bank within set tolerances.

7. Pitch Attitude Capture: From the trim condition, the aircraft was pitched up or down 5 degrees from the trim attitude in approximately 1 second while maintaining fixed power. The objective was to capture pitch attitude within set tolerances.

8. Maximum Level Flight Airspeed: Maximum continuous power (100% Q) was applied and the cyclic was manipulated to maintain altitude and wings-level as the airspeed stabilized.

9. Roll Reversal: From the trim condition with power and airspeed held constant, a single cycle roll reversal was conducted to 45 degrees angle of bank left or right at approximately 30 deg/sec. At each 45 degree angle of bank, the angle of bank was held approximately for 2 seconds prior to reversal continuation.

10. Steep Approach: The maneuver was initiated with the aircraft wings-level at 300-ft AGL at 50 KIAS. The aircraft altitude was kept constant until the steep glide-slope was intercepted (landing point viewed just above the glare shield). Collective was adjusted as necessary to maintain glide-slope and the airspeed was reduced as necessary to touchdown at the landing point with zero groundspeed.

11. Quickstop: The aircraft was stabilized wings-level on airspeed at 50-ft AGL. The collective was lowered to approximately one inch from full down in approximately 5 seconds. The longitudinal cyclic was manipulated to maintain altitude as the airspeed decreased, up to approximately 15° nose up attitude. The aircraft was decelerated until a normal glide-slope was intercepted.

12. Vertical Takeoff: From a hover with an external load attached to the cargo hook, the collective was applied to achieve a rate of climb of approximately 2-ft/sec. Heading and hover position were maintained until the external load was lifted off the ground.

13. Vertical Landing From In-Ground-Effect Hover: From an in-ground effect hover, the collective was reduced while maintaining ground position and heading. Rate of descent was up to 6-ft/sec. For landings with an external load, a 1-ft/sec rate of descent was targeted.

14. Air Taxi: From a hover, the longitudinal cyclic was applied to establish the aircraft in forward, left, right, and rearward air-taxies while maintaining constant groundspeed, altitude and heading.

15. Lateral Sidestep: From an in-ground effect hover, lateral cyclic was applied to achieve an angle of bank of 15° in approximately 1/2 second. The angle of bank was maintained until the new hover point was reached; at which time the aircraft was recovered to a hover. Heading and altitude were kept constant throughout the maneuver.

16. Power Recovery Autorotation: From the trim condition, the collective was lowered to establish rotor speed greater than power turbine speed. Collective was then increased to maintain rotor speed at 100%. Engine power control levers remained in fly. Airspeed was maintained during the descent until the flare altitude was reached (175-ft AGL for 80 KIAS autorotations, 225-ft AGL for 110 KIAS autorotations). Pitch attitude was increased as necessary to reduce the rate of descent and decrease airspeed. At 75-ft AGL, the aircraft was leveled and then collective increased to recover the aircraft in a 25-ft AGL hover.

17. Shipboard Approach: From the trim condition of 300-ft AGL and 60 KIAS, the aircraft was decelerated while maintaining the final approach course with an approximate 3 to 5 degree glide-slope to the simulated ship landing spot. The aircraft's final approach course was 45 degrees to the right of the simulated ship's direction of travel. At approximately 20 kt and 100-ft from the point of landing, the aircraft was yawed to the left at approximately 20 deg/sec to align the aircraft's nose with the simulated ship's direction of travel. As the aircraft was being yawed, the aircraft's final approach course was maintained. The aircraft was then brought to an in-ground effect hover over the helicopter pad aligned with the simulated ship's direction of travel.

INSTRUMENTATION AND DATA PROCESSING

Control positions were recorded by the use of control position transducers. Cyclic longitudinal and lateral positions were referenced from the center of the pilot's cyclic grip. Collective position was referenced from the center of the pilot's collective grip. Directional pedal position was referenced from the left center edge of the pilot's right directional pedal. Total control travel was measured from each control's reference and used to calibrate the control position transducers. Longitudinal cyclic total control travel was 12.0 inches. Lateral cyclic total control travel was 9.2 inches. Directional-pedal total control travel was 6.5 inches.

An onboard instrumentation package was installed in the test aircraft which measured basic aircraft parameters, flying quality parameters, pitch extensible link actuator positions, and DASH actuator positions. Sideslip was not calibrated due to fiscal and time considerations, as an instrumentation boom would be required. A list of instrumented parameters is presented in Table B- 2. All data was recorded in Pulse Coded Modulation (PCM) format on magnetic tape onboard the test aircraft and manually on kneeboard cards. The test team using Real-time Telemetry Processing System (RTPS) conducted post-flight data processing. Data files for each flight were archived on optical disk and 8mm tape. Files on 8mm tape were in Vax Binary format for use by BHC personnel. The data files were written to the Sun Workstation at RTPS for data reduction.

The data was accessed via the local network from desktop computers with workstation emulation software. Project data management and data reduction were accomplished using Advanced Scientific Analysis Package (ASAP) software resident on

a Sun Workstation. The ASAP software allowed calculation of event statistics for flying qualities tests, dynamic time-history data-presentations, and spectral analysis of PELA positions. Flying qualities data were plotted using Microsoft Excel 5.0 graphing software.

TEST STANDARDS

The -5 AFCS computers were evaluated against the following criteria:

- a. For similar test conditions and maneuvers, the -5 AFCS computers were to reduce the PELA response as compared to PELA response with the -4 AFCS computers.
- b. For similar test conditions, the flying qualities of the aircraft configured with the -5 AFCS computers were to be similar to the flying qualities of the aircraft configured with the -4 AFCS computers and the optimized -4 AFCS computers.

CHAPTER V

DISCUSSION

GROUND TESTS

Stiffened AFCS Shelf Minimum Resonant Frequencies

The stiffened AFCS shelf rap tests were conducted with -5 AFCS computers installed using the same force application and accelerometer locations as those used during the baseline testing (DON 1994). The testing was performed with the rotors static. Minimum resonant frequencies are presented in Table V- 1. The stiffened shelf increased the minimum resonant frequency from approximately 25 Hz to a minimum of 39 Hz; sufficient to detune the AFCS computers from the aircraft's natural frequencies of 13.2 Hz and 26.4 Hz, 3 per rev and 6 per rev, respectively.

Table V- 1
AFCS Shelf Minimum Resonant Frequencies

Accelerometer/ Force Location (1)	Minimum Resonant Frequency (Hz)			
	Stiffened Shelf		Non-Stiffened Shelf	
	-5 AFCS Computers (2)	Modified -4 AFCS Computers (3)(4)	Modified -4 AFCS Computers (3)(4)	-4 AFCS Computers (3)(4)
A1 / F1	39.1	36.0	25.5	25.0
A2 / F1	39.1	38.5	25.3	25.2
A3 / F1	39.0	37.5	24.8	25.0
A4 / F1	42.0	37.5	25.0	25.0
A1 / F2	39.9	-	-	-
A2 / F2	39.9	-	-	-
A3 / F2	39.0	35.5	25.0	25.0
A4 / F2	42.9	-	-	-
A1 / F3	39.5	-	-	-
A2 / F3	39.8	-	-	-
A3 / F3	41.9	-	-	-
A4 / F3	39.0	-	-	-

(1) Refer to Figure A- 13

(2) Buno 157692

(3) Buno 153355

(4) Results from DON 1994

Extensible Link Actuator Response

The frequency response characteristics of both the pitch and roll extensible link actuators (ELA) on the test aircraft were measured during ground testing with the rotors static (BHC 1995). ELA position responses were evaluated with 0.01 and 0.025-inch discrete command amplitudes between the frequency ranges of 0.10 to 100.0 Hz.

During the preliminary testing (DON 1994), the test aircraft, Buno 153355, exhibited differing pitch and roll extensible link actuators responses. The ELAs should have had the same response characteristics, as they are identical but installed in different control channels. The ELA responses, presented in Figure C- 1, reveal that the pitch ELA had a smaller bandwidth than that for the roll ELA. The pitch and roll ELAs of the current test aircraft, Buno 157692, exhibited approximately the same bandwidth, as presented in Figure C- 2, which was expected for identically manufactured ELAs.

Comparing the pitch ELA response characteristics of the two test aircraft, as presented in Figure C- 3 through Figure C- 8, the preliminary testing aircraft, Buno 153355, exhibited differing frequency response characteristics, including a lower bandwidth, than that of the current test aircraft, Buno 157692, over the entire tested range. Of particular note are the differences in the responses at the aircraft's natural frequencies of 13.2 and 26.4 Hz. Throughout the testing, these differing response characteristics were evident when comparing the preliminary testing data to the current testing data. The lower bandwidth of the preliminary testing aircraft may have resulted in less frequency content than that of the current test aircraft when comparing frequency response characteristics for the same test conditions. Therefore, the optimized -4

computer (rate gyros incorporated) data obtained from Buno 153355 was not suitable for comparison with the data obtained from Buno 157692.

Comparing the roll ELA response characteristics of the two test aircraft, as presented in Figure C- 9, both aircraft exhibited the same frequency response characteristics and approximately the same bandwidth. This resulted in approximately the same roll ELA responses for the test aircraft as for the preliminary testing aircraft.

FLIGHT TESTS

General

Qualitative and quantitative data obtained with the aircraft configured with -5 AFCS computers was compared to data obtained with the aircraft configured with -4 AFCS computers and data from the optimized -4 AFCS computer testing (DON 1994). Pitch extensible link actuators (PELA) response levels were compared utilizing aircraft maneuvers that had been determined by previous testing (DON 1994) to cause PELA chatter. These maneuvers consisted of maximum level flight airspeed (V_h), out-of-ground effect hover (HOGE), and quickstops. Due to the differences in PELA response characteristics between the current test aircraft and the preliminary testing aircraft, PELA response comparisons were limited to the current aircraft configured with the -5 and -4 AFCS computers. The mission maneuvers and flying qualities testing were conducted during the conditions as presented in Table B- 1.

The term "hover" denotes the aircraft stabilized over a hover point, at desired altitude, the aircraft's nose oriented into the prevailing wind, and the control forces trimmed to zero.

Neither PELA chatter nor increased airframe vibrations were experienced during the scope of the testing.

PELA Comparison Flights

Maximum Level Flight Airspeed

The aircraft's PELA responses and handling qualities were evaluated during maximum level flight airspeed, V_h , at 135 KIAS and 3,000-ft Hp in both the -5 and -4 AFCS computer configurations.

The representative time histories of the aircraft at V_h (Figure C- 10 and Figure C-11) reveal the PELAs of the -4 computer configuration as having greater overall actuator amplitude deviations than PELAs of the -5 computer configuration. Throughout much of the frequency range, the frequency responses of the -5 computer configuration PELAs (Figure C- 12) were of lower amplitude. Though at 13.2 Hz (3 per rev), the -5 computer configuration PELA #1 and PELA #2 responses were two and four times greater than the -4 computer configuration, respectively.

When comparing the same -5 computer configuration at different aircraft gross weights (Figure C- 13), the PELA response appeared to be affected by the aircraft's gross weight. Over most of the frequency range, gross weight did not appear to have much of an effect, as the amplitudes were approximately the same. However at 13.2 Hz, the

higher gross weight configuration PELA #1's amplitude was approximately 30% greater than the lighter gross weight configuration. It looks as if the heavier gross weight configuration creates greater amplitudes at the lower frequencies (less than 20 Hz).

The pilots were not able to discern any differences in the handling qualities between the -4 and -5 AFCS computer configurations when operating at maximum level flight airspeed.

The -5 AFCS computer configuration sufficiently reduced the PELA frequency content from that of the -4 AFCS computer configuration and did not affect the aircraft's handling qualities while operating at maximum level flight airspeed.

Out-of-Ground Effect Hover

The aircraft's PELA responses and handling qualities were evaluated during out-of-ground effect hover (HOGE) at 100-ft and 500-ft AGL in the -4 and -5 AFCS computer configurations.

The time histories of the two computer configurations at the 100-ft HOGE (Figure C- 14 and Figure C- 15) show that the -4 computer configuration PELAs as being more active and with larger PELA amplitudes than the -5 computer configuration. Overall, the -5 computer configuration PELA frequency responses (Figure C- 16) were approximately half the amplitude of those of the -4 computer configuration over the majority of the frequency ranges. At the natural frequency of 26.4 Hz (6 per rev), both PELA responses for the -5 configuration were miniscule as compared to the -4 configuration PELA responses. Although there was no improvement in the PELA #1 response at 13.2 Hz,

PELA #2's response improvement was dramatic, approximately a 92% reduction from the -4 computer configuration's amplitude.

Comparing the PELA frequency responses at a 500-ft HOGE (Figure C- 17) shows the -5 computer configuration with less of a frequency response reduction over much of the frequency range. However, at the natural frequencies of 13.2 and 26.4 Hz, there was a significant amplitude reduction for the -5 computer configuration. At 13.2 Hz, the -5 computer configuration's #1 and #2 PELA responses were 70% and 25% that of the -4 computer configuration responses, respectively. At 26.4 Hz, the -5 computer configuration's #1 and #2 PELA responses were 10% and 35% that of the -4 computer configuration responses, respectively.

When comparing the same -5 AFCS computer configuration at different aircraft gross weights (Figure C- 18), it appears that the PELA response may be affected by the aircraft's gross weight. Although over much of the frequency range the PELA response was not significantly affected; at the aircraft's natural frequencies of 13.2 and 26.4 Hz, the frequency responses at the heavier gross weight were approximately 35% to 60% that of the lighter gross weight. Overall, it appears that an increase in gross weight reduces the PELA response.

While conducting HOGE operations at 100 and 500-ft, the pilots were not able to discern any differences in the handling qualities between the -4 and -5 computer configurations.

The -5 AFCS computer configuration substantially reduced the PELA frequency response from that of the -4 AFCS computer configuration and did not affect the aircraft's handling qualities while in a HOGE.

In-Ground Effect Hover

The aircraft's PELA responses and handling qualities were evaluated during in-ground effect hover (HIGE) at 30-ft AGL in the -4 and -5 AFCS computer configurations.

The time histories of the two configurations for the HIGE (Figure C- 19 and Figure C- 20) show that the -4 computer configuration PELAs as being more active and with larger PELA amplitudes than the -5 computer configuration. Both PELAs' frequency responses with the -5 computer configuration (Figure C- 21) exhibited approximately a 50% amplitude reduction over a majority of the frequency range as compared to the -4 computer configuration's responses. At the natural frequency of 13.2 Hz, the PELAs' amplitudes with the -5 computers were 35% to 75% less than with the -4 computers installed. At 26.4 Hz, there was a significant amplitude reduction for both PELAs with the -5 computer configuration as the amplitudes were miniscule when compared to the amplitudes with -4 computers.

During HIGE operations, the pilots were not able to discern any differences in the handling qualities between the -4 and -5 computer configurations.

The -5 AFCS computer configuration substantially reduced the PELA frequency response from that of the -4 AFCS computer configuration and did not affect the aircraft's handling qualities while in a HIGE.

Quickstops

The aircraft's PELA responses and handling qualities were evaluated during quickstop maneuvers in the -5 and -4 AFCS computer configurations. The maneuver was commenced between 90 and 110 KIAS and 50-ft AGL.

The time histories of the two AFCS computer configurations (Figure C- 22 and Figure C- 23) reveal that the PELAs with the -4 computers as extremely active and having greater amplitude oscillations as compared with the -5 computer configuration. The -5 computer configuration significantly reduced PELA frequency response amplitudes to basically zero for frequencies above 13.2 Hz, as shown in Figure C- 24. For frequencies below 13.2 Hz, the PELA amplitudes were reduced by approximately 50%. Although there was no improvement in the PELA #1 response at 13.2 Hz, PELA #2's response improvement was remarkable, 20% of the -4 computer configuration's response.

The pilots were not able to discern any differences in the handling qualities between the -4 computer configuration and the -5 computer configuration when conducting quickstops.

The -5 AFCS computer configuration significantly reduced the PELA amplitudes from that of the -4 AFCS computer configuration and did not affect the aircraft's handling qualities while conducting quickstops.

Steep Approaches

The aircraft's PELA responses and handling qualities were evaluated during steep approaches in the -5 and -4 AFCS computer configurations. The maneuver was commenced at 60 KIAS and 300-ft AGL.

The representative time histories of the aircraft during a steep approach (Figure C- 25 and Figure C- 26) show the PELAs of the -4 computer configuration exhibiting about the same overall actuator amplitude deviations as the -5 computer configuration. However, the -4 computer configuration PELAs displayed an increase in activity as compared to the PELAs in the -5 computer configuration. Over a majority of the frequency range, the -5 computer configuration PELAs' frequency responses (Figure C- 27) were of much lower amplitude than the -4 computer configuration PELAs. Though at 13.2 Hz, the -5 computer configuration's PELA #1 had the same amplitude as the -4 computer configuration. PELA #2's amplitude at 13.2 Hz in the -5 computer configuration was approximately 35% that of the -4 computer configuration's amplitude. At 26.4 Hz, both -5 computer configuration PELAs' amplitudes were 20% to 15% that of the -4 computer configuration's amplitudes.

During the steep approaches, the pilots were not able to discern any differences in the handling qualities between the -4 and -5 computer configurations.

The -5 AFCS computer configuration satisfactorily reduced the PELA frequency content from that of the -4 AFCS computer configuration and did not affect the aircraft's handling qualities while conducting steep approaches.

Flying Qualities

Static Longitudinal Stability

The forward flight static longitudinal stability characteristics were evaluated at 100 KIAS trim airspeed and 3,000-ft Hp. Comparison data for aircraft configured with -5, -4, and optimized -4 AFCS computers (-4 with rate gyro) are presented in Figure C-28. The aircraft demonstrated positive static stability in that increasing forward cyclic from trim was required for increasing airspeeds and increasing aft cyclic from trim was required for decreasing airspeeds. All three computer configurations exhibited approximately the same cyclic position change to airspeed change gradient of 0.1%/kt. The pilot was unable to discern any differences in the static longitudinal stability characteristics between the -5 and -4 computer configurations.

For both the -5 and -4 computer configurations, capturing airspeed required slowly moving the cyclic in the desired direction until the desired airspeed was achieved. To maintain the airspeed required the pilot to easily hold pressure against the control force. Maintaining level flight trim airspeed within ± 2 KIAS during level flight was easy, requiring no pilot compensation (HQR 2). The optimized -4 computer configuration exhibited the same easy airspeed maintenance (DON 1994).

Within the scope of these tests, the forward flight static longitudinal stability characteristics and handling qualities of the aircraft configured with -5 AFCS computers exhibited the same characteristics as those of the aircraft configured with the -4 and optimized -4 AFCS computers.

Maneuver Stability

The maneuver stability was evaluated at 100 KIAS and 3,000-ft Hp utilizing constant level-flight power and increasing bank angle up to 45 degrees left and right at constant altitude; commonly referred to as windup turns. Data are presented in Figure C- 29. Although normal acceleration data were not available due to a failure in the instrumentation, for constant altitude turns an increase in angle of bank corresponds to an increase in normal acceleration. Both the -5 and -4 computer configurations exhibited the same characteristic of requiring slightly increasing forward cyclic for increasing angles of bank, therefore with increasing normal acceleration.

The amount of cyclic that was required with increasing angle of bank was symmetrical for both left and right angles of bank. The cyclic that was required for increasing normal acceleration was in the opposite direction that associated with stable maneuvering characteristics. However, the control travel of the cyclic from level flight to 45 degrees angle of bank was only 3% of the control travel (1/3 inch), and was imperceptible to the pilot. The cyclic migrating forward with increasing normal acceleration was caused by the washout function in the longitudinal channel control laws. The purpose of the washout is to maintain pitch rate damping when the aircraft is in a

constant angle of bank turn. Pitch rate damping is maintained by the PELAs re-centering once they are saturated during a constant turn, which causes the cyclic to migrate forward.

Within the scope of these tests, the aircraft configured with -5 AFCS computers exhibited the same maneuver stability characteristics as those of the aircraft configured with -4 AFCS computers.

Longitudinal Short Term Response

The forward flight longitudinal short term response characteristics were evaluated at a 100 KIAS trim airspeed and 3,000-ft Hp. The longitudinal short term response was excited by one-half inch longitudinal cyclic control doublets at an approximate frequency of one hertz. The longitudinal short term response was deadbeat as evident by no observable pitch attitude overshoots once the input was completed and the longitudinal cyclic returned to the trimmed position.

Pitch attitude captures were conducted by using longitudinal cyclic to pitch the aircraft to the desired attitude and then moving the cyclic once back to original trim position. Capturing a 5° nose-up pitch attitude change within $\pm 1^\circ$ at a pitch rate of approximately 10 deg/sec was very easy requiring only one longitudinal cyclic input back to trim (HQR 2). The aircraft's response was the same for both -5 and -4 computer configurations. The optimized -4 configuration also exhibited the same pitch attitude capture handling characteristics (DON 1994). The pilot was unable to discern any

differences in the longitudinal short term response characteristics between the -5 and -4 computer configurations.

Within the scope of these tests, the longitudinal short term response characteristics and the pitch attitude capture characteristics of the aircraft configured with -5 AFCS computers exhibited the same characteristics as those of aircraft configured with the -4 and the optimized -4 AFCS computers.

Longitudinal Long Term Response

The longitudinal long term response characteristics of the aircraft were qualitatively evaluated at 100 KIAS and 3,000-ft Hp. Natural disturbances (flight in light turbulence) did not easily excite the long term response. The long term response was artificially excited by reducing the trim airspeed by 10 KIAS using only longitudinal cyclic but without retrimming, then returning the cyclic to the 100 KIAS trim position. The wings were held level throughout the maneuver and balanced flight was maintained by the AFCS.

The pitch attitude response was heavily damped, aperiodic convergent as indicated by one overshoot of approximately one degree before returning to the trim attitude. The pitch attitude returned to the trim attitude in approximately 65 seconds. The airspeed returned to the trim airspeed in approximately 80 seconds. The aircraft's response was the same for both -5 and -4 computer configurations. The pilot was unable to discern any differences in the longitudinal long term response characteristics between the -5 and -4 computer configurations.

Within the scope of these tests, the longitudinal long term response characteristics of the aircraft configured with -5 AFCS computers exhibited the same characteristics as those of aircraft configured with the -4 AFCS computers.

Longitudinal Control Response

The aircraft's longitudinal control response characteristics were evaluated during forward flight at 100 KIAS and 3,000-ft Hp. Longitudinal cyclic step inputs from trim into a control fixture were used to initiate the aircraft's response. Data are presented in Figure C- 30.

Inputs of less than approximately one inch produced characteristics of an attitude command system. Inputs greater than approximately one inch produced characteristics of a rate command system. The response characteristics of the -5 computer configuration exhibited the same characteristics of the -4 computer configuration. The responses were generally linear and very predictable with larger cyclic inputs creating larger pitch rates. No time delay between input and aircraft response was noticeable to the pilot. The pilot was unable to discern any differences in the longitudinal control response characteristics between the -5 and the -4 computer configurations.

Within the scope of these tests, the forward flight longitudinal control response characteristics of the aircraft configured with -5 AFCS computers exhibited the same characteristics as those of aircraft configured with the -4 AFCS computers and the optimized -4 AFCS computers.

Lateral Control Response

The aircraft's lateral control response characteristics were evaluated during forward flight at 100 KIAS and 3,000-ft Hp and during a 500-ft AGL HOGE. Although no major changes were made in the AFCS's lateral axis, one project pilot thought that the lateral control in the -5 computer configuration was more sensitive than in the -4 computer configuration. Several flights were dedicated to investigate the lateral axis. Lateral cyclic step inputs from trim into a control fixture were used to initiate the aircraft's response. Data are presented in Figure C- 31 and Figure C- 32.

The response characteristics of the -5 computer configuration exhibited the same characteristics of the -4 computer configuration for both forward flight and in a hover. The responses were generally linear and very predictable with larger lateral cyclic inputs creating larger roll rates and roll acceleration. The pilots perceived no time delay between input and aircraft response. The pilots were unable to discern any differences in the lateral control response characteristics between the -5 and the -4 computer configurations while conducting the lateral control response testing.

Within the scope of these tests, the forward flight and hovering lateral control response characteristics of the aircraft configured with -5 AFCS computers exhibited the same characteristics as those of aircraft configured with the -4 AFCS computers.

External Load Flight

The handling qualities of the aircraft were evaluated while conducting mission maneuvers with an external load attached to the cargo hook with a 30-ft pendant. The

external load consisted of a 2,000-lb iron beam. The flight regime ranged from 0 to 80 KIAS and from 0 to 1,000-ft AGL. Flight tasks included vertical takeoff, hovering in-ground effect, hovering pedal turns at 5 deg/sec, 10 KGAS air taxi, maximum power climb, constant altitude turns up to 30 degrees angle of bank, simulated shipboard approach, and vertical landing.

The pilots' qualitative opinions were that the aircraft's handling qualities appeared no different than an aircraft with an external load configured with the -4 computers, though the -4 computer configuration was not tested with an external load. The pilots relied on past external load flights to formulate their opinions.

For the external load flight regimes tested, the -5 AFCS computer configured aircraft appeared to exhibit similar handling qualities as aircraft configured with the -4 AFCS computers.

Mission Maneuvers

The handling qualities of the aircraft were evaluated while conducting typical mission maneuvers. The maneuvers not previously discussed consisted of low speed, low altitude flight; maximum power climbs and descents with turns; constant altitude turns; angle of bank capture; roll reversals; power recovery autorotations; vertical landings; and lateral sidesteps.

For the mission maneuvers tested, the project pilots were not able to discern any differences in the aircraft's handling characteristics between the -5 and -4 computer

configurations. For the mission maneuvers tested, the aircraft's handling qualities did not change when flown in the -5 AFCS computer configuration.

CHAPTER VI

CONCLUSIONS

During flight conditions that had been determined to cause pitch chatter and during other mission representative flight maneuvers, the -5 AFCS computers reduced pitch extensible link actuator (PELA) oscillations and attenuated high frequency amplitudes from the levels exhibited by the test aircraft configured with -4 AFCS computers. The -5 AFCS computers have been installed in over half of the three hundred fleet CH-46Es. To date, no incidents of PELA chatter have been reported from the approximately 160 fleet aircraft configured with the -5 AFCS computers; therefore, the -5 AFCS computers appear to eliminate pitch chatter.

Within the scope of the tests flown, the flying qualities of the CH-46E with the -5 AFCS computers installed were the same as those exhibited with -4 AFCS computers and optimized -4 AFCS computers installed. The -5 AFCS computers did not improve nor deteriorate the aircraft's flying qualities.

Though not proved conclusively, the aircraft's weight appeared to have an influence on PELA response, a heavier aircraft causing less PELA frequency content.

CHAPTER VII

RECOMMENDATIONS

It is recommended that the -5 AFCS computers replace the -4 AFCS computers. Additionally, recommend that further testing be conducted with -5 AFCS computers:

- 1) at differing aircraft gross weights to determine the extent that aircraft weight influences PELA frequency responses for all mission maneuvers, and
- 2) on an aircraft with a known history of pitch chatter while in the -4 AFCS computer configuration to confirm pitch chatter elimination.

