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Preliminary Design Study

of a Higher Harmonic Blade

Feathering Control System

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PRELIMINARY DESIGN STUDY
OF A HIGHER HARMONIC
BLADE FEATHERING CONTROL SYSTEM

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PRELIMINARY DESIGN STUDY OF A HIGHER HARMONIC
BLADE FEATHERING CONTROL SYSTEM

By Richard W. Powers

SUMMARY

The objective of the work performed was to demonstrate the design feasibility of incorporating an active higher harmonic control (HHC) system on an OH-6A rotorcraft. The introduction of continuously-modulated low amplitude 4P feathering has shown substantial potential for reducing rotor-transmitted oscillatory loads. The design implementation of such a system on a baseline OH-6A required generation of a hydraulic power system, control actuator placement and design integration of a electronic subsystem comprised of an electronic control unit (ECU) and digital microcomputer. Various placements of the HHC actuators in the primary control system were evaluated. Assembly drawings of the actuator concepts and control rigging were prepared and are presented. This study confirms the advantages of generating both hydraulic power and 4P control motions in the nonrotating system.

INTRODUCTION

Over the past several years, the Structures Laboratory, U. S. Army Research and Technology Laboratory (AVRADCOM) has undertaken a comprehensive program of evaluating active controls for achieving lowered airframe vibration levels on rotorcraft. A candidate approach currently being explored by NASA/Langley Research Center in conjunction with the Structures Laboratory is higher harmonic rotor blade pitch control. Spurred by recent developments in digital and microcircuit technology, higher harmonic control (HHC) utilizes continuously-modulated, low-amplitude high frequency blade feathering to alter the aerodynamic rotor loads before being transmitted to the airframe. Thus, harmonic control cannot be attributed with any damping or absorption capabilities, as in References 1 and 2; rather, modulation of high frequency components of rotor downwash influence blade motion and the modified motions and airloads together control harmonic blade root shear forces transmitted from the hub to the fuselage.

Desired harmonic blade feathering is imparted to the rotor by oscillating the stationary swashplate collectively, and in pitch and roll. Considering a four-bladed helicopter rotor, the frequency of such oscillations is the fourth harmonic of rotor rotational frequency, denoted 4P, nominally 32 Hz for the Army/Hughes OH-6A. Controlled 4P accelerations in the aircraft are monitored and fed back through a shaping network, thereby deriving 4P swashplate oscillations required to regulate hub accelerations.

Modern evolution of higher harmonic pitch control began with an open-loop flight demonstration aimed at minimizing fuselage vibrations and expanding the helicopter's load-limited flight envelope, Reference 3. Although the investigation achieved limited success, the program did demonstrate that a reduction in vertical vibration can be achieved through proper application of harmonic feathering. Subsequent work by researchers, Reference 4 through 6, analytically confirmed the ability of HHC to reduce rotor-transmitted accelerations.

Although analytical treatments of higher harmonic control have established its feasibility, much of what is presently known about the technique has evolved through wind tunnel test programs, reported on in Reference 7 through 11. Experience gained from such investigations has led researchers to conclude:

- Blade pitch requirements for vibration alleviation vary as a function of advance ratio; generally less than a degree for $0.2 \leq \mu \leq 0.35$.

- Periodic variation of primary controls for vibratory hub load reduction carries no attendant penalty in terms of blade flap bending or rotor performance.

- Simultaneous reduction of several vibratory hub load components is possible due to superposition of harmonic pitch effects.

Having functionally characterized the effects of HHC, recent attention, References 12 through 16, has turned to developing optimal control algorithms wherein each new response measurement leads to an updated estimate of the system transfer matrix, thereby permitting optimum inputs to be calculated and applied to the rotor control system. Continuous computer execution of the control algorithms maintains fuselage vibratory load reduction over the majority of helicopter flight regimes.

Apart from an earlier preliminary design study of mechanical and hydraulic second-harmonic pitch control systems, Reference 17, there exists no prior

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definition of design requirements for implementing a 4P higher harmonic pitch control system for vibration reduction on current rotorcraft. The objective of the current study, therefore, is to demonstrate the design feasibility of incorporating an active higher harmonic control system on an OH-6A rotorcraft, Figure 1. Performed under NASA contract NAS1-14552, the study represents an extension of work originally reported on in References 18 and 19 which explored control laws and design concepts. The current study defines and contrasts alternative servoactuator configurations and placements. In addition, preliminary design concepts for the electronic subsystem comprised of a flightworthy digital microcomputer and electronic control unit (ECU) are presented and discussed in terms of functional and fail-safety requirements.

1. INSTRUMENTED STATIC MAST
2. MINICOMPUTER SUBSYSTEM
3. HIGH FREQUENCY HHC ACTUATORS (3)
4. MINICOMPUTER
5. ECU
6. HYDRAULIC PUMP
7. COLLECTIVE PITCH CONTROL STICK
8. FLIGHT CONTROL SYSTEM (CYCLIC CONTROL)
9. MAIN GEARBOX
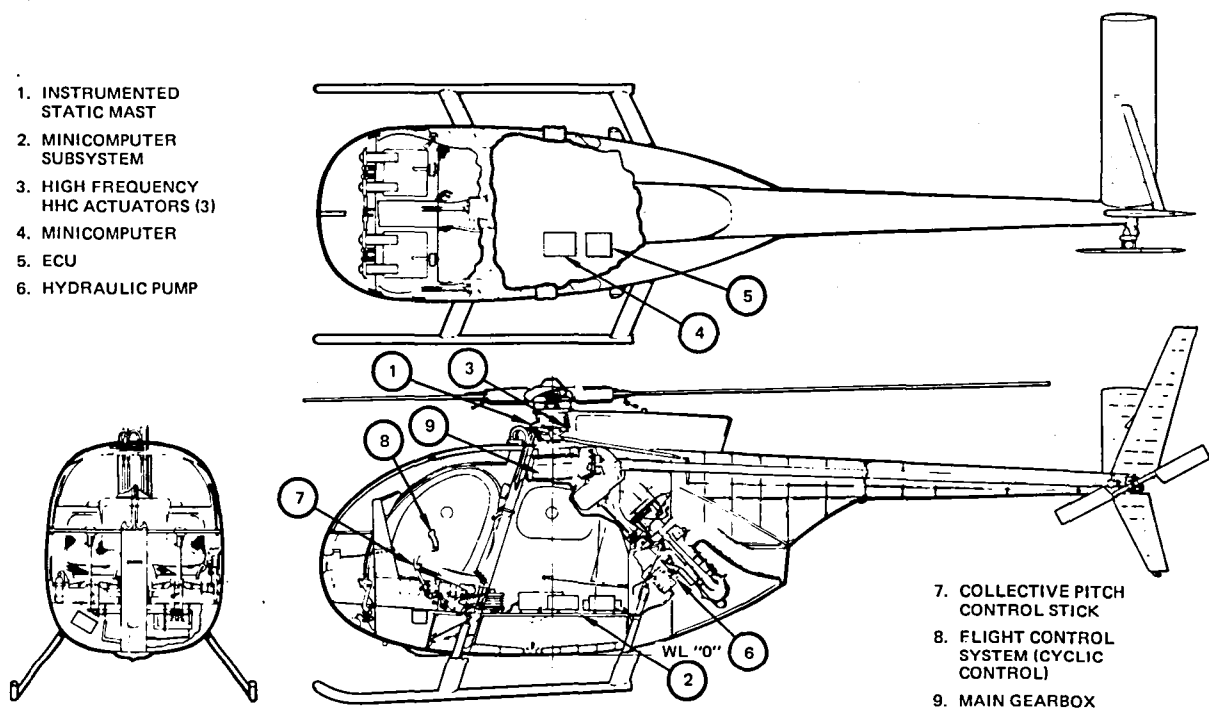
Figure 1. - OH-6A installation of active higher harmonic control system.

1.0 HIGHER HARMONIC CONTROL SYSTEM DESIGN REQUIREMENTS

1.1 General

For the four-bladed OH-6A helicopter, higher harmonic pitch control (HHC) as a means of vibration reduction requires superimposing 4P swashplate motion upon collective and cyclic control inputs required for aircraft trim. When external 4P control of the swashplate is present, third, fourth and fifth harmonic pitch variations may be obtained independently of the pilot's requirements for collective and cyclic pitch trim settings. Fourth harmonic (4P) blade root pitching can be induced by oscillating the swashplate vertically as a fourth harmonic about the collective trim setting. Third (3P) and fifth (5P) harmonic pitching can be imparted to the blade root by tilting the swashplate at 4P. Thus any phase and magnitude combination of blade root 3P and 5P pitch is possible by proper choice of amplitude and phase of fourth harmonic swashplate tilting about longitudinal and lateral axes.

In order to implement higher harmonic pitch control for vibration reduction in an active control network, there exist additional system requirements above and beyond the conventional OH-6A flight control system. A $2069 N/cm^2$ (3,000 psi) hydraulic subsystem comprised of servo-actuators, pump, reservoir and accumulator is specified for providing high-fidelity 4P blade pitching. A flightworthy digital computer is needed to meet the requirement of continuous vibratory load suppression over a variety of flight regimes. An electronic control unit (ECU) interfaces feedback transducers and servo-actuators with the microcomputer control functions. Control and system identification algorithms are required for computer solution of optimal 4P feathering phase and amplitude. Feedback acceleration transducers, optical encoder, control mixer modifications are additional incumbent needs of a higher harmonic control system.

Thus, as an element of preliminary design, system requirements and design criteria, as they relate to an active OH-6A vibration suppression system, will be identified and discussed in the paragraphs and sections that follow.

1.2 Hydraulic Subsystem Requirements

The standard OH-6A helicopter in the Army inventory relies on a purely mechanical primary flight control system. Thus, it is required to provide a complete hydraulic power and supply system capable of generating 4P swashplate oscillations in response to inputs from the electronic control unit.

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Preliminary design actuator concepts and hydraulic power subsystem design parameters are presented in Section 3.0 HIGHER HARMONIC CONTROL SYSTEM MECHANICAL DESIGN. A preliminary design specification for an HHC servo-actuator is given in Appendix B. For the proposed harmonic control system, a single hydraulic control system is utilized. All actuator configurations outlined in Section 3.0 revert to basic mechanical primary control under loss of hydraulic pressure, thus, precluding the need for a fully duplicated control system.

1.3 Electronic Subsystem Requirements

1.3.1 Electronic Control Unit

The electronic subsystem, comprised of an electronic control unit (ECU) and digital microcomputer, is required to provide three separate 4P sinusoidal drive currents of optimal phase and amplitude to each of three electrohydraulic servo-valves such that 4P aircraft accelerations are minimized. Specifically, it is required that the ECU perform the following functions:

1. Track main rotor RPM by means of one of the following; an optical encoder, magnetic interrupter, or multi-pole drive-mounted commutator.

2. Generate a reference signal at the fourth harmonic of the current blade passage frequency.

3. Perform cross correlation to extract the 4P components of rotor hub feedback forces. This requirement is in lieu of a microcomputer-performed FFT routine, due to record length and sample time constraints.

4. Generate appropriate 4P servo-valve drive signals utilizing computer-derived phase and amplitude data.

5. Perform basic self-test functions.

Section 2.0 ELECTRONIC SUBSYSTEM PRELIMINARY DESIGN presents preliminary ECU design concepts which specifically address the above functional requirements.

1.3.2 Digital Microcomputer

1.3.2.1 Microcomputer Hardware

The digital microcomputer will be required to employ specific control algorithms for calculating optimal 4P feathering inputs in addition to performing self-test, limit checking and error recovery functions. Flexibility afforded by the digital computer is required to facilitate control algorithm development and modification. A multi-redundant digital system is unnecessary for the current approach by virtue of the low authority (~ 10 percent) associated with the harmonic pitch control system. Candidate digital systems are presented and contrasted in Section 2.3.

1.3.2.2 Control Software

The manner in which HHC for vibratory hub load suppression is employed constitutes an adaptive feedback control system. That is, certain system parameters (i.e., hub transfer matrix) vary in an unknown manner. The parameter variations being large, it is desirable to design for the ability to continuously estimate them and change the control parameters such that system performance criteria are met. Therein exists a requirement for an identification algorithm to estimate hub transfer matrices from sampled data and a control algorithm to calculate optimal control inputs and optimize performance criteria.

Data requirements for estimating and controlling algorithms generally consist of little more than hub 4P acceleration responses in vertical pitch and roll and command control inputs yielding the measured hub responses. In the general case, three control degrees of freedom exist to optimize the performance criteria associated with at most three hub responses. Available control degrees of freedom can be identified as either (1) swashplate vertical, pitch and roll motion, or (2) blade root 3P, 4P and 5P blade feathering, or (3) rod end displacements of three individual servo actuators. Although there is reason to believe that blade root feathering frequency is the most effective control parameter in terms of its ability to minimize 4P hub responses, it is proposed that actuator rod end displacements be employed in HHC control algorithms. There is no attendant coordinate transformation required to implement calculated rod end displacement, as is required in the other control degrees of freedom.

Comprehensive treatment of candidate estimating and control algorithms is beyond the scope of the current study. The reader is referred to References 13 and 15 for a discussion of Kalman filtering and other techniques currently being evaluated for HHC application.

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1.4 Loads

1.4.1 Main Rotor Blade Dynamic Analysis

In order to define and quantify the effects of small amplitude 4P blade feathering upon control loads and blade stresses, a single-blade, transient response aeroelastic simulation analysis was performed. Calculations were made with the Hughes Helicopter DART program. A ten-station fully coupled model employing non-linear aerodynamic characteristics was used to obtain natural frequencies as well as blade and hub fatigue loads in high speed level flight. The control system stiffness used was that associated with cyclic torsion modes, while collective drive system stiffness was used for the edgewise boundary condition. Higher harmonic feathering was effectively introduced in the rotating system by imparting 3P, 4P and 5P feathering individually to the blade root.

A paramount consideration in assessing the impact of higher harmonic blade feathering on rotor loads is ensuring there exist no resonant blade frequencies near 3P, 4P and 5P. A calculated main rotor resonance diagram for the OH-6A, Figure 2, illustrates that except for the coupled second flapwise bending/5P vertical plane pendulum absorber mode, the majority of blade modes are reasonably removed from the frequencies of interest.

Prior to obtaining rotor blade responses under the influence of HHC, a cursory comparison of measured and calculated trends in baseline blade loads with advance ratio was prepared. Good agreement was observed for alternating chord bending, flap bending and pitch link loads at moderate advance ratios, as seen in Figures 3, 4 and 5. Lesser degrees of correlation were observed for these parameters at low advance ratios possibly due to potential blade-vortex interaction phenomena at transitional speeds. Thus, trends in blade stresses and control loads observed during flight test can be simulated using the current model over the majority of level flight regimes with a reasonable degree of confidence.

The influence of arbitrary HHC inputs on alternating blade stresses and control loads was next explored. Separate sine-phased 0.5 degree amplitude inputs at the pitch link were made at 3P, 4P, and 5P. The arbitrary nature of the control inputs was due, in part, to the lack of a definitive rotor wake induced velocity profile. The benefit of HHC is derived from its ability to redistribute high frequency air loads around the rotor azimuth. Thus, there is only nominal hub 4P excitation when such rotor wake effects are omitted leading to a tendency by many to underpredict required optimal control amplitudes. Based upon published wind tunnel studies, References 9 and 11, it was reasonable to assume 0.5 degree HHC amplitudes for this category of rotorcraft.
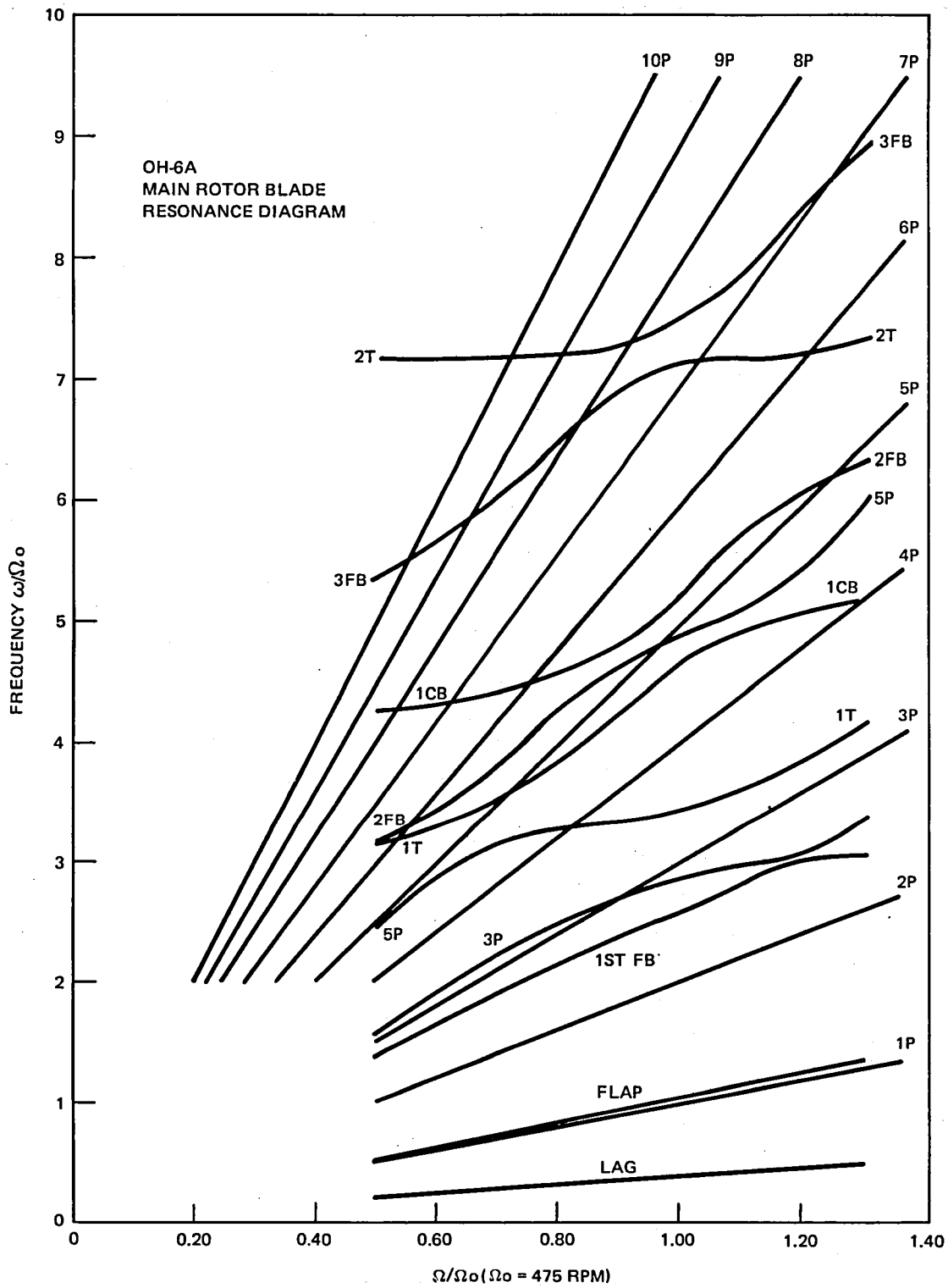
Figure 2. - OH-6A main rotor blade cyclic mode resonance diagram.
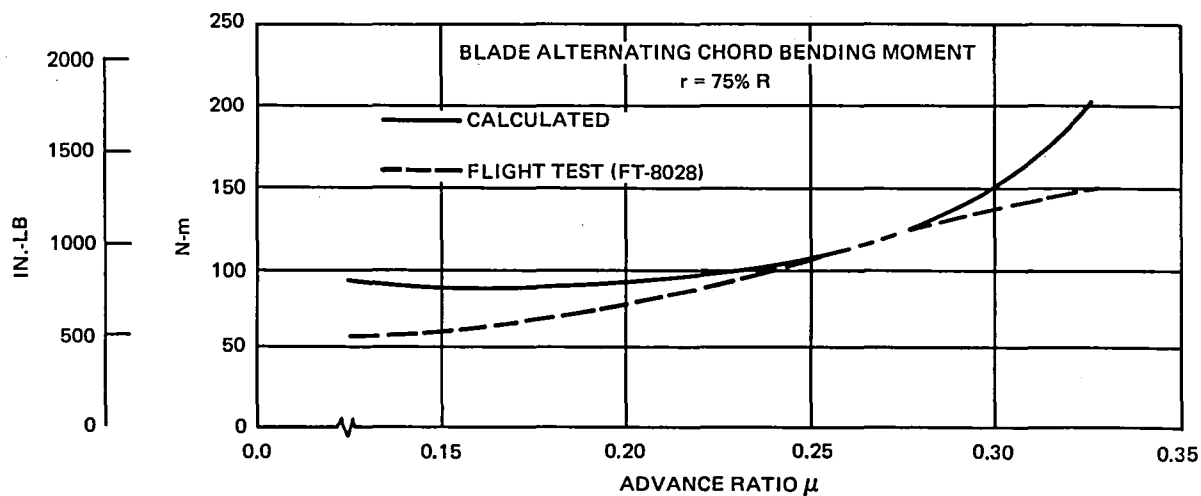
Figure 3. - Calculated vs. measured blade chord bending moments.
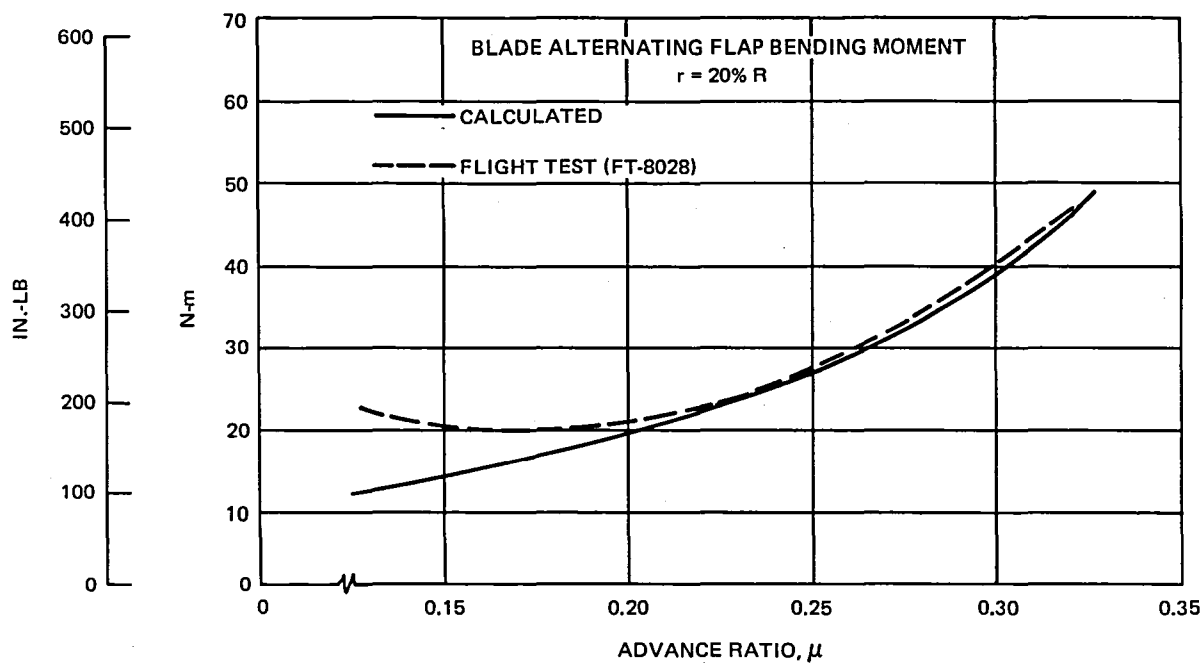
Figure 4. - Calculated vs. measured blade flap bending moments.
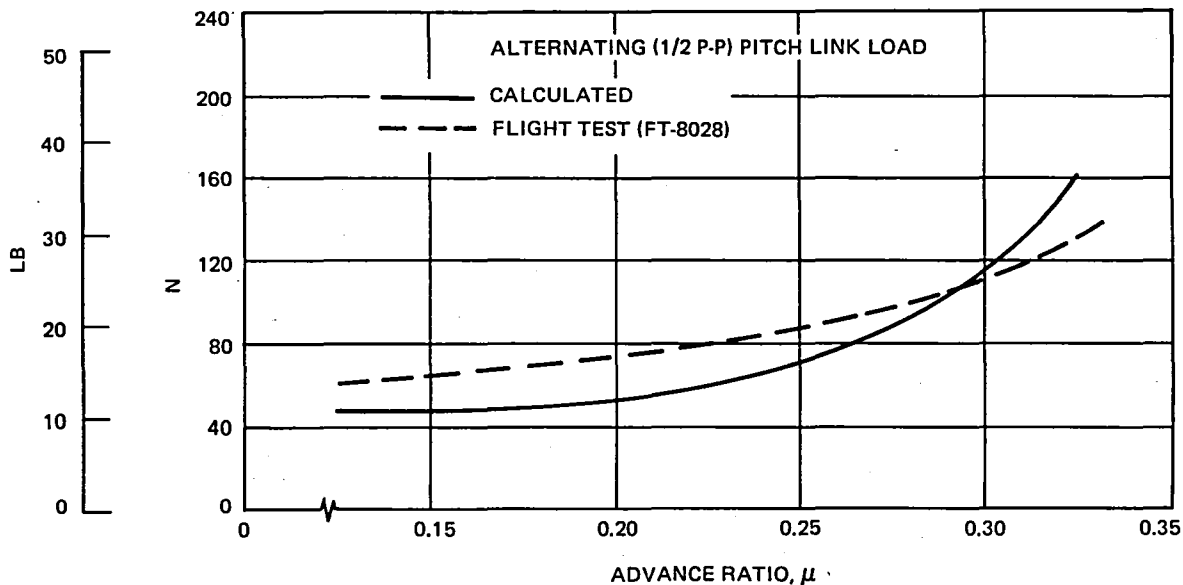
Figure 5. - Calculated vs. measured pitch link loads.

Figure 6 shows the effect of 3P, 4P and 5P HHC blade feathering on alternating chord bending moments. The strongest sensitivity to HHC inputs is demonstrated at 5P feathering which induced less than a ten percent increase in edgewise moments at midspan. Inputs at 3P and 4P produced only minor changes in edgewise moment distribution, owing to the proximity of first chord bending to 5P, Figure 2. Figure 7 compares the harmonic content of chordwise bending moment response for the baseline condition and HHC input conditions. As can be seen interharmonic coupling is slight, the major influence of 3P, 4P and 5P feathering being seen in the respective harmonics of chordwise response. The source of increased chord bending moments can be attributed to HHC-induced flapping motion coupled into lag motion via the twisted retention straps and Coriolis accelerations.

Figure 8 similarly illustrates the increase in blade flap bending stresses as produced by the various HHC inputs. Inboard flap bending moments increase by about 37 percent for all three forms of HHC. The increase is to be expected in view of the 4.8P second flap bending mode receiving strong inertial forcing at 3P, 4P and especially 5P.

Further insight is gained in reviewing the flap bending moment harmonic breakdown in Figure 9. Little harmonic coupling is seen for the frequency
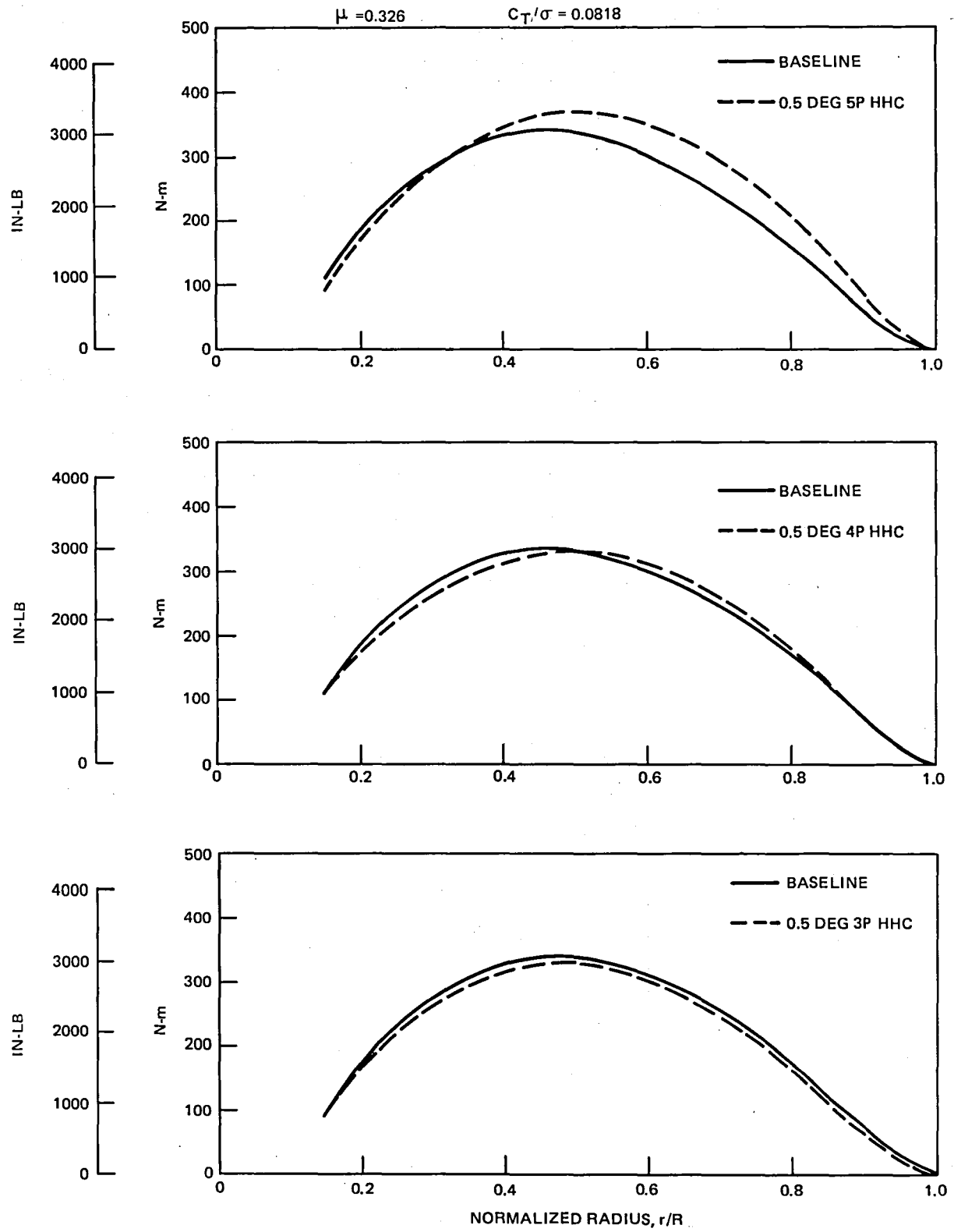
Figure 6. - Blade alternating chord bending moments.