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APPENDICES

APPENDIX A

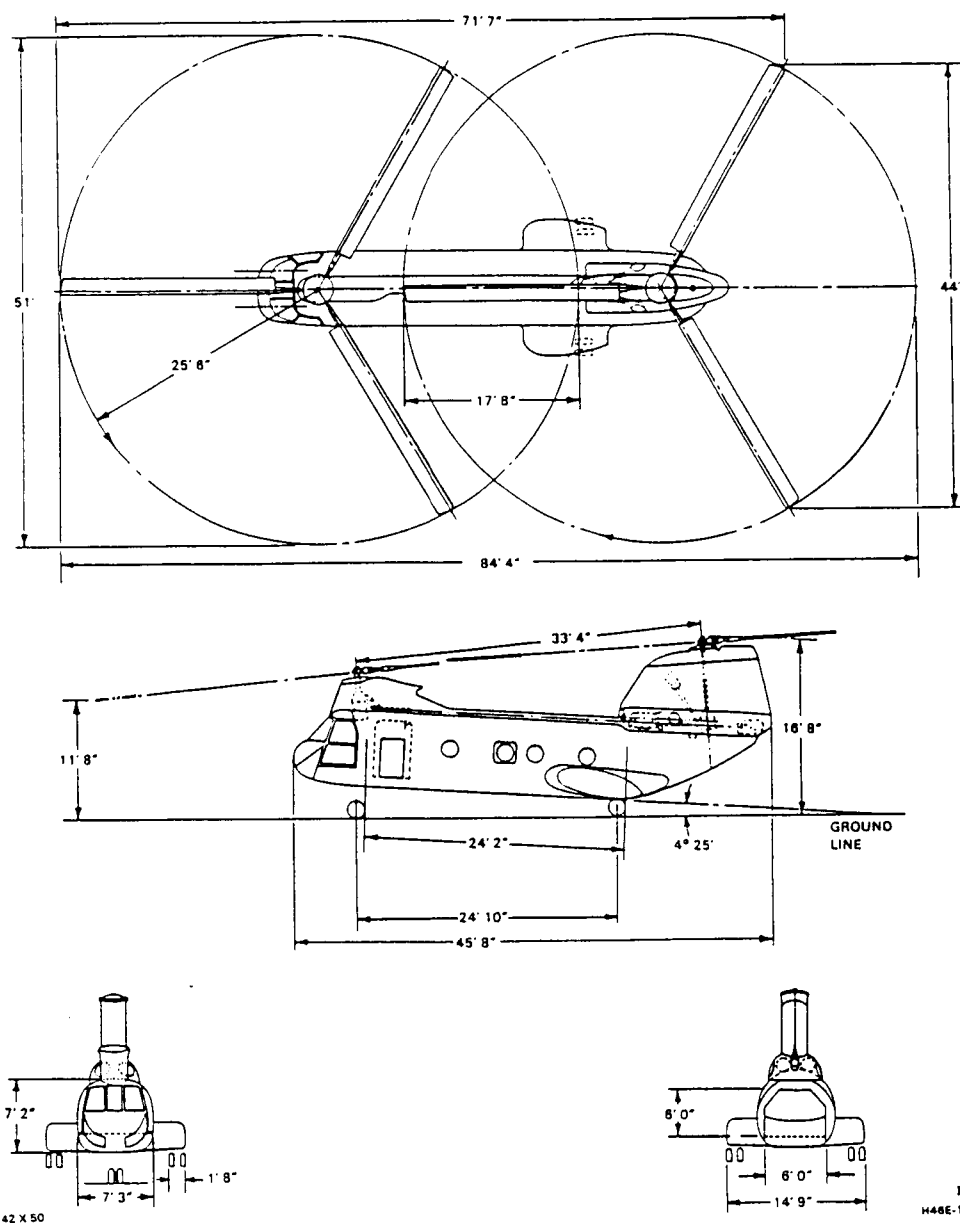
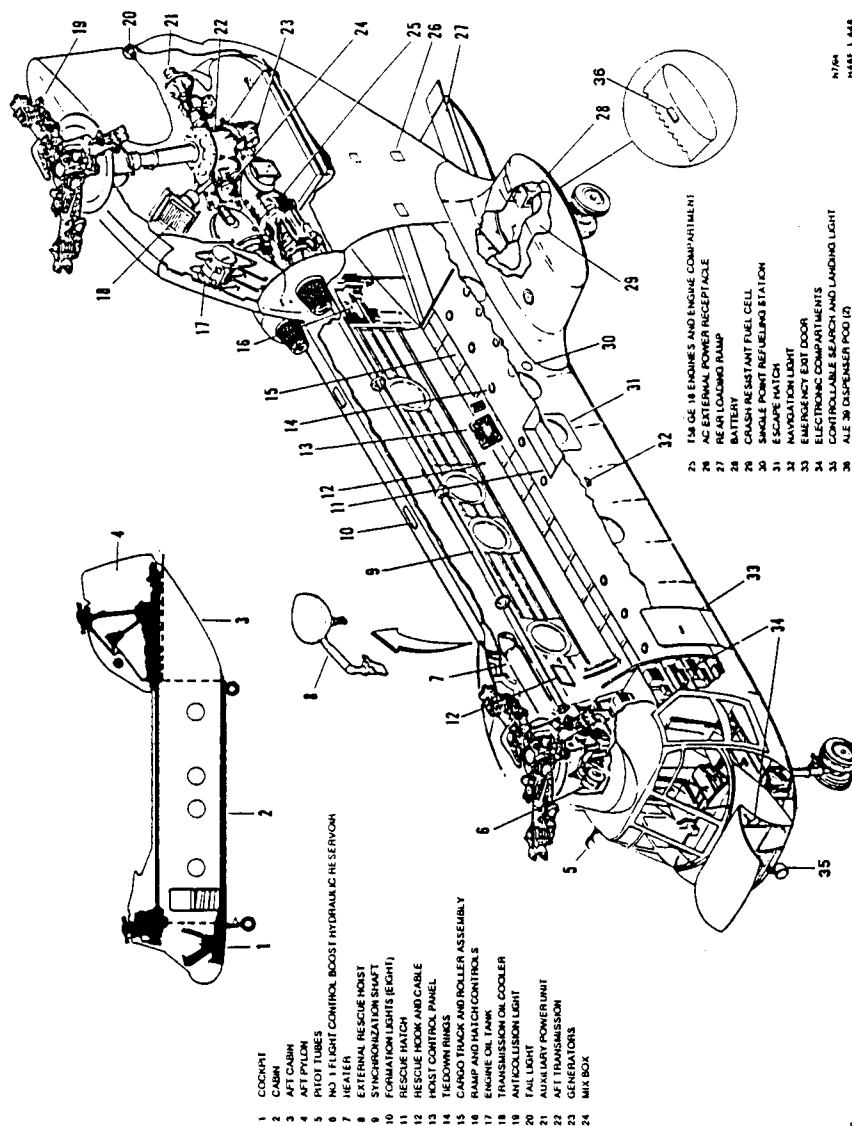


Figure A- 1
CH-46E Principle Dimensions

Source: DON 1992



N744
PAGE 1 448

54 x 38

Figure A-2
CH-46E General Arrangement

Source: DON 1992

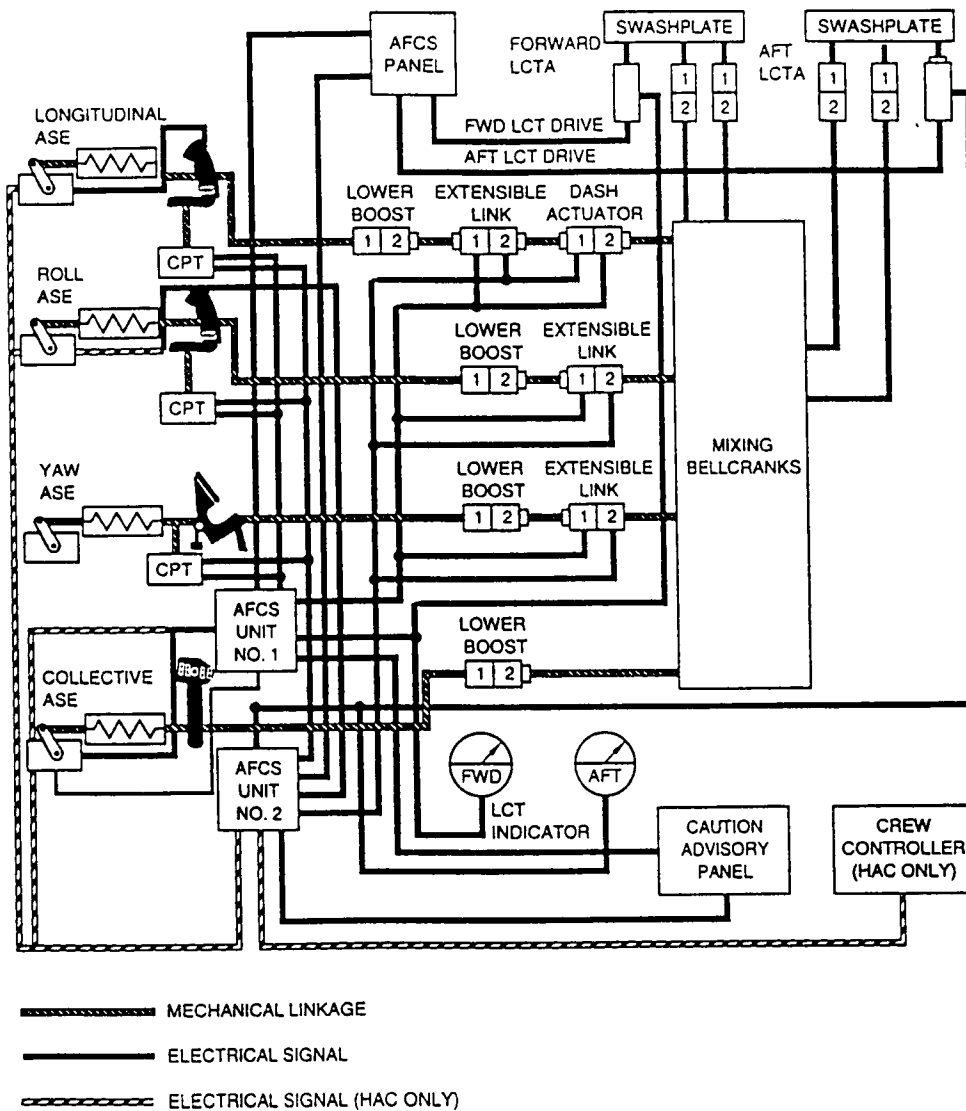


Figure A- 3
AFCS Interface Block Diagram

Source: DON 1995

1. Automatic Stabilization Equipment (Centering Springs, Control Position Transducers, Longitudinal and Lateral Actuators)
2. Dual Extensible Link Actuators
3. Differential Airspeed Hold Actuator
4. Forward Longitudinal Cyclic Trim Actuator
5. Aft Longitudinal Cyclic Trim Actuator
6. Crew Controller
7. Automatic Flight Control Computers
8. Sideslip Ports (Left and Right Sides)
9. Static Ports (Left and Right Sides)
10. Pitot Ports
11. Weight-On-Wheels Switch

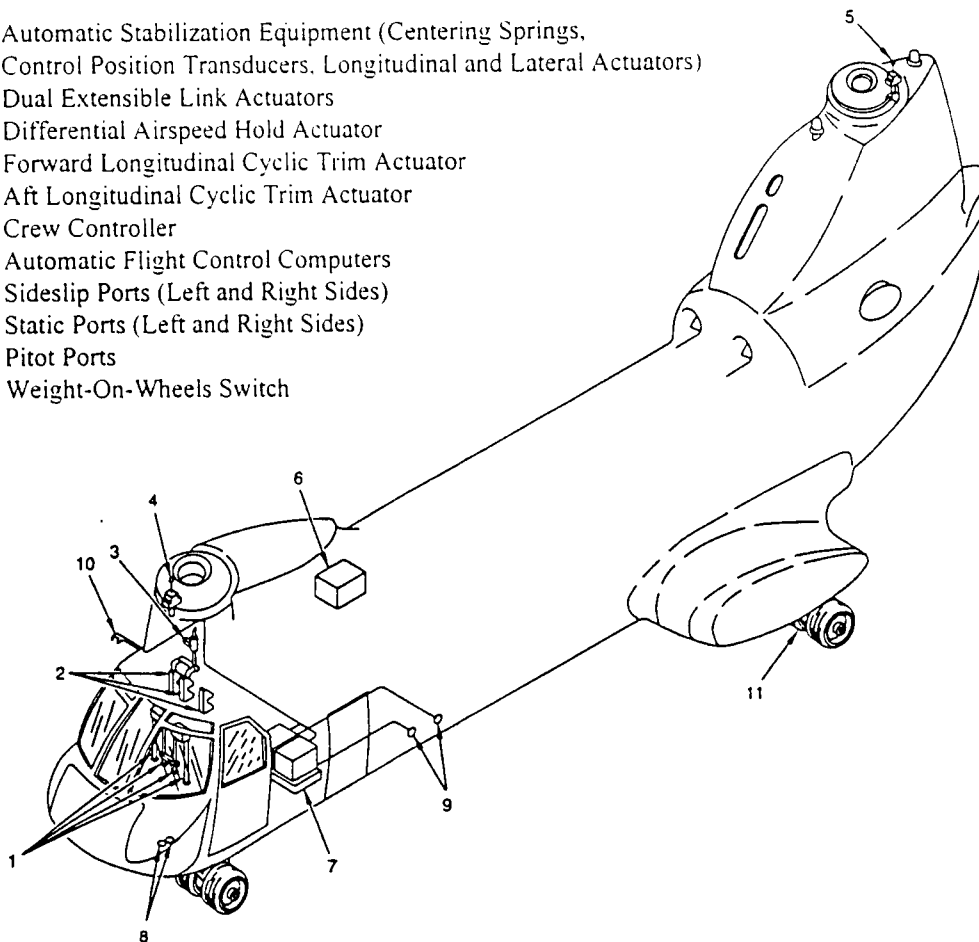
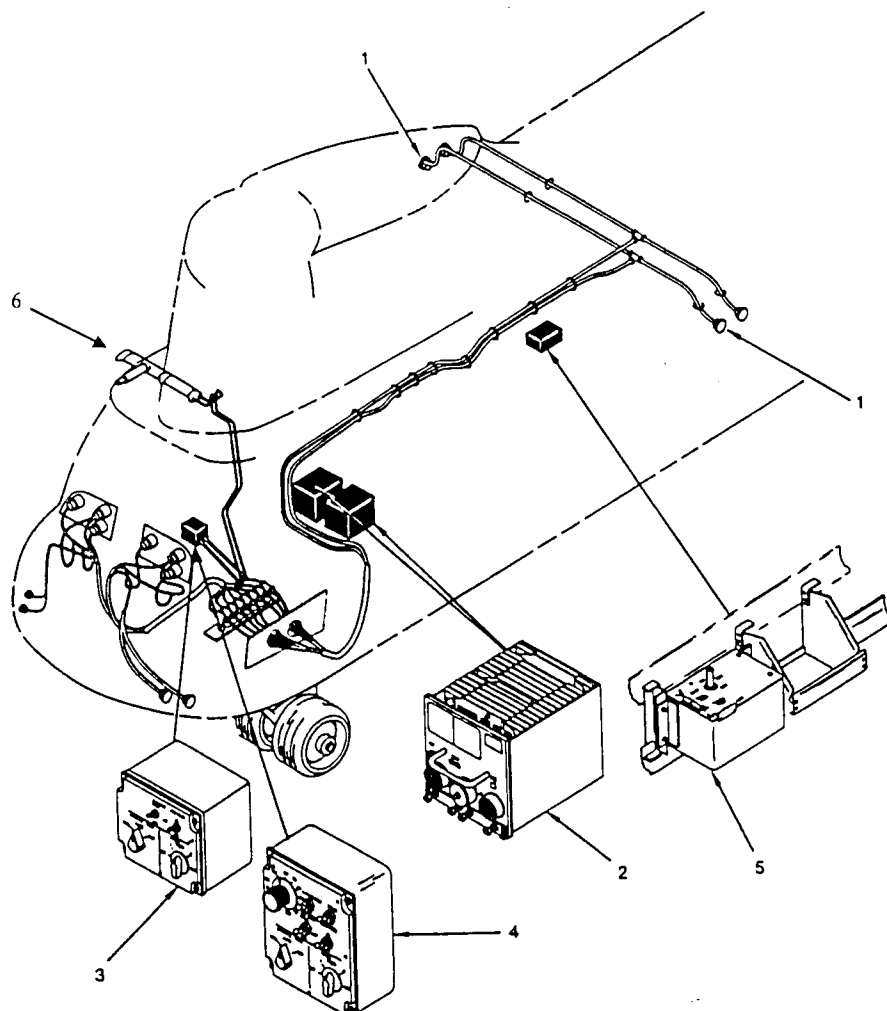


Figure A- 4
AFCS Component Location

(Sheet 1 of 3)
Source: DON 1995



1. Static Ports
2. AFCS Computer
3. AFCS Control Panel (Non Doppler Configuration)
4. AFCS Control Panel (Doppler Configuration)
5. Aircrew Station Hover Control Panel
6. Pitot Ports

Figure A-4
AFCS Component Location
 (Sheet 2 of 3)

Source: DON 1995

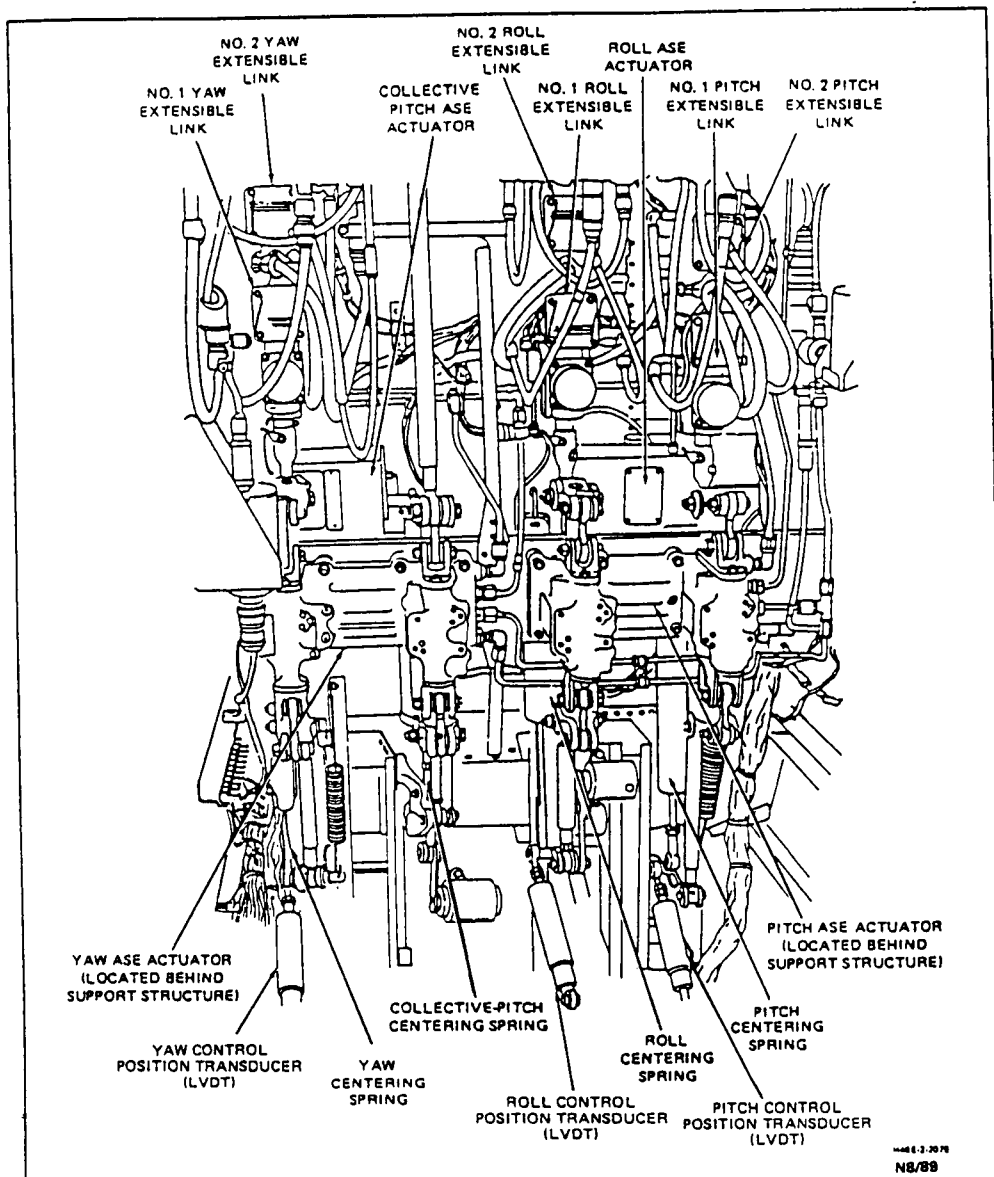
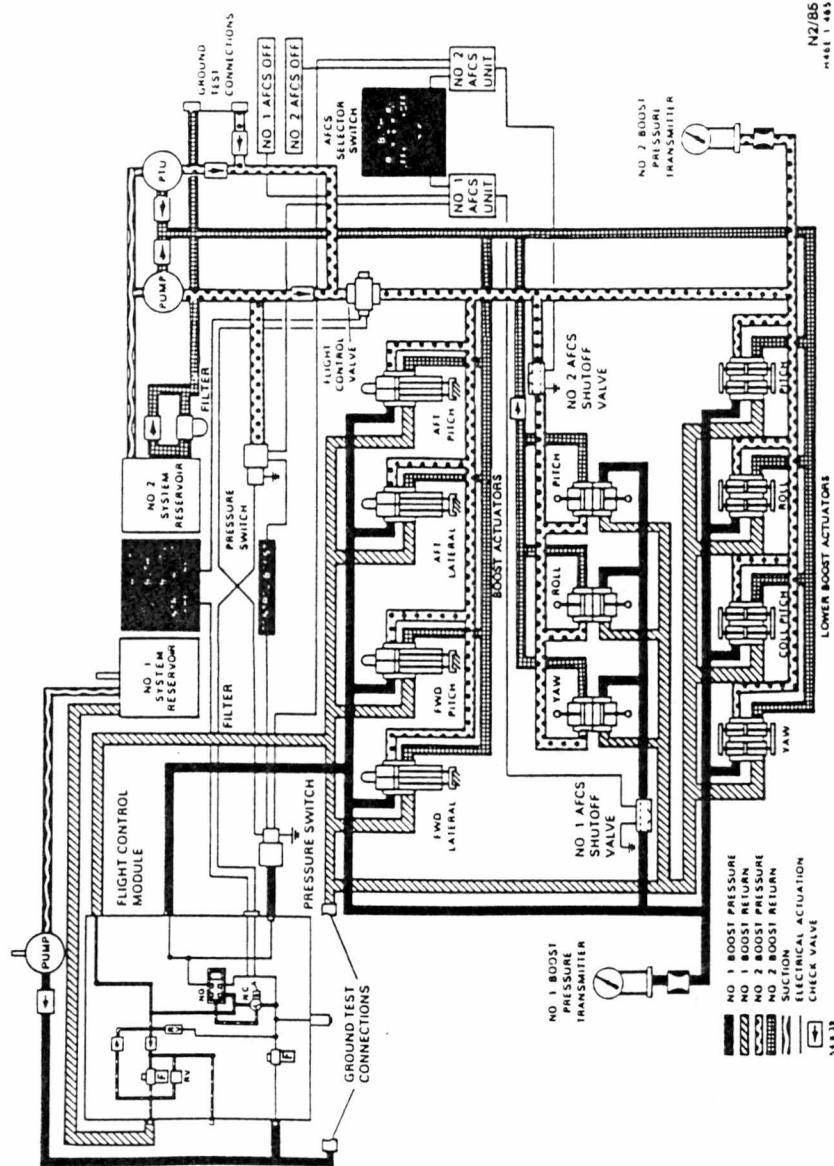


Figure A-4
AFCS Component Location
 (Sheet 3 of 3)

Source: DON 1992



N2/85
H451 1 453

Figure A-5
Flight Control Boost System

Source: DON 1992

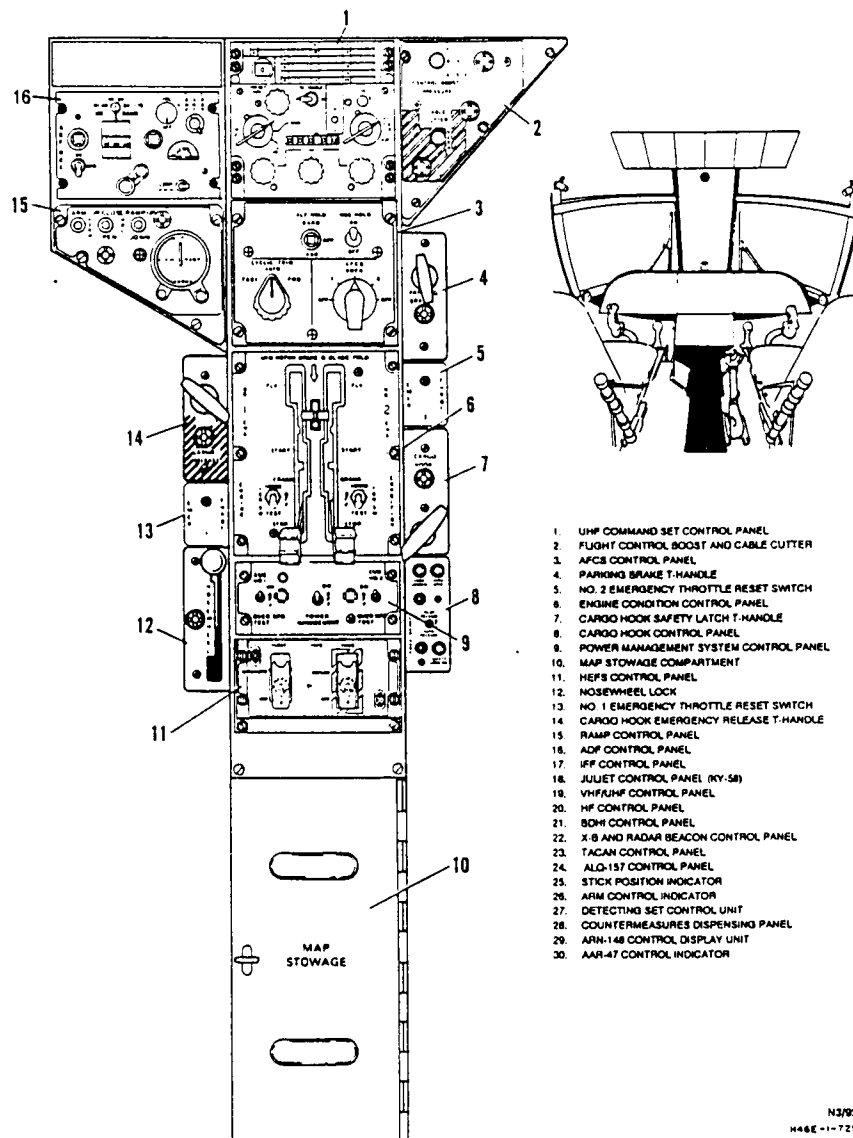


Figure A- 6
AFCS Control Panel Location

Source: DON 1992

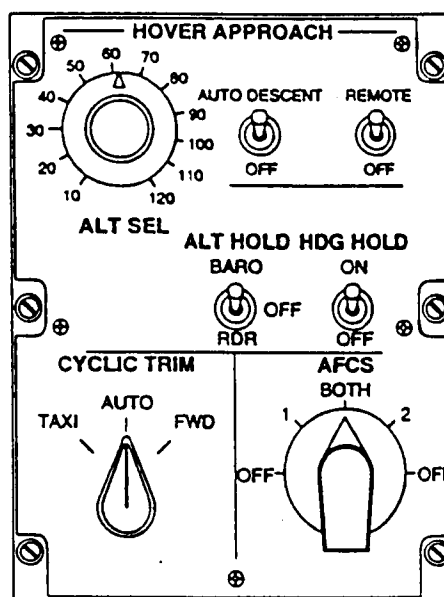


Figure A-7
AFCS Control Panel

Source: DON 1995

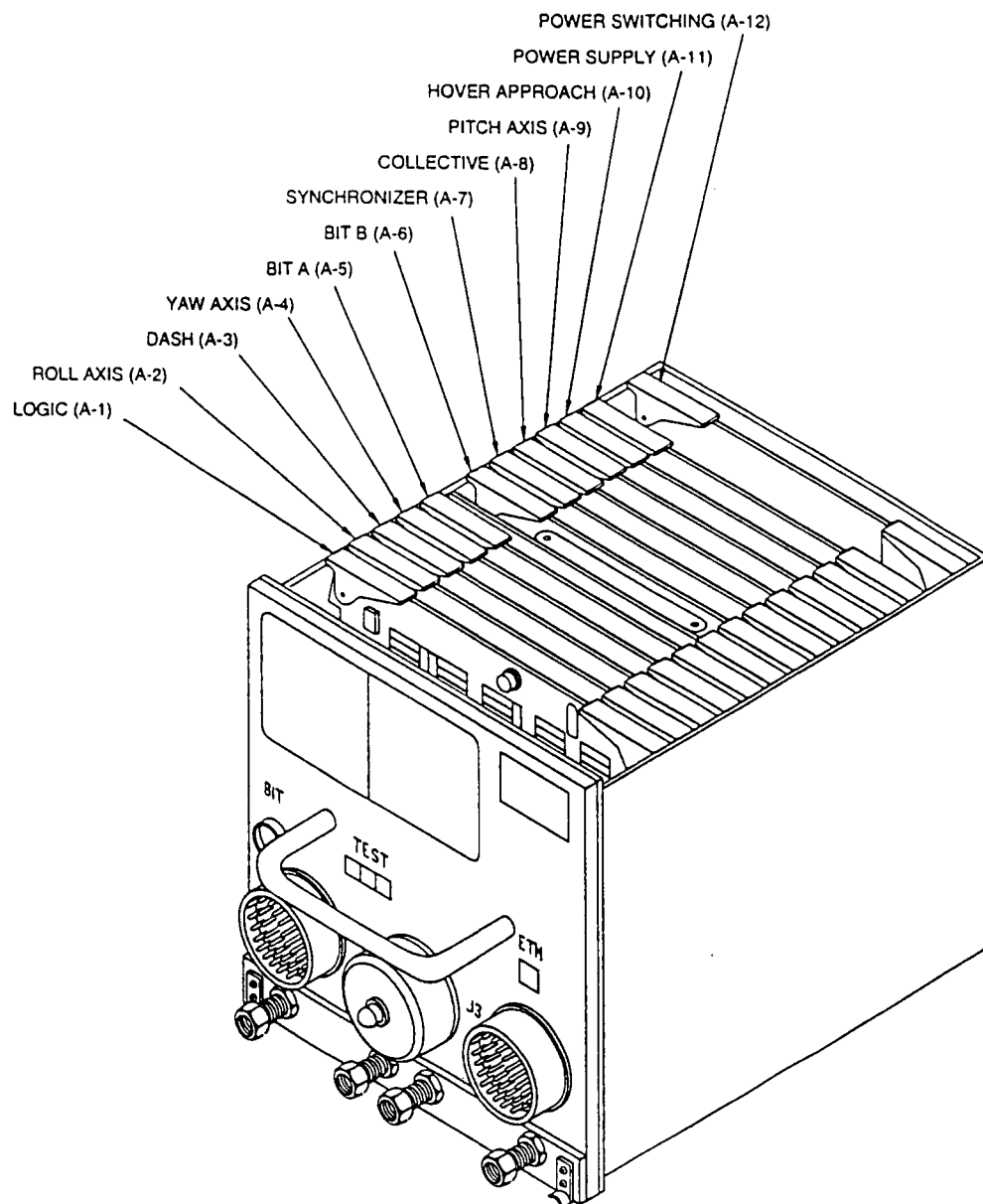


Figure A- 8
AFCS Computer

Source: DON 1995

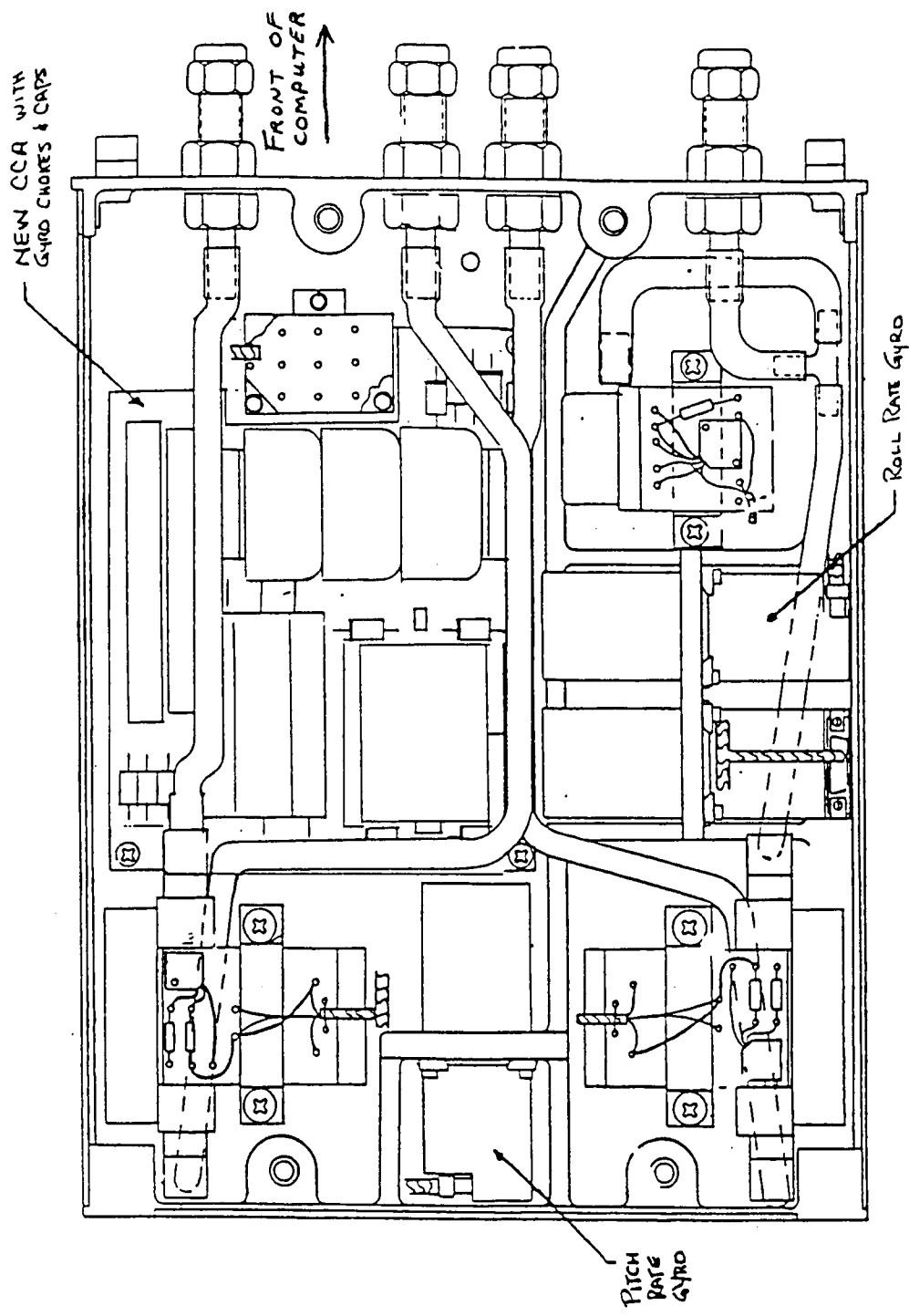


Figure A-9
Pitch Rate Gyro And Roll Rate Gyro Locations

Source: Boeing Helicopter Corporation

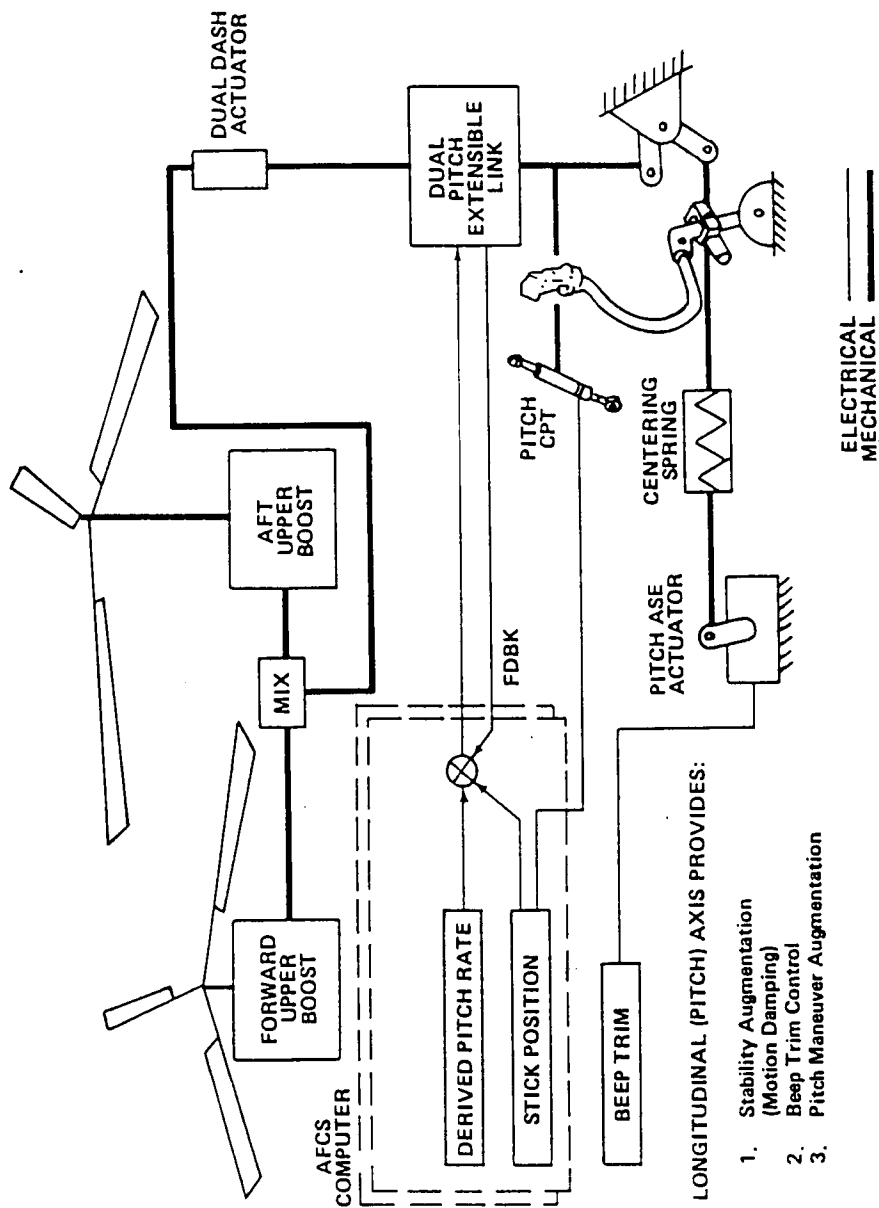


Figure A-10
Pitch Channel Interface

Source: DON 1992

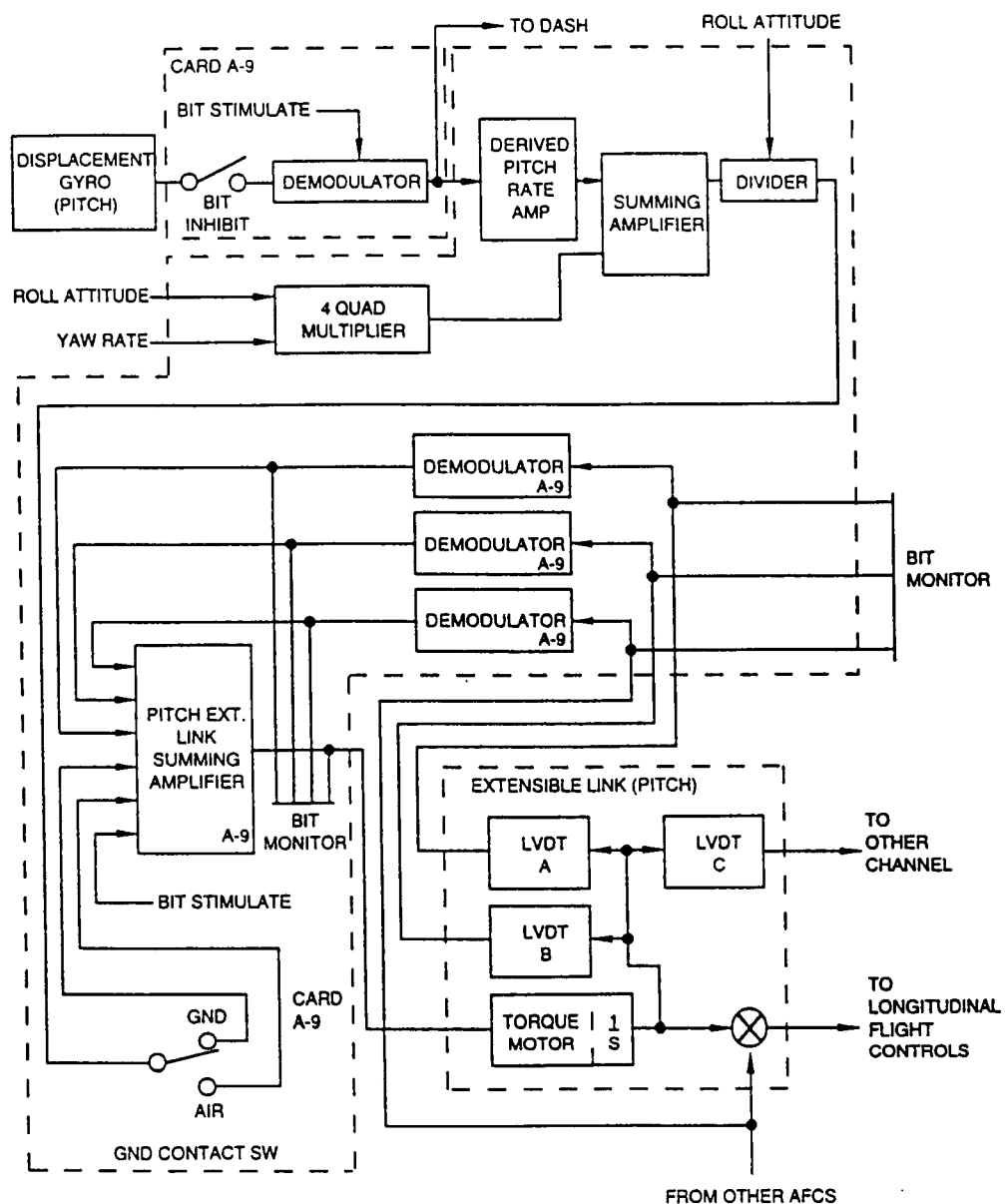


Figure A- 11
AFCS-4 Pitch Axis Stabilization Block Diagram

Source: DON 1995

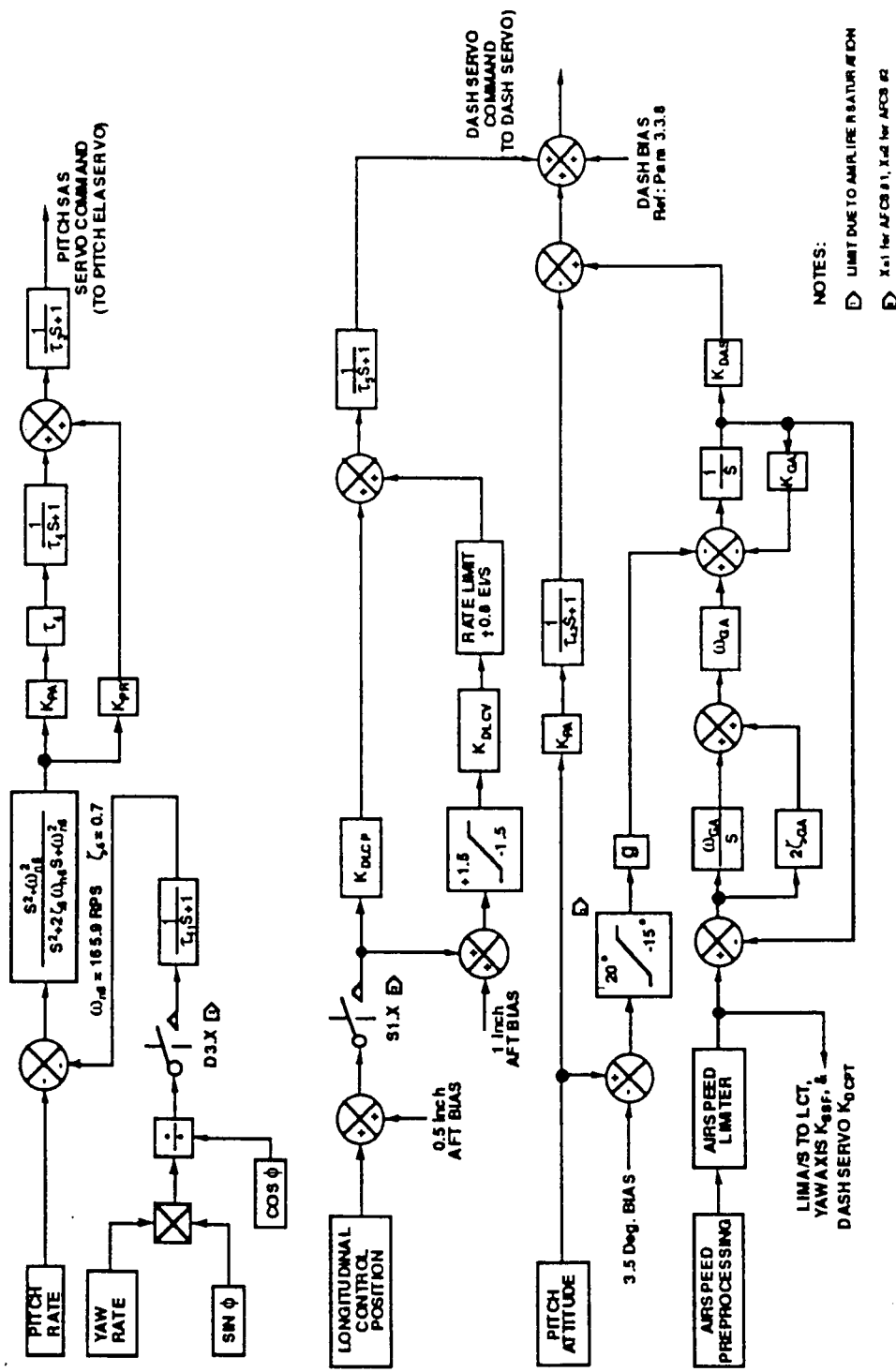
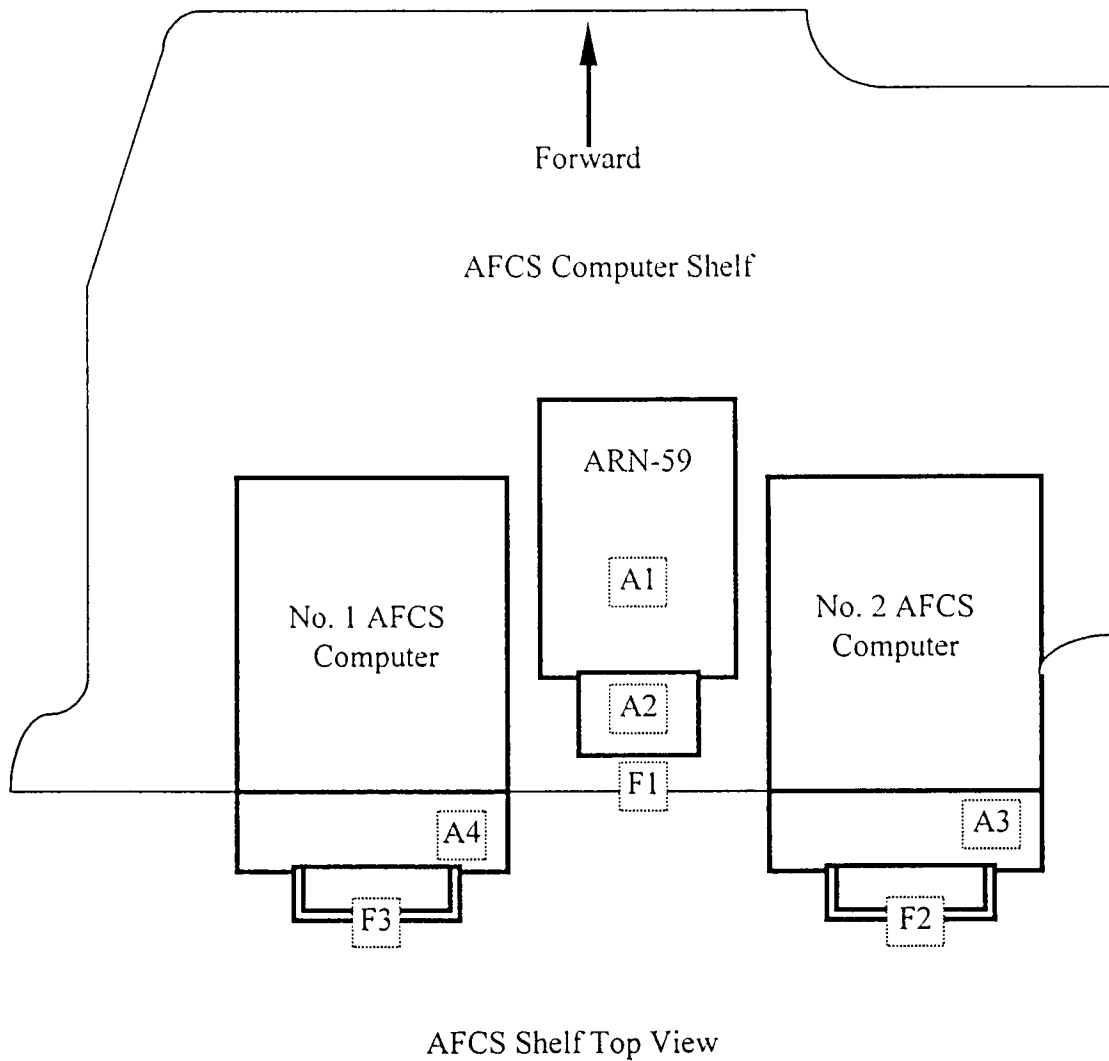


Figure A-12
AFCS-5 Longitudinal Axis Control Laws

Source: BHC 1993

MODEL CH-46E HELICOPTER



A: Accelerometer location
F: Rap hammer force application location

Figure A- 13
AFCS Computer Rap Test Accelerometer and
Force Application Locations

HANDLING QUALITIES RATING SCALE

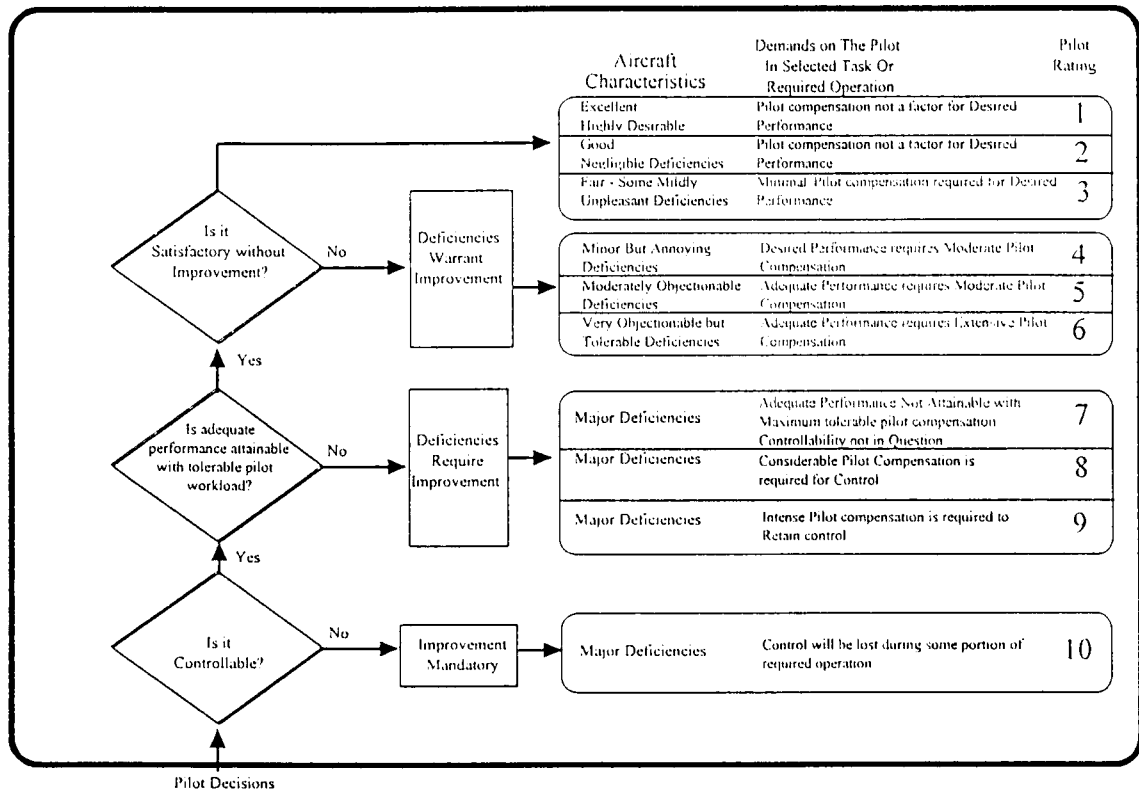


Figure A-14
Cooper-Harper Handling Qualities Rating Scale

Source: NASA TND-5153

APPENDIX B

Table B- 1
CH-46E Test And Test Conditions
(Sheet 1 of 6)

Test ⁽¹⁾	AFCS Computer Configuration ⁽²⁾	GW (lb)	CG (in)	Pressure Altitude (ft)/ Temperature (°C) ⁽³⁾	Airspeed (KIAS)	Remarks
Ground Events						
AFCS Functional Tests and Frequency Response Tests	N/A	N/A	N/A	On Deck	0	External hydraulic power and electrical power applied, engines secured, rotors static. Tests conducted by Boeing personnel (BHC 1995). The command frequency range was 0.10 to 100.00 Hz. Discrete command amplitudes were 0.01 and 0.025 inches to evaluate the responses of the pitch extensible link actuators positions.
Rap Tests on Stiffened AFCS Shelf	N/A	N/A	N/A	On Deck	0	Used dynamic signal analyzer for data reduction. Applied hammer rap at selected mount locations. Compared data with data from DON 1994.
Flying Qualities Tests						
Longitudinal Static Stability	A02VS100-5	20,780 22,690	305.5 310.4	3,000 / 5 3,000 / -3	100	Varied airspeed ± 5, 10, 15 kt from trim. Collective fixed. No retrimming.
	A02VS100-4	20,690	305.1	3,000 / -6		
Maneuvering Stability	A02VS100-5	20,130 21,620	303.6 307.7	3,000 / 5 3,000 / -3	100	Constant power, steady turns at ±15, 30, 45 deg AOB.
	A02VS100-4	20,240	303.8	3,000 / -7		

TABLE B-1
CH-46E Test And Test Conditions
(Sheet 2 of 6)