11
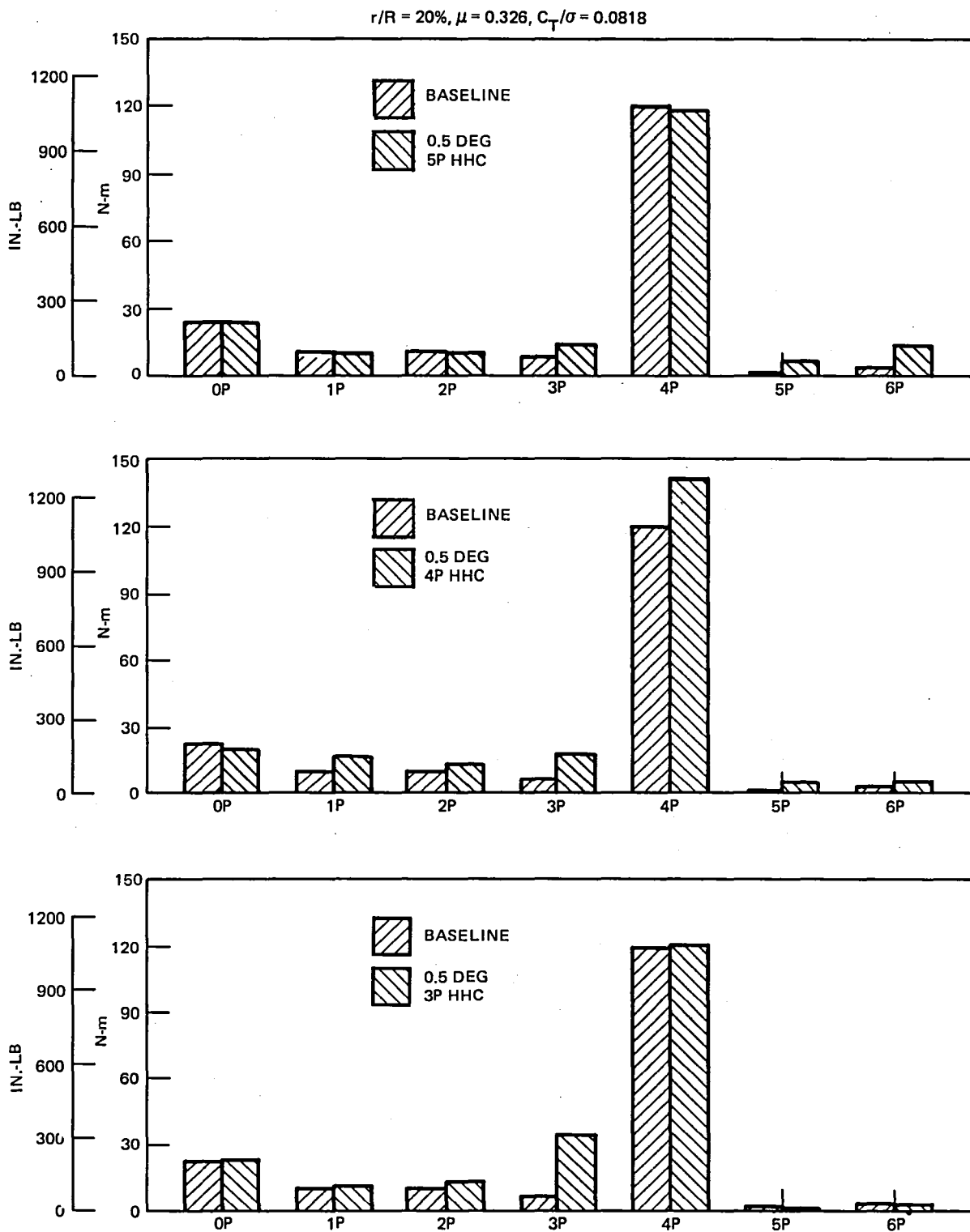
Figure 7. - Harmonic decomposition blade chord bending moments.

$\mu = 0.326$ $C_T/\sigma = 0.0818$
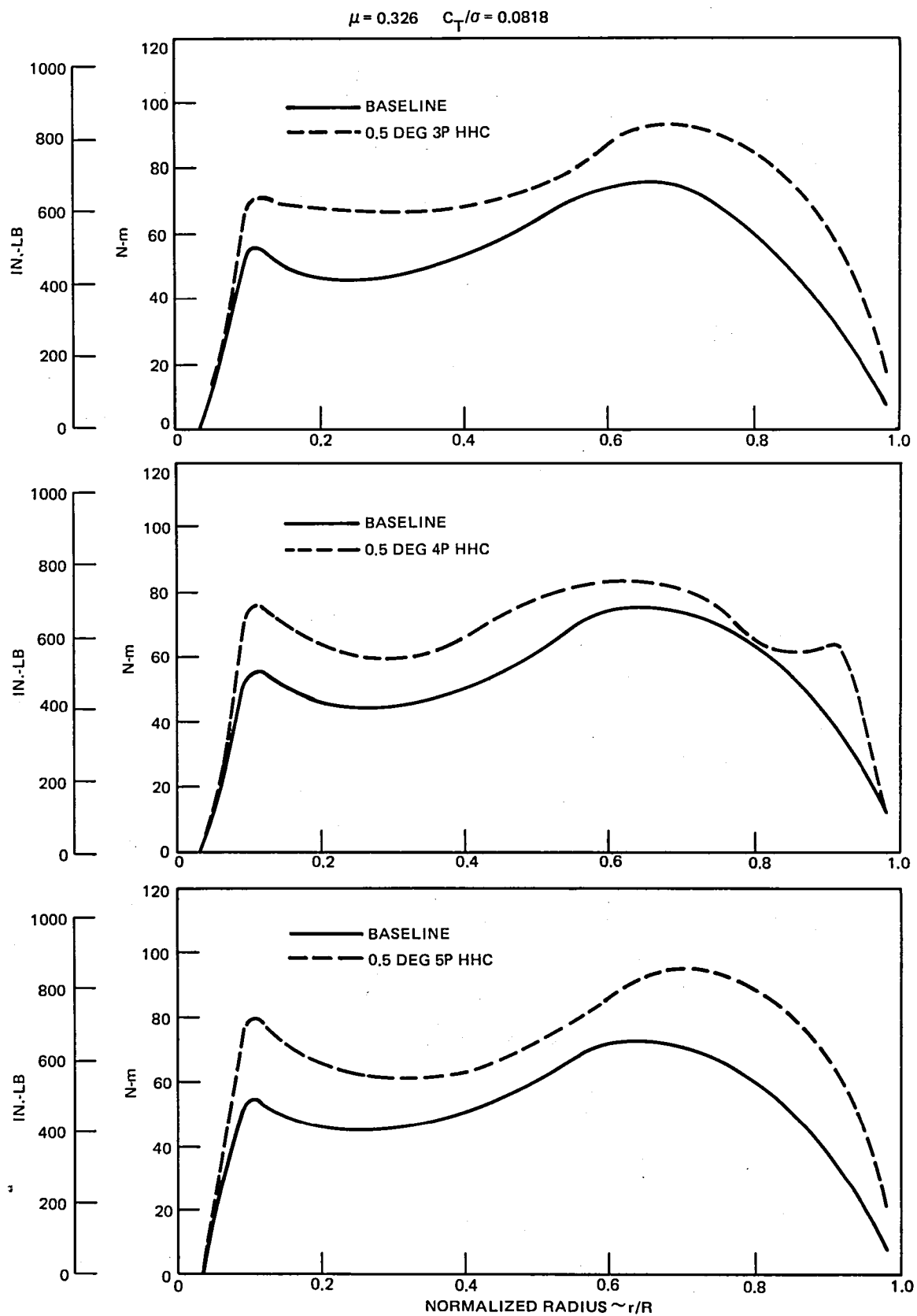
Figure 8. - Blade alternating flap bending moments.

13
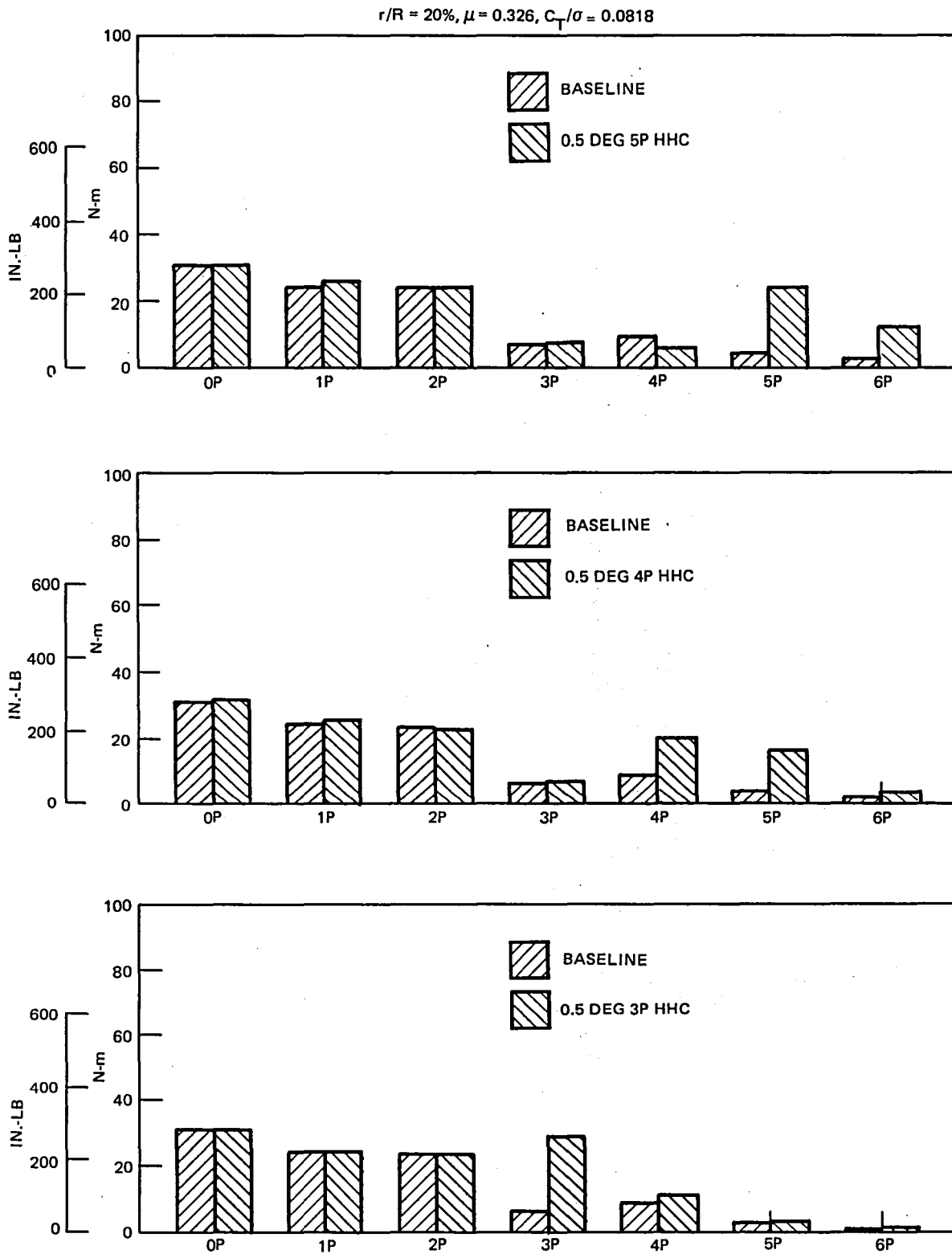
Figure 9. - Harmonic decomposition blade flap bending moments.

components below the HHC feathering frequencies. Slight interharmonic coupling is found, however, for components above the HHC frequencies of interest. It is suggested that the small 5P/6P, 4P/5P, and 3P/4P interdependencies are more a function of nonlinear aerodynamics rather than typically stronger structural or kinematic coupling.

Of consuming interest in terms of control system structural integrity is the increase in pitch link load under the influence of HHC. A 33 percent increase in alternating pitch link load was experienced using 4P HHC, with the load varying linearly with pitch amplitude and with the square of feathering frequency. A decomposition of the harmonic content of pitch link load is presented in Figure 10. As in previous response studies the degree of interharmonic coupling is small compared to the content at the primary forcing frequencies.

The source of increase in pitch link loads due to high frequency feathering is primarily the nose-down pitching moment associated with the centrifugal restoring moment (tennis racket effect), and inertial forcing contributions made by the 3.4P first torsion mode. Thus, the 5P response due to 5P HHC is dominated by the centrifugal restoring moment, while the 4P response to 4P HHC is a result of both tennis racket and first torsion mode contributions. The 3P response component due to 3P HHC is primarily due to first torsion mode inertial forcing, the tennis racket moment decreasing to one-third of that in the 5P HHC case.

A comprehensive assessment of the impact of increased fatigue loads due to HHC on the flight control system is dictated in detail design. It is not anticipated, however, that control system design loads or endurance limits will be exceeded due to HHC inputs. For example, the maximum measured pitch link load, 996N (224 pounds), is associated with a 120-knot 2g pull-up maneuver. The pitch link design limit load is 2,255N, (507 pounds), and ultimate strength is 3,380N, (760 pounds). Thus, in light of a 100 percent factor of safety for a flight condition outside the normal flight envelope, Figure 11, the spectre of exceeding design limit loads using HHC is not foreseen.

In terms of fatigue damage, there currently exist only two life-limited components in the baseline OH-6A flight control system; a longitudinal control rod from the cyclic stick to the mixer assembly, with a life of 48,000 hours, and a Uni-lock device for preventing longitudinal control load feedback to the cyclic stick, with a life of 88,000 hours.

Pitch links, for example, have an unlimited life for oscillatory loads below the 1,143N, (257 pound) endurance limit, by definition. The maximum
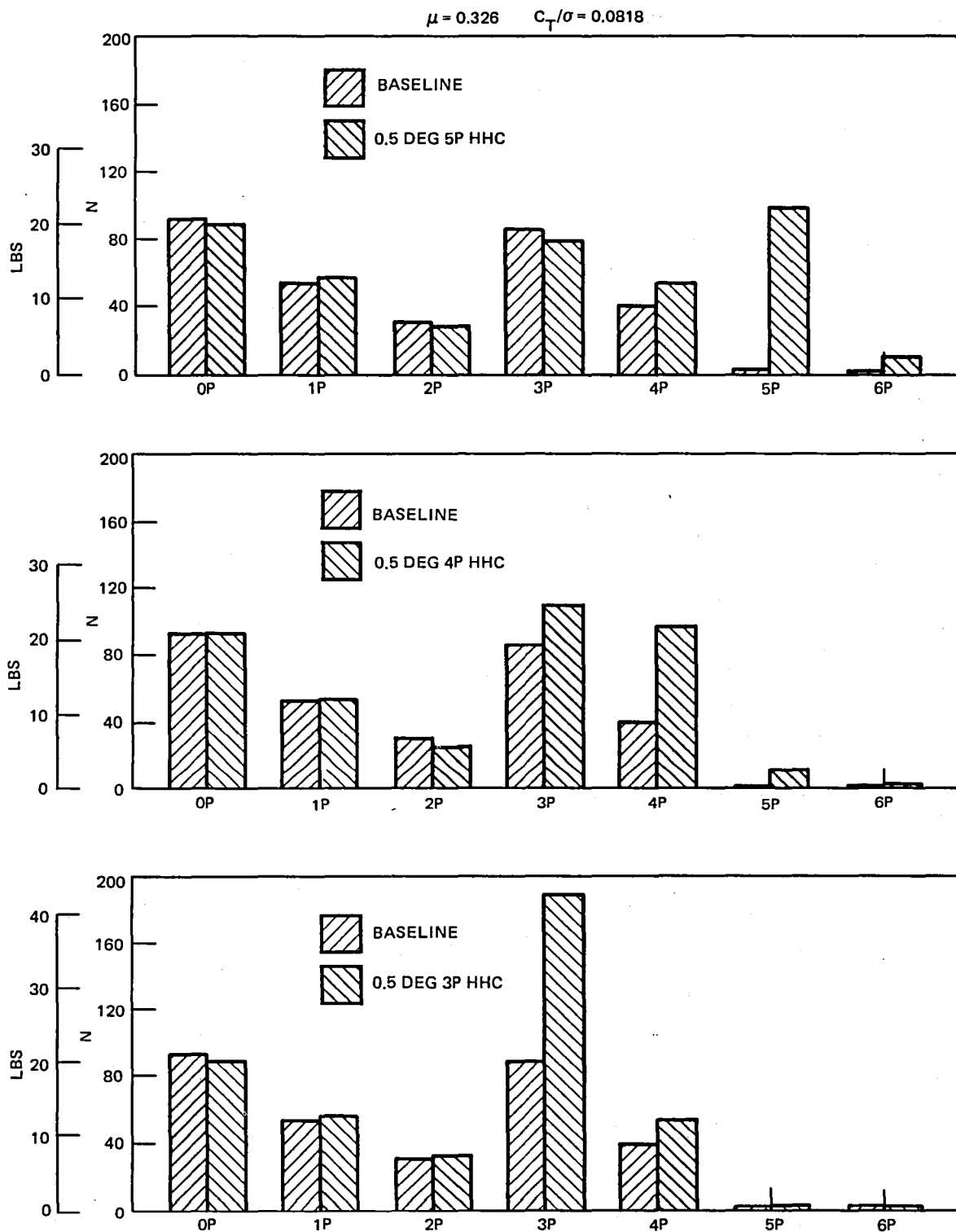
Figure 10. - Pitch link load harmonic decomposition.
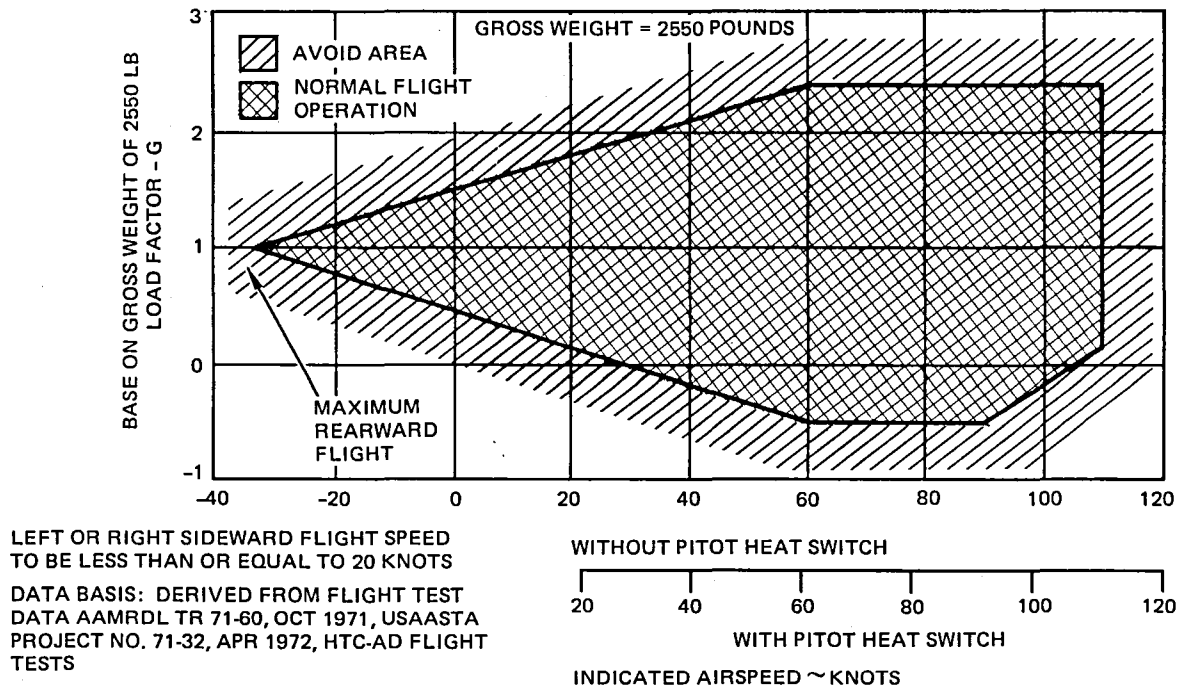
Figure 11. - Flight envelope chart.

measured alternating load, again associated with a 120-knot 2g pullup maneuver is 872N, (196 pounds), yielding a 31 percent margin for a flight condition outside the normal flight envelope. Were the endurance limit exceeded, no fatigue damage would be incurred since the load is encountered only once per maneuver. It is proposed that a comprehensive fatigue analysis be performed in detail design to assess any life-limiting impact of HHC on primary control system elements.

2.0 ELECTRONIC SUBSYSTEM PRELIMINARY DESIGN

2.1 System Requirements

A fundamental requirement of active harmonic control for vibration reduction is an ability of the control network to adapt to changes in the helicopter's

operating environment. In this case, adaptation to changing external flight loads requires modulation of 4P feathering phase, amplitude and frequency such that a minimum oscillatory load condition is maintained over the spectrum of flight conditions. Thus, the control network must possess the ability to (1) sense vibratory hub shear forces and moments at the rotor, (2) process past and present load data to permit calculation of optimal 4P feathering phase and amplitude, and (3) develop appropriate sinusoidal servovalve drive currents. Tasks pursued during preliminary design, therefore, included:

a. Evaluating flightworthy general purpose digital microcomputers for HHC applications. The device is required to perform adaptive recalculation of optimal 4P feathering signals in response to changes in flight conditions in addition to limited system self-testing.

b. Development of a device to suitably interface microcomputer functions with feedback transducers while providing basis self-test functions. This device, the Electronic Control Unit (ECU), coupled with the microcomputer above, comprises what is termed the Electronic Subsystem.

2.2 ECU System Definition

2.2.1 System Overview

The System block diagram is shown in Figure 12. The ECU incorporates a reference source synchronized with main rotor rotation to develop a 16P square wave or reference pulse plus a 4P synch reference signal. A drive-mounted 16-pole commutator or optical encoder are two such reference sources currently being considered for this application. Reference signals are utilized in the ECU correlator section to derive 4P components of vertical, pitch and roll accelerations. Outputs from the correlators are provided to the digital computer as DC analog signals. Note that all data provided to and generated by the microcomputer is in DC form. This both simplifies interface circuitry and eliminates the need for the computer to construct AC drive signals.

Two signals are derived from each acceleration transducer, with one signal being directly proportional to the in-phase component of measured 4P acceleration and the other being proportional to the quadrature component. The ECU also provides a DC analog signal to the computer which is directly proportional to rotor rpm. Two other DC analog signals, one each proportional to the magnitude of the sine and cosine reference 4P signals are generated by the
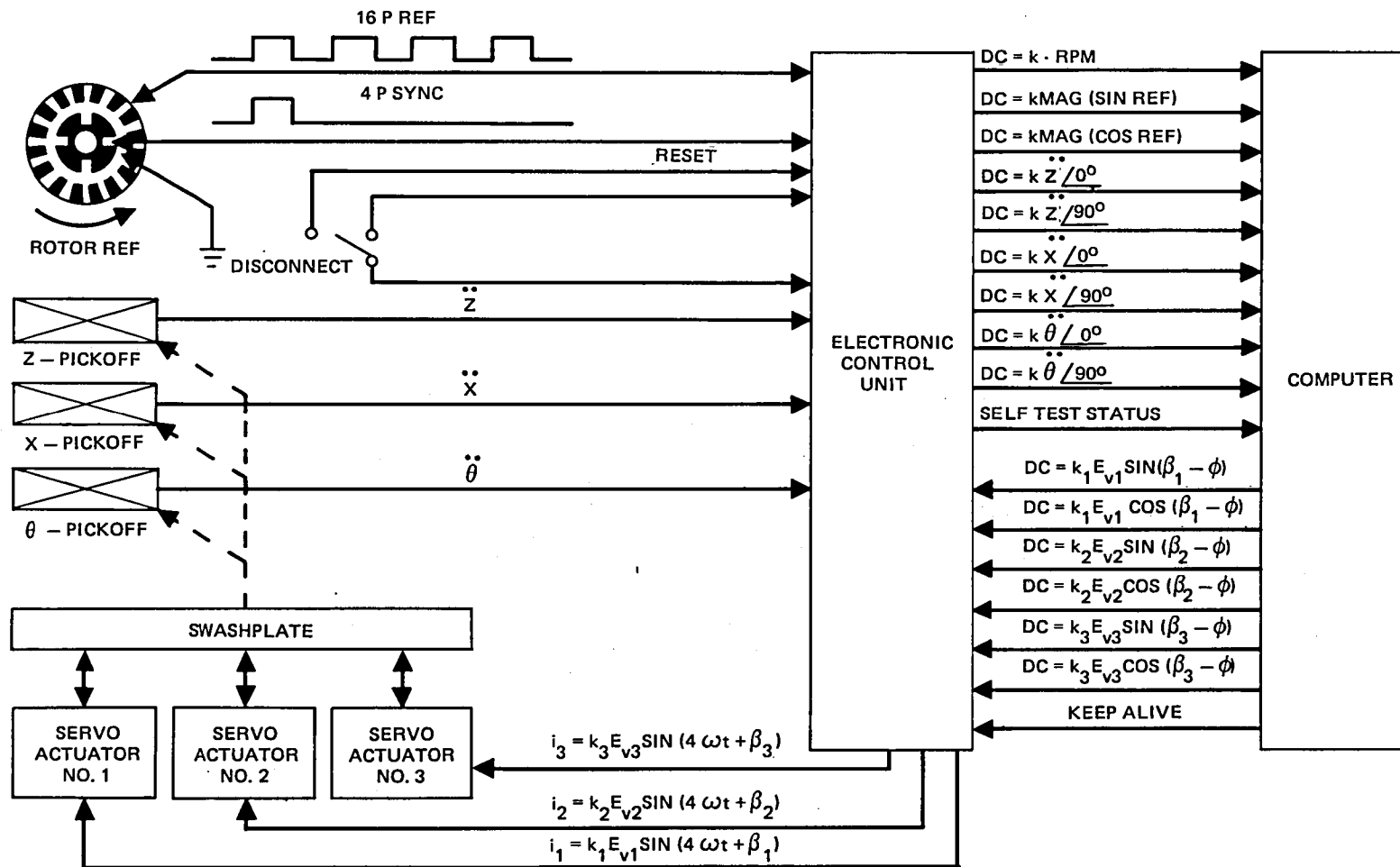
Figure 12. - Electronic subsystem block diagram.

ECU. The remaining signal provided by the ECU is a self-test discrete signal which indicates the presence or absence of any internally detected processing errors.

The computer, in turn, provides DC analog signals to the ECU to control phase and magnitude of the drive signal output to each of the three servo-actuator units. The computer similarly provides a "keep alive" signal to the ECU to indicate the processor is performing desired program operations in a normal manner.

2.2.2 ECU Block Diagram

The ECU block diagram is shown in Figure 13. The unit consists of a sine/cosine generator, three identical correlator sections, one for each feedback transducer, and three identical servo-actuator drivers. The last section of the ECU is the self-test portion which monitors proper performance of each of the other sections.

2.2.3 Sine/Cosine Generator

A block diagram of the sine/cosine generator is shown in Figure 14. The section contains a two-bit gray code counter, two identical Butterworth filter sections, two identical voltmeter sections and a frequency-to-voltage converter.

2.2.3.1 Gray Code Counter

The gray code counter is shown in block diagram form in Figure 15. The 16P reference signal clocks two D-type flip flops. The 4P synch reference is used to preset the exact state of the flip flops at each quarter revolution of the main rotor. Resultant wave shapes are illustrated in Figure 16. Output of the first flip flop, Q_A, becomes a 4P sine-phased square wave, while Q_B becomes a cosine-phased 4P square wave. Thus a 4P sine reference and cosine reference have been generated and can be used for further processing.

2.2.3.2 Filter Section

Each of the two 4P reference signals from the gray code counter are passed through a filter section. Each filter consists of a four-pole Butterworth low pass filter section and two single-pole RC high-pass filter sections, as shown schematically in Figure 17. The combination provides a band pass filter with a band width of 26 to 38 Hz, accommodating a ±20 percent variation in main rotor RPM.

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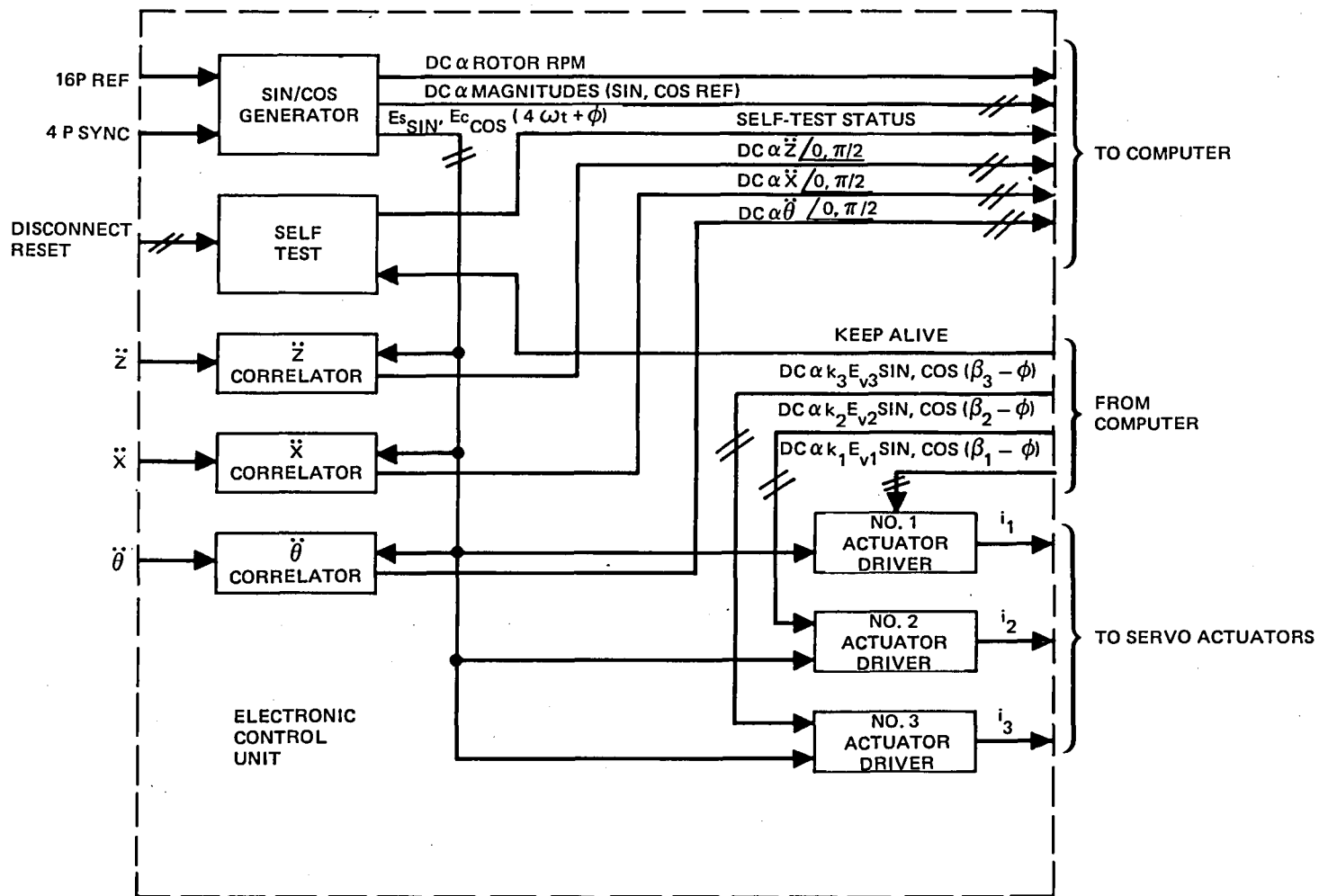
Figure 13. - Electronic control unit block diagram.
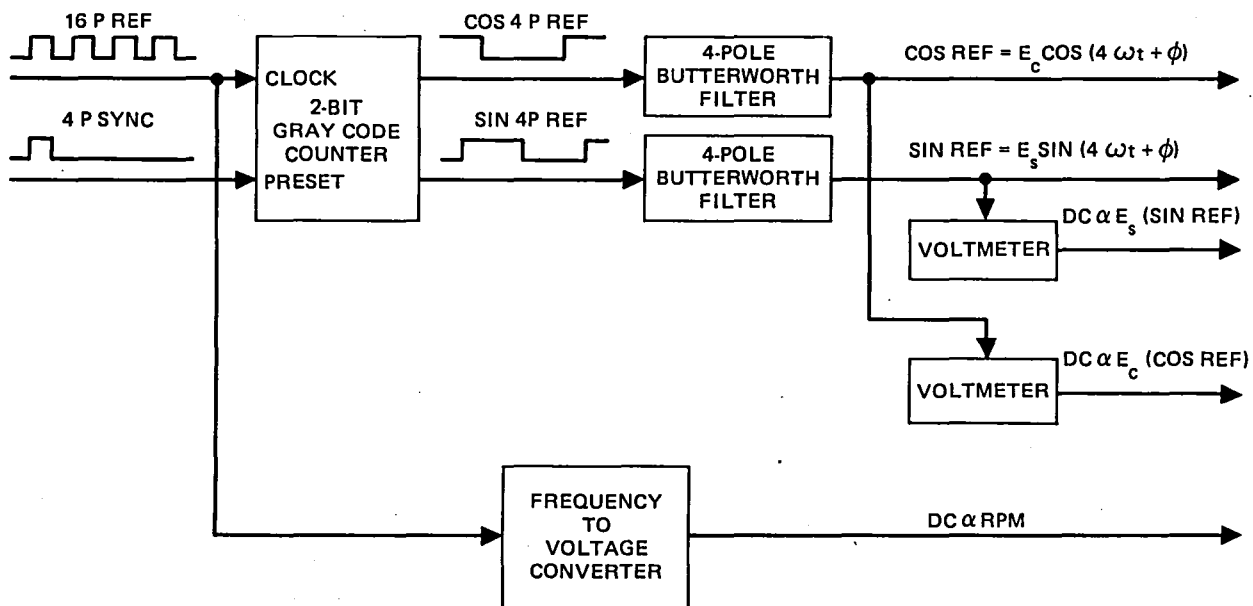
Figure 14. - SIN/COS generator block diagram.
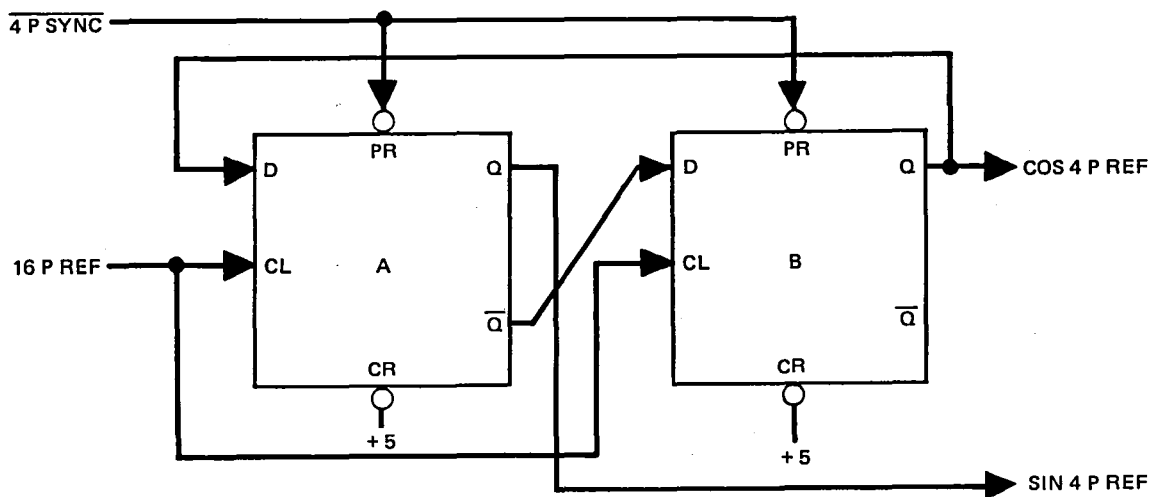
● D-FLIP/FLOP CHARACTERISTICS

  ● WHEN PR GOES LOW, Q LATCHES HIGH (TRUE)
  ● $\overline{Q}$ = Q INVERTED
  ● ON THE RISING EDGE OF THE CL PULSE, Q COPIES D

Figure 15. - Two-bit gray code counter.
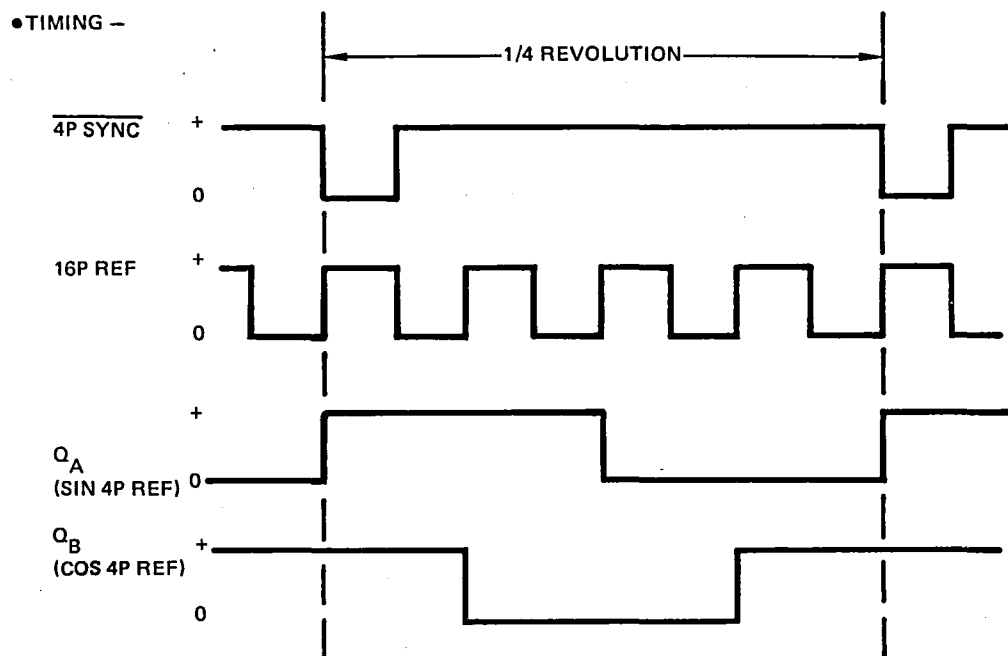
Figure 16. - Two-bit gray code counter timing diagram.
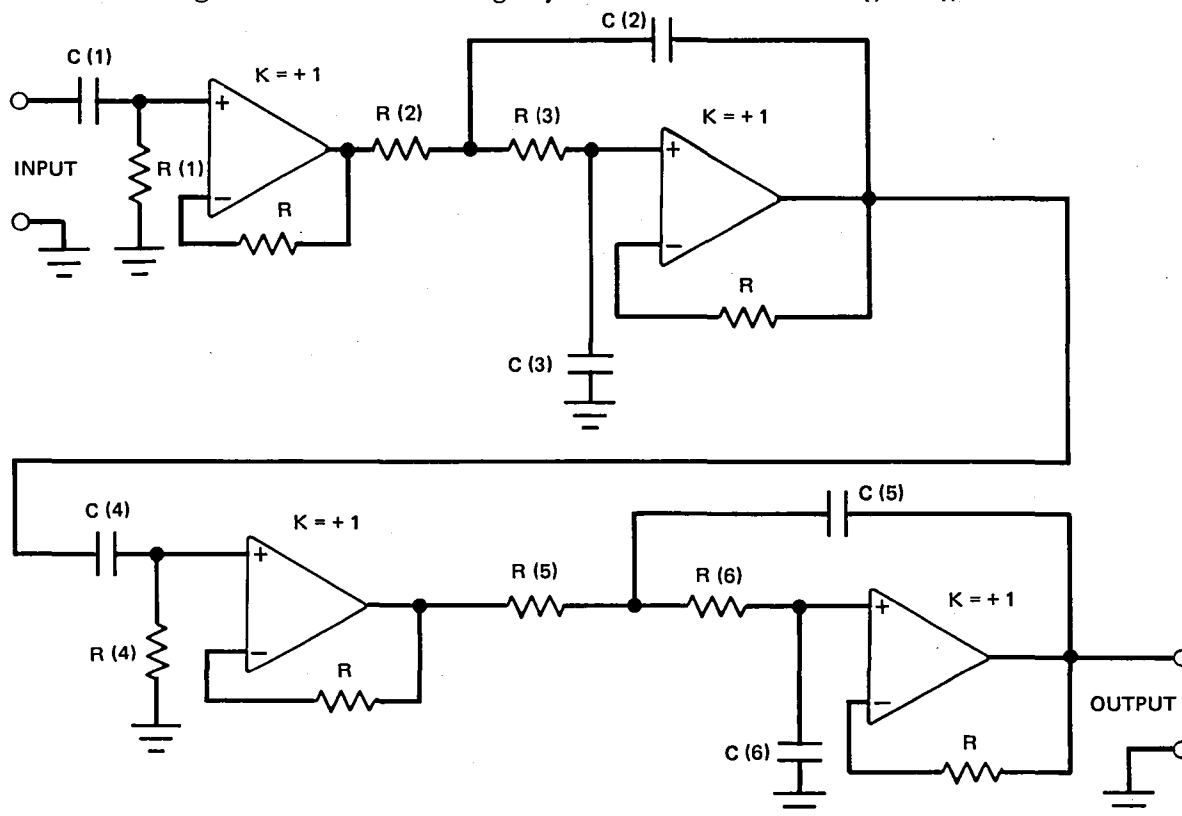
Figure 17. - 4-pole Butterworth and 2-pole RC band pass filter
pass band — 26 to 38 Hz.

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Figures 18 through 21 present the results of a computer simulation of the filter section. The input/output signal wave shapes at the nominal mid-band frequency (32 Hz) are plotted in Figure 18. The output signal is nearly sinusoidal and was computed as having approximately 1.2 percent third harmonic (12P) distortion. There is no 8P harmonic distortion since a square wave contains only odd harmonics.

There exists considerable phase shift (100 degrees) between the input and output signals, as seen in Figure 18. Figure 18 illustrates further that peak output amplitude is very nearly equal to peak input amplitude. Figures 19 and 20 depict the trends in gain and phase shift as a function of frequency. Filter gain varies from nearly unity to 0.8 as frequency increases, while phase varies almost linearly from 60 degrees to 140 degrees as frequency increases. The trend in third harmonic distortion with frequency is plotted in Figure 21, and varies from 2.4 percent at 26 Hz to less than 1 percent at 38 Hz. Thus, signals generated by the filter sections are very nearly pure sinusoids. Since phase shift and amplitude gain exhibit known variations with frequency, the computer can be programmed to compensate for these variations as a function of frequency. The frequency converter section generates an analog signal proportional to rotor RPM, hence permitting such computer compensation over the entire RPM operational band of the system.
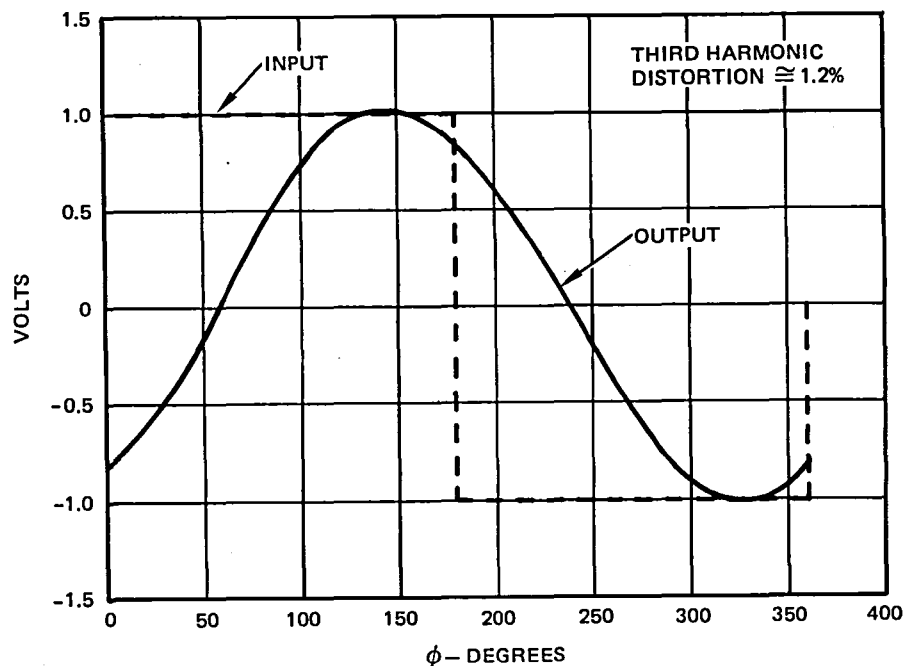
Figure 18. - Simulated bandpass filter - input frequency - 32 Hz.
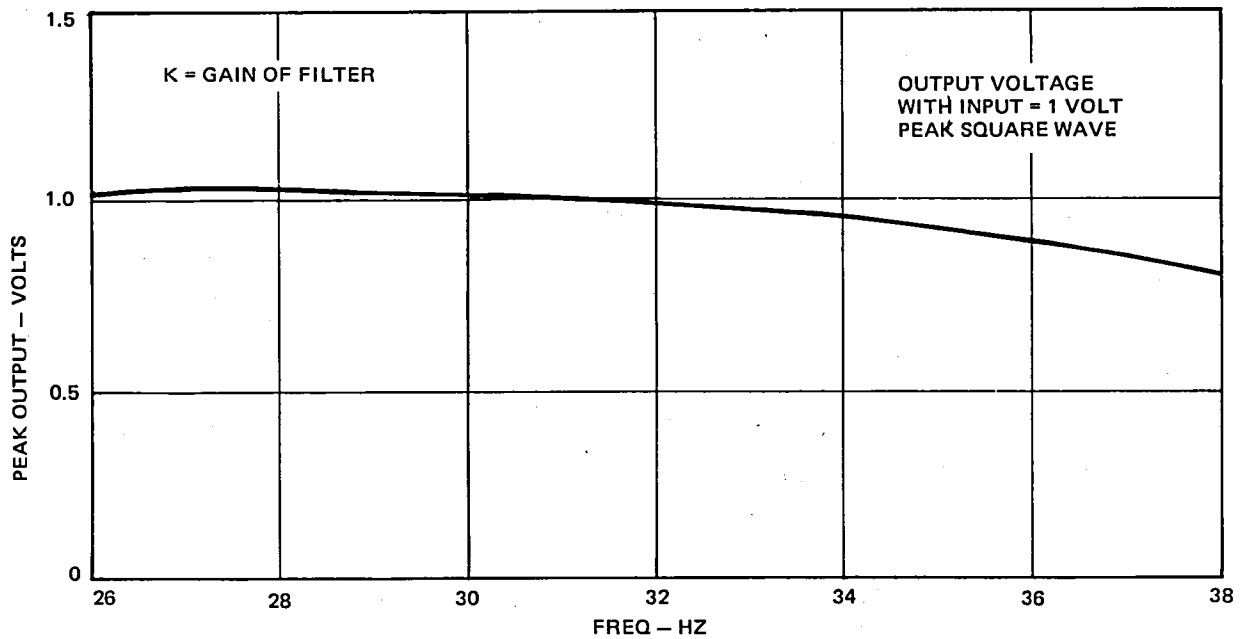
Figure 19. - Simulated bandpass filter gain - 26 to 38 Hz.
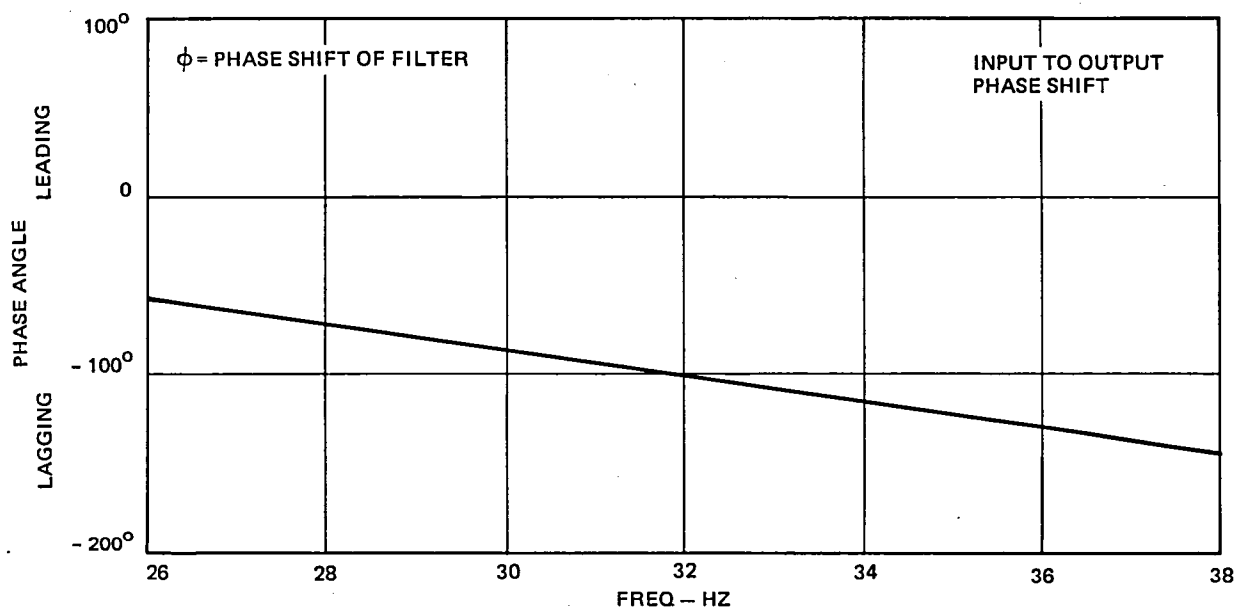
Figure 20. - Simulated bandpass filter phase shift - 26 to 38 Hz.
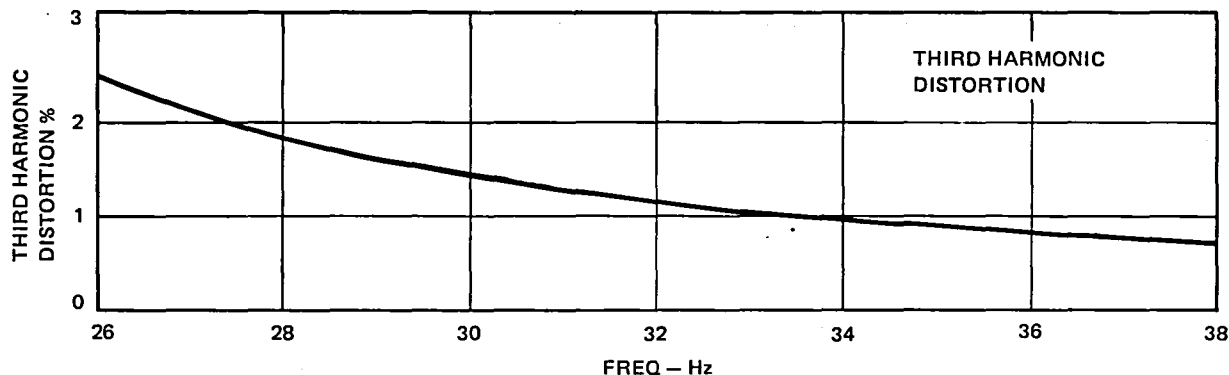
Figure 21. - Simulated bandpass filter distortion - 26 to 38 Hz.

2.2.3.3 Voltmeter Section

It is proposed that the voltmeter section be an off-the-shelf commercially available device. Such a unit develops a DC analog output signal which is proportional to the RMS value of input signal.

2.2.3.4 Frequency-to-Voltage Converter

It is proposed that a commercially available frequency-to-voltage converter be utilized. The device will develop a DC analog output signal directly proportional to rotor RPM. A typical frequency to voltage converter is schematically shown in Figure 22.

2.2.4 Correlator Section

The block diagram for each of the correlator sections is shown in Figure 23. The correlator consists of a bandpass filter (identical in design to that used in the reference generator section), two multipliers and two integrator sections. By using a filter in the vibration signal path which is identical to that used in the sine/cosine generator section, the phase shift introduced between

26
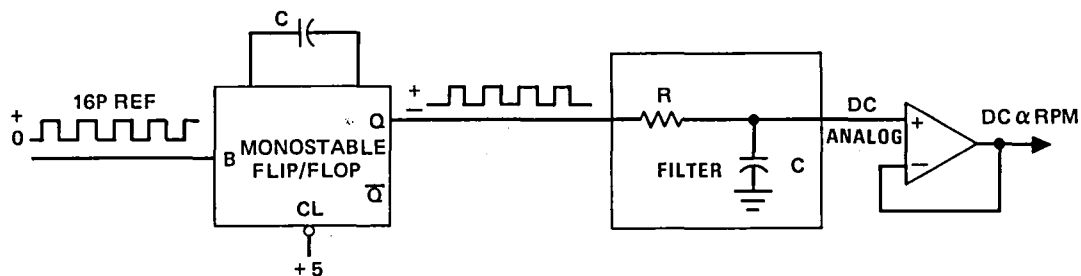
- MONOSTABLE FLIP/FLOP CHARACTERISTICS

  - EACH TIME THE INPUT TRANSITIONS FROM 0 TO + VOLTAGE, Q GOES TO A PLUS OUTPUT FOR A FIXED PERIOD OF TIME
  - THE EXTERNAL CAPACITOR, C, IS CHOSEN TO GIVE THE DESIRED DURATION OF THE Q OUTPUT PULSE

- FILTER CHARACTERISTICS

  - THE FILTER BLOCKS THE AC SIGNAL AND ALLOWS ONLY THE DC COMPONENT TO PASS THE AMPLIFIER

- TIMING
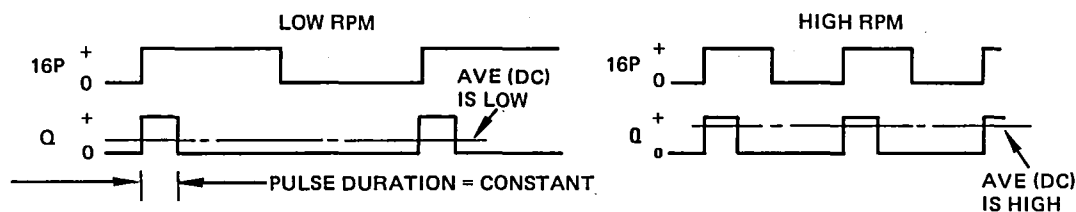
- DC OUTPUT IS DIRECTLY PROPORTIONAL TO FREQUENCY

Figure 22. - Typical frequency-to-voltage converter.
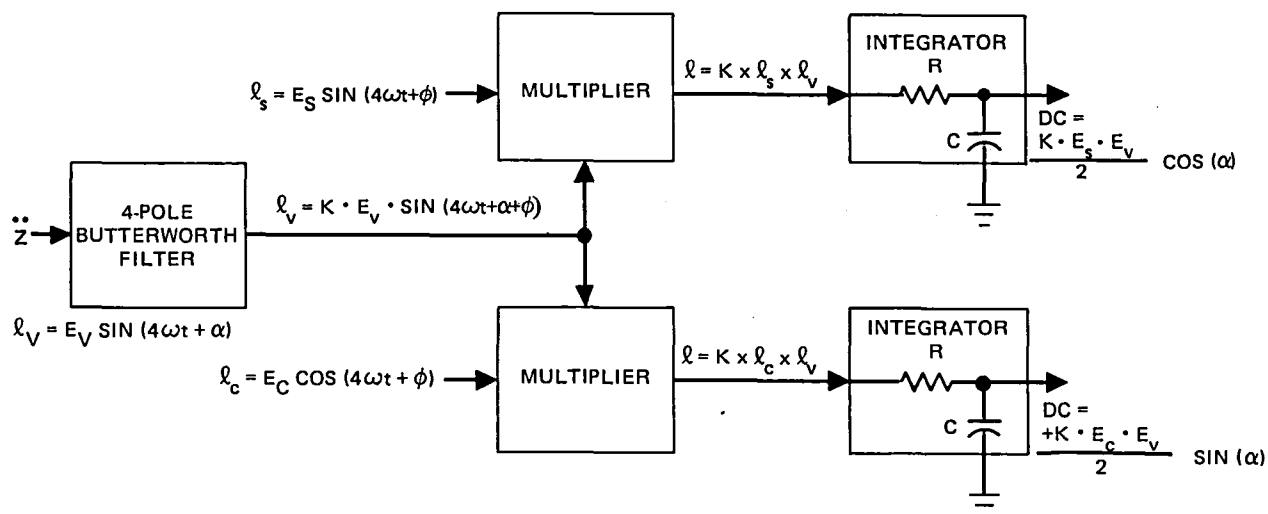
Figure 23. - ECU correlator block diagram.