Test ⁽¹⁾	AFCs Computer Configuration ⁽²⁾	GW (lb)	CG (in)	Pressure Altitude (ft)/ Temperature (°C) ⁽³⁾	Airspeed (KIAS)	Remarks
Dynamic Longitudinal Stability	Long Term	20,630	305.0	3,000 / 5	100	Natural and artificial excitation (slowed airspeed by 10 kt, returned cyclic to trim).
		22,520	310.0	3,000 / -3		
	Short Term	20,590	304.8	3,000 / -7	100	Conducted $\pm 1/2$ in. doublets at 1 Hz.
		20,530	304.8	3,000 / 5		
Static Lateral-Directional Stability		22,320	309.5	3,000 / -3	100	Conducted steady heading sideslips: $\pm 1/4$ and $1/2$ ball deflections.
		20,390	304.2	3,000 / -7		
Dynamic Lateral-Directional Stability		19,280	301.0	3,000 / -7	100	Evaluated LDO behavior. Release from steady heading sideslip from $1/2$ right ball.
		22,230	309.3	3,000 / -3		
Forward Flight Control Response		19,280	301.0	3,000 / -7	100	Longitudinal cyclic step inputs up to $\pm 1/2$ inches from trim in approximately .2 seconds or less. Increment size $1/4$ inch.
		20,380	304.3	3,000 / 12		
		20,080	303.5	3,000 / 12	100	Lateral cyclic step inputs up to $\pm 1/2$ inches from trim in approximately 0.2 seconds or less. Increment size $1/4$ inch.
		20,740	305.3	3,000 / -2		
Hover Control Response		19,800	302.5	3,000 / -7	0	From a stable hover into the wind. Wind 3 kt. Lateral cyclic step inputs up to $\pm 1/4$ inches from trim in approximately .2 seconds or less. Increment size $1/4$ inch.
		20,340	304.1	3,000 / 7		
		20,090	303.4	460 (500) / 3	0	
		19,840	302.6	380 (500) / 12		

TABLE B-1
CH-46E Test And Test Conditions
(Sheet 3 of 6)

Test	AFCS Computer Configuration (2)	GW (lb)	CG (in)	Pressure Altitude (ft)/ Temperature (°C)(3)	Airspeed (KIAS)	Remarks
Level Flight with Copilot's (MD-1) Attitude Gyro CB Pulled.	A02VS100-5	20,400	304.4	3,000 / 5	100	In-flight test to verify that AFCS logic allows SAS servo loops to remain active with VG invalid. Test only conducted with proposed production, A02VS100-5 AFCS units installed in test aircraft.
Mission Maneuvers						
In-Ground Effect Hover	A02VS100-5	20,280	304.4	-270 (30) / -1	0	Stable hover into wind with trim engaged and disengaged. Conducted 90, 180, 360 deg pedal turns at approximately 24 deg/sec. Wind 4 -16 kt.
		21,120	306.4	-350 (30) / 6		
		19,720	302.2	-10 (30) / 4		
Out-Of-Ground Effect Hover	A02VS100-4	19,370	301.1	170 (30) / 5	0	Stable hover into wind. Wind 3 -16 kt.
		20,890	305.6	-80 (30) / 5		
		20,700	305.6	-200 (100) / -2	0	
		21,120	306.4	-250 (100) / 5		
Low Speed Flight	A02VS100-4	19,935	302.9	460 (500) / 3		Used doppler to determine groundspeed. Maintained wind off the nose. Wind 4 kt.
		19,370	301.1	240 (100) / 2	0	
Max Power Climb with Turns	A02VS100-5	19,735	302.3	380 (500) / 12		Maintained 100% Q during climb for angle of banks of ± 0, 15, 30, 45 degrees.
		20,080	303.4	-270 (30) / -1	20	
		20,820	305.6	-350 (30) / 6	KGAS	
		21,300	306.9	1,500 / 2	100	
	A02VS100-5	19,180	300.6	3,000 / -7		
		19,680	302.2	3,000 / 12		
	A02VS100-4	20,990	305.9	3,000 / -6	80	

TABLE B-1
CH-46E Test And Test Conditions
(Sheet 4 of 6)

Test	AFCS Computer Configuration (2)	GW (lb)	CG (in)	Pressure Altitude (ft)/ Temperature (°C)(3)	Airspeed (KIAS)	Remarks
Min Power Descent with Turns	A02VS100-5	20,030	303.3	3,000 / 5	100	Maintained Nf/Nr needles joined with collective for angle of banks of ± 0 , 15, 30, 45 degrees.
	A02VS100-4	19,180	300.6	3,000 / -7	80	
Constant Altitude Turns	A02VS100-5	20,230	303.9	3,000 / 5	100	$\pm 15, 30, 45$ deg AOB turns while maintaining altitude.
	A02VS100-4	21,720	308.0	3,000 / -3	100	
Max Level Flight Airspeed	A02VS100-5	20,430	304.5	3,000 / 5	140	100% torque applied.
	A02VS100-4	19,480	301.6	3,000 / -7	135	
	A02VS100-5	22,110	309.0	3,000 / -3	131	
	A02VS100-4	20,310	304.0	3,000 / -7	135	
Quickstops	A02VS100-5	19,030	300.2	-250 (50) / -2	90	From initial airspeed and 50 ft AGL, reduced collective as necessary to maintain altitude while decelerating in a approximately 15 degree nose up attitude.
	A02VS100-4	18,630	298.8	-240 (50) / 15	110	
AOB Capture	A02VS100-5	19,830	302.7	200 (500) / -2	100	Constant power, 45 degrees angles of bank left & right with trim engaged in 2 seconds.
	A02VS100-4	21,070	306.1	3,000 / -2	100	
Pitch Attitude Capture	A02VS100-4	20,690	305.1	3,000 / 7	100	Constant power, 10 degrees angle of bank left and right with trim release engaged and disengaged.
	A02VS100-5	20,580	304.9	3,000 / 5		
	A02VS100-4	22,380	309.6	3,000 / -3		
Roll Reversal	A02VS100-4	20,540	304.7	3,000 / -7	80	Constant power, ± 45 deg max AOB. Stabilized for 2 seconds at each banked condition. Conducted single cycle reversals up to 30 deg/sec.
	A02VS100-5	19,730	302.4	200 (500) / -2		

TABLE B-1
CH-46E Test And Test Conditions
(Sheet 5 of 6)

Test	AFCS Computer Configuration (2)	GW (lb)	CG (in)	Pressure Altitude (ft)/ Temperature (°C)(3)	Airspeed (KIAS)	Remarks
Steep Approach	A02VS100-5	20,030	303.3	0 / -2	50-0	Approach to IGE hover LAW HMT-204 Standardization Manual from 300 ft to 10 ft AGL. Wind 3 - 13 kt.
	A02VS100-4	19,490	301.5	475 (300) / 2		
Power Recovery Autorotations Vertical Landing From HIGE	A02VS100-5	19,280	301.0	2000 / 14	80, 110	ECLs in FLY. Maintained Nr at 100%.
	A02VS100-4	20,090	303.4	2,000 / -5	80, 110	Recovered by 1,000 ft AGL.
	A02VS100-5	19,590	301.8	-10 (30-0) / 4	0	From 30 ft hover, descended at approximately 6 ft per second. Hover into wind. Wind 0- 6kt.
	A02VS100-4	20,790	305.4	-80 to -110 (30-0) / 5		
Lateral Sidestep	A02VS100-5	19,290	300.6	-10 (30) / 4	0	Started from hover into 6 kt wind. Applied 15 angle of bank in approximately 1/2 second. Stopped sidestep after approximately 100 ft
	A02VS100-4	20,670	305.0	-80 / 5		
Flight with External Load(4)						
Max Power Climb	A02VS100-5	23,080	306.0	700 (500) / 2	80	Used 100% Q to climb to 1000 ft AGL.
Constant Altitude Turns	A02VS100-5	22,890	305.6	1,200 (1000) / 0	80	Conducted constant altitude, turns at ± 0, 15,30 deg AOB.
Shipboard Approach	A02VS100-5	22,730	305.1	500 to 200 (300-0) / 1	80-0	From 300 ft commenced AGL approach shipboard approach to NAWCAD helo pad to end up at a hover.
In-Ground Effect Hover	A02VS100-5	22,480	304.5	200 (35) / 3	0	Hover into wind.
Hover Pedal Turns	A02VS100-5	22,240	303.8	200 (35) / 3	0	Conducted 180 degree pedal turns left and right at approximately 5 deg/sec. Wind 5 kt

TABLE B-1
CH-46E Test And Test Conditions
(Sheet 6 of 6)

Test	AFCS Computer Configuration (2)	GW (lb)	CG (in)	Pressure Altitude (ft)/ Temperature (°C)(3)	Airspeed (KIAS)	Remarks
Air Taxi	A02VS100-5	22,180	303.7	200 (35) / 3	10 KGAS	Conducted forward, rearward, left and right sideward taxi using doppler for groundspeed. Nose into wind. Wind 5 kt.
Vertical Takeoff	A02VS100-5	23,080	306.0	200 (35) / 3	0	Conducted vertical takeoff at approximately 2 ft per second. Wind 5 kt.
Vertical Landing From Hover	A02VS100-5	21,920	303.0	200 (35) / 3	0	Conducted vertical descent at approximately 1 ft per second. Wind 5 kt.

Notes: (1) All flight tests performed at 100% Nr and AFCS in "Both." Hatch and ramp closed, cabin hatch open, windows closed.
(2) Average values.
(3) Altitudes in parenthesis are AGL.
(4) External load of 2,000 lb.

Table B- 2
CH-46E AFCS Evaluation Instrumented
Aircraft And Flight Parameters

PARAMETER	UNITS	MEASUREMENT RANGE	SIGNAL ORIGIN
#1 Longitudinal Cyclic Control Position	%	0-100	AFCS #1
#2 Longitudinal Cyclic Control Position	%	0-100	AFCS #2
#1 Lateral Cyclic Control Position	%	0-100	AFCS #1
#2 Lateral Cyclic Control Position	%	0-100	AFCS #2
#1 Directional Pedal Position	%	0-100	AFCS #1
#2 Directional Pedal Position	%	0-100	AFCS #2
Collective Control Position	%	0-100	Instrumented
#1 Pitch Attitude	deg	± 45	AFCS #1
#2 Pitch Attitude	deg	± 45	AFCS #2
#1 Pitch Rate ⁽¹⁾	deg/sec	± 30	AFCS #1
#2 Pitch Rate ⁽¹⁾	deg/sec	± 30	AFCS #2
#1 Roll Attitude	deg	± 90	AFCS #1
#2 Roll Attitude	deg	± 90	AFCS #2
#1 Roll Rate ⁽¹⁾	deg/sec	± 30	AFCS #1
#2 Roll Rate ⁽¹⁾	deg/sec	± 30	AFCS #2
#1 Sideslip (Differential Pressure)	psi	± 0.75	AFCS #1
#2 Sideslip (Differential Pressure)	psi	± 0.75	AFCS #2
#1 Yaw Rate	deg/sec	± 30	AFCS #1
#2 Yaw Rate	deg/sec	± 30	AFCS #2
Pilot's Indicated Airspeed	kts	0-170	Instrumented
Barometric Altitude	ft	-1,000 to 10,000	Instrumented
#1 Pitch ELA Position	inch	± 0.2	Instrumented
#2 Pitch ELA Position	inch	± 0.2	Instrumented
#1 DASH Actuator Position	%	0-100	Instrumented
#2 DASH Actuator Position	%	0-100	Instrumented
AFCS Shelf Vertical Acceleration	G's	± 3.5	Instrumented
Record Counter	--	--	Instrumented
Event Marker	--	--	Instrumented
Audio Voice	--	--	Instrumented

Note: (1) Signal was derived rate with AFCS A02VS100-4 computers installed in aircraft.

APPENDIX C

MODEL CH-46E HELICOPTER

PELA #1 Command to PELA #1 Position
 RELA #1 Command to RELA #1 Position
 Buno 153355

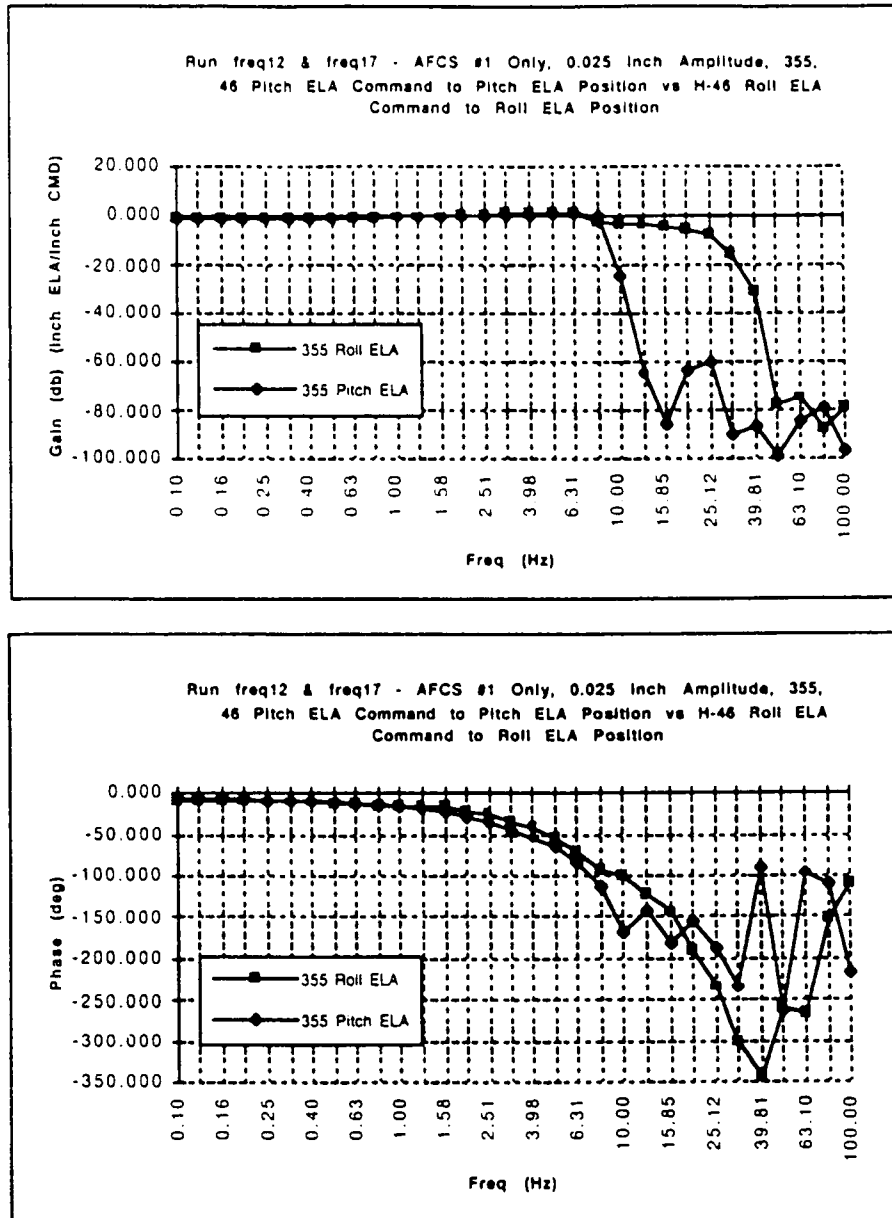


Figure C- 1
 Pitch and Roll Extensible Link Actuators (PELA, RELA) Responses ~
 AFCS #1, 0.025 Inch Command Amplitude

MODEL CH-46E HELICOPTER

PELA #1 Command to PELA #1 Position
 RELA #1 Command to RELA #1 Position
 Buno 157692

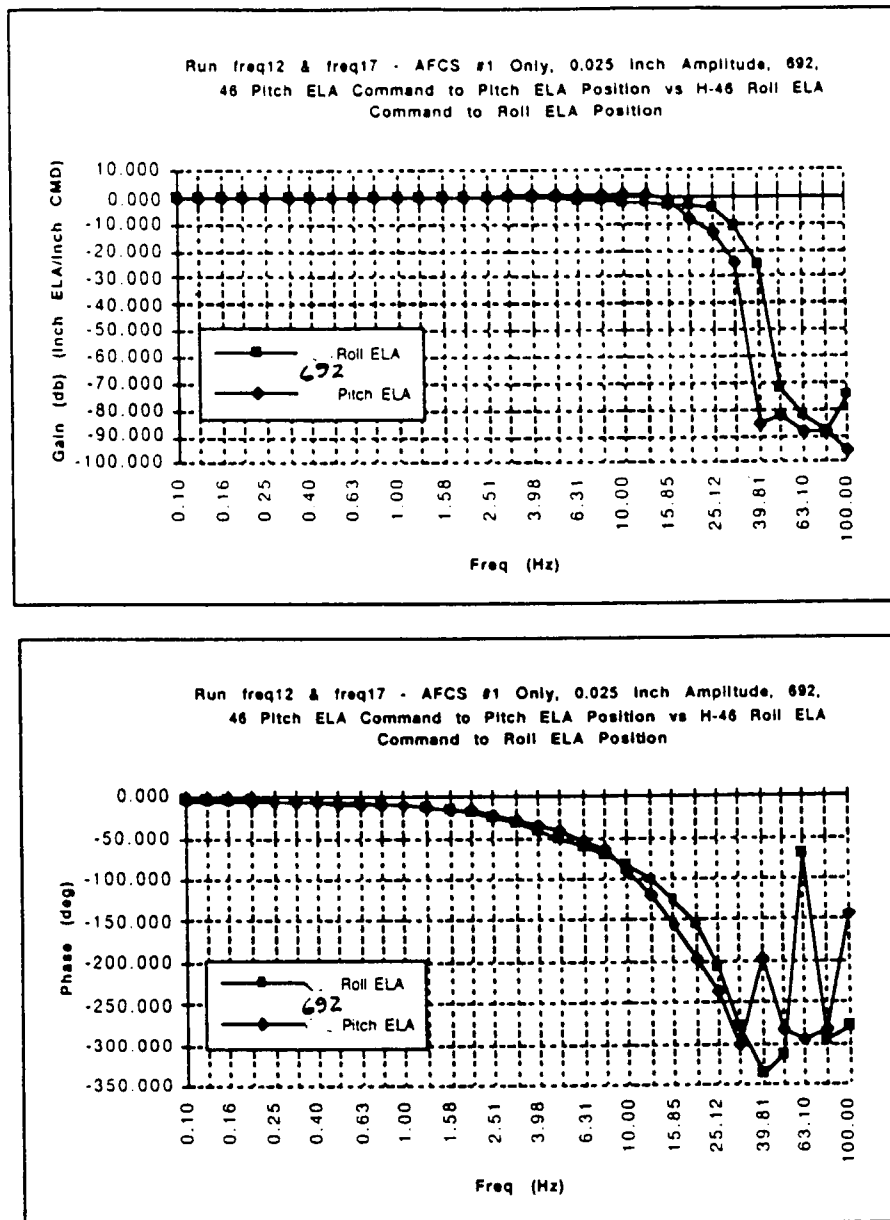


Figure C- 2
 Pitch and Roll Extensible Link Actuators (PELA, RELA) Responses ~
 AFCS #1, 0.025 Inch Command Amplitude

MODEL CH-46E HELICOPTER

PELA #1 Command to PEL A #1 Position
Buno 157692 vs. 153355

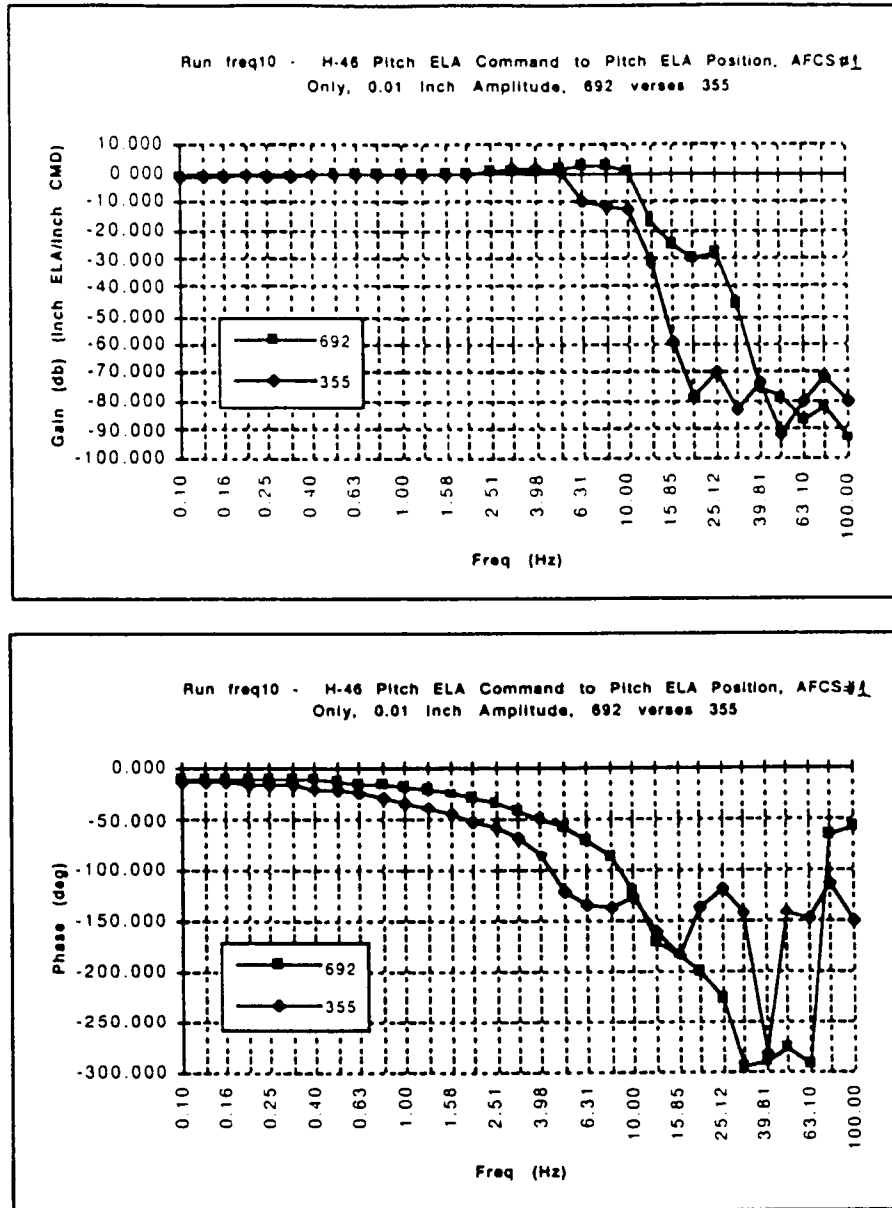


Figure C-3
Pitch Extensible Link Actuator (PELA) Response ~
AFCS #1, 0.01 Inch Command Amplitude

MODEL CH-46E HELICOPTER

PELA #1 Command to PELA #1 Position
Buno 157692 vs. 153355

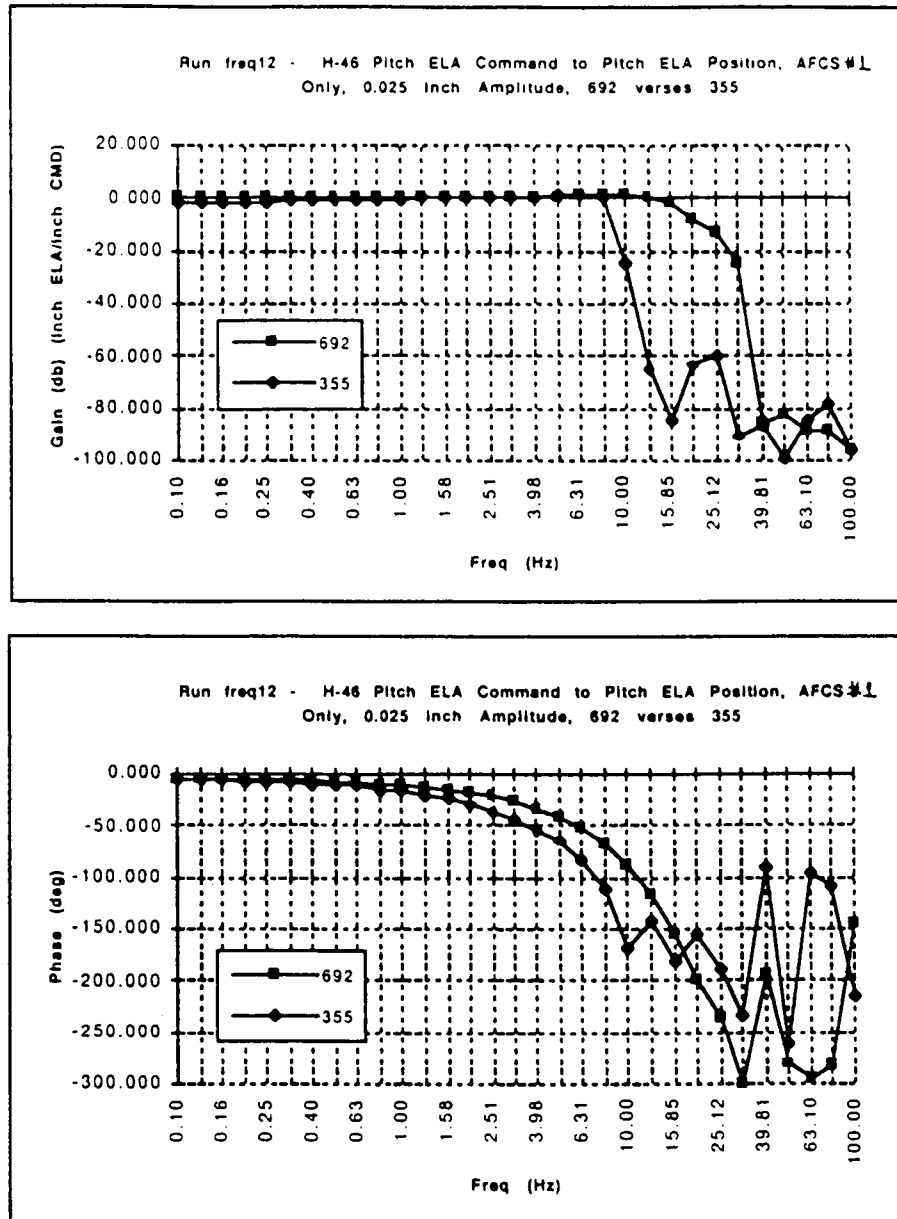


Figure C- 4
Pitch Extensible Link Actuator (PELA) Response ~
AFCS #1, 0.025 Inch Command Amplitude

MODEL CH-46E HELICOPTER

PELA #2 Command to PELA #2 Position
Buno 157692 vs. 153355

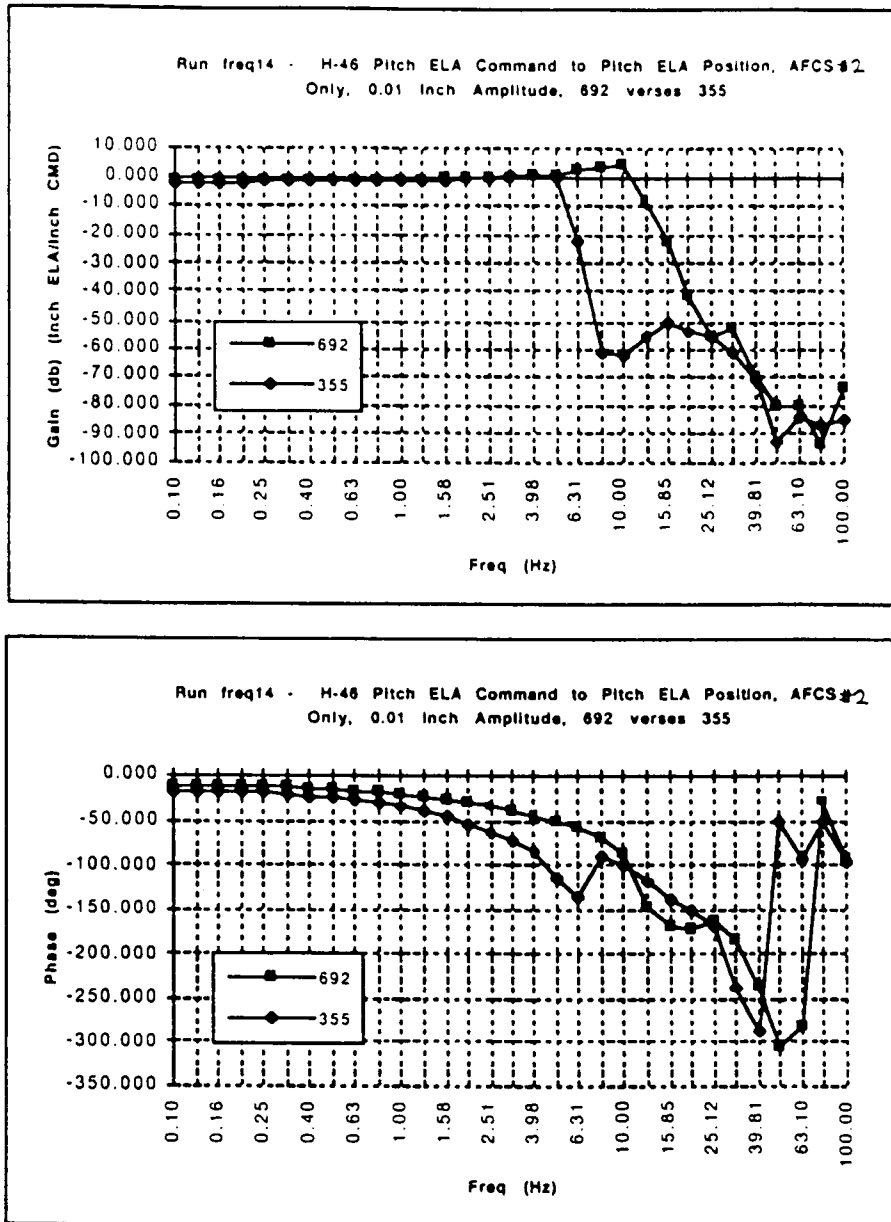


Figure C-5
Pitch Extensible Link Actuator (PELA) Response ~
AFCS #2, 0.01 Inch Command Amplitude