27

the square wave reference signal and sinusoidal reference signal is equal to the phase shift introduced in the feedback transducer signal as it passes through the filter. Thus, when the 4P vibration component is multiplied by the sine and cosine reference signals, the effect of the phase shift generated by the filters is nullified. Specifically, let the 4P sine reference with Butterworth filter phase shift $\emptyset$, be represented by,

$$e_s = E_s \sin (4\Omega t + \emptyset) \tag{1}$$

and similarly let the 4P component of vertical acceleration be written as

$$e_v = E_v \sin (4\Omega t + \alpha) \tag{2}$$

where α is an arbitrary phase. Butterworth filtering of e_v induces an amplitude gain K and phase shift $\emptyset$.

$$e_v' = KE_v \sin (4\Omega t + \emptyset + \alpha) \tag{3}$$

Multiplying (1) and (3) and invoking trigonometric identities yields

$$e_s \cdot e_v' = \frac{KE_s E_v}{2} \left[\cos (\alpha) - \cos (8\Omega t + 2\emptyset + \alpha) \right] \tag{4}$$

Thus, integrating the multiplier output filters out the double frequency (8P) term while passing the DC term. Appendix A contains an analysis to determine an optimum integrator time constant such that sample and solution rates may be maximized. The analysis concludes that optimal solution update rates are achieved with correlator integration time constants approaching 1/1P, or roughly 125 milliseconds.

2.2.5 Actuator Driver Section

The block diagram for one of three identical actuator drivers is illustrated in Figure 24. This section consists of two multipliers, a summing junction and an output current driver.

$$DC = \frac{E_v k_1}{E_s} \, COS \, (\beta_1 - \phi)$$

$$\ell_s = E_S \, SIN \, (4\omega t + \phi)$$

MULTIPLIER

$$\ell = \frac{E_v k_1}{2} \, [SIN \, (4\omega t + \beta_1) + SIN \, (4\omega t - \beta_1 + 2\phi)]$$

$$\ell = E_v k_1 \, [SIN \, (4\omega t + \beta_1)]$$

$$i = E_v k_i \, SIN \, (4\omega t + \beta_1)$$

CURRENT DRIVER

EXTERNAL LOAD

$$DC = \frac{E_v k_1}{E_c} \, SIN \, (\beta_1 - \phi)$$

$$\ell_c = E_C \, COS \, (4\omega t + \phi)$$

MULTIPLIER

R

$$\ell = \frac{E_v k_1}{2} \, [SIN \, (4\omega t + \beta_1) - SIN \, (4\omega t - \beta_1 + 2\phi)]$$
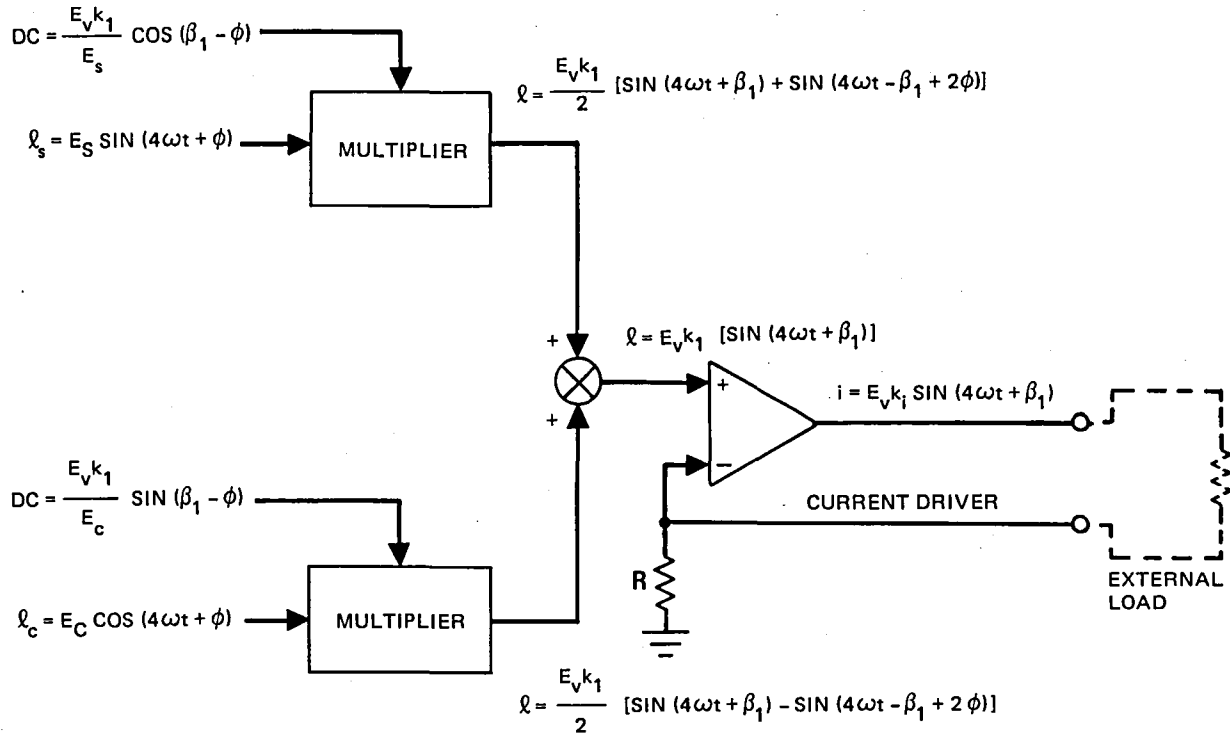
Figure 24. - Actuator driver block diagram.

Having cancelled the filter-induced phase shift in the correlator section, there still exists a need to compensate for the phase shift induced in the reference generator, since reference signals are used to develop actuator driver signals. A method for backing out the filter phase shift in the actuator driver circuitry has been developed, and is summarized for a single driver as follows:

1. The microcomputer is provided 4P transducer signal amplitude E_v, 4P sine and cosine reference amplitudes, E_s and E_c, and reference frequency, thus permitting table look-up of filter phase shift, ϕ.

2. Sampled-data recursive calculations yields optimal 4P feathering gain and phase, κ_1, and β_1.

3. Two DC signals are generated by the computer for each actuator driver and passed to the ECU;

$$DC_1 = \frac{E_v \kappa_1}{E_s} \quad \cos (\beta_1 - \emptyset) \tag{5}$$

$$DC_2 = \frac{E_v \kappa_1}{E_c} \quad \sin (\beta_1 - \emptyset) \tag{6}$$

4. Within the actuator driver, DC_1 is multiplied by the sine reference, DC_2 is multiplied by the cosine reference, and the results are summed to yield

$$DC_1 \cdot e_s + DC_2 \cdot e_c = E_v k_1 \sin (4 \Omega t + \beta_1). \tag{7}$$

In this manner, computer-generated optimal 4P feathering phase and amplitude for each actuator is converted into an appropriate sinusoidal driving current.

2.2.6 Actuator LVDT Demodulator

Provisions have been made in actuator preliminary design (Section 3.) for LVDT installations. Figure 25 schematically depicts one channel of one LVDT demodulator, required if feedback information is desired. Although further study is dictated in detail design, elimination of LVDT requirements is proposed. Sampled-data recursive calculation of state transfer matrices (Reference 13) precludes the need for actuator displacement data or even blade feathering response data. Data required for such calculations are simply commanded feathering phase and amplitude in DC analog form, and resultant 4P acceleration phase and amplitude.

2.2.7 ECU Self Test Provisions

Several preliminary self test mechanizations have been provided for in the current phase of design, and are outlined schematically in Figure 26. Although subject to revision in subsequent design phases, it is proposed to include, as a minimum, the following self test capabilities:

1. Main rotor RPM exceeds ECU limits

2. Power supply out of limits

30
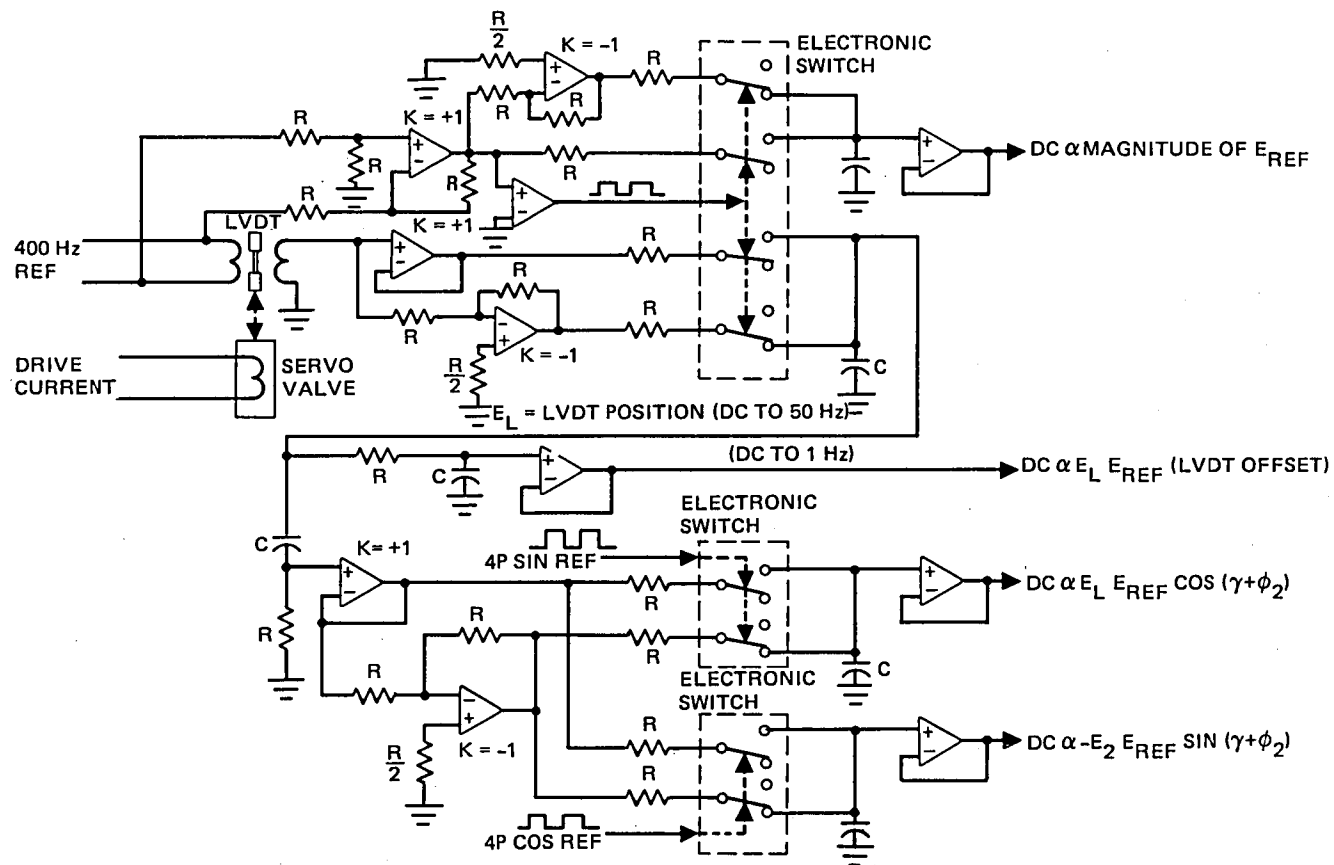
Figure 25. - LVDT demodulator (one channel).
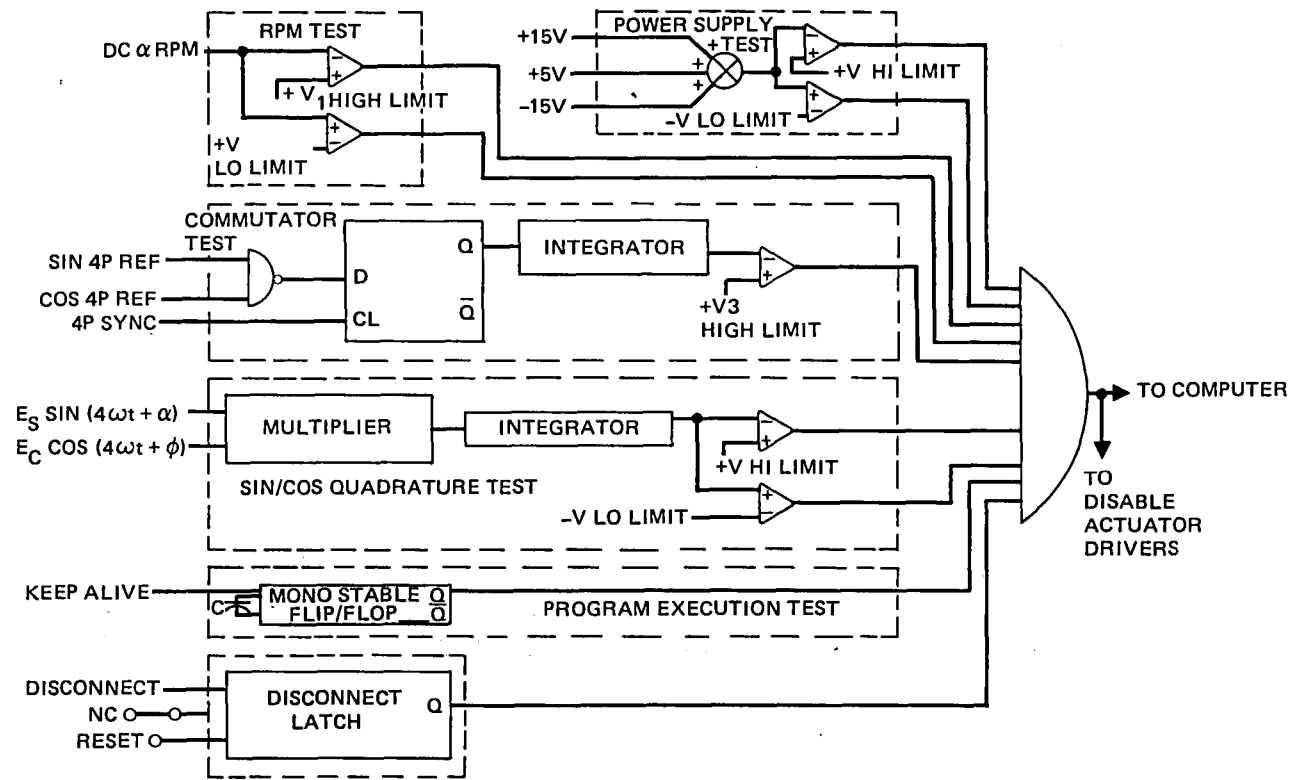
Figure 26. - Self test block diagram.

3. Commutator/Optical encoder missed pole

4. Sine and cosine reference signals not orthogonal

5. Failure to receive a "keep alive" from microcomputer (Program execution test)

6. Pilot disconnect latch.

If a failure condition is encountered in any of the tests above, all three actuator drivers are disabled.

It is important to note that the return to a normal state of any self-test functions results in a system reset and at pilot option, resumption of the HHC feathering control sequence.

2.3 Microcomputer Hardware

The requirement for some form of computational processor is born out of the need to not only compute optimal 4P feathering inputs, but also to provide phase shift and amplitude gain compensation and to conduct limit checking on blade and control system loads. Although an analog approach offers advantages in size, weight reliability and speed, such factors are over-shadowed by the digital processor's flexibility, ease of programming and ability to process arrays. It is, therefore, proposed that a general purpose digital computer be employed to determine and periodically update 4P feathering amplitude and phase required to minimize 4P rotor-transmitted loads. The computer is also to provide secondary functions that include error checking, error recovery, compensation, and limit checking for loads.

Within the scope of the preliminary design study, two different types of processors were evaluated:

1. A potentially flightworthy LSI-11-based Andromeda 11/B laboratory microcomputer

2. A completely flightworthy Sperry SDP-175[1] airborne data system

Both systems are illustrated in Figures 27 and 28.

---

[1]Sperry Flight Systems, A Division of Sperry Rand
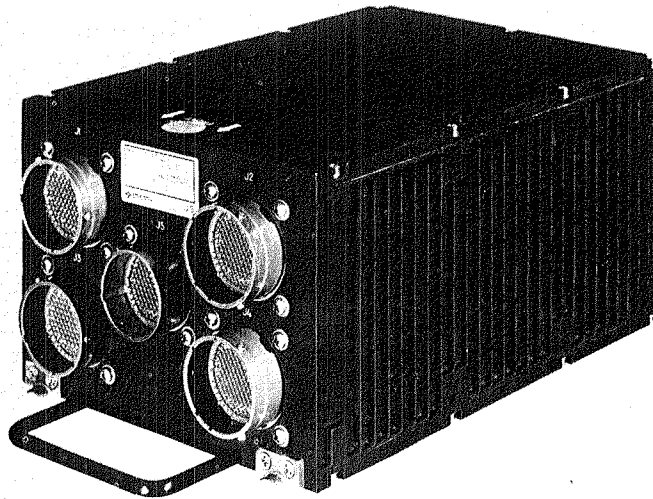
Figure 27. - Sperry SDP-175 digital computer.
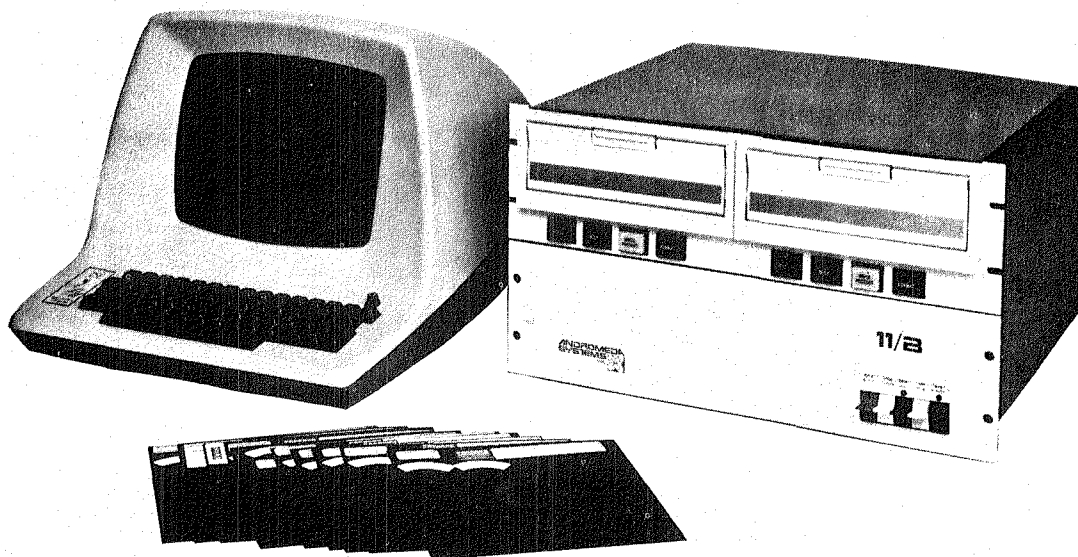
Figure 28.   Andromeda 11/B microcomputer system.

The Andromeda 11/B system is a dual floppy disk-based microcomputer built around the DEC_{TM}[1] LSI-11 CPU. Intended primarily as a laboratory data acquisition system, this system supports RT-11, a DEC real time operating system, which in turn supports Basic, FORTRAN, a Macro Assembler, and text editor.

The Sperry SDP-175 is a flightworthy 16-bit computer generally employed in multiplex bus control, fire control, stores management and guidance and navigation on a time-shared basis. Programming is accomplished in Assembly under a floppy disk operating system. Available I/O includes AC analog (400 Hz), DC analog, discrete (off-on), switch, serial, and synchro. Additional controller characteristics for both systems are provided in Table 1.

Data in Table 1 clearly establishes the SDP-175 as the superior airborne controller. Mitigating factors such as cost, ease of programming and lead time make the Andromeda system a viable competitor. Although the SDP-175 is 5-6 times faster than the LSI-11 CPU, recent HHC wind tunnel tests at the NASA/Langley Research Center demonstrated the ability of the Andromeda to provide updated HHC solutions at rates near 1P, using highly non-optimal control algorithms. An area of concern, however, is the ability of the laboratory controller to withstand a flight environment. Further trade-off studies in this important area are dictated in detail design.

3.0 HIGHER HARMONIC CONTROL SYSTEM MECHANICAL DESIGN

3.1 Preliminary Considerations

As stated in earlier sections, 3P, 4P and 5P harmonics of blade pitch needed to modulate blade root shear forces of like frequency can be generated using 4P swashplate collective, pitch and roll inputs. Accordingly, a primary design goal of the OH-6A HHC configuration was to accomplish both sensing of feedback forces and corrective blade feathering in the stationary system. Such an approach avoids the need for generating multiple frequencies (3P, 4P and 5P) of blade feathering, as would be the case in a rotating actuator system.

Confining the HHC hydraulic subsystem to the stationary system facilitates the adaptation of hydraulic power subsystems to the OH-6A. There is no requirement for a rotating hydraulic manifold and slip-ring assembly in this approach. Further, turbine power take-off generation of hydraulic power,

---

[1]Trademark Digital Equipment Corporation

TABLE 1. - CONTROLLER CHARACTERISTICS DATA

	11/B	SDP-175
Weight:	35 Kg, (78 pounds)	7 Kg, (15 pounds)
Power Dissipation:	175 Watts	40 Watts
Memory:	32K-words by 16 bit	24K ROM, 2K RAM by 16 bit Expandable to 64K
Processor:	General purpose, 16 bit data and instructions	General Purpose, 16-bit data and instructions
	Instruction times:	Instruction times:
	3.5 μsec add, subtract	0.5 μsec add, subtract
	30.0 μsec multiply	5.5 μsec multiply
	78.0 μsec divide	12.0 μsec divide
	3.5 μsec test, set bit	0.75 μsec test, set bit 800 KOPS for standard airborne mix
Software:	Assembly, Basic, FORTRAN Real time operating system	Assembly, Floppy disk operating system

I/O:	16 A/D Channels	Input	Output
	6 D/A Channels	4 AC	0 AC
		20 DC	20 DC
		16 28V Discrete	16 28V Discrete
		48 5V Discrete	56 5V Discrete
Functions:	Laboratory data acquisition	Airborne remote terminal bus, control avionics processing	

to be discussed in later sections, has no impact on the aircraft's autorotation capabilities, as would a rotor-mounted hydraulic power source Reference 2. Also, actuators and pressure lines are not required to operate in a centrifugal force field with this approach.

3.2 Actuator Subsystem Parametric Study

Parametric preliminary design studies were conducted for defining, evaluating and selecting a higher harmonic blade pitch control system for prototype flight tests on an OH-6A helicopter. Early in preliminary design, it was decided to limit the scope of study to those approaches not requiring a change in rotor mast height. This precluded the use of direct-connecting (stationary swashplate to mast base) vertical actuators.

Although attractive in that primary and HHC control inputs are implemented with a minimum of mechanical mixing, a single system actuator required a 17.78cm (7-in.) mast height increase, while a dual tandem actuator necessitated a 27.94cm (11-in.) mast height increase. Therefore, it was necessary to limit modifications to the non-rotating primary flight control system keeping intact the concept of a mixer.

Three approaches to HHC power and mechanization were studied. The approaches provide for different actuator configurations, locations, and control linkages. The systems selected for comparison studies can be characterized as follows:

1. Integrated series control actuators, Section 3.2.2

2. Swashplate link control actuators, Section 3.2.3

3. Combined primary and HHC actuators, Section 3.2.4

Each system is discussed and evaluated with design installation and assembly drawings. The three approaches are significantly different in their basic configurations to provide a basis for trade-off evaluations. Other forms of "prime mover" power were briefly considered (pneumatic, electrical, all mechanical). However, no alternatives to hydraulic power offer the advantages of frequency response and proven technology. Thus, all HHC actuator concepts are based on 2,069 N/cm^2 (3,000 psi), Type II hydraulic power, in order to meet the rigorous high frequency response requirements.

3.2.1 Baseline Primary Control System

The OH-6A primary flight control system is shown schematically on Figures 29, 30, and 31. The fully articulated main rotor utilizes a NACA 0015 blade. Low friction during blade pitch actuation is provided by a laminated steel strap retention system coupled with self-lubricating feathering bearings. Pilot inputs are mixed using a bellcrank system to provide the required collective and cyclic commands. Trim inputs are transmitted to the lower (non-rotating) swashplate by two push-pull links and an anti-torque link which serves to locate the lower swashplate. A hydraulic one-way lock device prevents longitudinal control loads from feeding back to the pilot.

The main rotor hub mounts via two angular contact roller bearings to a stationary mast, and is driven by the main rotor driveshaft within the mast. The mast carries all rotor loads, except torsion, and provides one means of obtaining HHC hub vertical, lateral and longitudinal feedback accelerations.

3.2.2 Integrated Series Control Actuators

This concept is an extension of a full power hydraulic stability augmentation system developed by Sperry in 1975. The control system was successfully flight tested on an Army OH-6A as part of the New Initiatives Aerial Scout (NIAS) Program. The Sperry configuration is illustrated in Figure 32.

Primary changes would involve the three boost actuators controlling blade pitch. Figure 33 illustrates modifications to the boost actuator to incorporate HHC capabilities. The SAS actuators as well as the tail rotor boost actuator are removed in this approach.

The primary change envisioned consists of adding a short-stroke independently-controlled servo to the upper end of the primary control actuator. The primary control actuator is a moving-body follow-up type having a piston rod and bearing attached to support structure. The upper end bearing of the body housing is attached to an output bellcrank arm. Thus, the augmented piston-housing-piston scheme operates in series. The lower main section of the actuator provides the primary control system travels, while the upper short-stroke piston generates HHC motions via the primary control mixer. The two actuating units are separate; however, there exists a common fluid supply routed from the same pressure and return ports, as illustrated in the sectional and porting schematics of Figures 34 and 35.

1. MAIN ROTOR DRIVE SHAFT
2. DAMPER
3. DAMPER ARM (LINK)
4. BLADE ATTACH PIN (TYP)
5. MAIN ROTOR BLADE
6. DAMPER LINK ATTACH PIN
7. MAIN ROTOR MAST BASE
8. ROTATING SWASHPLATE
9. MAIN ROTOR CONTROLS (MIXER, IDLER BELLCRANKS AND LINKS)
10. LATERAL MIXER CONTROL ROD (CYCLIC)
11. LONGITUDINAL MIXER CONTROL ROD (CYCLIC)
12. COLLECTIVE MIXER CONTROL ROD
13. STATIONARY SWASHPLATE
14. COLLECTIVE CONTROL TORQUE TUBE
15. PILOT'S COLLECTIVE PITCH STICK
16. DROOP CONTROL OVERRIDE LINK
17. PILOT'S CYCLIC STICK
18. LATERAL CYCLIC FRICTION KNOB
19. CYCLIC LATERAL CONTROL ROD
20. LATERAL CYCLIC TRIM ACTUATOR
21. LONGITUDINAL CYCLIC TRIM ACTUATOR
22. LONGITUDINAL CYCLIC FRICTION KNOB
23. ONE-WAY LOCK (LONGITUDINAL CYCLIC CONTROL)
24. CYCLIC CONTROL TORQUE TUBE
25. GAS PRODUCER CONTROL ROD
26. COLLECTIVE BUNGEE

Figure 29.- Main rotor and control system.

39
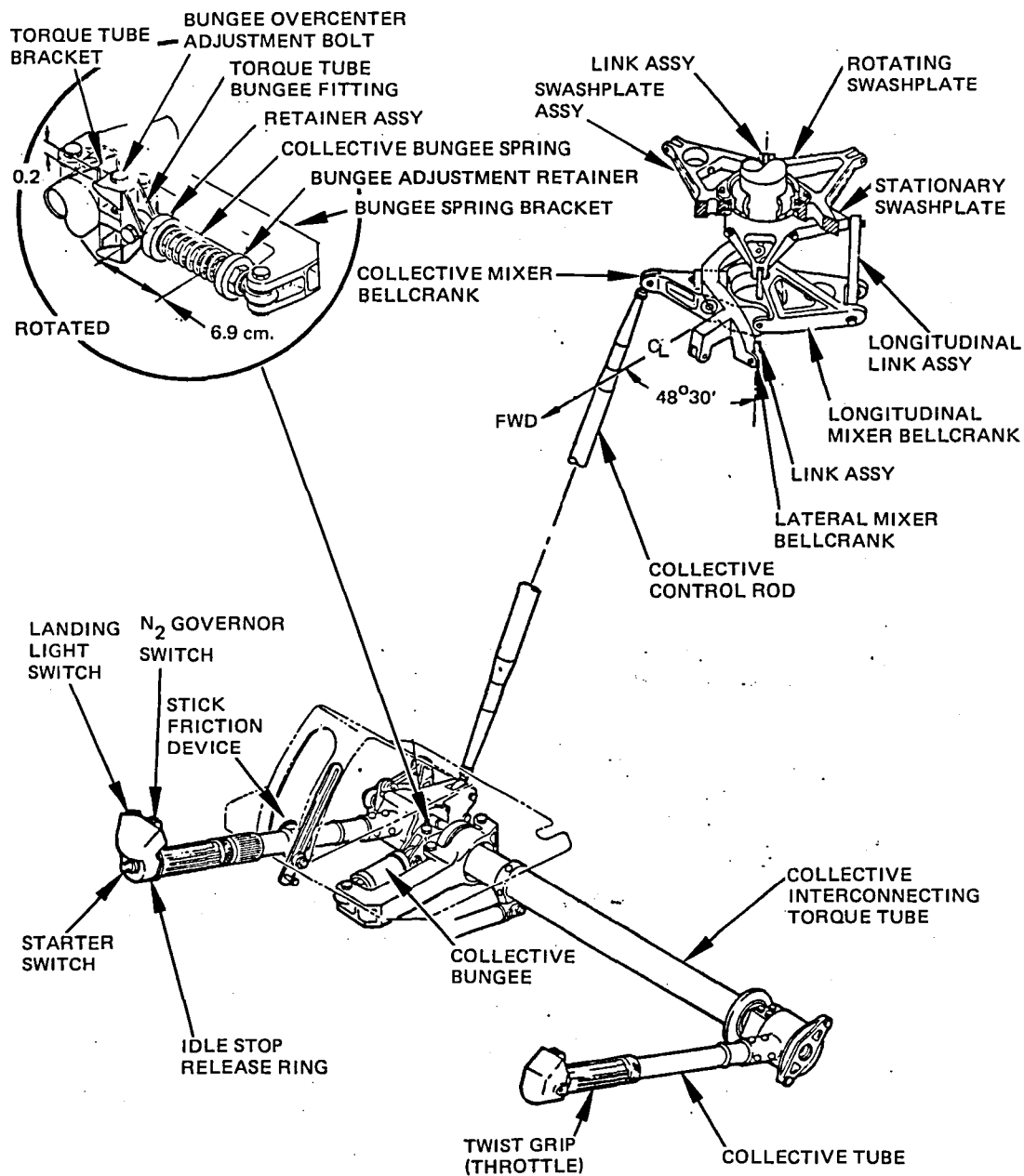
Figure 30. - Existing OH-6A main rotor collective control system.

CYCLIC PITCH CONTROL SYSTEM

LONGITUDINAL IDLER

LATERAL PITCH MIXER BELLCRANK

LONGITUDINAL CONTROL ROD

ONE-WAY LOCK

LONGITUDINAL TRIM ACTUATOR

LATERAL CONTROL ROD

PILOT'S CYCLIC CONTROL STICK

LATERAL TRIM ACTUATOR

LATERAL FRICTION DEVICE

LONGITUDINAL FRICTION DEVICE
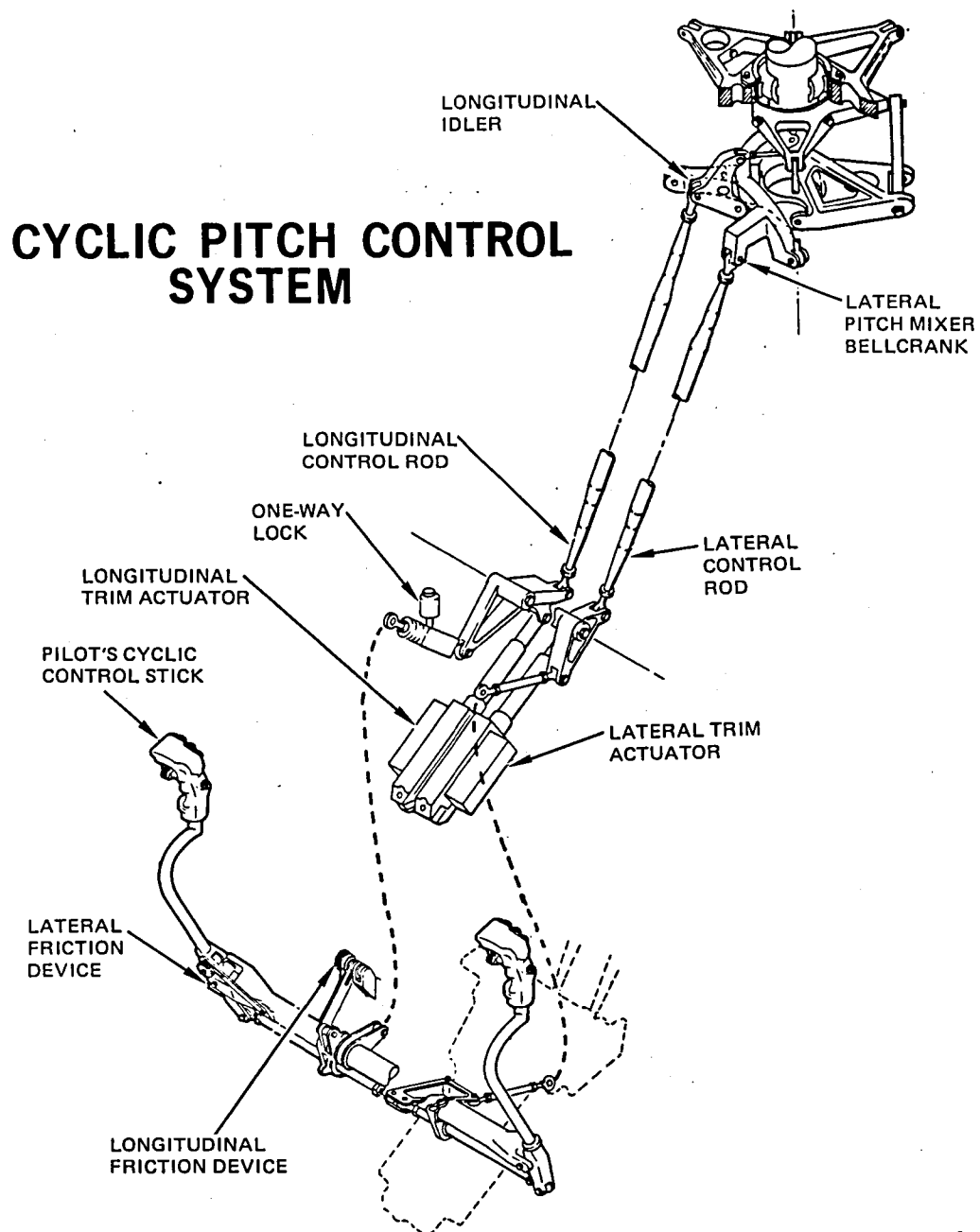
Figure 31.- Existing OH-6A main rotor cyclic control system.

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HYDRAULIC ACTUATORS

ELECTROMECHANICAL STABILITY
AUGMENTATION ACTUATORS
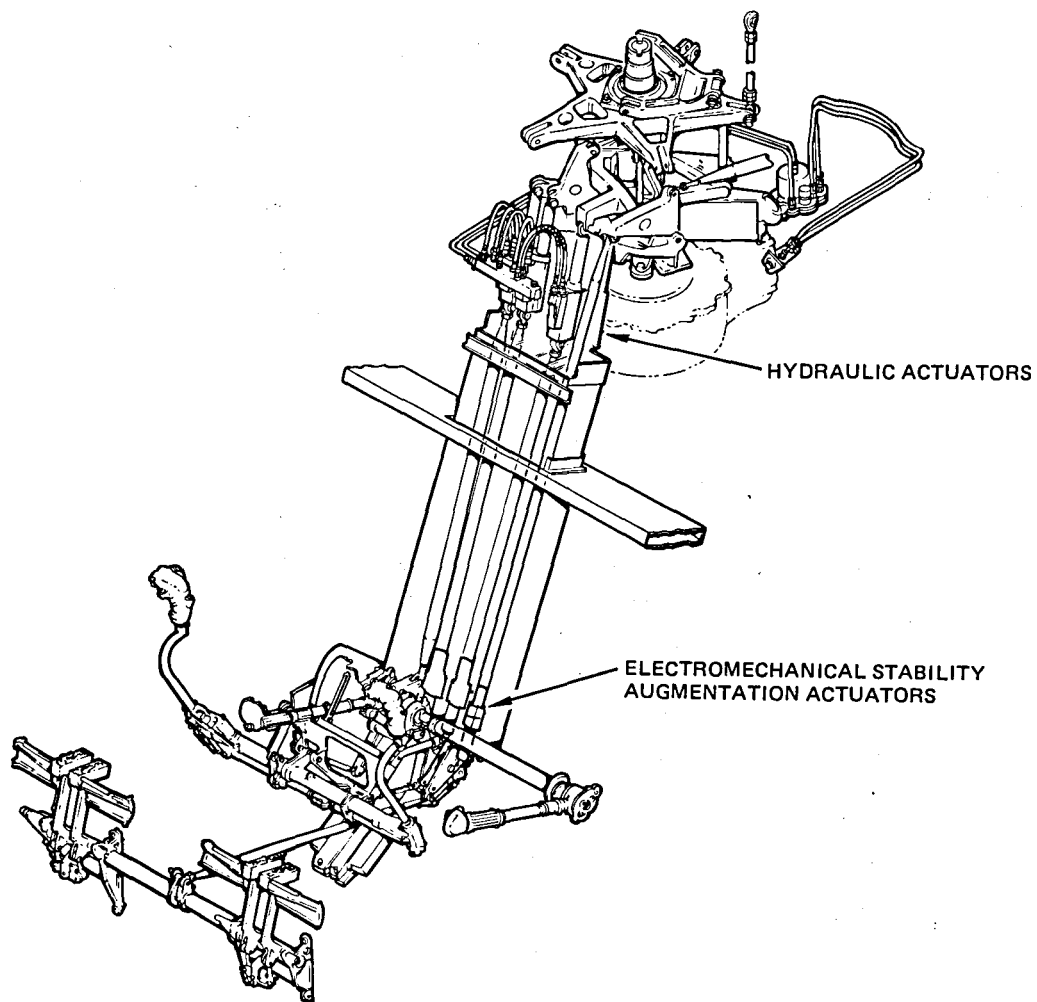
Figure 32. - OH-6A "Sperry" SAS flight control system.

MAIN SERVO VALVE, LOW
PRESSURE BYPASS DEVICE
AND " STUCK VALVE"
BYPASS DEVICE

₵ PISTON ROD TO BODY
LOCK MECHANISM

LVDT ROD
(OFFSET
TOWARD
FAR SIDE)

RETURN

PRESSURE

PILOT
INPUT

HIGHER HARMONIC MOTION PISTON
(ELECTROHYDRAULIC SERVOVALVE CONTROLLED)
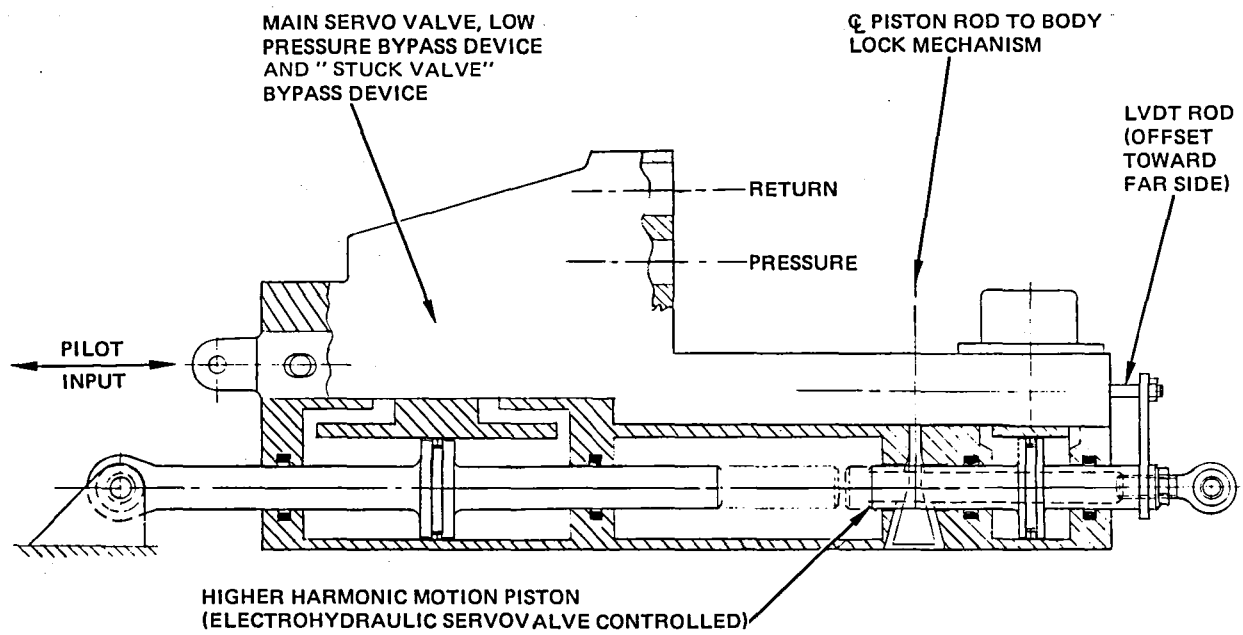
Figure 33. - Integrated series actuator schematic.

NOTE: ALL DIMENSIONS
IN CENTIMETERS

MOUNTING A MANIFOLD
PATTERN FOR SIZE IB
ELECTROHYDRAULIC SERVOVALVE
(SAE-ARP490C)

LVDT INSTALLATION

MAIN PRIMARY FLIGHT CONTROL ACTUATOR SECTION
SIMILAR TO HH SOURCE CONTROL DRAWING 369C7210
(USED FOR HUGHES OH-6A/SPERRY STABILITY
AUGMENTATION SYSTEM DEMONSTRATION HELICOPTER)
CONTAINS SERVOVALVE, LOW PRESSURE BYPASS DEVICE,
AND "STUCK VALVE" BYPASS DEVICE

RETURN PORT

PRESSURE
PORT

B

HHC PISTON ROD TO ACTUATOR BODY LOCK
MECHANISM (SHOWN IN UNLOCKED POSITION)

MAX HIGHER HARMONIC CONTROL STROKE

₵ MAIN CONTROL
VALVE SPOOL

A

0.955

0.478

0.478

35.56 AT MIDSTROKE

0.046 (NOMINAL) EACH SIDE OF
NEUTRAL (POSITIVE STOP AT
VALVE INPUT SPOOL)

B

1.19 LOCK TRAVEL

2.54

5.08

2.54

A

MAX PRIMARY
CONTROL STROKE

Figure 34. - Integrated series actuator design.

CL HYDRAULIC UNLOCK
MECHANISM PISTON
(CONNECTED TO
PRESSURE PORT)

INTERNAL HYD
FLUID RETURN
PORTING

INTERNAL HYD
FLUID PRESSURE
PORTING

VIEW A-A

SECTIONAL VIEW B-B

Figure 35. - Integrated series actuator porting diagram.

The major design, installation and operating features of adapting the series
actuator approach to the OH-6A are:

a. Actuators become larger (especially lengthwise) and are
   likely to intrude into the cockpit space in the upper tunnel
   area.

b. Flexible hydraulic lines are required which protrude out
   the top of the fuselage above the control tunnel.

c. A means or device for stablizing this type of actuator
   about its long axis is required. This may take the form
   of scissors, roller and track, or links.

This configuration cycles the mixer bellcranks in addition to the swashplate
pair and remaining upper blade pitch control linkage. A potential problem
of degraded frequency response exists, born out of control system inertia
and structural compliance. Efforts are best directed to placing the actuator
system "higher" in the control system if 4P frequency response is a
problem.

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In the event hydraulic pressure is lost, integrity of the control system is maintained. The primary side of the actuator is fail-safe since valve inputs under pressure are converted into body inputs with loss of pressure. This is accomplished with a low pressure bypass valve in the primary body. The HHC side of the actuator is similarly fail-safe under loss of pressure due to a piston rod-to-actuator body locking mechanism. The fail-safe features of the device, coupled with its low authority (<10 percent) HHC capability, precludes the need of a redundant capability.

An installation schematic of the HHC system utilizing series control actuators is provided in Figure 36.

3.2.3 Swashplate Link Control Actuators

Figure 37 illustrates in perspective how three short stroke electrohydraulic actuators are attached directly to the lower swashplate to generate the required 4P blade feathering. The components to be replaced are two lateral cyclic system push-pull links and a longitudinal cyclic system scissors links. Two flexible hydraulic hose assemblies and a small wire harness assembly are required for each actuator to accommodate vertical translations due to primary inputs.
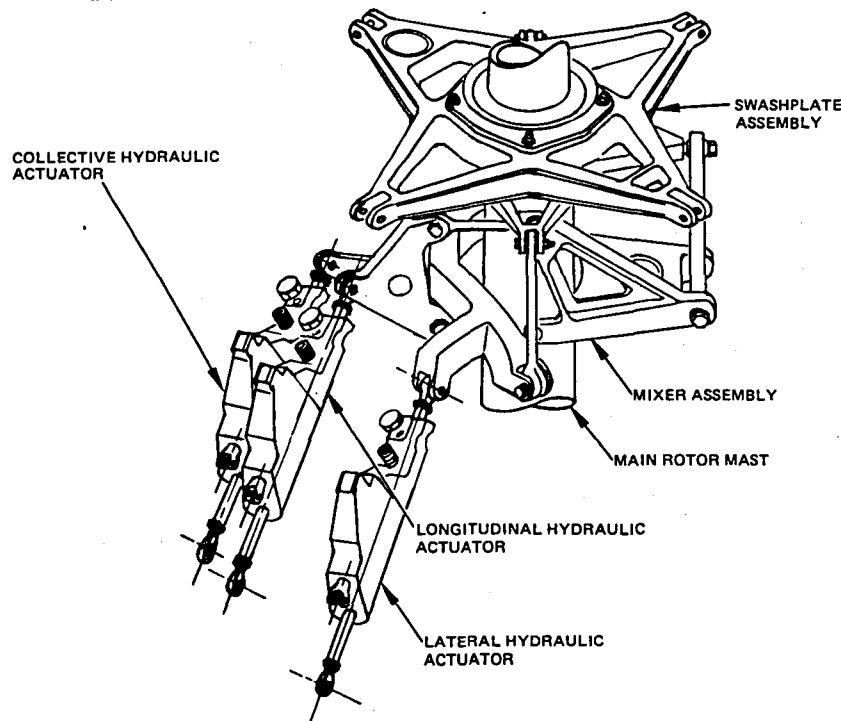
Figure 36. - Integrated series control actuator configuration.

UPPER (ROTATING) SWASHPLATE

LOWER (NONROTATING) SWASHPLATE

HIGHER HARMONIC HYDRAULIC ACTUATOR ASSEMBLY (3 PLACES)

MIXER ASSEMBLY
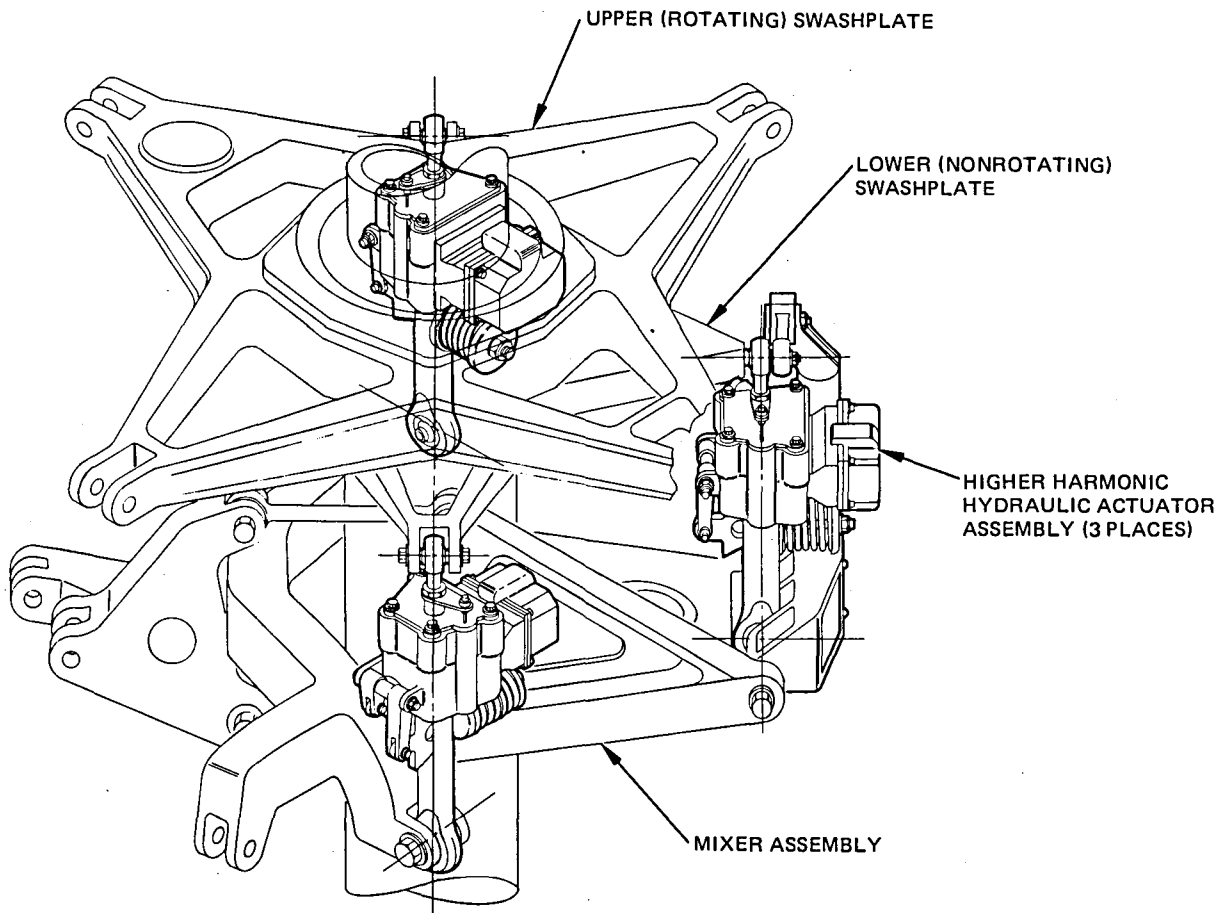
Figure 37. - Swashplate link control actuators.

A detailed system design layout is illustrated in Figure 38. The main view along with the auxiliary view projections were done to establish clearances among components. As can be seen, the drawing substantiates the feasibility of the approach by verifying that the small (mid-stroke length is 15.875cm (6.25 in.)) actuators can be utilized in the available space. The major design and porting details of the actuator and its functional components are given in Figures 39, 40, and 41. Except for angular relationships of the rod end bearings, the actuator configuration is the same for all three locations. This has been accomplished by tailoring the locations of the LVDT, servovalve and pressure and return port bosses to suit all three locations. A piston-to-body lock mechanism described in Section 3.2.2 is again provided for control system integrity under loss of pressure.

Anti-torque restraint for the lower swashplate is provided by a link assembly detailed in Figure 42. The lower fitting of the scissors assembly provides the center support point for the lower actuator bearing, and supports the vertical tube that mounts the short upper link. The upper link extends horizontally to connect with the lower swashplate.

VIEW D-D
(LOOKING AFT)

¢ HELICOPTER

¢ LH LATERAL ACTUATOR (REF)

VIEW C-C

LH LATERAL + COLLECTIVE
HHC ACTUATOR ASSEMBLY

¢ RH LATERAL
ACTUATOR (REF)

(369C7210 ACTUATOR-HYDRAULIC
SERVO, PRIMARY FLIGHT CONTROL
TYP 3 PLACES

369A7605 BELLCRANK,
LONGITUDINAL PITCH-MIXER
(REF)

AEROQUIP AE206 HIGH
PRESSURE HOSE (OR
EQUIV) TYP

C

D FORWARD

369A7606 BELLCRANK,
LATERAL CYCLIC-MIXER
(REF)

A

41°30'

B

RH LATERAL +
COLLECTIVE HHC
C ACTUATOR ASSEMBLY

D

369A2015 TUBE-MAST
(REF)

369A2016 FITTING, MAST
BASE (REF)

WL 60.00

PLAN
VIEW

WL 62.94

STA 100.00

369A7516 BELLCRANK,
TAIL ROTOR CONTROL
STA 95.00 (REF)

B

A

B

VIEW B-B

VIEW A-A
(LOOKING INBOARD
FROM LH SIDE)

LONGITUDINAL +
COLLECTIVE HHC
ACTUATOR ASSEMBLY

ANTI-TORQUE LINK
ASSEMBLY — NON-ROTATING
SWASHPLATE

¢ MAIN ROTOR

Figure 38. – Design layout – swashplate link actuators installation.

REX-SHAFER TGA-204 BEARING
(OUTER RING GROOVED FOR
ROLL SWAGE RETENTION)

LOCK MECHANISM BELLCRANK

⊄ HYDRAULIC UNLOCK
MECHANISM PISTON
(CONNECTED TO
PRESSURE PORT)

15.88

(0.376 TOTAL STROKE)

0.478

0.478

1.19 LOCK TRAVEL

PISTON ROD TO
BODY LOCK
MECHANISM
(SHOWN IN
UNLOCKED
POSITION)

REX-SHAFER TFM-404W
ROD END BEARING
(MIL-B-8948) OR EQUIV

MOUNTING AND MANIFOLD
PATTERN FOR SIZE IB
ELECTROHYDRAULIC SERVOVALVE
(SAE-ARP490C)

NOTE: ALL DIMENSIONS
IN CENTIMETERS

Figure 39. - Swashplate link actuator assembly.

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LVDT ARMS TO HAVE KEY TABS PER
NAS-513 LOCK WASHERS FOR DIFFERENT
ROD END BEARING ORIENTATIONS (SEE
INSTALLATION LAYOUT)

PRESSURE PORT

RETURN PORT

A

SECTION A-A

A

LVDT INSTALLATION

LOCK MECHANISM BELLCRANK

15.88 cm
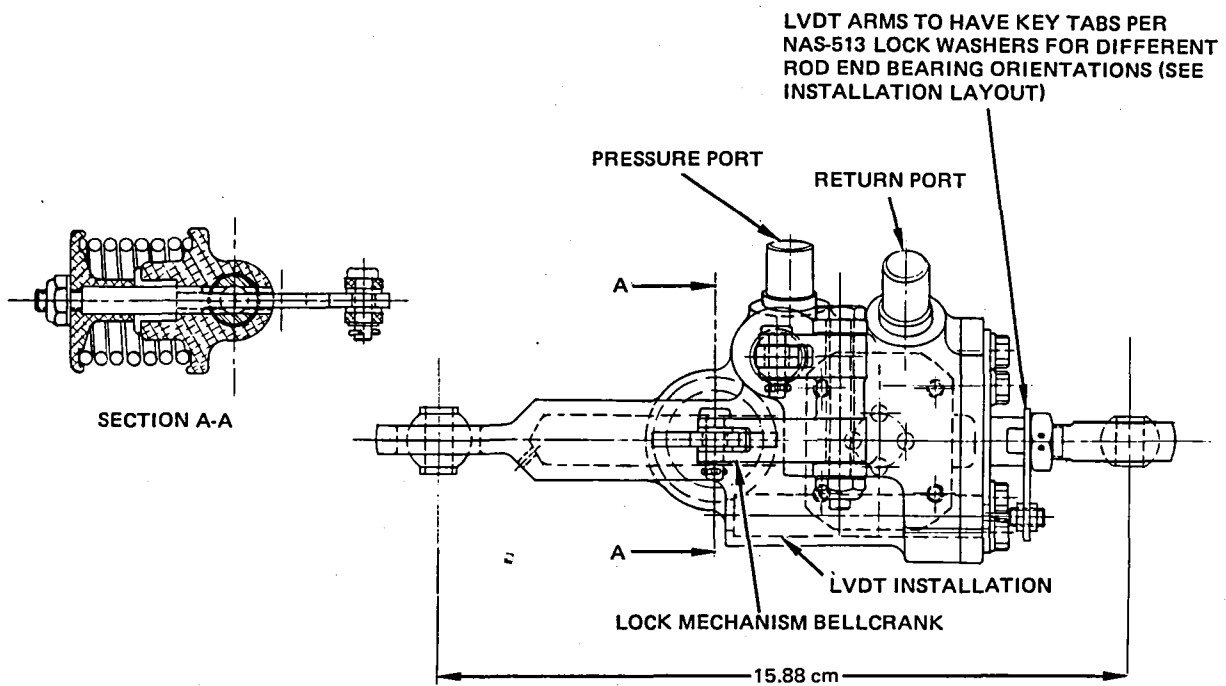
Figure 40. - Swashplate link actuator assembly.