MODEL CH-46E HELICOPTER

PELA #2 Command to PELA #2 Position
Buno 157692 vs. 153355

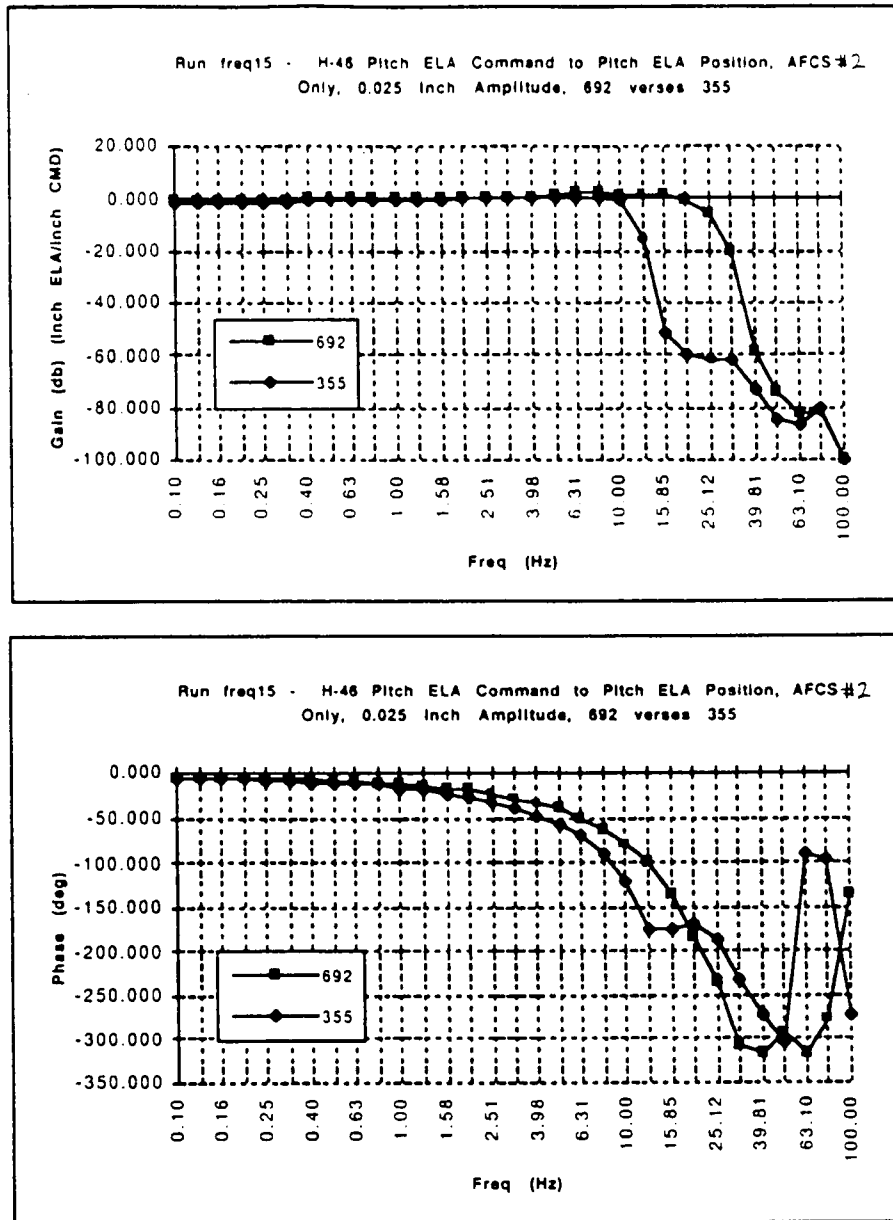


Figure C- 6
Pitch Extensible Link Actuator (PELA) Response ~
AFCS #2, 0.025 Inch Command Amplitude

MODEL CH-46E HELICOPTER

PELA #1 and PEL A #2 Command to PEL A #1 Position
Buno 157692 vs. 153355

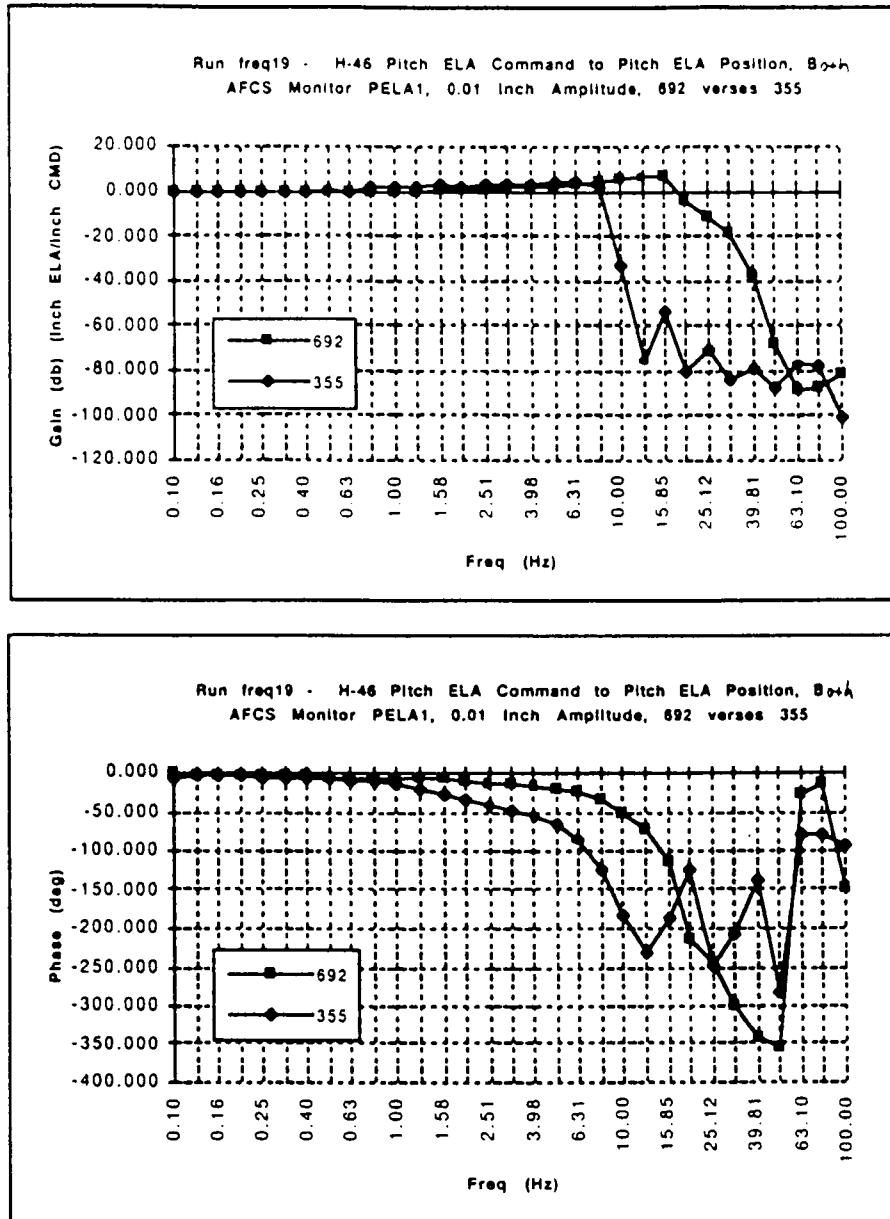


Figure C- 7
Pitch Extensible Link Actuator (PELA) Response ~
Both AFCS, 0.01 Inch Command Amplitude

MODEL CH-46E HELICOPTER

PELA #1 and PEL A #2 Command to PEL A #1 Position
Buno 157692 vs. 153355

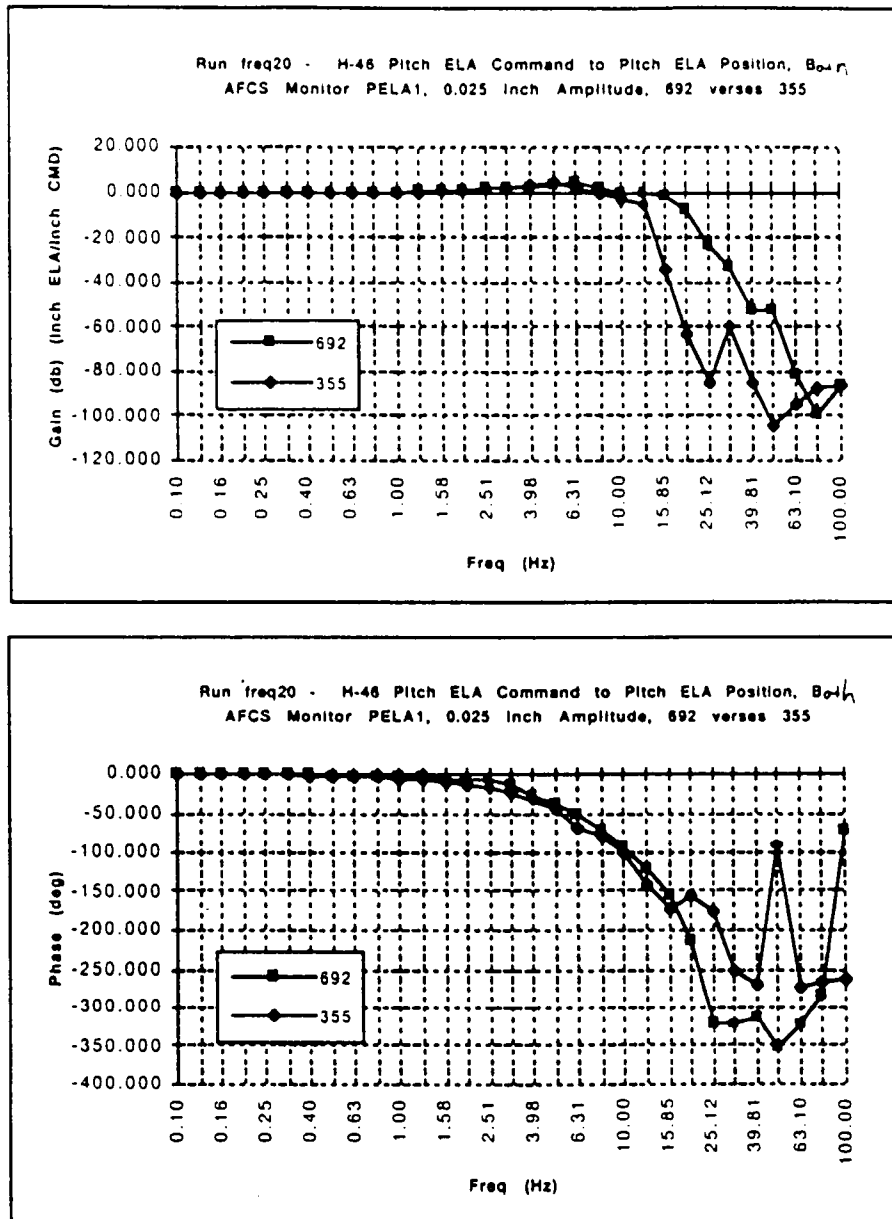


Figure C- 8
Pitch Extensible Link Actuator (PELA) Response ~
Both AFCS, 0.025 Inch Command Amplitude

MODEL CH-46E HELICOPTER

RELA #1 Command to RELA #1 Position
Buno 157692 vs. 153355

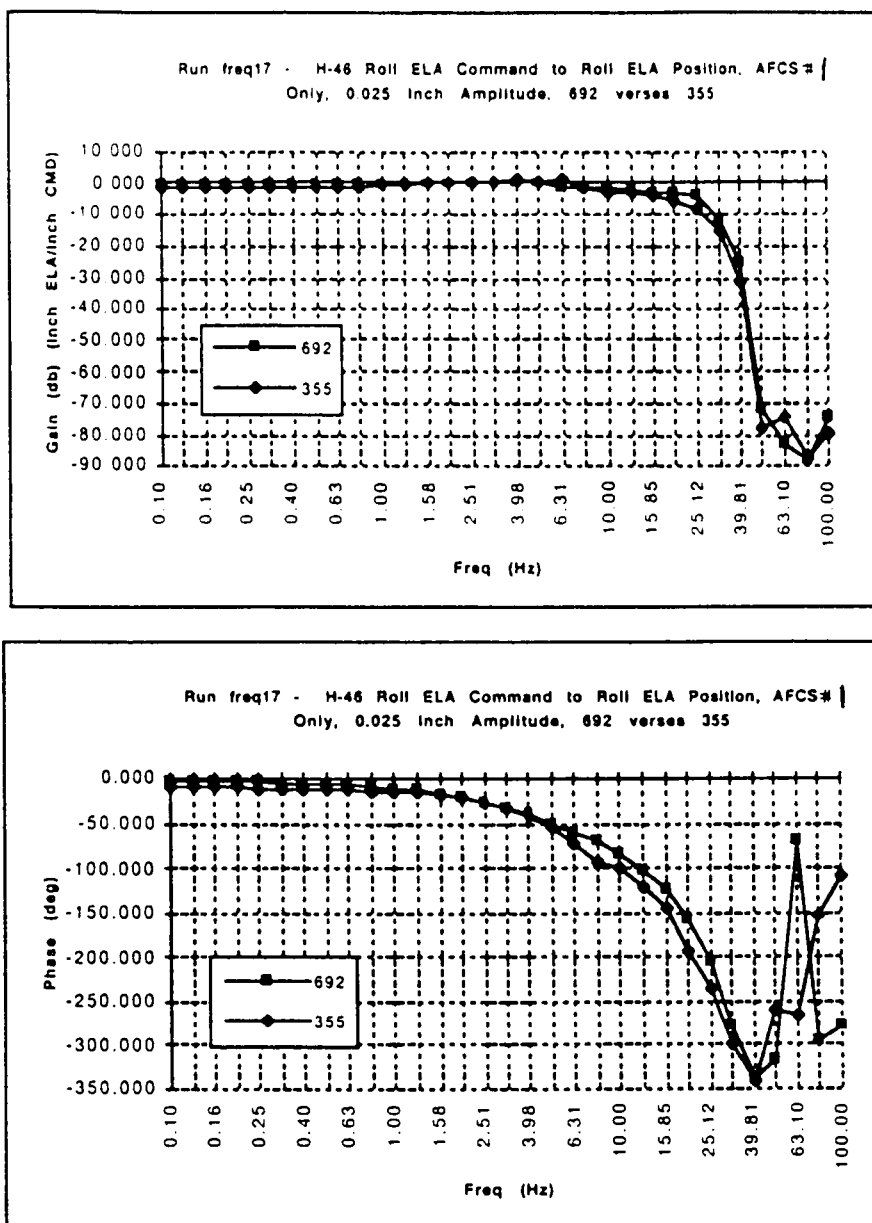


Figure C- 9
Roll Extensible Link Actuator (RELA) Response ~
AFCS #1, 0.025 Inch Command Amplitude

MODEL CH-46E HELICOPTER

Aircraft Buno	GW (lb)	C.G. (in)	Hp (ft)	OAT (°C)	Nr (%)	Airspeed (KIAS)	AFCS Configuration
157692	20,305	304.0	3,000	-7	100.0	135	AFCS-4

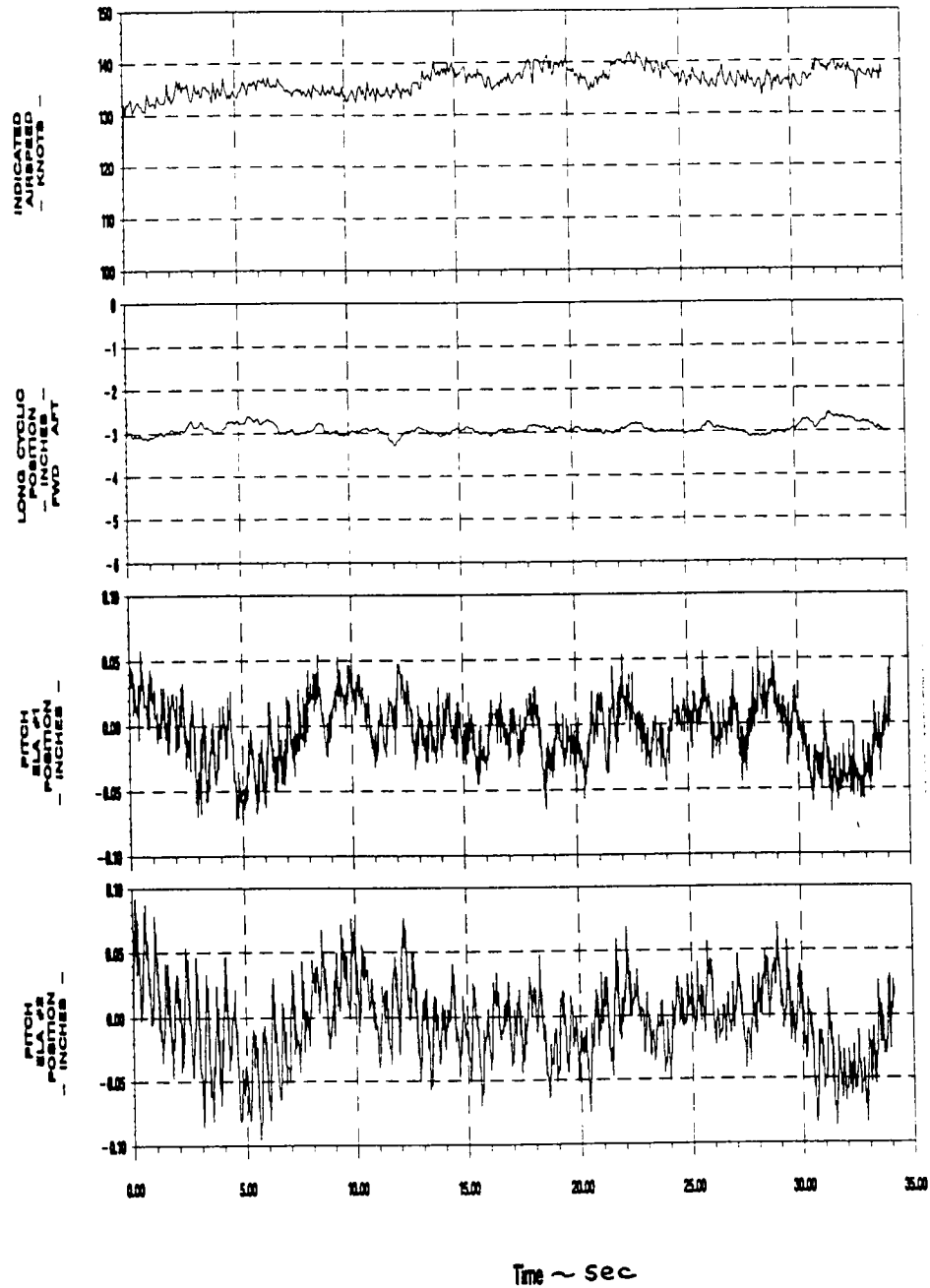


Figure C- 10
Maximum Level Flight Airspeed, AFCS-4 Computers

MODEL CH-46E HELICOPTER

Aircraft Buno	GW (lb)	C.G. (in)	Hp (ft)	OAT (°C)	Nr (%)	Airspeed (KIAS)	AFCS Configuration
157692	19,480	301.6	3,000	-7	100.0	135	AFCS-5

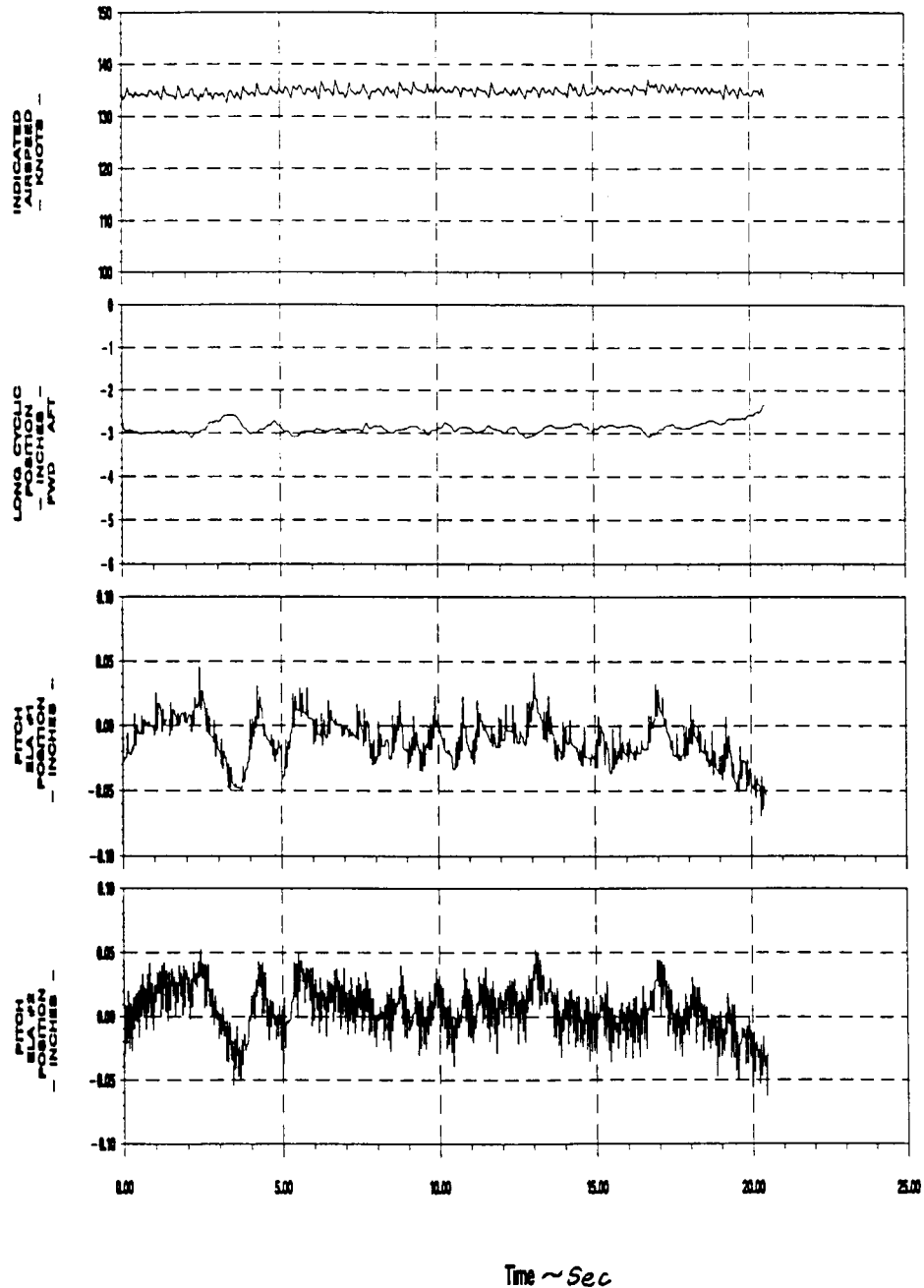


Figure C- 11
Maximum Level Flight Airspeed, AFCS-5 Computers

MODEL CH-46E HELICOPTER

Symbol	Aircraft Buno	GW (lb)	C.G. (in)	Hp (ft)	OAT (°C)	Nr (%)	Airspeed (KIAS)	AFCS Configuration
—	157692	19,480	301.6	3,000	-7	100.0	135	AFCS-5
---	157692	20,310	304.0	3,000	-7	100.0	135	AFCS-4

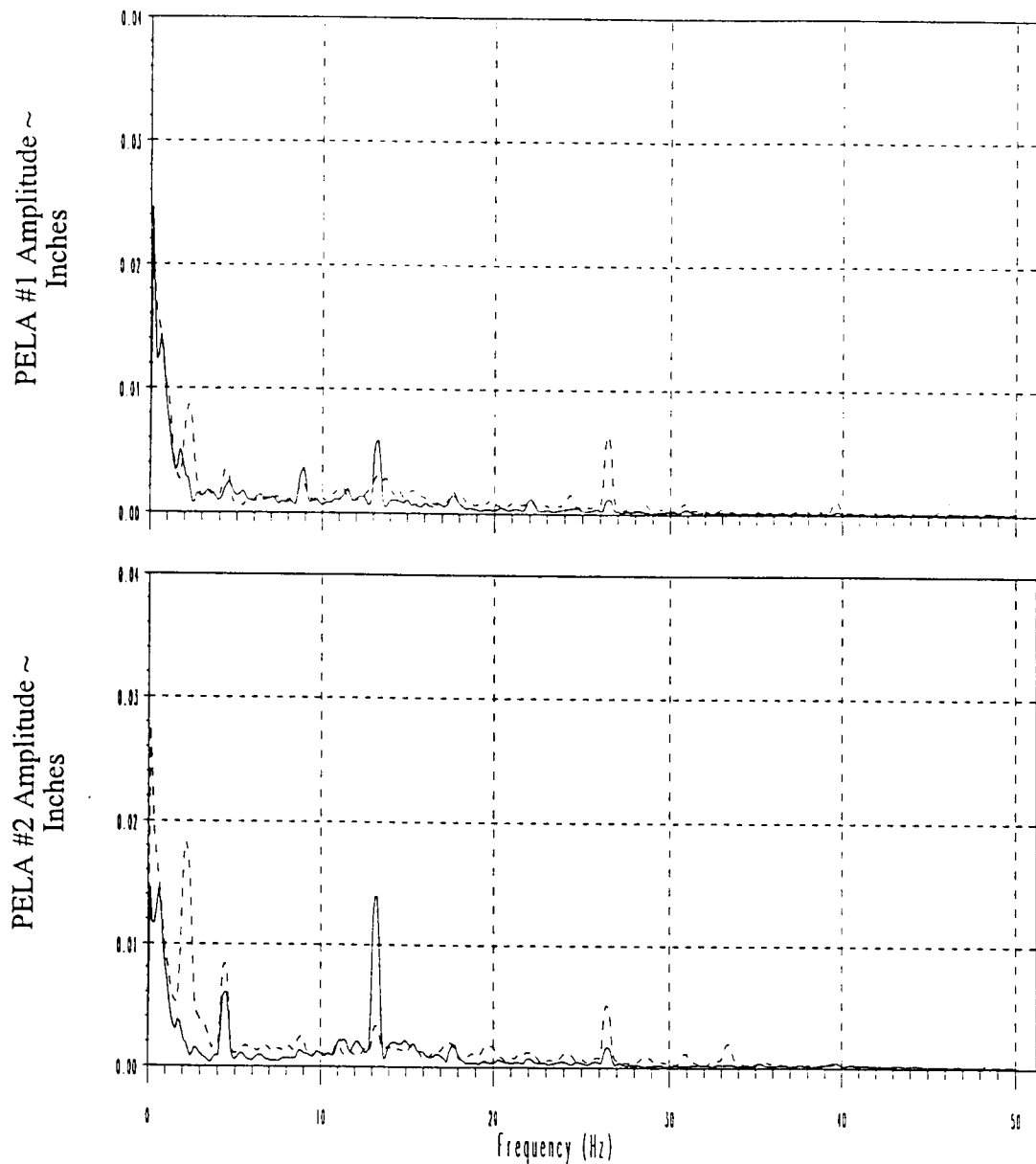


Figure C- 12
Maximum Level Flight Airspeed PELA Frequency Response,
AFCS Computer Comparison