CONTROL PORT C_1

NAS 1395 C06L INSERTS (OR EQUIV TYPE) 4 PLACES

PRESSURE PORT

LOWER END BEARING ROTATED 90° FROM MAIN VIEW FOR TWO LATERAL SYSTEM ACTUATORS

CONTROL PORT C_2

LOWER END BEARING ORIENTATION SHOWN FOR LONGITUDINAL SYSTEM ACTUATOR

NAS 1394 C3L INSERTS (OR EQUIV TYPE) 4 PLACES IN MAIN BODY, CLEARANCE HOLES THRU END CAP
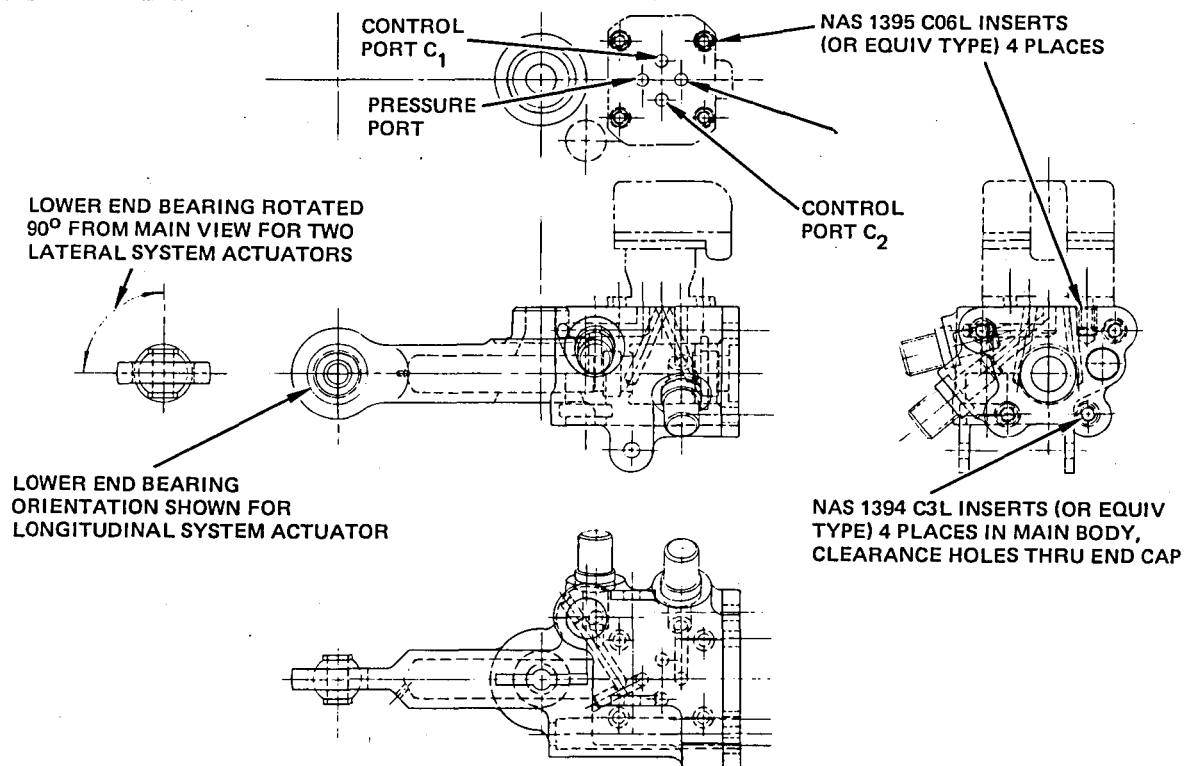
Figure 41. - Swashplate link actuator porting diagram.
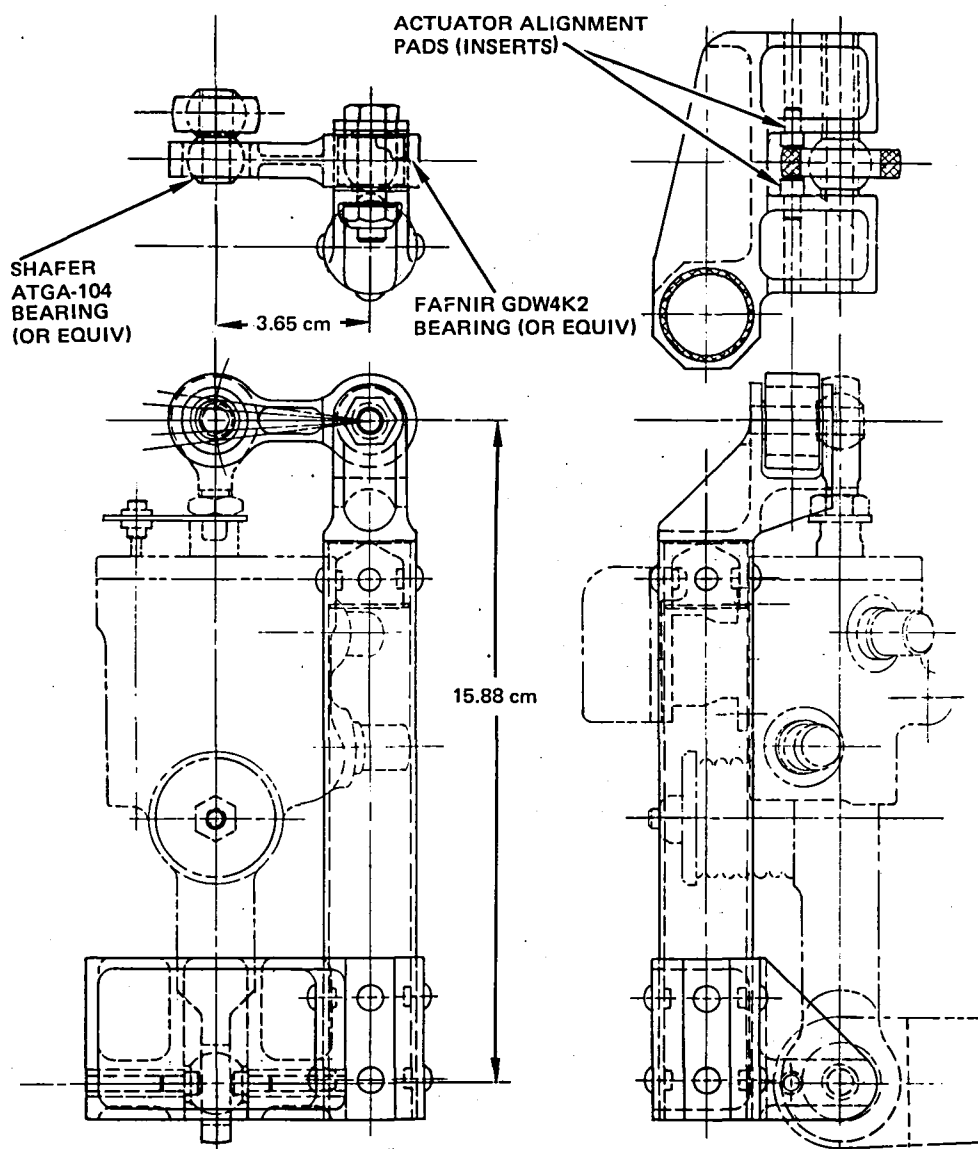
Figure 42. - Swashplate link anti-torque scissors assembly.

As compared to the series actuator approach, installation of the actuators nearer the swashplate assembly reduces the weight of mixer components cycled during HHC blade feathering by 48 percent. This not only reduces the actuator workload and reaction forces, but greatly improves the 4P feathering frequency response.

3.2.4 Combined Primary and HHC Actuators

This system study evolved as a result of an attempt to locate total primary and HHC hydraulic control as near to the blade pitch arm as is functionally feasible. As discussed earlier, there existed constraints precluding actuators or other hydraulic components in the rotating system. Thus a preliminary kinematic study was performed wherein three servo-actuators are mounted near to and parallel to the present main rotor mast support fitting, Figure 43. Having defined the preliminary travels and kinematic points, a means by which longitudinal, lateral and collective control motions are converted into mixed control valve travels was devised. Figures 44 and 45 provided overall and exploded view schematics of the resultant primary boost control and HHC system. As noted in the drawings, the three hydraulic actuator/linkage assemblies are appropriately located around the swashplate azimuth in order that the vertical push-pull links correctly position the swashplate.

The mixer configuration is designed to take advantage of the necessary change in direction of linkages required at the top of the control tunnel. The collective control travels can be combined with cyclic control travels, as noted in Figure 46, without introducing additional bellcranks into the system.

The actuator at each point has a single piston rod for all output control functions. The main valve assembly is a multi-functional unit with concentric spools as shown in Section B-B of Figure 47. The inner spool is for the fluid bypass function. Fluid is bypassed when supply pressure falls below a predetermined value, thus permitting control inputs to be made manually. An intermediate main spool is controlled mechanically by the pilot for primary control motions as noted in Section A-A of Figure 48. The outer spool is controlled by the electrohydraulic servovalve and thereby performs the higher harmonic control functions in the valve assembly. Each of the actuators is functionally identical, and except for right-hand and left-hand linkages, has identical components.

Primary control motions are governed and mixed by two groups of bellcranks: the mixer collective bellcranks and the valve input bellcrank assembly. The combination bellcrank assembly at the top of the control tunnel, Figure 45, is a major mixer component. The main collective bellcrank supports three cyclic bellcranks, each mounted on a common axis raised vertically above its

NOTE: SEE PLAN VIEW (SHEET 3) FOR
      FURTHER CONTROL LINKAGE
      ROUTING INFORMATION, ACTUATOR
      LOCATIONS, AND ANGULAR
      RELATIONSHIPS

WL 83.00

₵ MAIN ROTOR MAST

3°00' (REF)

FORWARD

UPPER (ROTATING) SWASHPLATE
SHOWN IN MID-COLLECTIVE AND
ZERO (NEUTRAL) CYCLIC POSITION

MIXER COLLECTIVE
BELLCRANK SUPPORT
PIVOT AXIS

₵ LATERAL PUSH-PULL
RODS (SEE SHEET 2)

₵ ACTUATOR

₵ ACTUATORS

TOP OF 369A2016
MAST FITTING ASSY

₵ COLLECTIVE
PUSH-PULL ROD

₵ LONGITUDINAL
PUSH-PULL ROD

COCKPIT CONTROL
TUNNEL AREA

FUS STA 100
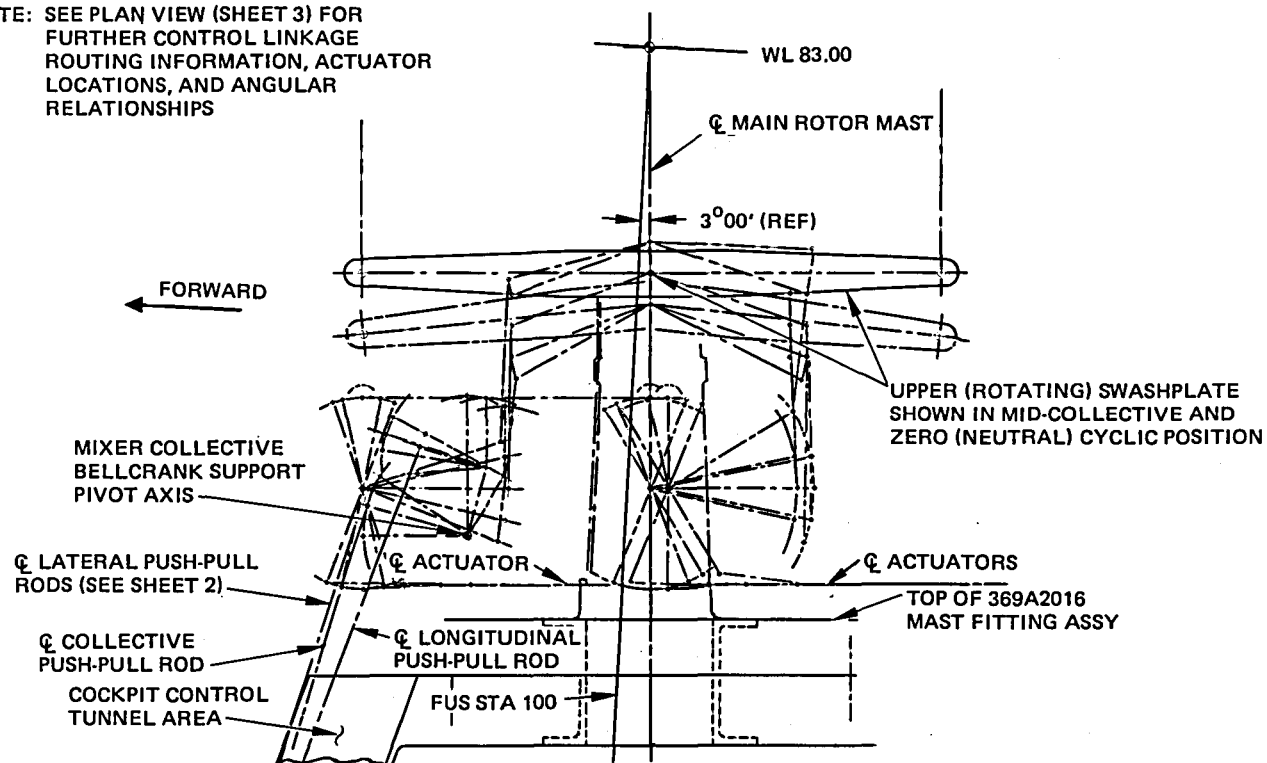
Figure 43. - Combined primary and HHC configuration kinematics.

1. MAIN ROTOR BLADE PITCH

2. ELECTROHYDRAULIC SERVOVALVE
   (TYPICAL, 3 PLACES)

3. R.H. LATERAL/COLLECTIVE ACTUATOR

4. LINK TO NONROTATING (LOWER) SWASHPLATE
   (TYPICAL, 3 PLACES)

5. LONGITUDINAL/COLLECTIVE ACTUATOR

6. L.H. LATERAL/COLLECTIVE ACTUATOR

7. ACTUATOR TO BELLCRANK LINK

8. L.H. LATERAL CYCLIC CONTROL ROD

9. LONGITUDINAL CYCLIC CONTROL ROD

10. TAIL ROTOR CONTROL ROD

11. COLLECTIVE CONTROL STICK (2 PLACES)

12. TAIL ROTOR CONTROL PEDALS

13. CYCLIC CONTROL STICK (2 PLACES)

14. R.H. LATERAL CYCLIC CONTROL ROD

15. COLLECTIVE CONTROL ROD

16. MIXER COLLECTIVE BELLCRANK ARM

17. PILOT INPUT ROD (FROM MIXER ASSEMBLY)

18. VALVE ACTUATING LINK

19. ROTATING
    SWASHPLATE
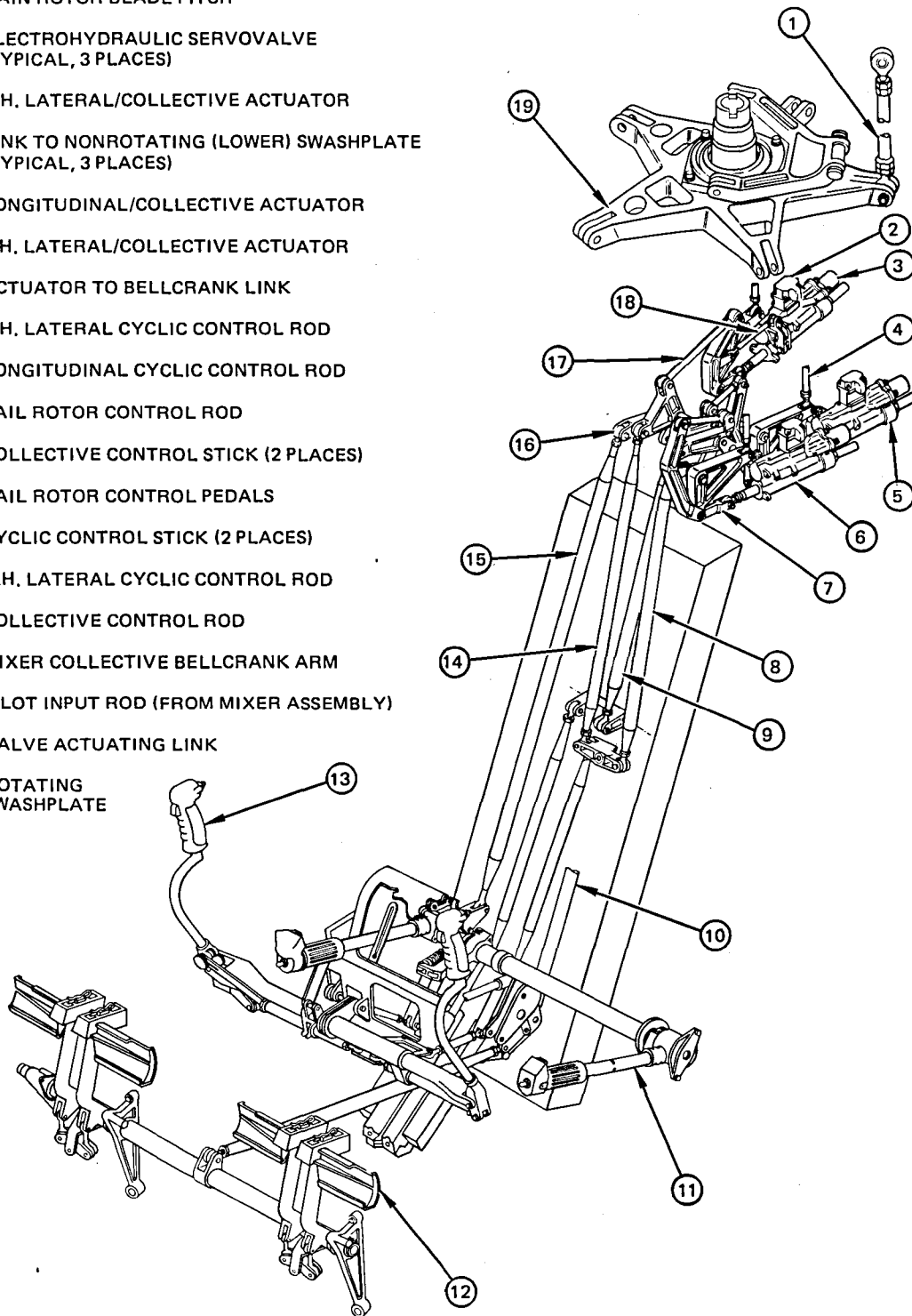
Figure 44. - Combined primary and HHC actuator configuration.

55

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PILOT TO VALVE ACTUATING
BELLCRANK (1 PLACE)

RETURN PORT

PRESSURE PORT

₵ LINK TO VALVE ARM
(TYPICAL)

LINK TO ACTUATOR
(TYPICAL)

VALVE ACTUATING ARM

₵ LINK TO RH LATERAL/
COLLECTIVE ACTUATOR

R.H. LATERAL/COLLECTIVE
ACTUATOR (TYPICAL)

LINK TO NONROTATING (LOWER)
PART OF SWASHPLATE

COLLECTIVE BELLCRANK
ARM

₵ LINK TO LONGITUDINAL/
COLLECTIVE ACTUATOR

₵ LINK TO LH LATERAL/
COLLECTIVE ACTUATOR

ACTUATOR TO LH CYCLIC/
COLLECTIVE SWASHPLATE LINK
BELLCRANK (TYPICAL)

MIXER COLLECTIVE BELLCRANK
(SUPPORTS SEPARATE ACTUATOR
VALVE INPUT BELLCRANKS)

MAIN BELLCRANK SUPPORT
AXIS (TYPICAL)

PILOT TO VALVE ACTUATING
BELLCRANK (TYPICAL 2 PLACES)

BELLCRANK SUPPORT LUGS

Figure 45. - Details of combined primary
HHC control system.

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CONTROL TUNNEL

MIXER COLLECTIVE
BELLCRANK

COLLECTIVE

RH LATERAL CYCLIC

⊄ HELICOPTER

LONGITUDINAL CYCLIC

LATERAL BELLCRANK
PIVOT AXIS   LH LATERAL CYCLIC

MIXER COLLECTIVE
BELLCRANK SUPPORT
BRACKET CLEVISES

COLLECTIVE, LONGITUDINAL AND LATERAL
PUSH-PULL CONTROL RODS FROM LOWER
TUNNEL BELLCRANKS

FORWARD

LONGITUDINAL BELLCRANK
PIVOT AXIS
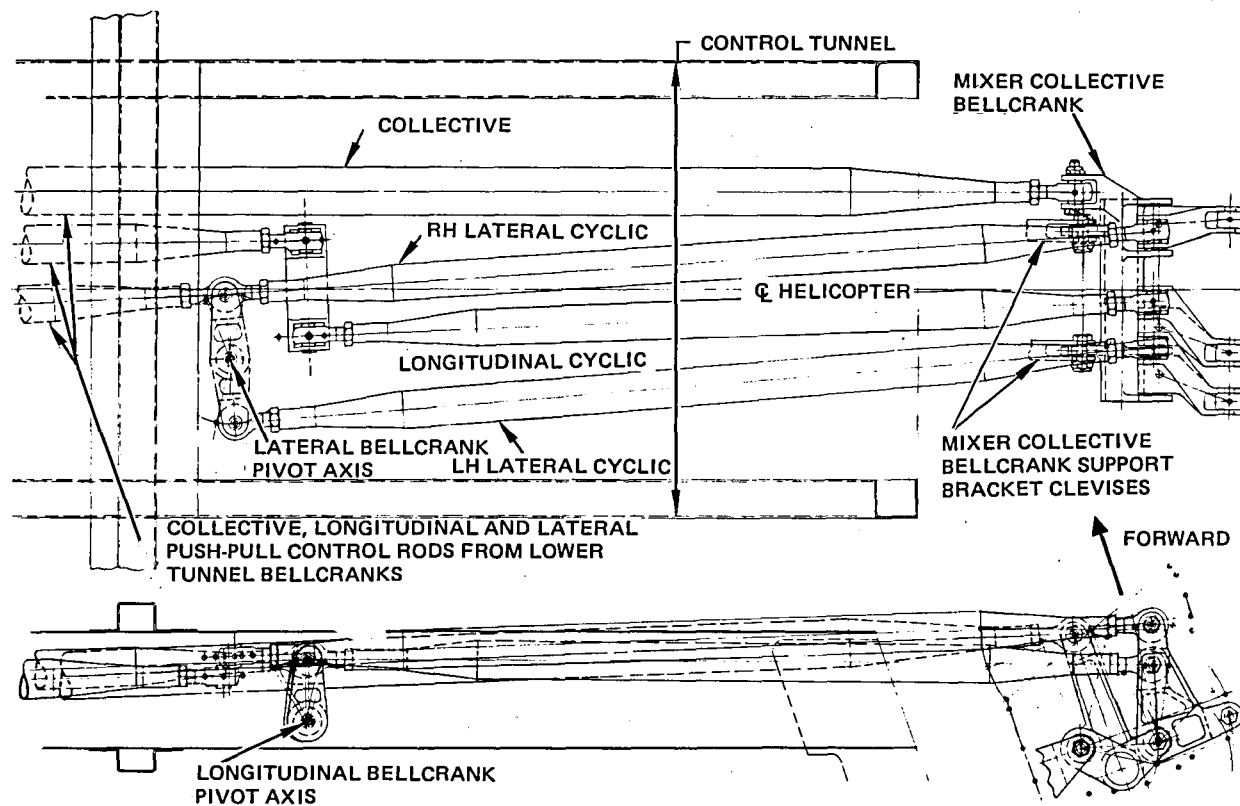
Figure 46. - Combined primary and HHC actuators - tunnel detail.

PRESSURE PORT

RETURN PORT

2.438 RETRACTED
6.248 NEUTRAL
9.042 EXTENDED

6.604
TOTAL STROKE

ELECTROHYDRAULIC
SERVOVALVE

NOTE: ALL DIMENSIONS
IN CENTIMETERS

2.54

5.398

0.356

0.953   2.380

17.027

4.763

22.05
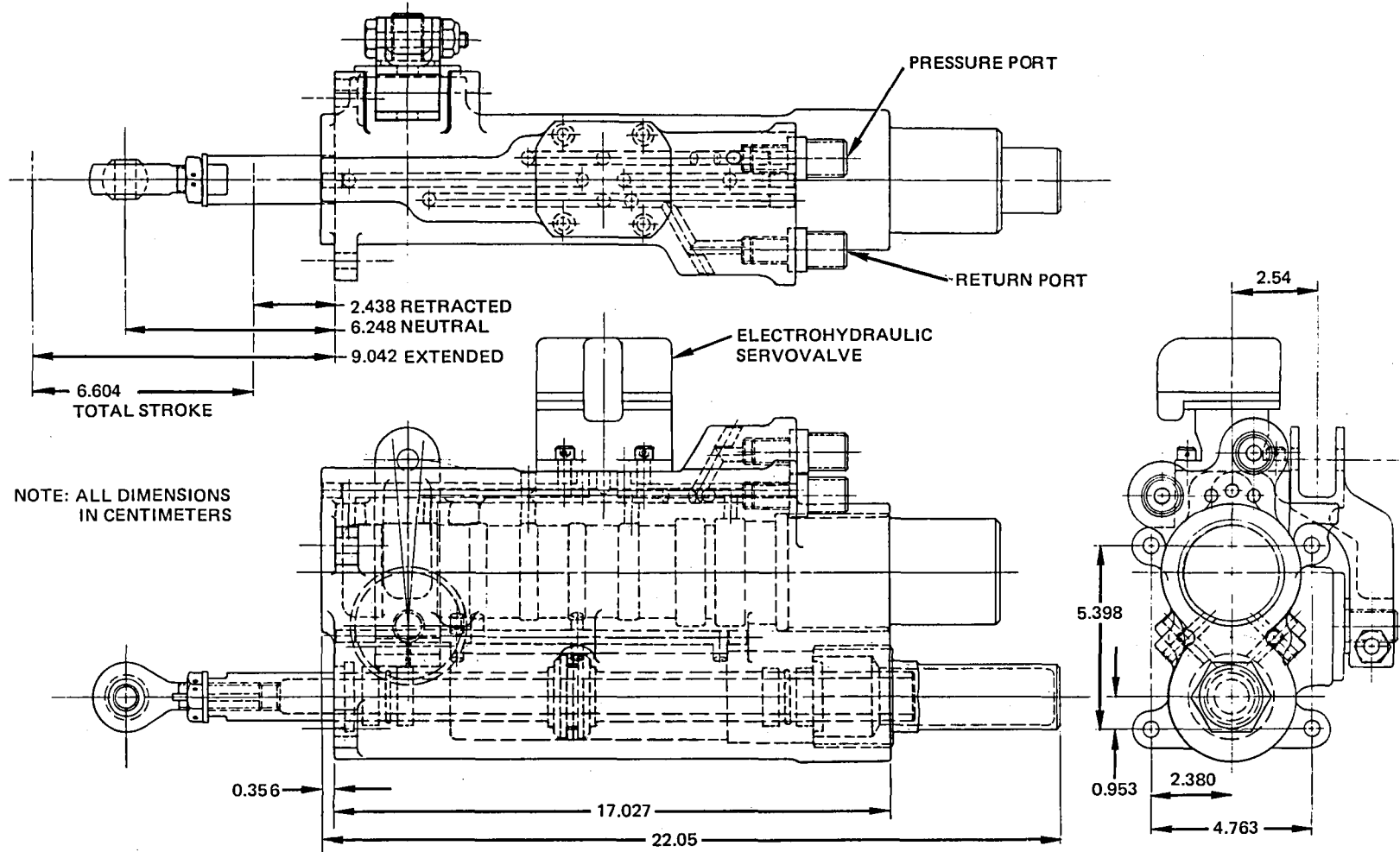
Figure 47. - Combined primary and HHC actuator design.
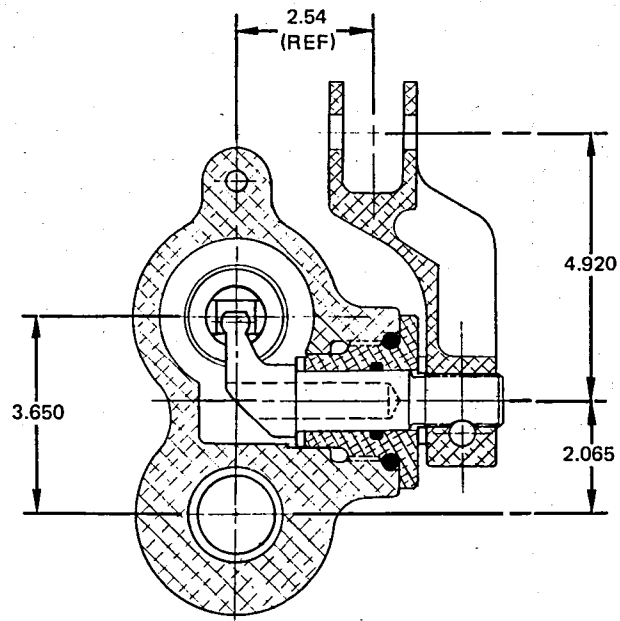
SECTION A-A

NOTE: ALL DIMENSIONS
IN CENTIMETERS
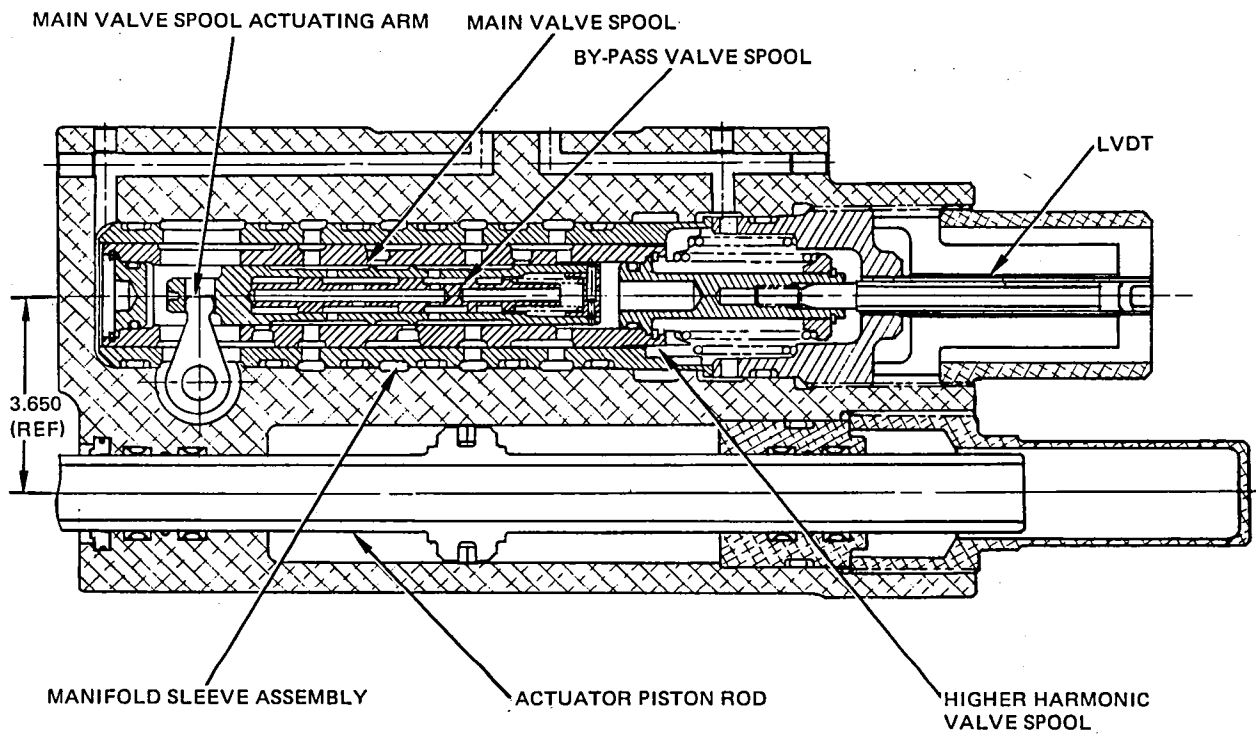
SECTION B-B

Figure 48. - Combined primary and HHC actuator design cutaway.

basic pivot axis. Controlled by a collective push-pull rod, an integral arm of this bellcrank controls bellcrank assembly angular motion. Such collective bellcrank motion translates all three cyclic bellcranks in a longitudinal direction, since the cyclic bellcranks are restrained at one end by the undeflected cyclic control rods. Thus uniform collective command motion is transmitted to all three actuator valve arms.