MODEL CH-46E HELICOPTER

Symbol	Aircraft Buno	GW (lb)	C.G. (in)	Hp (ft)	OAT (°C)	Nr (%)	Airspeed (KIAS)	AFCS Configuration
—	157692	22,210	309.0.0	3,000	-3	100.0	131	AFCS-5
---	157692	19,480	301.6	3,000	-7	100.0	135	AFCS-5

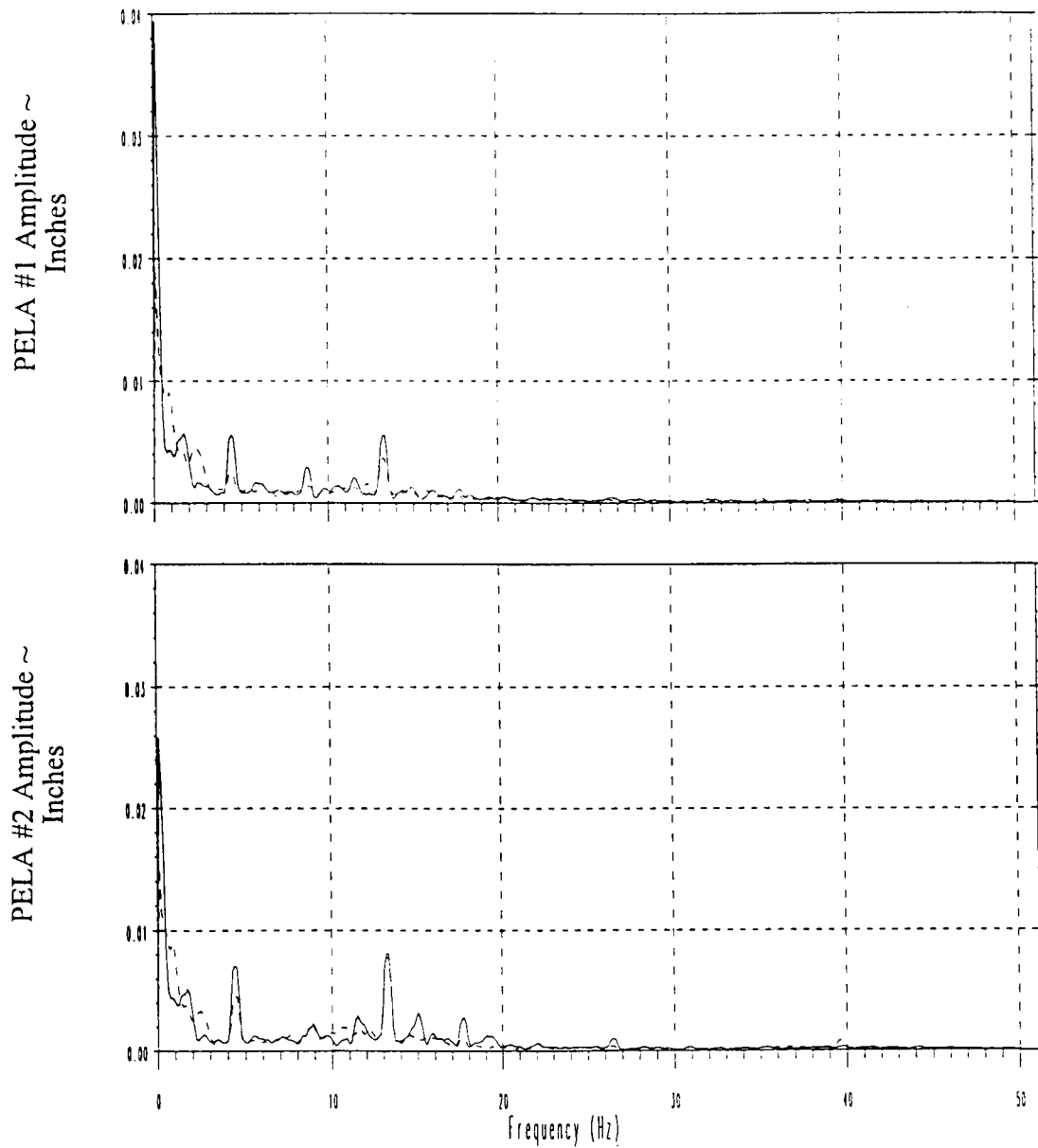


Figure C- 13
Maximum Level Flight PELA Frequency Response,
Aircraft Weight Comparison

MODEL CH-46E HELICOPTER

Aircraft Buno	GW (lb)	C.G. (in)	Hp / AGL (ft)	OAT (°C)	Nr (%)	Wind (KT)	AFCS Configuration
157692	19,370	301.1	240 / 100	2	100.0	15	AFCS-4

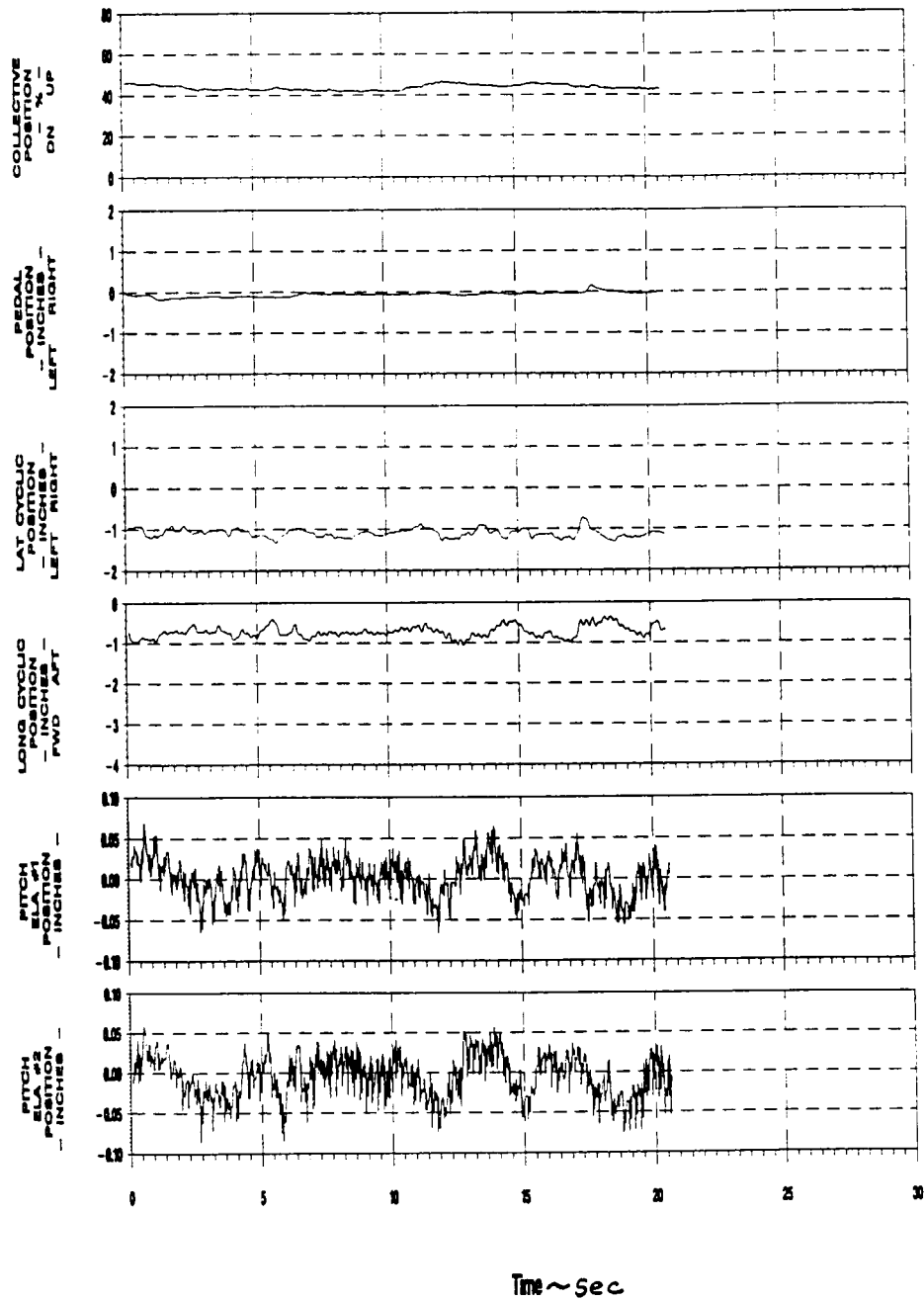


Figure C- 14
100-ft Out-of-Ground Effect Hover , AFCS-4 Computers

MODEL CH-46E HELICOPTER

Aircraft Buno	GW (lb)	C.G. (in)	Hp / AGL (ft)	OAT (°C)	Nr (%)	Wind (KT)	AFCS Configuration
157692	20,280	304.4	-200 / 100	-2	100.0	4	AFCS-5

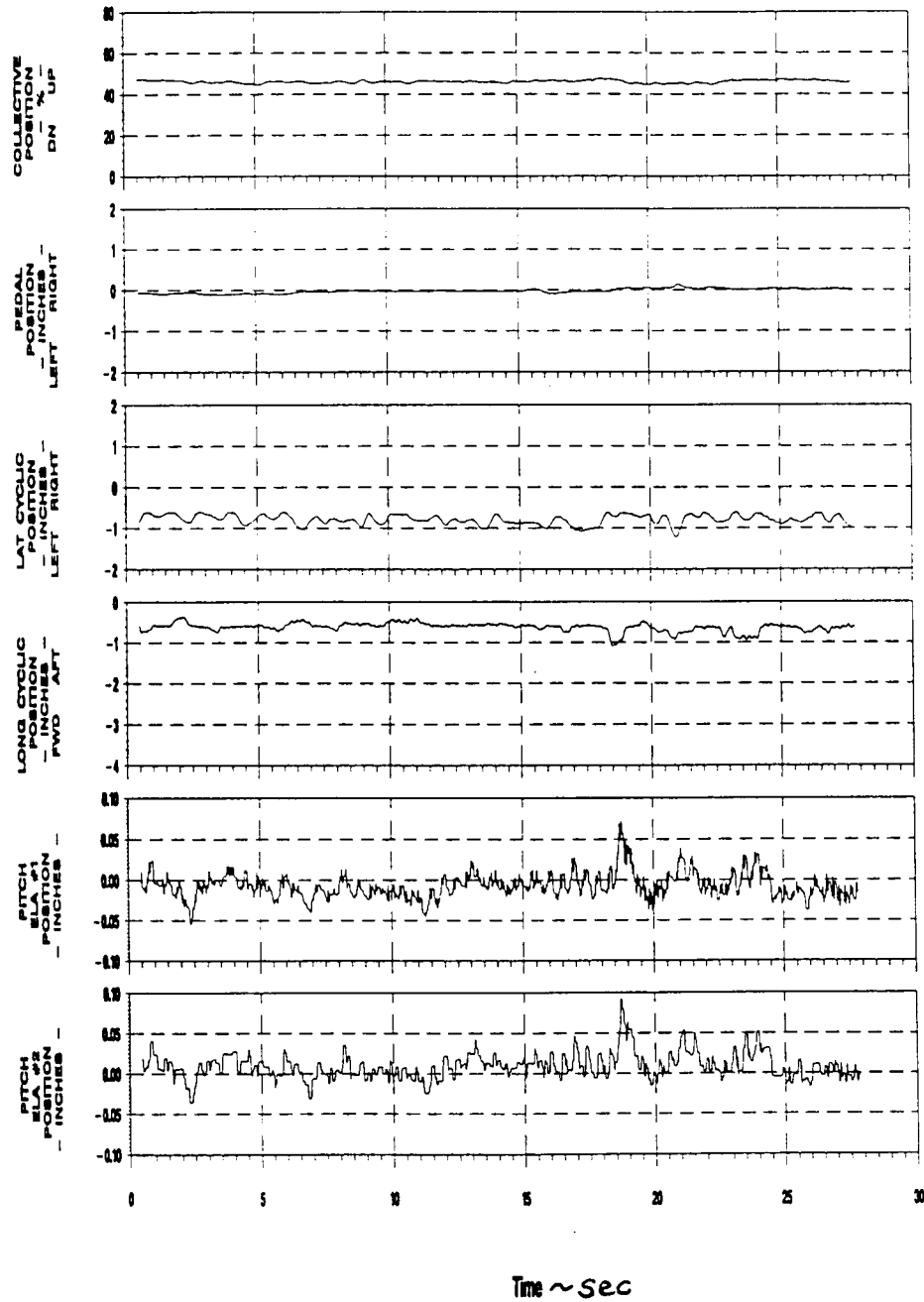


Figure C- 15
100-ft Out-of-Ground Effect Hover , AFCS-5 Computers

MODEL CH-46E HELICOPTER

Symbol	Aircraft Buno	GW (lb)	C.G. (in)	Hp / AGL (ft)	OAT (°C)	Nr (%)	Wind (KT)	AFCS Configuration
—	157692	20,280	304.4	-200 / 100	-2	100.0	4	AFCS-5
- - -	157692	19,370	301.1	240 / 100	2	100.0	15	AFCS-4

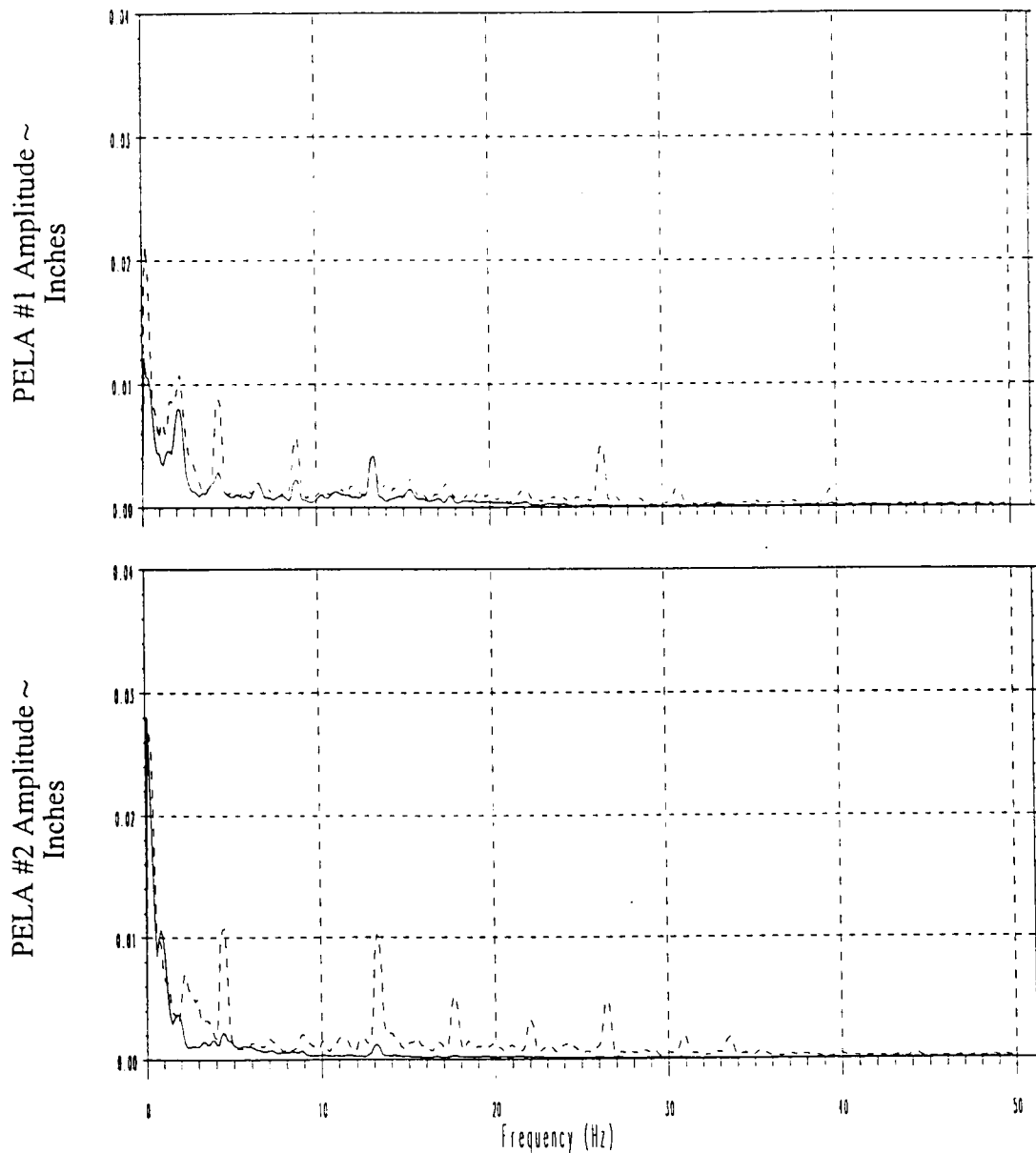


Figure C- 16
100-ft Out-of-Ground Effect Hover, AFCS Computer Comparison

MODEL CH-46E HELICOPTER

Symbol	Aircraft Buno	GW (lb)	C.G. (in)	Hp / AGL (ft)	OAT (°C)	Nr (%)	Wind (KT)	AFCS Configuration
—	157692	19,935	302.9	460 / 500	3	100.0	3	AFCS-5
---	157692	19,735	302.3	380 / 500	12	100.0	3	AFCS-4

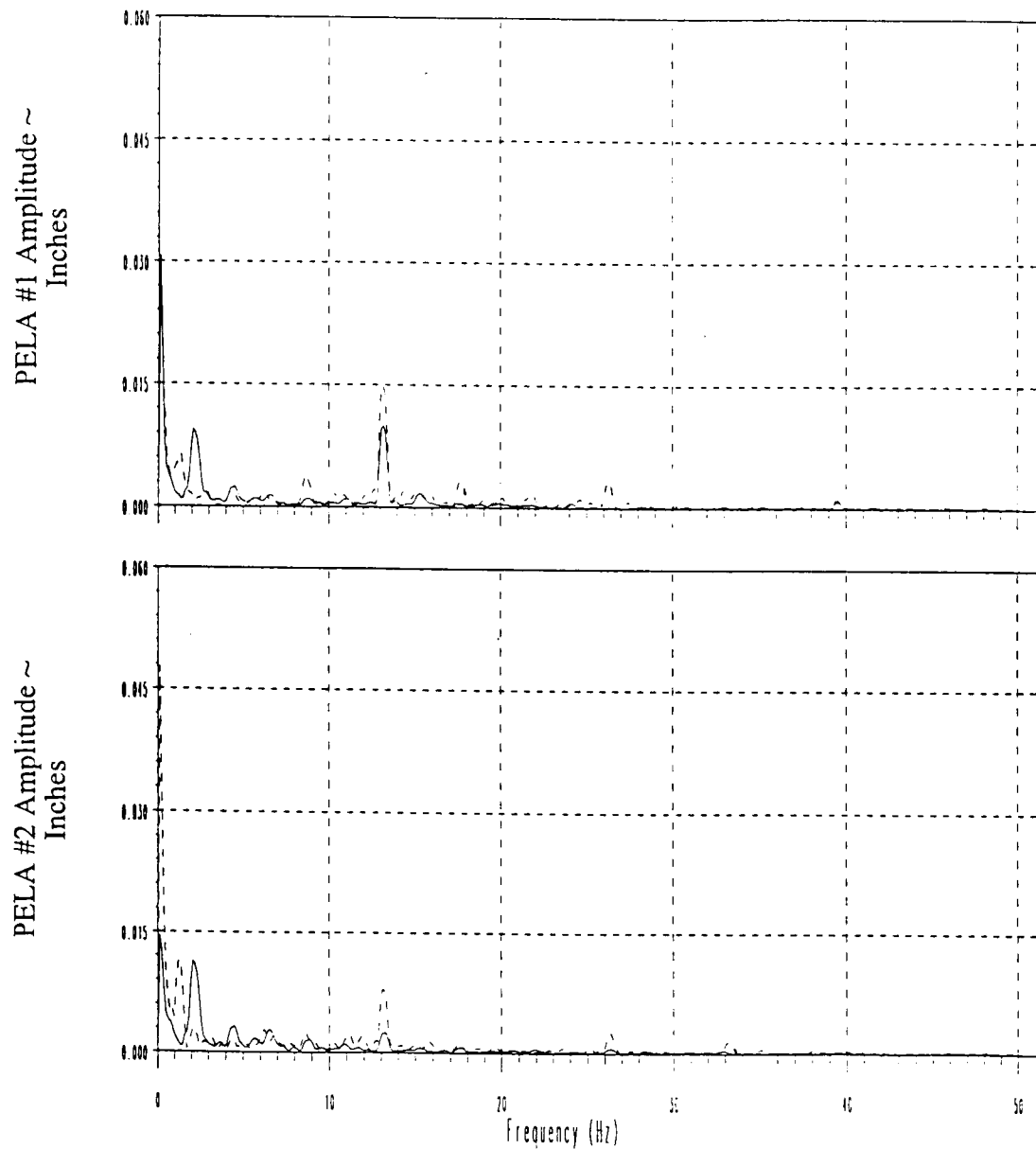


Figure C- 17
500-ft Out-of-Ground Effect Hover, AFCS Computer Comparison

MODEL CH-46E HELICOPTER

Symbol	Aircraft Buno	GW (lb)	C.G. (in)	Hp / AGL (ft)	OAT (°C)	Nr (%)	Wind (KT)	AFCS Configuration
—	157692	21,120	306.4	-250 / 100	5	100.0	9	AFCS-5
---	157692	20,280	304.4	-200 / 100	-2	100.0	4	AFCS-5

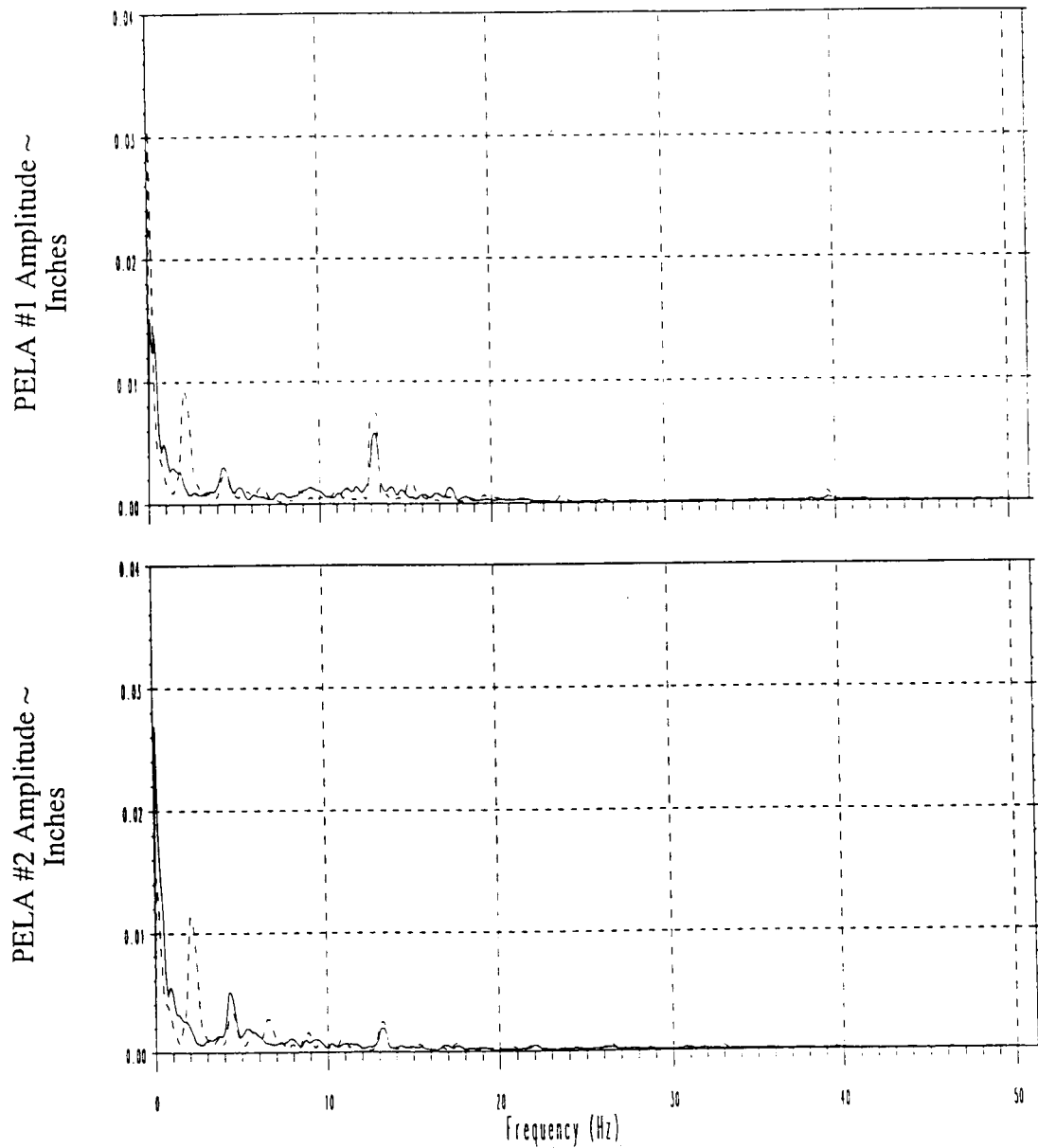


Figure C- 18
Out-of-Ground Effect Hover, Aircraft Weight Comparison

MODEL CH-46E HELICOPTER

Aircraft Buno	GW (lb)	C.G. (in)	Hp / AGL (ft)	OAT (°C)	Nr (%)	Wind (KT)	AFCS Configuration
157692	20,785	305.4	-80 / 30	5	100.0	6	AFCS-4

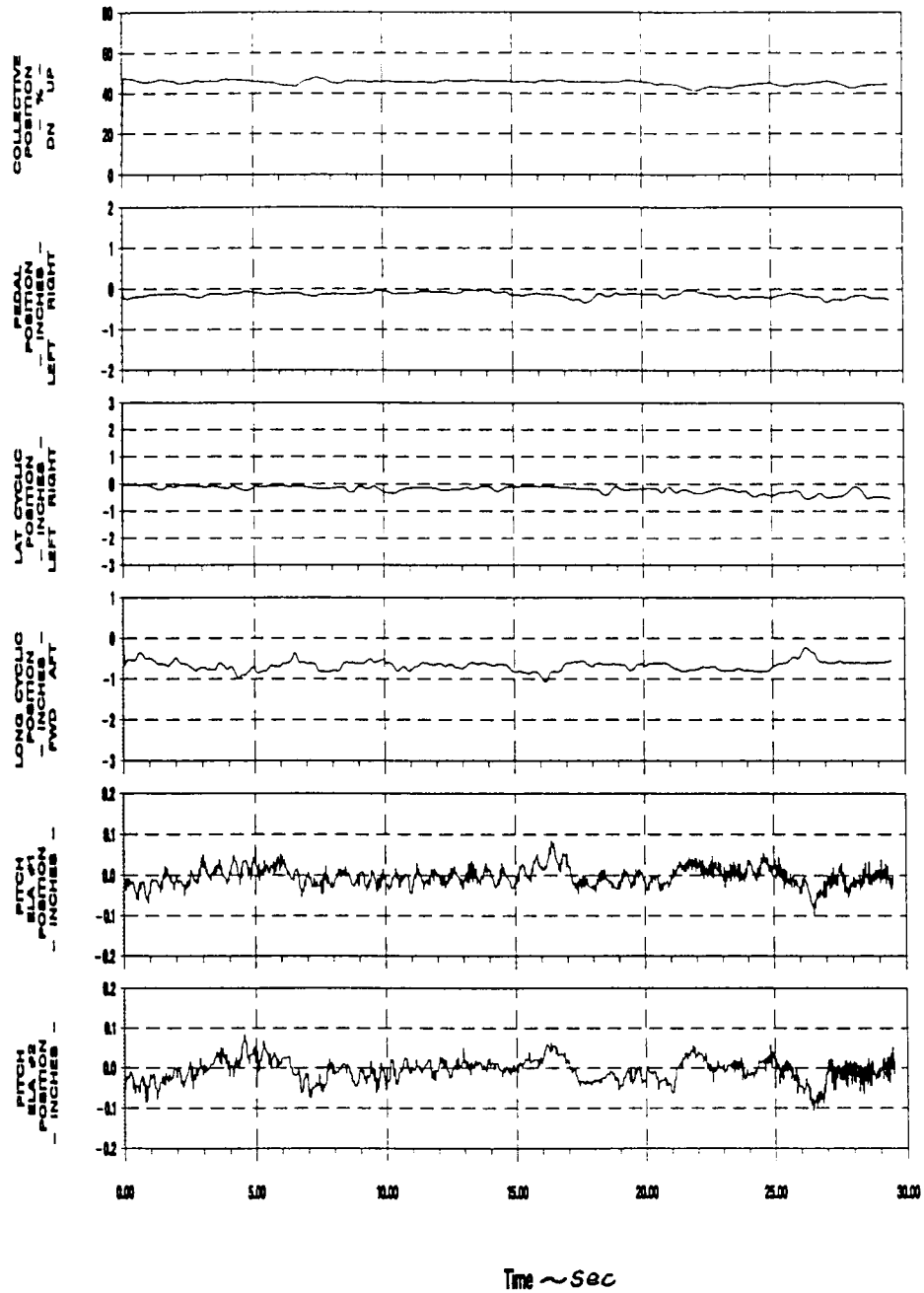


Figure C- 19
In-Ground Effect Hover, AFCS-4 Computers

MODEL CH-46E HELICOPTER

Aircraft Buno	GW (lb)	C.G. (in)	Hp / AGL (ft)	OAT (°C)	Nr (%)	Wind (KT)	AFCS Configuration
157692	21,120	306.4	-350 / 30	6	100.0	9	AFCS-5

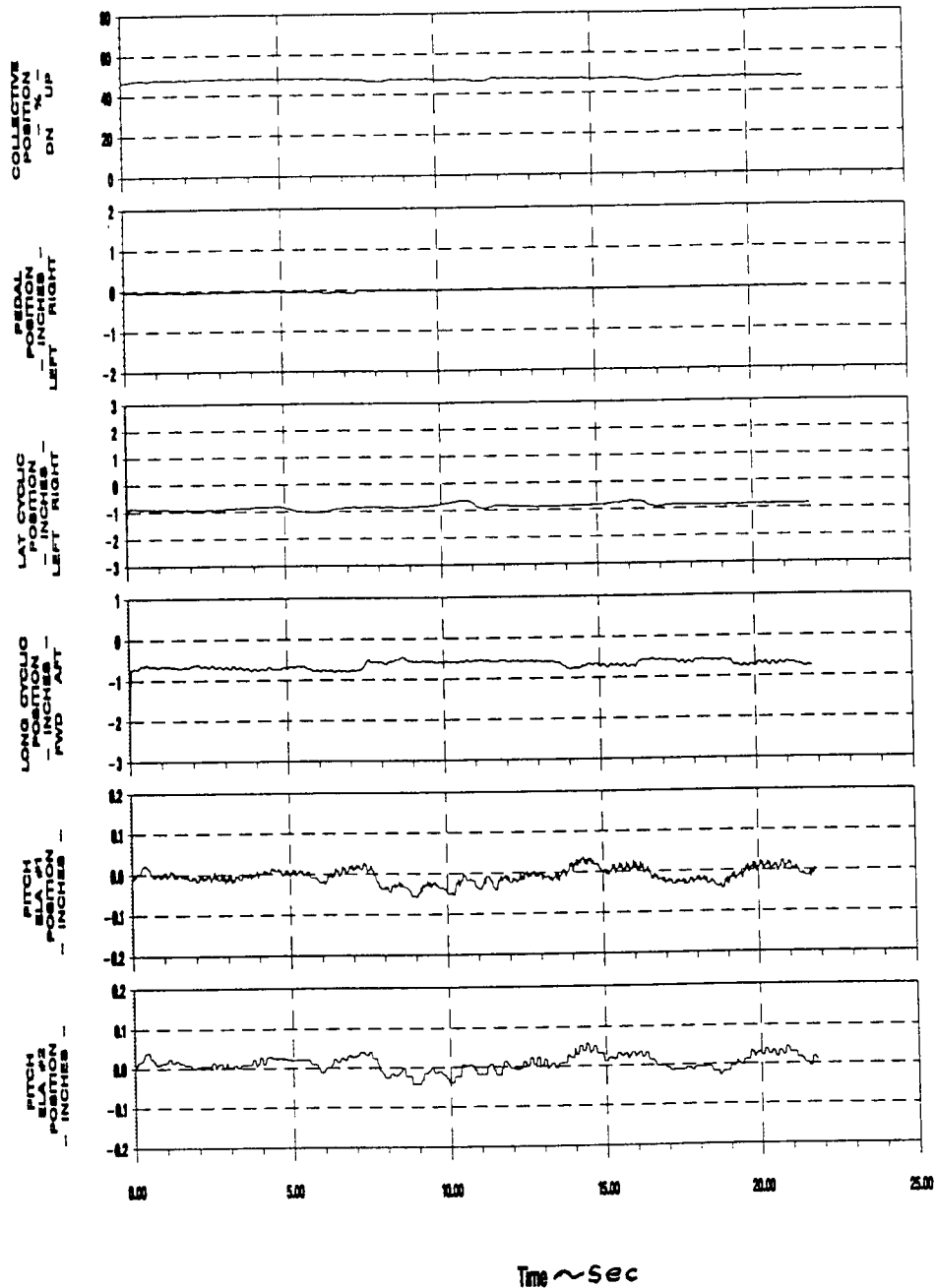


Figure C- 20
In-Ground Effect Hover, AFCS-5 Computers

MODEL CH-46E HELICOPTER

Symbol	Aircraft Buno	GW (lb)	C.G. (in)	Hp / AGL (ft)	OAT (°C)	Nr (%)	Wind (KT)	AFCS Configuration
—	157692	21,120	306.4	-350 / 30	6	100.0	9	AFCS-5
---	157692	20,785	305.4	-80 / 30	5	100.0	6	AFCS-4

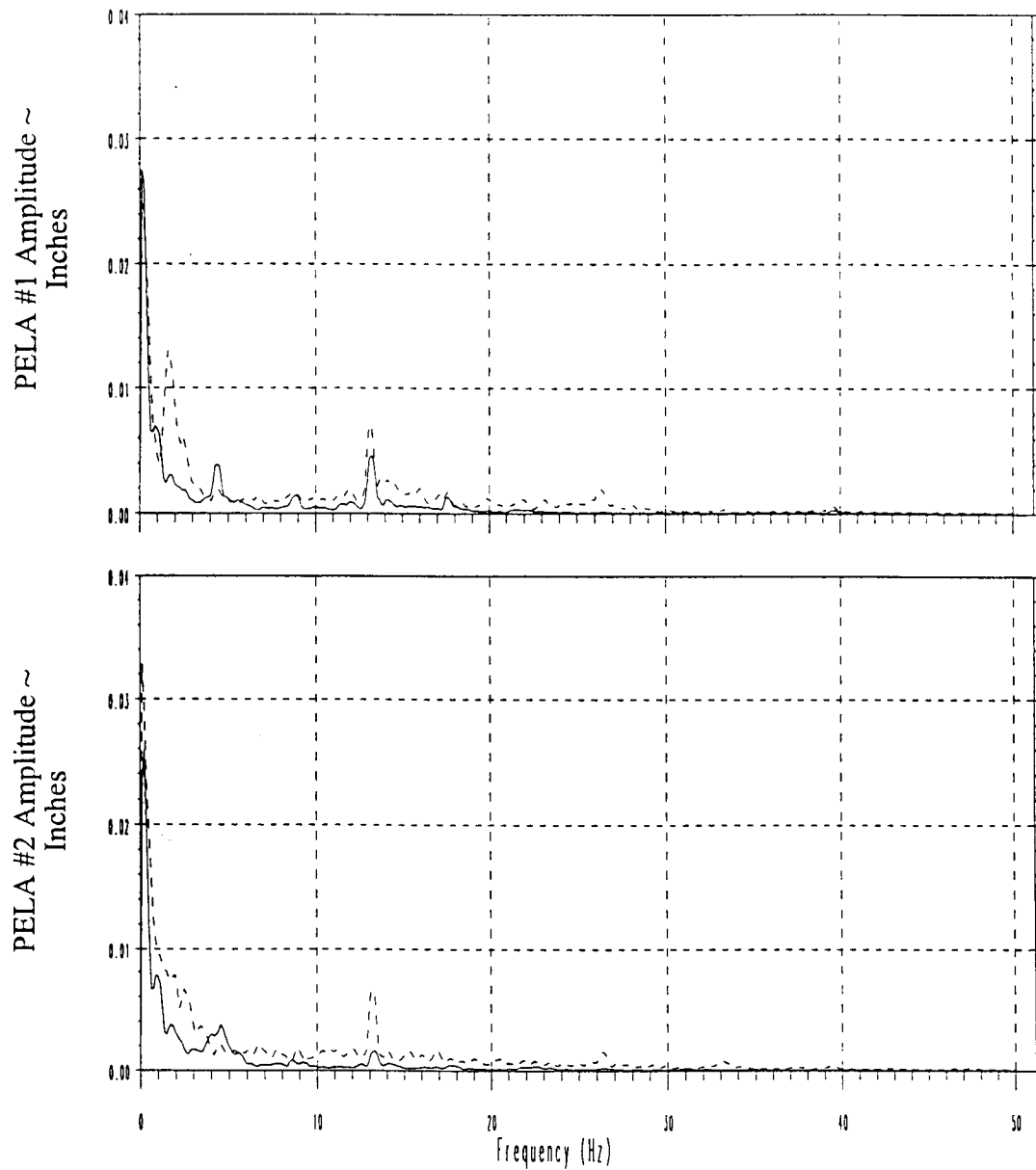


Figure C- 21
In-Ground Effect Hover, AFCS Computer Comparison

MODEL CH-46E HELICOPTER

Aircraft Buno	GW (lb)	C.G. (in)	Hp / AGL (ft)	OAT (°C)	Nr (%)	Wind (KT)	AFCS Configuration
157692	19,370	301.1	240 / 100	2	100.0	15	AFCS-4

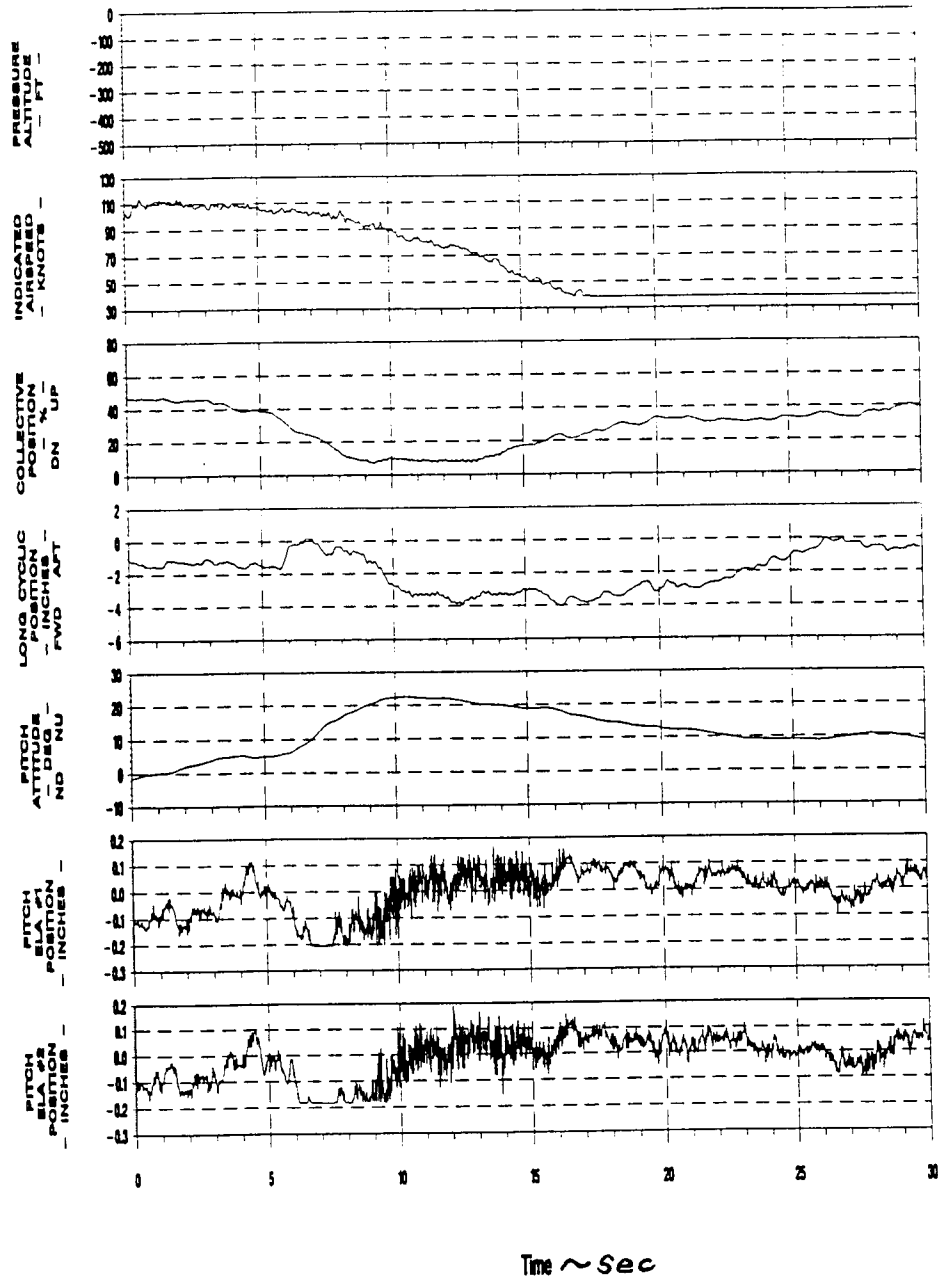


Figure C- 22
Quickstop, AFCS-4 Computers

MODEL CH-46E HELICOPTER

Aircraft Buno	GW (lb)	C.G. (in)	Hp / AGL (ft)	OAT (°C)	Nr (%)	Wind (KT)	AFCS Configuration
157692	20,280	304.4	-200 / 100	-2	100.0	4	AFCS-5

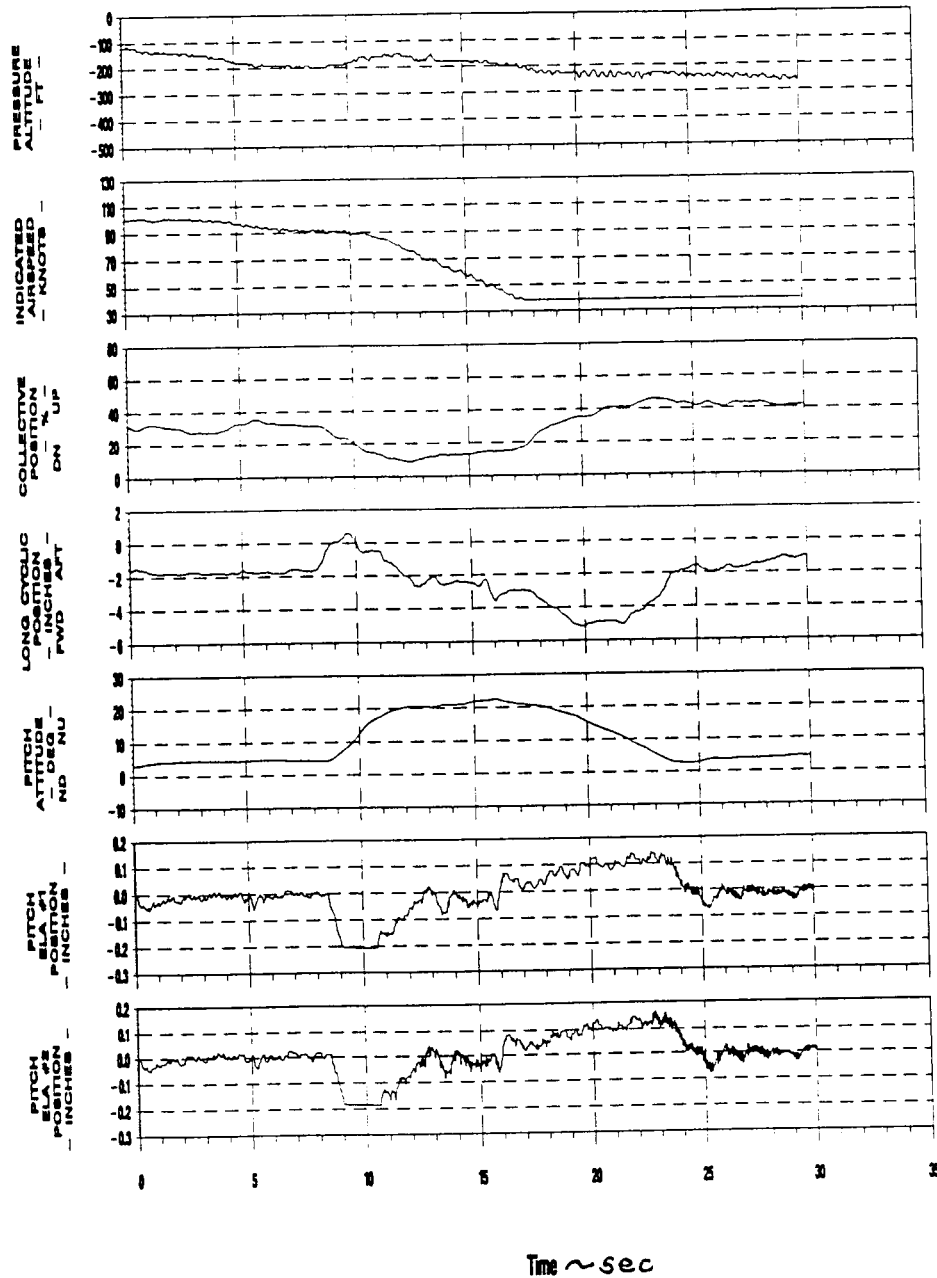


Figure C- 23
Quickstop, AFCS-5 Computers

MODEL CH-46E HELICOPTER

Symbol	Aircraft Buno	GW (lb)	C.G. (in)	Hp / AGL (ft)	OAT (°C)	Nr (%)	Wind (KT)	AFCS Configuration
—	157692	20,280	304.4	-200 / 100	-2	100.0	4	AFCS-5
---	157692	19,370	301.1	240 / 100	2	100.0	15	AFCS-4

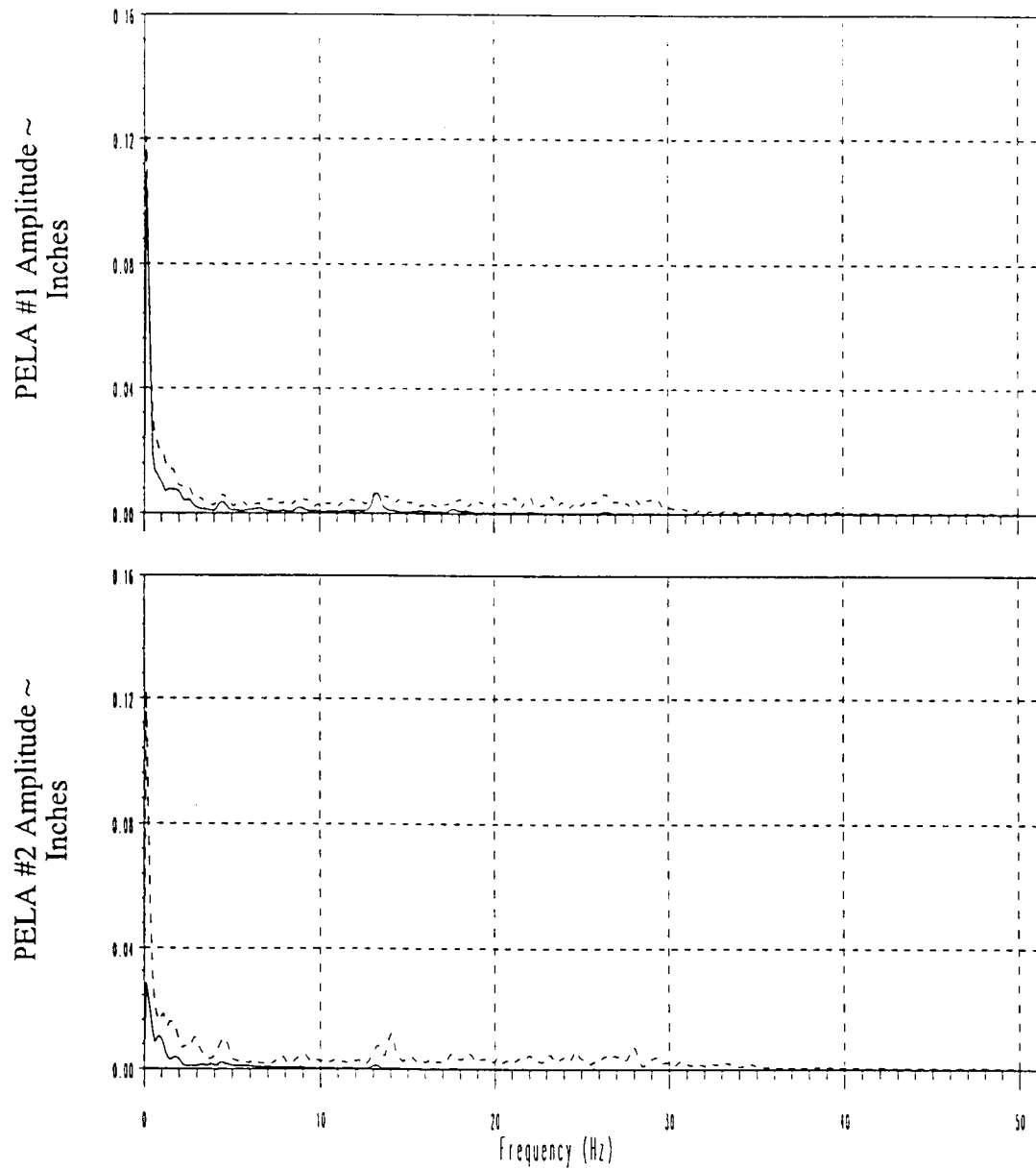


Figure C- 24
Quickstop, AFCS Computer Comparison

MODEL CH-46E HELICOPTER

Aircraft Buno	GW (lb)	C.G. (in)	Hp / AGL (ft)	OAT (°C)	Nr (%)	Wind (KT)	AFCS Configuration
157692	19,490	301.5	475 / 300	2	100.0	13	AFCS-4

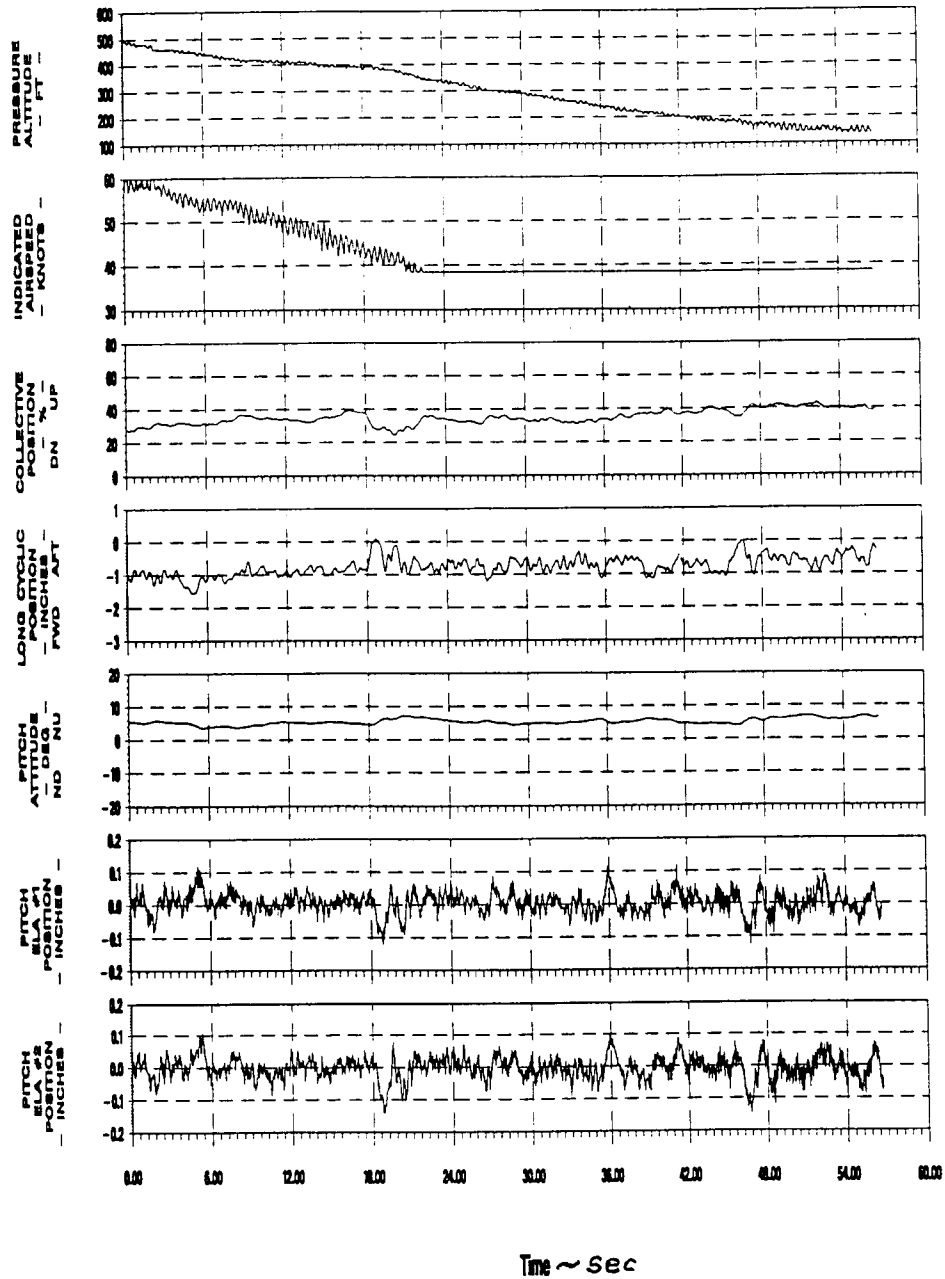


Figure C- 25
Steep Approach, AFCS-4 Computers

MODEL CH-46E HELICOPTER

Aircraft Buno	GW (lb)	C.G. (in)	Hp / AGL (ft)	OAT (°C)	Nr (%)	Wind (KT)	AFCS Configuration
157692	20,030	303.3	0 / 300	-2	100.0	3	AFCS-5