Longitudinal cyclic command inputs operate the center cyclic bellcrank. These motions sum in series with any collective motions transmitted to the actuator. Lateral cyclic travels differentially operate the left and right outer cyclic bellcranks, similarly summed with collective control motions.

A second bellcrank assembly exists for each actuator installation. Main collective bellcrank command inputs are transmitted to the actuator valve arm via the actuating bellcrank. This initial rotation of the valve arm proportionally displaces the valve spool. The displaced valve spool ports fluid under pressure to the appropriate side of the actuator piston and opens the opposite chamber return port. Responding in the commanded direction to differential fluid pressure, piston rod movement returns the valve actuating link to its neutral position, thereby centering the valve spool and stopping fluid flow. Higher harmonic blade motions are controlled by an electrohydraulic servovalve and accomplished using the dual concentric spool/main hydraulic slide valve assembly, discussed earlier.

This control system configuration offers several advantages over the configurations discussed earlier. Control loads are reacted by an area of high structural impedance (i. e. , the mast base). The actuators are mounted upstream of the mixer for improved 4P feathering frequency response. There exist no practical space constraints, thus permitting the use of dual tandem or higher redundancy, multi-function actuator concepts. A greater potential exists for reducing the control system flexibility since a minimum of linkage is required in the load-carrying or reacting path. It is noted, however, the configuration under discussion requires a substantially higher level of effort in terms of design, manufacturing and proof testing for its implementation.

3. 3 Design Features of HHC Actuation System

The three HHC power actuation concepts discussed earlier are equally capable of providing blade root pitch at a frequency equal to the third, fourth and fifth harmonics of blade passage. All three HHC actuation system approaches are sized to provide a maximum collective 4P pitch amplitude of two degrees, including control system flexibility. The four degree increase in total blade

pitch requirements are well within main rotor retention strap limits of twist. Design studies utilizing data obtained from NASA/Langley Research Center scale model test indicate two degrees of 4P pitch amplitude to be an adequate margin of control at moderate advance ratios ($\mu = 0.3$). Recent studies, however, Reference 15, have demonstrated potentially higher amplitude requirements at low and high advance ratios, underscoring the need for further study in the detail design cycle.

It is proposed that fluid flow and horsepower requirements be based on average, rather than peak, flow associated with full stroke collective 4P feathering at 523 RPM (10 percent above nominal). A 163.9 cc, (10 cu. in.), accumulator pre-charged with 1,379 N/cm^2 (2,000 psi) nitrogen pressure provides sufficient pump inlet pressure under peak demand conditions. Table 2 presents a comparison of physical characteristics for the three HHC system concepts. The actuator configurations providing primary flight control motions are assessed an additional 20 percent workload in terms of fluid flow.

TABLE 2. - ACTUATOR PHYSICAL CHARACTERISTICS

| Characteristic	Configuration		
	Series Integrated	Swashplate Link	Combined Primary and HHC
Function	Primary/HHC	HHC only	Primary/HHC
Actuator Length, cm., (in.)	35.56 (14.0)	15.88 (6.25)	28.30 (11.14)
Total stroke, cm., (in.)	5.08 (2.0)	0.97 (0.38)	6.60 (2.60)
Actuator thrust, N., (lbs.)	2606.00 (586.0)	2606.00 (586.00)	2606.00 (586.00)
Peak total flow, lpm, (gpm)	42.80 (11.3)	35.60 (9.40)	42.80 (11.30)
Average total flow, lpm, (gpm)	27.30 (7.2)	22.70 (6.00)	27.30 (7.20)
Total required horsepower	12.60 HP	10.50 HP	12.60 HP
Fuel consumption, Kg/hr, (lb/hr)	+3.58 (+7.9)	+2.99 (+6.60)	+3.58 (+7.90)

Fuel consumption figures are based on C18 installation losses of
0.5 lb/horsepower/hour at sea level standard, and plant efficiency of
80 percent. Design provisions have been made for using Shamban-type dual
teflon/steel piston rings and advanced piston rod wiper seals on all three
actuator concepts. Thus, problems with seal wear and reliability as a
result of high piston velocities are not anticipated.

Safety of flight considerations require that each configuration be capable of
maintaining control system integrity under loss of hydraulic pressure. The
integrated series actuator and combined primary and HHC actuator provide
for fluid bypass when pressure falls below a predetermined value. Thus,
pilot commands to the main control valve simply bottom out the value by
0.46 cm, (.018 in.), and translate the main body as a rigid link. The HHC
portion of the integrated series actuator and the swashplate link actuator
provide for pressure loss with a piston rod-to-body locking mechanism. The
fully-depressed locking spring exerts a 872 N., (196 pound) force on a locking
cylinder piston, equilibrated by ambient hydraulic pressure acting on the
1.27 cm, (1/2 in.) diameter locking piston. As hydraulic pressure falls
below 690 N/cm^2, (1,000 psi), equilibrium is lost and the locking wedge
centers and locks the main piston rod assembly. Lock-up pressure is varied
by changing spring rates.

3.4 Hydraulic Power Subsystem

Features of the hydraulic power subsystem as required for introducing 4P
swashplate motion on an OH-6A are illustrated schematically in Figure 49.
A single hydraulic supply system is proposed, since for all actuator con-
figurations discussed earlier, loss of hydraulic pressure causes the control
system to revert to normal operation. The system is pressurized by a
variable delivery, pressure-compensated hydraulic pump as used in Type II,
2,069 N/cm^2 (3,000 psi) hydraulic systems in accordance with MIL-H-5440.
Driven by a spare power take-off pad on the Allison 250-C18 turboshaft
engine, the pump is rated at approximately 30.3 Lpm, (8 gpm) at 6,000 RPM.
Pressure compensation insures the minimum discharge pressure at maximum
flow will be no less than 90 percent of the discharge pressure at zero flow
demand under all conditions.

The fluid reservoir is an integrated reservoir/manifold assembly identical
to that developed by Bertea for the Phase I YAH-64 AAH. The 491.6 cc,
(30 cu. in.) fluid reservoir is pressurized with regulated 20.7 N/cm^2,
(30 psi) bleed air to prevent pump cavitation. A 163.9 cc, (10 cu. in.)
accumulator charged with 1,379 N/cm^2 (2,000 psi) nitrogen provides
sufficient pump inlet pressure under peak demand conditions. The reservoir/
manifold assembly incorporates ground service connectors, fill restrictors,

7-2116210002 RESERVOIR/MANIFOLD

P→6S-020

TO RTN

7-211810011 PUMP

P→6S-020

GROUND SERVICE CONN

S←6A-028

R→4A-028

R←6A-028
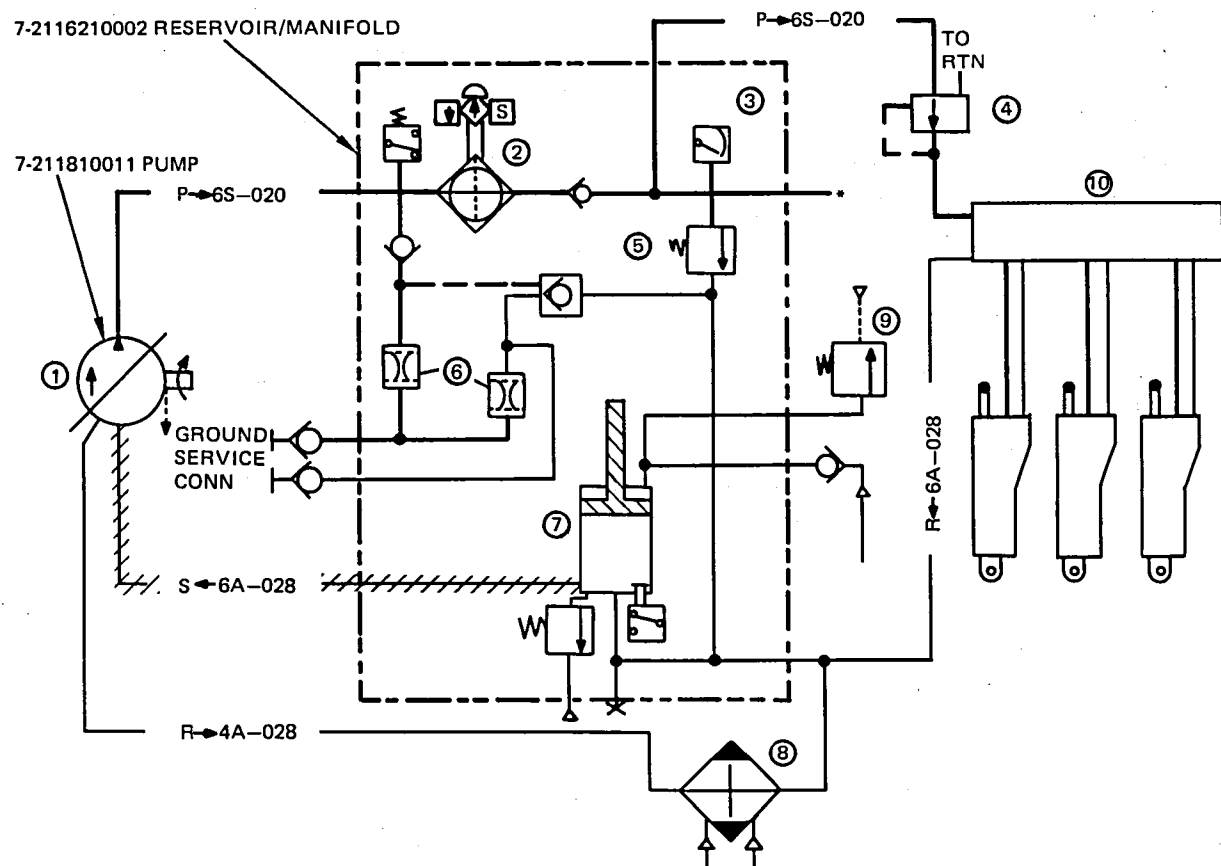
KEY

① ENGINE DRIVEN, VARIABLE DELIVERY PUMP

② TEMPERATURE-COMPENSATED FILTER

③ PRESSURE TRANSDUCER

④ PRESSURE REGULATOR

⑤ HIGH PRESSURE RELIEF VALVE

⑥ FLOW RESTRICTERS

⑦ FLUID RESERVOIR (AIR PRESSURIZED)

⑧ HEAT EXCHANGER

⑨ AIR PRESSURE RELIEF VALVE

⑩ HHC ACTUATORS

Figure 49. - HHC system hydraulic schematic.

temperature compensated 15-micron pressure line filter with dirty filter indicator, a high pressure relief valve, reservoir low fluid level switch, and a pressure transducer. A turbine bleed air-supplied oil cooler is installed in the pump return line to ensure fluid temperatures do not exceed the +275°F design limit. All stationary pressure lines are 3/8 in., .020 wall steel, return lines are 1/4 in., .028 wall aluminum, and suction lines are 3/8 in., .028 wall aluminum. Translational motion of the swashplate link actuator is accommodated by braided stainless steel flexible hoses.

3.5 System Weight

Listed below are estimated weights for the elements of the Swashplate Link Actuator configuration. This HHC actuator configuration is felt to represent the optimal approach to providing HHC capabilities to the OH-6A in terms of the minimum extent of aircraft modifications. The impact of instrumentation, additional fuel requirements and the air coolers effect on performance were not considered.

Actuators (3)	2.99 Kg	(6.6 lbs.)
Reservoir/Manifold	2.77 Kg	(6.1 lbs.)
Hydraulic Pump	0.91 Kg	(2.0 lbs.)
Heat Exchanger	0.41 Kg	(0.9 lbs.)
ECU	5.35 Kg	(11.8 lbs.)
SDP-175	6.80 Kg	(15.0 lbs.)
Fluid	1.81 Kg	(4.0 lbs.)
Hydraulic Lines	2.22 Kg	(4.9 lbs.)
Misc. Wiring, Brackets and Hardware	2.72 Kg	(6.0 lbs.)
Total	25.98 Kg	(57.3 lbs.)

Thus, the prototype HHC system imposes an estimated 2.2 percent weight penalty on a 1,179 Kg, (2,600 lb.) OH-6A rotorcraft. It is important to recognize, however, that there exists no need for a general purpose digital microcomputer in a production version of the HHC active control system. Control tasks may therefore be delegated to a microprocessor-based device.

Signal processing and self-test functions currently performed by the ECU are similar candidates for microprocessor implementation. It is not unreasonable to expect that the total electronic subsystem weight of 12.16 Kg (26.8 lbs.) may, therefore, be reduced in a production version to one-third of its prototype value, yielding a production OH-6A weight penalty of 1.5 percent. Linearly relating actuator weight to 0.3 percent aircraft gross weight yields a total system weight penalty for the 6,577 Kg (14,500 lb.), YAH-64 of 0.5 percent.

3.6 System Failure Modes

The results of a Failure Modes, Effects and Criticality Analysis (FMECA) for selected system components are presented in Appendix C and summarized below for a production version of the active control system.

Component	Predicted Failure Rate
Hydraulic Power Subsystem	
Pump	0.3571
Manifold	0.2157
3-Servo-valves	0.0675
3-Actuators	0.7500
Total	1.3903
MTBF	719.0 hours
Electronic Subsystem	
ECU	0.4000
SDP-175 and Power Supply	0.4148
Total	0.8148
MTBF	1,227.0 hours

From a summary of the subsystems above, the addition of the HHC system to an OH-6A is predicted to reduce vehicle reliability by two percent. It is noted that the HHC system would only marginally degrade vehicle reliability for a rotorcraft already employing primary hydraulic boost controls and/or stability augmentation system. Further, with the reduction in airframe vibratory levels indicated in Reference 13, the stress factor (K) utilized in reliability predictions is estimated to be reduced by ten percent. This means that an overall improvement in vehicle reliability would be achieved to offset the reduction caused by the additional components of the HHC system.

3.7 Areas Requiring Further Research

Within the constraints of the preliminary design study, several potential problem areas have been defined and explored. To a large degree, attention is drawn to the capability of the OH-6A mechanical control system to generate high quality 4P blade feathering of sufficient amplitude. Specifically, the following areas have been identified as requiring additional research during the detail design phase.

a. Minimization of Control System Compliance. Potential approaches to the task include buildup of parts from aluminum alloy with increased section moduli where clearances permit. Alternatively, cold bonding of high strength, high modulus graphite/epoxy composites to existing castings has been proposed as a means of increasing control system stiffness. Candidate components for increased stiffness include the lateral, longitudinal and collective pitch mixer bellcranks as well as both stationary and rotating swashplates.

b. As discussed earlier, a need exists to define maximum 4P feathering amplitude requirements over a given flight envelope. The ability to accurately model and predict 4P pitch amplitude requirements is generally constrained by high frequency downwash modelling techniques. The problem of analytically defining 4P pitch requirements would be aided by the availability of cost effective and meaningful rotor wake models. Thus, alternative techniques such as model rotor wind tunnel data coupled with dimensional analysis need be explored in detail design.

REFERENCES

1. O'Leary, J.J., "Reduction in Vibration of the CH-47C Helicopter Using Variable Tuning Vibration Absorber", presented at the 1969 Shock and Vibration Symposium.
2. Ham, Norman D., McKillip, Robert M., Jr., "A Simple System for Helicopter Individual-Blade-Control and Its Application to Gust Alleviation", Preprint 80-65, presented at the 36th Annual Forum of the AHS, May, 1980.
3. Wernicke, R.K., and Drees, J.M., "Second Harmonic Control", Proceedings of the American Helicopter Society, Inc., 19th Annual National Forum, Washington, D.C., May 1-3, 1963.
4. Daughaday, H., "Supression of Transmitted Harmonic Rotor Loads by Blade Pitch Control", USAAVLABS TR-67-14, November, 1967.
5. Balcerak, J.C., and Erickson, J.C. Jr., "Supression of Transmitted Harmonic Vertical and Inplane Rotor Loads by Blade Pitch Control", USAAVLABS TR-69-30, July, 1969.
6. Yen, J.G., "Vibration Reduction with Higher Harmonic Blade Feathering for Helicopters with Two-Bladed Teetering and Four-Bladed Hingeless Rotors", Preprint 80-69 presented at the 36th Annual Forum of the American Helicopter Society, Washington, D.C., May, 1980.
7. London, R.J., Watts, G.A., and Sissingh, G.J., "Experimental Hingeless Rotor Characteristics at Low Advance Ratio with Thrust", NASA CR-114684, December, 1973.
8. Sissingh, G.J., and Donham, R.E., "Hingeless Rotor Theory and Experiment on Vibration Reduction by Periodic Variation of Conventional Controls", presented at the AHS/NASA - Ames Specialists' Meeting on Rotorcraft Dynamics, February 13-15, 1974.
9. McHugh, F.J., and Shaw, J., Jr. "Benefits of Higher-Harmonic Blade Pitch: Vibration Reduction, Blade-Load Reduction and Performance Improvement", presented at the American Helicopter Society Mideast Region Symposium on Rotor Technology, August, 1976.
10. McHugh, F.J., and Shaw, J., Jr., "Helicopter Vibration Reduction with Higher Harmonic Blade Pitch", presented at the Third European Rotorcraft and Powered-Lift Aircraft Forum, Aix-en-Provence, France, September 7-9, 1977.
11. Hammond, C.E., "Helicopter Vibration Reduction Via Higher Harmonic Control", Structures Laboratory, U.S. Army Research and Technology Laboratories, presented at Rotorcraft Vibration Workshop, NASA Ames Research Center, February 22-23, 1978.
12. Wood, E.R., Powers, R.W., and Hammond, C.E., "On Methods for Application of Harmonic Control", Proceedings of the Fourth European Rotorcraft and Powered-Lift-Aircraft Forum, Stresa, Italy, September 13-15, 1978, Paper No. 22.

REFERENCES (Cont.)

13. Hammond, C. E., "Wind Tunnel Results Showing Rotor Vibratory Loads Reduction Using Higher Harmonic Blade Pitch", Preprint 80-66, Structures Laboratory, U.S. Army Research and Technology Laboratories (AVRADCOM), presented at the 36th Annual Forum of the AHS, Washington, D.C., May, 1980.

14. Hammond, C. E. and Cline, J. H., "Active Higher Harmonic Blade Pitch Control for Helicopter Vibration Reduction", Structures Laboratory AVRADCOM, presented at 12th Army Science Conference, West Point, New York, June 17-20, 1980.

15. Shaw, J., and Albion, N., "Active Control of the Helicopter Rotor for Vibration Reduction", Preprint 80-68, presented at the 36th Annual Forum of the AHS, Washington, D.C., May, 1980.

16. Taylor, R. B., Farrar, F. A., and Miao, W., "An Active Control System for Helicopter Vibration Reduction by Higher Harmonic Pitch", Preprint 80-71, presented at the 36th Annual Forum of the AHS, Washington, D.C., May, 1980.

17. Ewans, J. R., "Higher Harmonic Control Systems for Helicopters", Prepared for Naval Air Systems Command, Contract No. N00019-75-C-0459, May 17, 1976.

18. Powers, R. W., "Application of Higher Harmonic Blade Feathering for Helicopter Vibration Reduction", NASA CR-158985, NASA Contract NAS1-14552, April, 1978.

19. Wood, E. R. and Powers, R. W., "Practical Design Considerations for a Flightworthy Higher Harmonic Control System", Preprint 80-67, presented at the 36th Annual Forum of the AHS, Washington, D.C., May 1980.

APPENDIX A

DEVELOPMENT OF HHC TIME RESPONSE CONSTRAINTS

The ECU representation considered in developing HHC time response constraints is shown in block diagram in Figure A-1. The objective of this analysis is to determine optimum filter time constants for rapid servo response but still achieve acceptable smoothing of DC analog output signals. For this application the vertical acceleration spectrum for the OH-6A was assumed to have equal magnitudes of 1P and 4P accelerations, as depicted in Figure A-2.

The bandpass filter now being considered for HHC application of the ECU consists of a 1-pole RC differentiator (Hi-pass), followed by a 4-pole Butterworth (Lo-pass) filter, followed in turn, by another RC differentiator section. Differentiator sections of the filter are designed to have a high-pass 3 dB cutoff frequency at 23 Hz. The two differentiator sections thus yield an attenuation of 6 dB/octave per section or 12 dB/octave total at frequencies below cut-off. Thus, the low frequency attenuation is:

at 23.0 Hz - 6 dB

at 11.5 Hz - 18 dB

at 5.75 Hz - 30 dB

and the filter section thus has 24 dB attenuation at 8 Hz (1P) when considered end-to-end.

The voltage ratio (in dB) of the 1P input and output signals is computed as follows:

$$\text{Ratio} = 20 \log \frac{E_{in}}{E_{out}} \qquad\qquad \text{A.1}$$
$$= 24$$

which yields

$$\frac{E_{in}}{E_{out}} = 15.85 \qquad\qquad \text{A.2}$$

further yielding a $1/15.85 = 0.0631$ volt output based on an assumed 1 volt input at 1P. The 4P filter attenuation, as shown in Section 2., is approximately 3 dB, yielding $E_{out} = 0.707$ volts based on an assumed 1 volt 4P input. Thus, for equal 1P and 4P frequency content in vertical acceleration, the 4P output of the filter section is 21 dB greater than the 1P output signal. Plotting the resultant spectrum for point B of Figure A-1, yields Figure A-4.

Now consider the multiplier portion of Figure A-1. Based on the assumption of pure multiplication, the general equation for instantaneous multiplier output is:

$$e_{out} = (e_{1_{in}}) \times (e_{2_{in}})$$
A. 3

It is reasonable to discard any cross product terms since the multiplier to be used is specified as having less than 1 percent non-linearity distortion, and the 4P reference input to the multiplier is expected to contain less than 2 percent harmonic distortion. Any measurable distortion present in the reference would occur at odd harmonics of 4P, thus having no components at 1 P or 8P. Thus it is reasonable to assume that the reference signal is a true sine wave centered at 4P frequency with a peak amplitude of 1 volt. The two inputs to the multiplier are acceleration and 4P reference, written respectively as;

$$e_1 = 0.063 \sin (16 \pi t) + 0.707 \sin (64 \pi t)$$
A. 4

$$e_2 = 1.0 \sin (64 \pi t), \text{ (t in seconds)}$$
A. 5

Performing the indicated multiplication yields

$$e_1 \cdot e_2 = .3536 + .0315 \cos 48 \pi t - .0315 \cos 80 \pi t -$$
$$.3536 \cos 128 \pi t$$
A. 6

Note the output consists of DC (the desired output correlation signal) plus 3P, 5P and 8P components. The 1P and 4P multiplier inputs are not present in the output signal. Note further that the 3P and 5P components have amplitudes an order of magnitude less than DC and 8P components. This clearly indicates the advantages gained by using the high-pass filter

A-2

section along with the low-pass Butterworth section. Omitting the high-pass section would yield 3P and 5P signals equal in amplitude to the 8P signals under the assumed conditions. Thus it remains only to specify a suitable integrator to attenuate the 8P component. Such an integrator has an inherently small time delay when compared with filters designed to attenuate lower frequency components since the cutoff frequency is inversely proportional to the RC integrator time constant.

The attenuation of an integrator is 3 dB at cutoff frequency with a slope of 6dB/octave at frequencies above cut-off. Using these data, the integrator response curves will be drawn for 3 case; (1) the integrator has a time constant equal to 125 milliseconds, (2) 63 milliseconds and (3) 31 milliseconds, (1/1P = 125 milliseconds). The ratio E_{out}/E_{in} versus frequency is plotted for the various time constants and presented in Figure A-5. Figure A-6 presents the same data on an expanded frequency scale.

A 3 dB attenuation point is plotted on Figure A-5, with the corresponding cut-off frequencies depicted. Figure A-5 demonstrates a potential 20 dB reduction in 1P frequency component with the $\frac{1}{P}$ time constant filter. The $\frac{1}{2P}$ and $\frac{1}{4P}$ time constant filters pass a correspondingly larger portion of 1P input component, thus indicating that time constants $\frac{1}{P}$ or greater are optimal in terms of 1P attenuation.

It is further seen from Figure A-5 how a $\frac{1}{P}$ filter has a 3 dB bandwidth of less than 1.5 Hz. Thus, any signal whose frequency is outside the band 4P ±1.5 Hz will be attenuated 3 dB or more. Applying the $\frac{1}{P}$ filter attenuation at 3P, 5P, 8P and DC, as shown in Figure A-6, to the multiplier output, Equation A.6, the resulting integrator outputs can be computed as follows:

at DC = .3536 volts

at 3P = (.0525) . (.0315) = .0017 volts

at 5P = (.0325) . (.0315) = .0010 volts

at 8P = (.0200) . (.3536) = .0071 volts

Given the worst case, total peak ripple would be;

.0017 + .0010 + .0071 = .0098 volts

This is less than a 3 percent peak ripple (for the $\frac{1}{P}$ filter), and as such this filter would appear to provide a DC input to the computer with very little AC ripple superimposed on the signal.