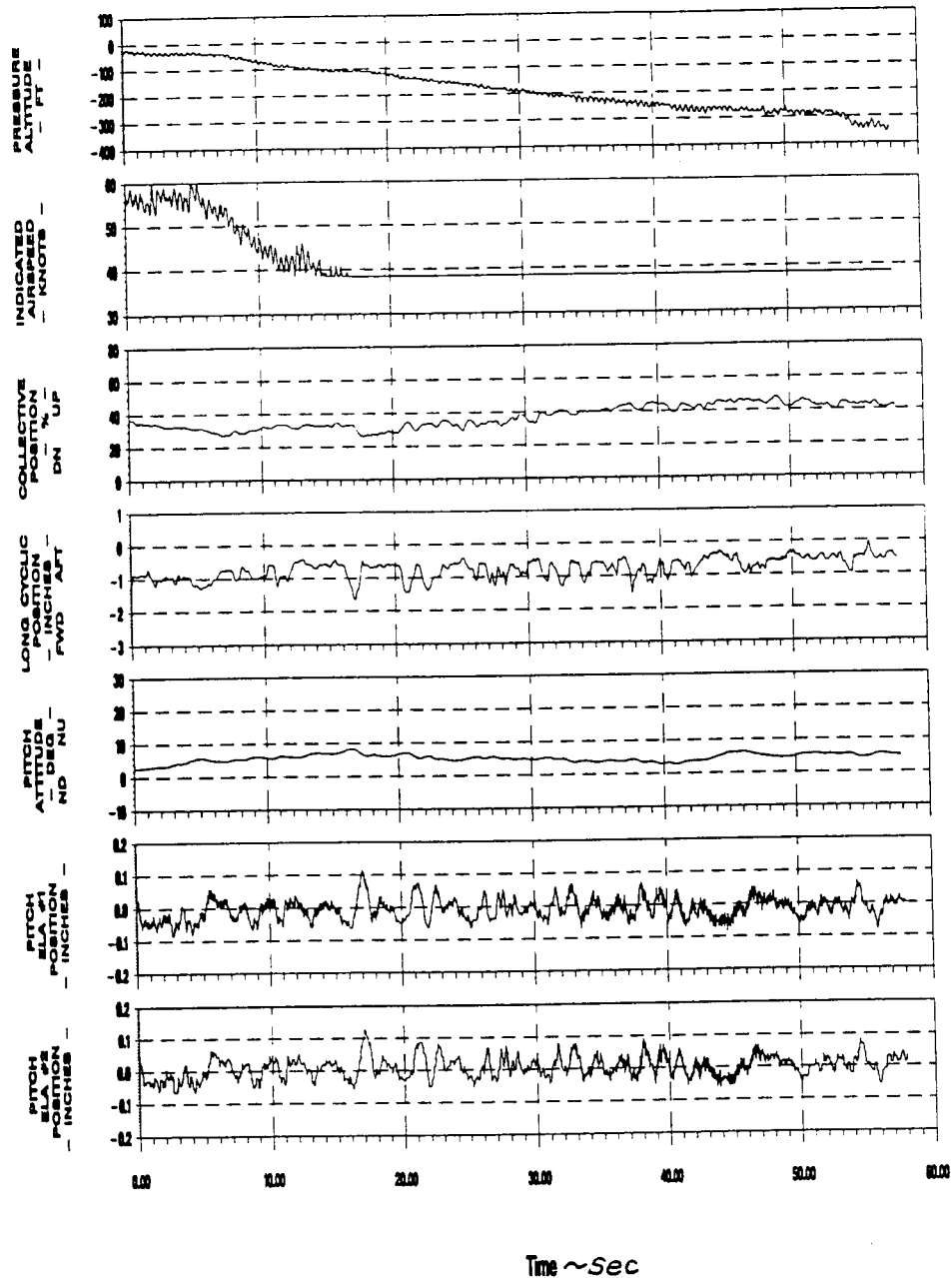


Figure C- 26
Steep Approach, AFCS-5 Computers

MODEL CH-46E HELICOPTER

Symbol	Aircraft Buno	GW (lb)	C.G. (in)	Hp / AGL (ft)	OAT (°C)	Nr (%)	Wind (KT)	AFCS Configuration
—	157692	20,030	303.3	0 / 300	-2	100.0	3	AFCS-5
---	157692	19,490	301.5	475 / 300	2	100.0	13	AFCS-4

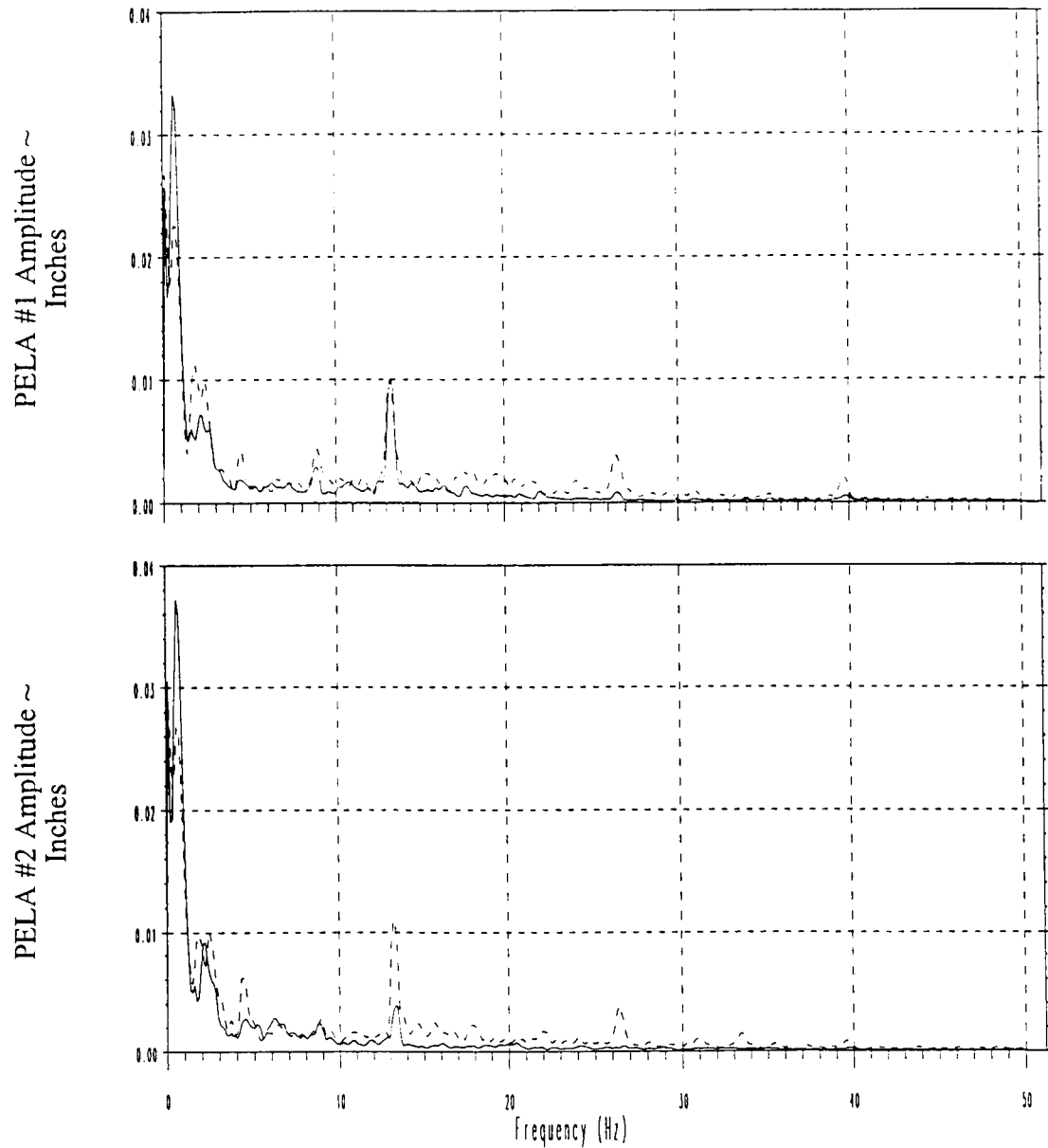


Figure C- 27
Steep Approach, AFCS Computer Comparison

MODEL CH-46E HELICOPTER

Symbol	Aircraft Buno	GW (lb)	C.G. (in)	Hp (ft)	OAT (°C)	Nr (%)	Airspeed (KIAS)	AFCS Configuration
□	153355	20,775	302.9	3,200	-3	100.5	106	AFCS-4 w/ Rate Gyro
○	157692	20,780	305.5	3,000	5	100.0	95	AFCS-5
△	157692	20,690	305.1	3,000	-6	100.0	100	AFCS-4

Shaded points denote trimmed flight condition.

Total longitudinal control travel is 12 inches.

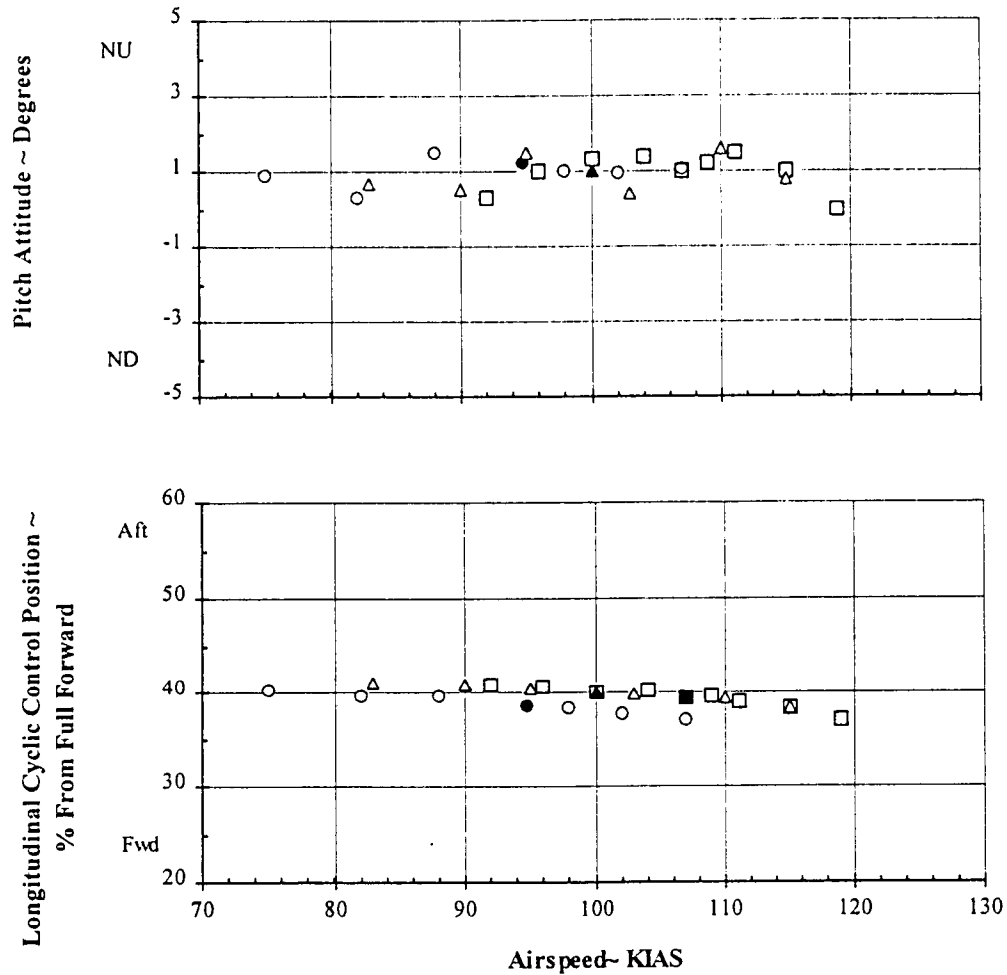


Figure C- 28
Static Longitudinal Stability

MODEL CH-46E HELICOPTER

Symbol	Aircraft Buno	GW (lb)	C.G. (in)	Hp (ft)	OAT (°C)	Nr (%)	Airspeed (KIAS)	AFCS Configuration
Δ	157692	20,130	303.6	3,000	5	100.0	100	AFCS-5
O	157692	20,240	303.8	3,000	-7	100.0	100	AFCS-4

Method: Constant collective turns.

Full longitudinal control travel is 12 inches.

Note: Nz not available.

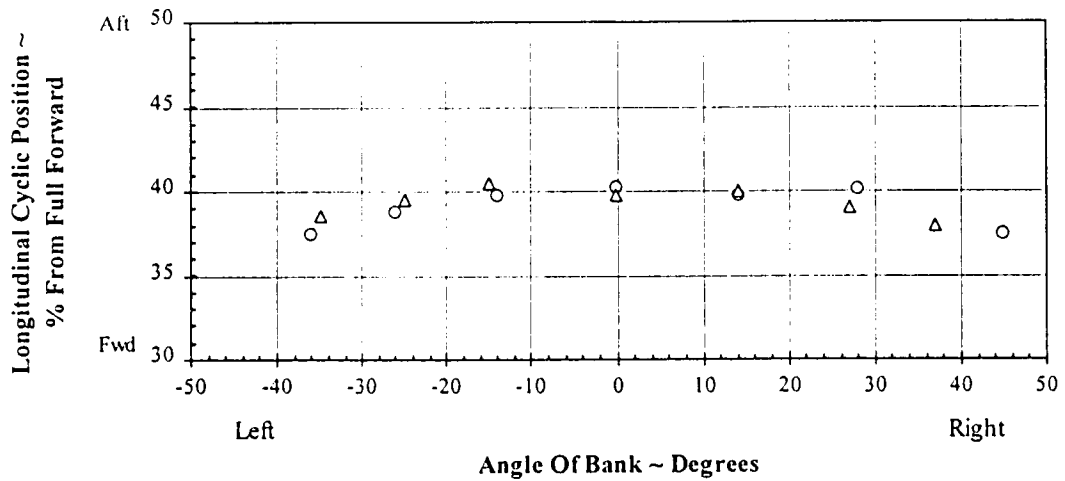


Figure C- 29
Maneuver Stability

MODEL CH-46E HELICOPTER

Symbol	Aircraft Buno	GW (lb)	C.G. (in)	Hp (ft)	OAT (°C)	Nr (%)	Airspeed (KIAS)	AFCS Configuration
□	153355	20,775	302.9	3,200	-3	99.7	102	AFCS-4 w/ Rate Gyro
Δ	153355	18,330	279.8	3,000	11	100.0	100	AFCS-4
○	157692	20,380	304.3	3,000	12	100.0	100	AFCS-5

Full longitudinal control travel is 12 inches.

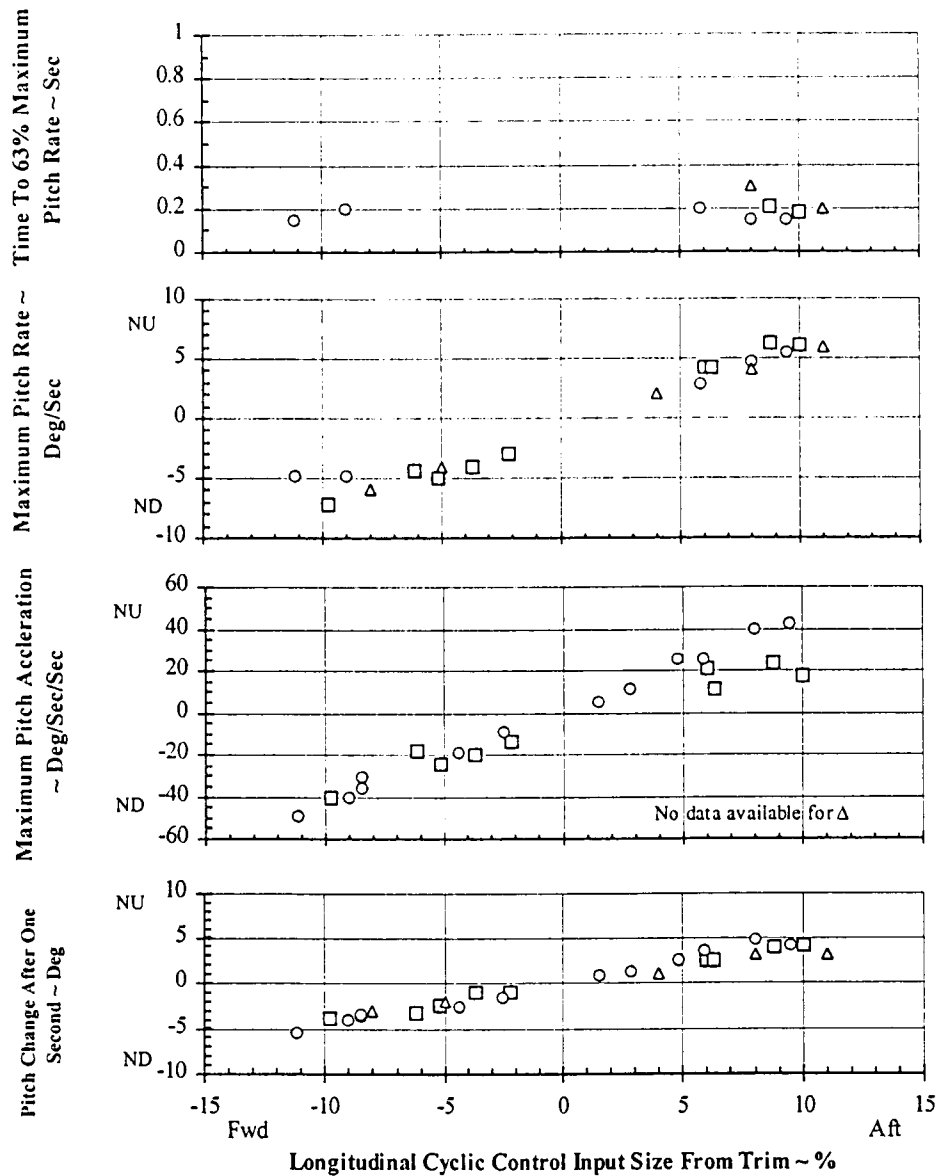


Figure C- 30
Longitudinal Control Response, Forward Flight

MODEL CH-46E HELICOPTER

Symbol	Aircraft Buno	GW (lb)	C.G. (in)	Hp (ft)	OAT (°C)	Nr (%)	Airspeed (KIAS)	AFCS Configuration
Δ	157692	19,800	302.5	3,000	-7	100.0	100	AFCS-4
O	157692	20,080	303.5	3,000	12	100.0	100	AFCS-5

Full lateral control travel is 9.2 inches.

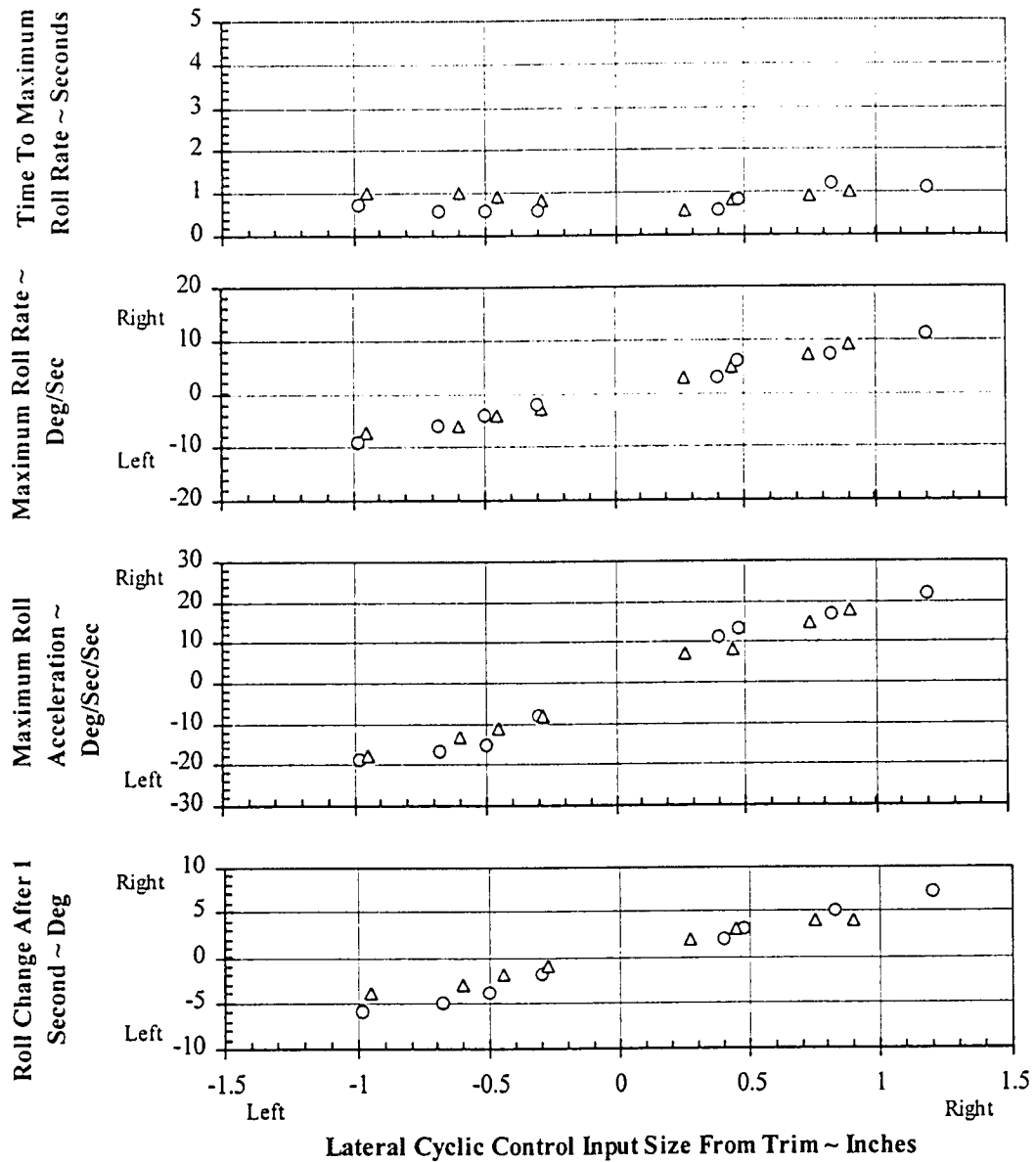


Figure C- 31
Lateral Control Response, Forward Flight

MODEL CH-46E HELICOPTER

Symbol	Aircraft Buno	GW (lb)	C.G. (in)	Hp (ft)	OAT (°C)	Nr (%)	Wind (KT)	AFCS Configuration
Δ	157692	19,840	302.6	380	12	100.0	3	AFCS-4
O	157692	20,090	303.4	460	3	100.0	3	AFCS-5

Full lateral control travel is 9.2 inches.

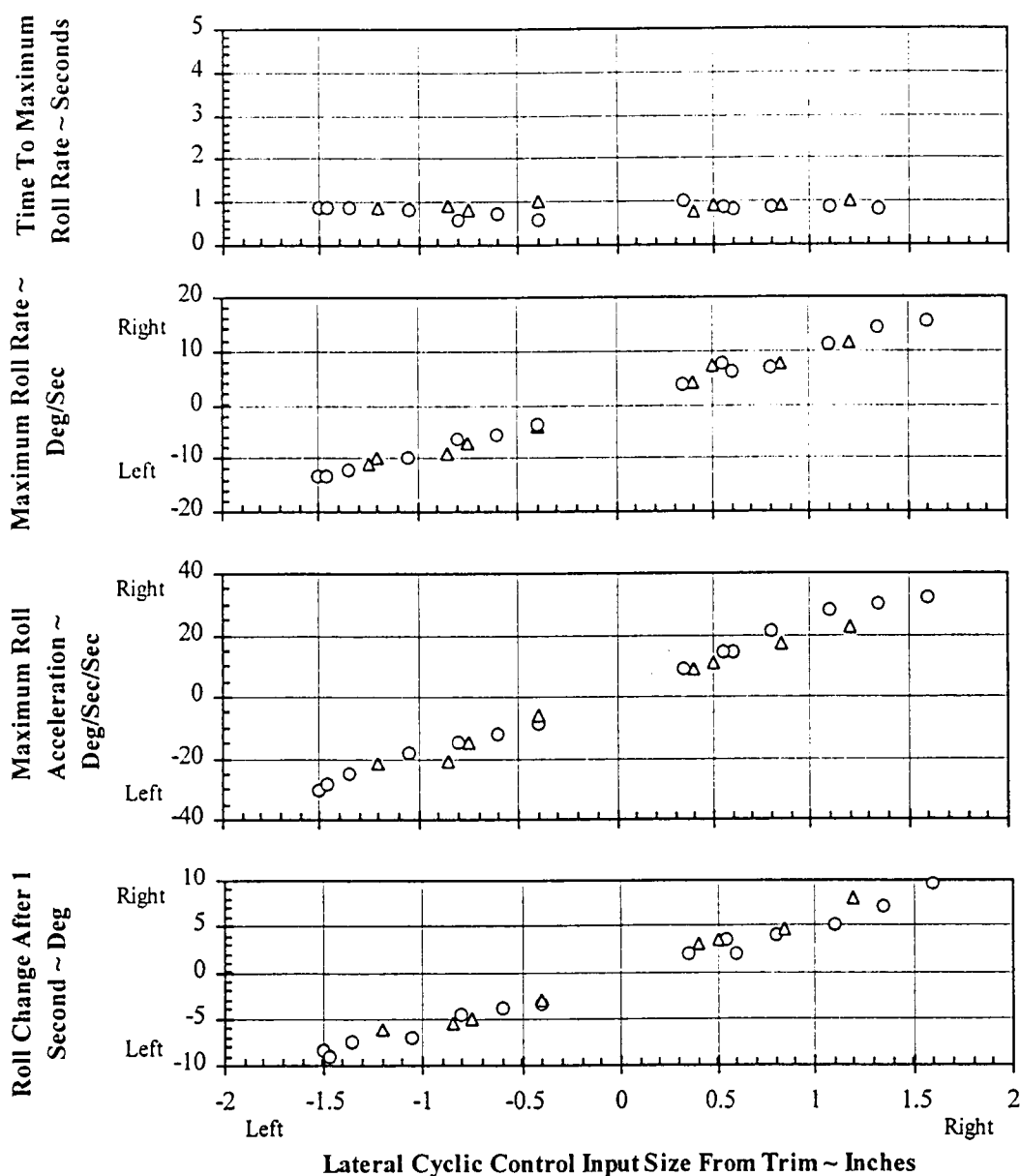


Figure C- 32
Lateral Control Response, Out-of-Ground Effect Hover

VITA

Eric Treworgy was born in Burlington, Vermont on February 17, 1963. He attended schools in the public school system of Rockville, Maryland where he graduated from Robert E. Peary High School in June 1981. He entered West Virginia University in August of 1981 where in August 1985 he received the Bachelor of Science in Aerospace Engineering. Upon graduation, he was commissioned a Second Lieutenant in the United States Marine Corps and attended The Basic School in Quantico, Virginia in April of 1986.

He entered primary flight training in Pensacola, Florida in October of 1986 where he was designated a naval aviator in December 1987. After becoming qualified in the CH-46E helicopter, he was assigned to Marine Corps Air Station New River, North Carolina. There he was sent on several overseas deployments that included Operations Desert Shield and Desert Storm. He flew combat missions out of Jubail, Saudi Arabia and from the expeditionary field Lonesome Dove during the air and ground campaigns. After the Gulf War, he attended the United States Naval Test Pilot School in Patuxent River, Maryland as a member of Class 107 where he graduated from the rotary wing curriculum in June of 1995.

Upon graduation from Test Pilot School, he was assigned to the Naval Rotary Wing Aircraft Test Squadron in Patuxent River, Maryland where he was the H-46 Platform Coordinator responsible for all developmental testing of H-46 model helicopters. In December of 1996, he was reassigned to the United States Naval Test

Pilot School where he is currently an instructor in the rotary wing curriculum. While stationed in Patuxent River, Maryland he entered The University Of Tennessee, Knoxville via the off-campus program to pursue the Master of Science in Aviation Systems. The master degree was received June 1998.

He is married to the former Serena Roh of Rockville, MD. They have two children, Tyler and Alicia.