Now consider the response time of the ECU as constrained by the $\frac{1}{P}$ integrator. Given an abrupt in-flight disturbance, within one time constant (125 milliseconds), 63 percent of the change in 4P accelerations would be sensed by transducers, processed by the correlation filter and made available as a DC input to the computer, Figure A-7. After three time constants (375 milliseconds) 95 percent of the change in 4P acceleration is made available as a DC input to the computer. This would appear to be adequate response to dynamic conditions occurring in flight. It is therefore concluded that $\frac{1}{P}$ filter time constant represents a good choice to provide sufficiently fast response times and adequate AC filtering.
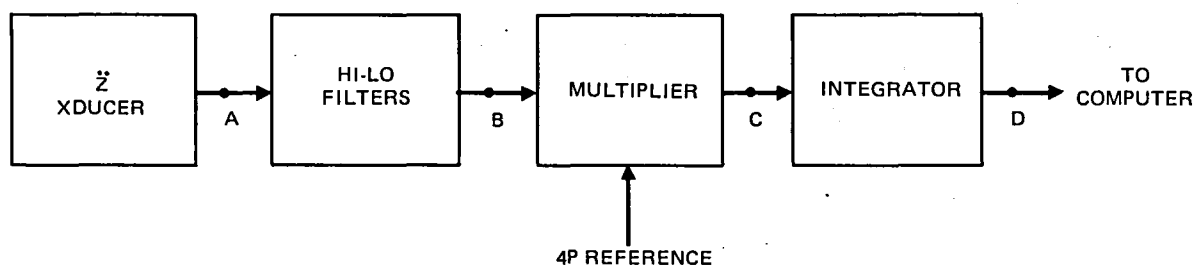
Figure A-1. - ECU signal processing block diagram (one channel).
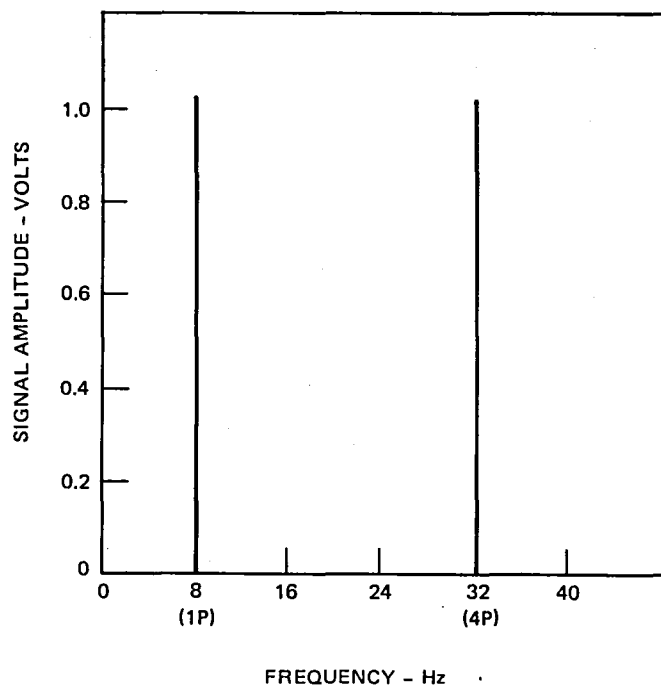
Figure A-2. - Assumed vertical acceleration spectrum.
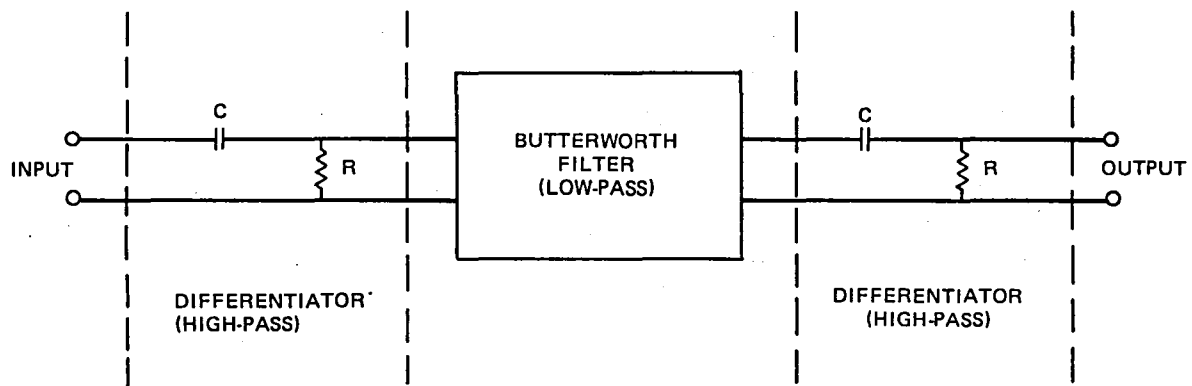
Figure A-3. - Filter block diagram.
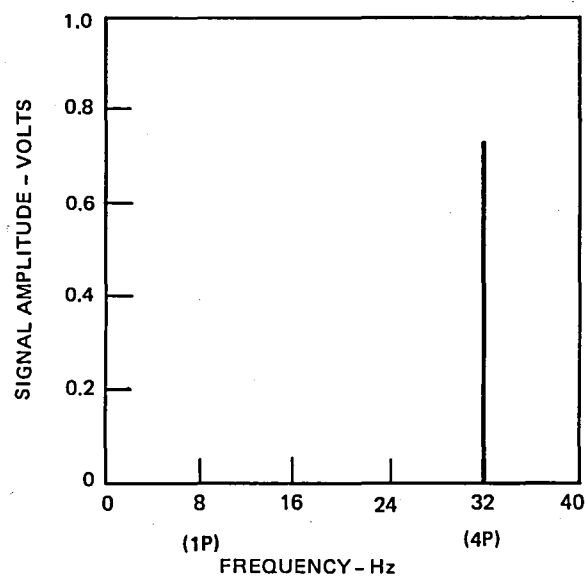
Figure A-4. - Filter output spectrum.
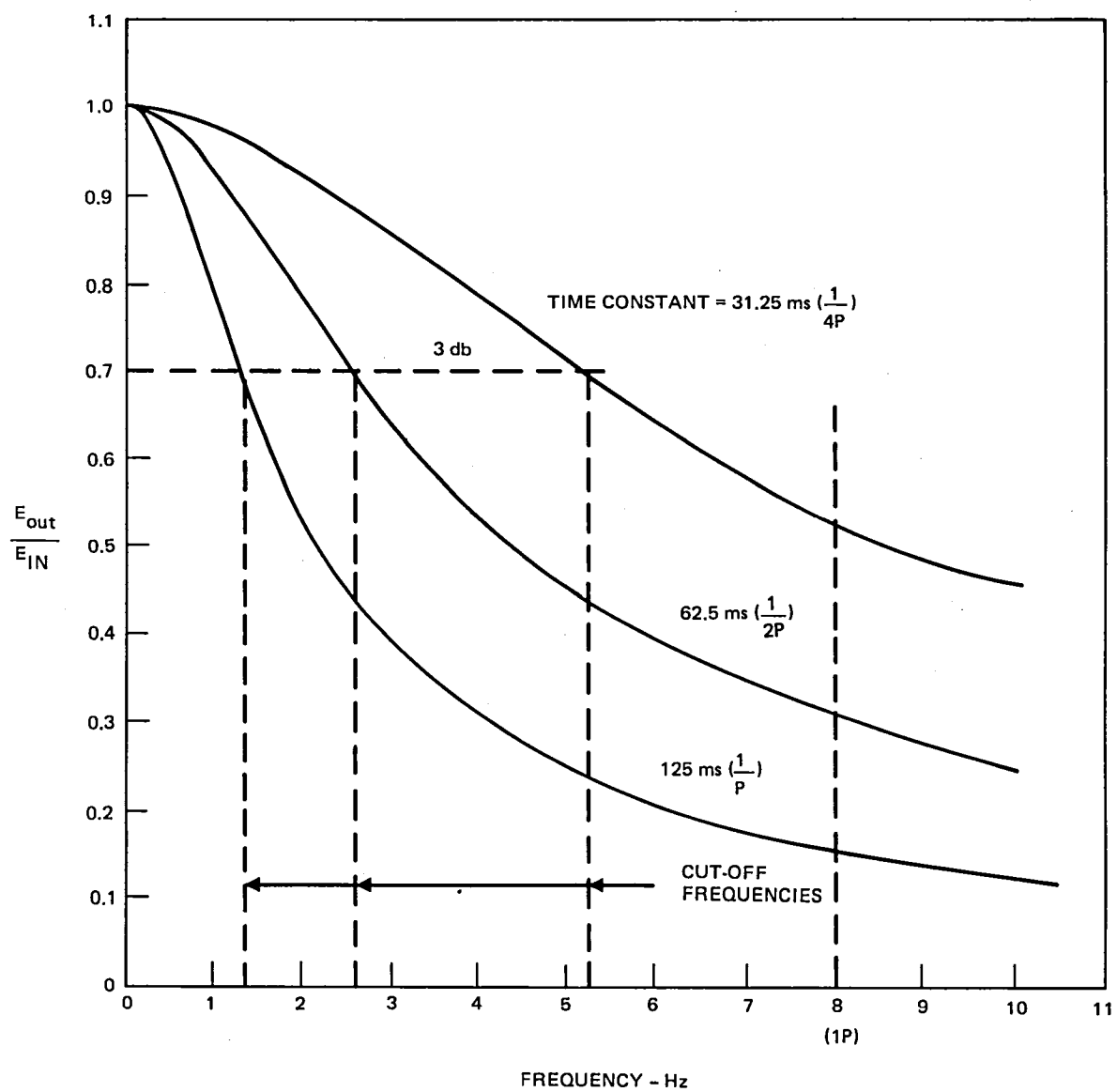
Figure A-5.-- Integrator frequency response.
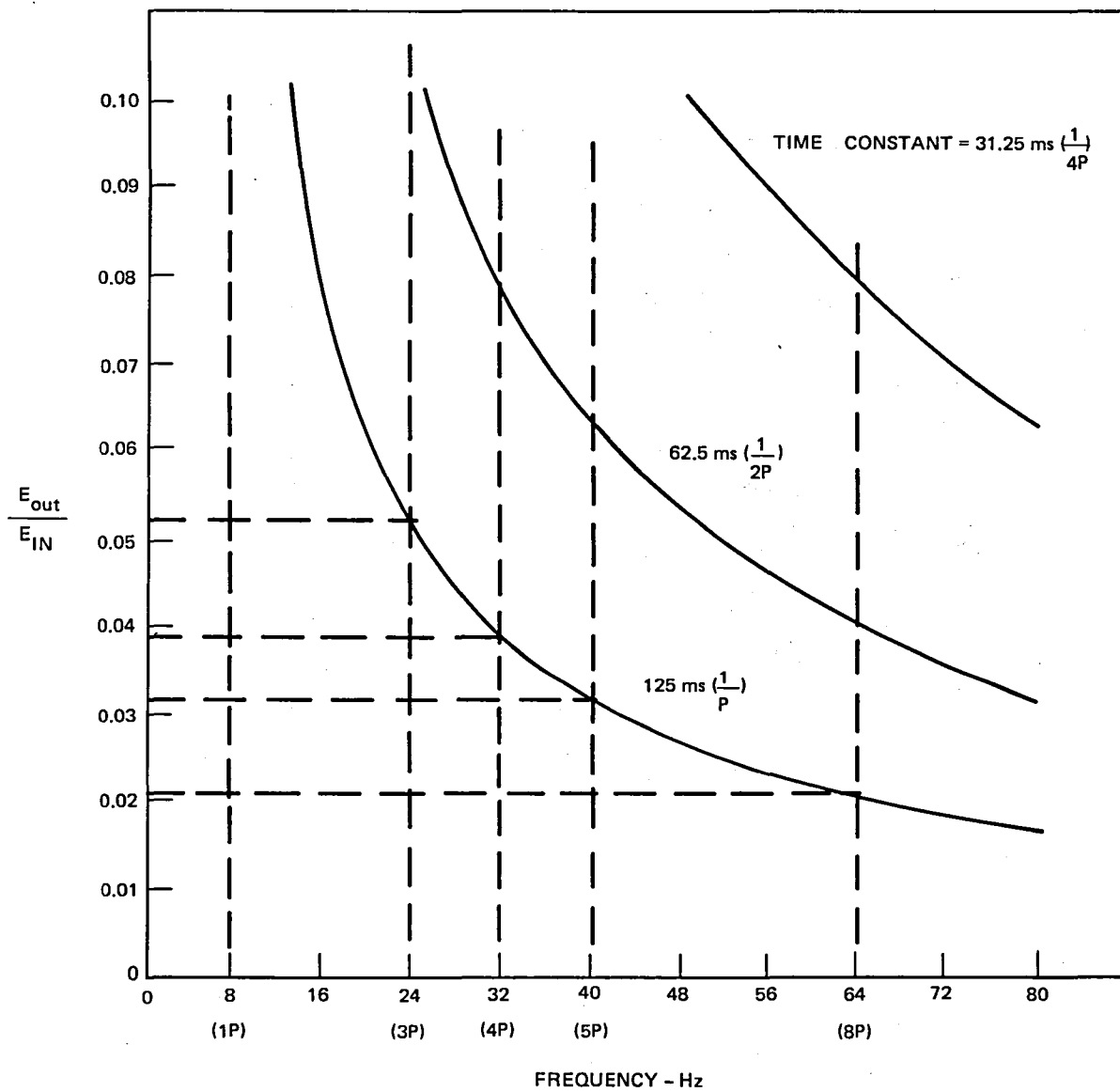
Figure A-6. - Integrator frequency response.

$$e_{OUT} = E_{IN} \left(1 - \epsilon \frac{-t}{(RC)} \right)$$
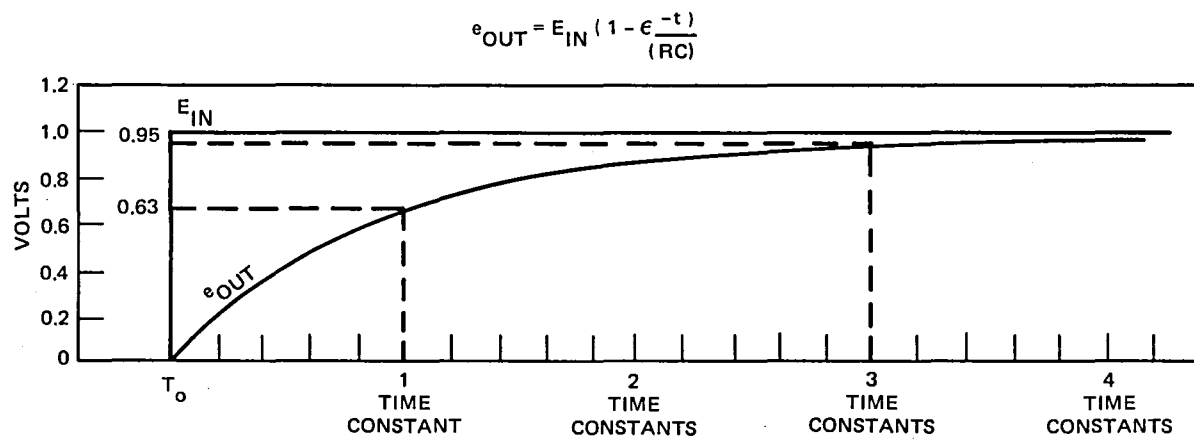
Figure A-7. - RC integration response curve.

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APPENDIX B

PRELIMINARY DESIGN SPECIFICATION FOR HIGHER
HARMONIC CONTROL SERVO-ACTUATOR

The design parameters for the Higher Harmonic Control servo-actuator are presented in this Appendix. They are presented in a design specification for the fabrication and qualification of a development article. The actuator is a swashplate link actuator assembly as described in Section 3.2.3 of this report.

1. SCOPE

1.1 General

This specification establishes the performance, design, and developmental requirements for an HHC Servo Actuator. This unit shall be designed to replace the Hughes OH-6A helicopter main rotor mixer links.

2. APPLICABLE DOCUMENTS

2.1 Documents

The following documents of the exact issue shown form a part of this specification to the extent specified herein. In the event of a conflict between a document referenced herein and the contents of this specification, the requirements of this specification shall be considered the superseding requirement. In other paragraphs of this specification, only the basic document number is stated. The revisions and changes for the applicable documents are identified only in this paragraph.

2.1.1 Specifications

2.1.1.1 Military

MIL-D-1000	Drawings, Engineering and Associated Lists
MIL-E-5400P	Electronic Equipment, Airborne, General Specification for
MIL-H-5440G	Hydraulic System, Aircraft Type I and II Design, Installation, and Data Requirements
MIL-C-5501E(1)	Caps and Plugs, Protective Dust and Moisture Seal
MIL-C-5503 C(3)	Cylinders, Aeronautical, Hydraulic Actuating, General Requirements for
MIL-G-5514F	Packings, Installation and Gland Design, Hydraulic, General Specification for
MIL-C-5541B(2)	Chemical Films for Aluminum and Aluminum Alloys
MIL-H-5606C	Hydraulic Fluid, Petroleum Base, Aircraft Missile and Ordinance
MIL-C-6021G	Casting, Classification and Inspection of
MIL-H-6083D	Hydraulic Fluid, Petroleum Base Preservative
MIL-I-6866B(2)	Inspection, Penetrant Method of
MIL-I-6868D	Inspection Process, Magnetic Particle
MIL-P-6906B	Plates, Identification

MIL-F-7179(D)	Finishes and Coatings, General Specification for Protection of Aircraft and Aircraft Parts
MIL-S-7742B	Screw Threads, Standard Optimum Selected Series, General Specification for
MIL-I-8500C	Interchangeability and Replaceability of Component Parts for Aerospace Vehicles
MIL-A-8625C(1)	Anodic Coatings for Aluminum and Aluminum Alloys
MIL-P-8651B	Plates: Identification and Modification (for Aircraft), Installation of
MIL-H-8775C	Hydraulic System Components, Aircraft and Missiles, General Specification for
MIL-S-8879A	Screw Threads, Controlled Radius Root with Increased Minor Diameter, General Specification for
MIL-Q-9858A	Quality Program Requirements
MIL-F-18372	Flight Control Systems: Design Installation and Test of, Aircraft (General Specification for)
MIL-P-25732B	Packing, Preformed, Petroleum Hydraulic Fluid Resistant 275°F
MIL-C-26074B(1)	Coating, Nickel-Phosphorus, Electroless Nickel, Requirements for
MIL-C-0026482F(1)	Connector, Electric, Circular, Miniature Quick Disconnect, Environment Resistant, General Specification for
MIL-H-83282A	Hydraulic Fluid, Fire Resistant Hydrocarbon Base, Aircraft
MIL-P-83461	Packing, Preformed, Petroleum Hydraulic Fluid Resistant, 275°F

2.1.2 Standards

2.1.2.1 Military Standards

MIL-STD-130D-1	Identification and Marking of U.S. Military Property
MIL-STD-143B	Standards and Specification, Order of Precedence for the Selection of
MIL-STD-453(1)	Inspection, Radiographic
MIL-STD-721B-1	Definitions of Effective Terms for Reliability, Maintainability, Human Factors, and Safety

MIL-STD-704A	Electric Power, Aircraft, Characteristics
(Notice 3)	and General Utilization of
MIL-STD-810B-4	Environmental Test Methods
MIL-STD-889A	Dissimilar Metals
MIL-STD-1472B(1)	Human Engineering Design Criteria for
	Military Systems, Equipment and Facilities

2.1.2.2 Federal Standards

| FED-STD-1028 | Preservation, Packaging and Packaging Levels |

2.1.2.3 Miscellaneous Standards

| ANS B46.1 | Surface Texture (American National Standards |
| | Institute, Inc.) |

2.1.3 Other Government Documents

AMCP 706-203	Engineering Design Handbook, Helicopters,
	Volume III, Qualification Assurance
QQ-C-320A	Chromium Platting (Electro-Deposited)
QQ-N-290A	Nickel Plating (Electro-Deposited)
MIL-HDBK-5A-4	Metallic Materials and Elements for Aerospace
	Vehicle Structures
AFFDL-TR-69-111	Fracture Mechanics Guidelines for Aircraft
	Structural Applications
USAMRDL-TR66-9	Fatigue Crack Propagation in Aircraft
	Materials

Drawings

Hughes Helicopters

| 369ASK3051 | Higher Harmonic Control Actuators Installa- |
| | tion (Design Layout) |

3. REQUIREMENTS

3.1 Item Definition

The actuator shall be a 3,000 psi unit with electronic servo-valve capable of
generating a small amplitude, high frequency (32 Hz) sinusoidal wave form

of high fidelity under load. The article shall be designed in accordance with MIL-H-5440, MIL-H-8775 and MIL-C-5503. All performance, design and qualification testing requirements shall be met under all environmental and operating conditions, including main rotor speeds from 80 to 120 percent of nominal rotor speed (475 RPM).

3.1.1 Item Schematic

The item schematic shall be defined by Figure 39.

3.1.2.1 Physical Interface

The physical interface shall be defined on this installation design layout drawing, Figure 38.

3.1.2.2 Electrical Interface

The electrical interface shall be defined in Figure 12.

3.1.2.3 Hydraulic Interface

The hydraulic interface shall be as defined in the hydraulic system schematic, Figure 49.

3.1.2.4 Functional Interface

3.1.2.4.1 Electric Power

The actuator shall meet the specified performance requirements when supplied with a ±10ma current source from the Electronics Control Unit, conforming to MIL-STD-704. The ECU utilizes 28 VDC electric power to generate an AC servo-valve drive signal. The servo-valve circuit schematic provided by the supplier shall include identification of each circuit function, the wire sizes and color codes, the connector pins, shielding and twisting.

3.1.2.4.2 Hydraulic Interface

The unit shall interface with a Type II, 3,000 psi hydraulic system in accordance with MIL-H-5440. Flexible pressure and return lines shall be used to accommodate actuator translational motion. The supplier shall insure pressure and return port orientation conforms to Figure 39.

3.1.2.4.3 Mechanical Interface

Actuators shall replace two OH-6A main rotor mixer links, 369A7613, and an anti-torque link, 369A7608, maintaining the clearances and orientation of the installation layout schematic, Figure 38, as required to properly provided 4P (32 Hz ±20 percent) feathering.

3.1.2.5 Rated Operating Pressure

The rated servo-valve operating pressure shall be proportional to the applied load. The maximum rated operating pressure shall be 3,000 psi at the servo-valve, which combined with a 1,000 psi drop across the valve, yields a maximum rated actuator pressure of 2,000 psi.

3.1.3 Components

The actuator shall contain, as a minimum, the following components and features:

    a.   Electrohydraulic servo-valve

    b.   Piston position transducer

    c.   Low pressure piston lock-out device

    d.   Teflon/steel-type piston rings

3.1.4 Government-Furnished Property List

Not applicable to the actuator.

3.2 Characteristics

3.2.1 Performance

Unless otherwise specified, values set forth to establish requirements for performance apply to performance under both standard conditions and all combinations of the environmental conditions specified herein. Compliance with Section 3 requirements shall not relieve the supplier of the responsibility of satisfying the performance requirements specified in the following paragraphs.

3.2.1.1 Standard Conditions

Unless otherwise specified, the actuator performance requirements apply under the following standard conditions:

Fluid: MIL-H-5606

Ambient Temperature: 100°F

Frequency: 32 Hz ±20 percent

Rated Electrohydraulic Servo-valve drive
Current: ±10 ma

3.2.1.2 Leakage

3.2.1.2.1 Static Seal Leakage

There shall be no leakage from any static seal under any operating, non-operating or environmental condition.

3.2.1.2.2 Internal Leakage

The internal leakage of the module shall be kept to an absolute minimum to minimize the unit's overall power consumption.

3.2.1.3 Stroke

The unit shall provide a stroke of ±0.20 inches.

3.2.1.4 Piston Velocity

The maximum peak piston velocity shall be 48 in./sec extend or retract with an externally applied load of one-half the unit's stall load. The peak velocity requirement corresponds to a full-stroke 20 percent overspeed conditions.

3.2.1.5 Output Force

The actuator shall provide an output force of 586 ±15 pounds at stall.

3.2.1.6 Piston Rod Lock-Out

The design of the piston rod lock-out mechanism shall ensure full disengagement at inlet pressures above 1,000 psig and full lock engagement at all inlet pressures below 700 psig. Operational signals to the electro-hydraulic valve are not to be applied at inlet pressures below 1,000 psig to prevent cyclic impacts on the lock mechanism when it is partially engaged.

3.2.1.7 Open-Loop Operation

The servo-valve pressure gain, velocity gain, hysteresis, threshold, null bias and null shift shall be consistent with current state-of-the-art servo-valves. The above parameters shall be so defined such that actuator performance requirements are met. The actuator frequency response characteristics shall be within the limits shown in Figure B-1. The actuator shall exhibit a linear output to input relationship over the entire stroke, to within 1 percent.

3.2.1.8 Design Pressures

The design pressures, in psig, shall be as follows:

a. Working Pressure

    1. Pressure   3,000 psi

    2. Return   12 to 65 psi

b. Proof Pressure

    1. Pressure   4,500 psi

    2. Return   2,250 psi

c. Burst Pressure

    1. Pressure   7,500 psi

    2. Return   4,500 psi

d. Impulse Pressure

    1. Pressure   4,500 psi

    2. Return   2,250 psi

3.2.1.9 Servo Dynamic Stiffness

The servo actuator's dynamic stiffness at frequencies above cut-off shall be 120,000 lbs/in/ using a bulk fluid modulus of 150,000 psi.

3.2.2 Physical Characteristics

3.2.2.1 Unit Weight

The actuator unit weight shall not exceed 3 pounds dry weight.

3.2.2.2 Envelope

The unit shall not exceed the envelope and outline dimensions shown in drawing 369ASK3051.

3.2.2.3 Fatigue Life

The actuator shall be capable of sustaining the loads listed below:

a. 10^8 cycles of a - 500 ±75-lb load applied externally in tension and compression and reacted hydraulically by the actuator without evidence of external leakage, performance degradation or permanent deformation.

b. 10^6 cycles of a - 100 ±450-lb load applied externally in tension in compression and reacted hydraulically by the actuator without evidence of external leakage, performance degradation or permanent deformation.

c. The unit shall withstand 10^8 impulse cycles in the hydraulic circuit of the actuator, resulting from high frequency servo-valve operation.

It is noted that items (a) and (c) shall not be required to be cumulative on a single unit.

3.2.2.4 <u>Load Factor</u>

During operation, the actuator shall be capable of sustaining a load factor of 10 g's in any direction without permanent deformation or performance degradation.

3.2.2.5 <u>Insulation Resistance</u>

The insulation resistance between each connector pin and the actuator body shall be greater than 100 megohms (measured with 500 volts dc applied for one minute) following a 1-minute application of 1,000 volts RMS at 60 Hz to each connector pin. Testing shall be performed at room temperature and humidity conditions.

3.2.2.6 <u>External Adjustments</u>

The unit shall be adjusted to meet this specification prior to installation. If external adjustments are utilized, they shall be sealed with inspection stamps. The unit shall not require adjustment once installed on the aircraft.

3.2.2.7 <u>Seal Glands</u>

All seal glands shall be in accordance with MIL-G-5514 except piston-head seal grooves.

3.2.2.8 <u>Seals</u>

Piston rings shall be used for the piston-head seals. The piston-rod seals shall use a filled Teflon slipper seal on the high-pressure side and an elastomer seal on the low-pressure side. Elastomer seals shall conform to MIL-P-25732, MIL-P-83461 or MS 28775. Sealing shall not be accomplished by crushing. O-rings designated by MIL-G-5514 for static application only shall not be used.

3.2.2.9 <u>Back-Up Rings</u>

Back-up rings, when used, shall be installed on both sides of the o-ring. Back-up rings shall conform to MS 28774 when used with MS 28775 o-rings. When MIL-P-83461 compound rings are used, MS 28774 back-up rings cannot be used.

3.2.2.10 Screw Threads

Screw threads shall be in accordance with MIL-S-8879 or MIL-S-7742.
Except for standard parts that are approved by Hughes, only MIL-S-8879
threads shall be used.

3.2.2.11 Lubrication

Only MIL-H-5606 or MIL-H-83282 hydraulic fluid shall be used to lubricate
seals during the installation and assembly of the module. The need for
lubrication during the normal service life of the module is prohibited.

3.3.2.12 Scraper Rings

The actuator shall incorporate scrapers or boots at the exposed ends of the
piston rod to preclude the introduction of external contamination in the seal
area.

3.3.2.13 Safetying

All threaded parts shall be securely locked or safetied with safety wire,
self-locking nuts or some other approved methods. Safety wire shall have
a minimum diameter of 0.032 inch and shall conform to MS 20995. Safety
wire shall be applied in accordance with MS 33540.

3.2.3 Reliability

The mean-time-between-failures as defined by MIL-STD-721 for the HHC
actuator assembly shall not be less than 2,000 hours. The mean-time-
between-corrective maintenance shall not be less than 1,000 hours. The
HHC actuator shall have a minimum total operating life of 6,000 hours.

3.2.4 Maintainability

3.2.4.1 Mean-Time-Between-Unscheduled-Maintenance

The unit shall have a minimum acceptable MTBUM of 500 hours.

3.2.5 Environmental Conditions

The actuator shall withstand the environmental conditions to the extent
specified herein.

3.2.5.1 Altitude

The actuator shall withstand altitudes from sea level to 20,000 feet.

3.2.5.2 Temperature

The actuator shall perform without degradation of performance over the following temperatures:

    a. Rated temperature (fluid inlet): +240°F

    b. Minimum start -65°F

    c. Ambient

        1. Minimum -65°F

        2. Maximum +160°F

3.2.5.3 Humidity

The actuator shall withstand relative humidities up to 95 percent.

3.2.5.4 Sea-Salt Fallout (salt fog)

The actuator shall withstand the conditions described in AR 70-38, Category 2.

3.2.5.5 Sand and Dust

The actuator shall withstand the conditions described in AR 70-38, Category 4, except that the concentration of sand and dust shall be representative of hover conditions for helicopters.

3.2.5.6 Shock

The unit shall exhibit no external leakage or indication of failure when subjected to 18 sawtooth shock pulses of 20 ±2 g peak value.

3.2.5.7 Vibrations

The unit shall withstand the vibrations herein, without evidence of external leakage, structural failure, or loosening of parts from an ambient temperature of -67° to +160°F.

Vibration Frequency (Hz)	Double Amplitude (da) or g level
4 to 20	0.10 da
20 to 33	±2 g
33 to 52	0.036 da
52 to 500	±5 g

3.3 Design and Construction

3.3.1 Materials, Processes, and Parts

Materials, processes and standard parts shall be chosen in accordance with MIL-C-5503C and ADS-13A in the order of precedence set forth in MIL-STD-143. In the event the supplier chooses to use his own company-oriented documentation or commercial specifications, the supplier shall submit such deviations for review and approval prior to implementation.

3.3.1.1 Materials

All metals used in the unit's construction shall be corrosion resistant. Ferrous alloys shall have a chromium content of not less than 12 percent or shall be internally and externally protected against corrosion. Dissimilar metal protection shall be provided to those parts in direct contact. Dissimilar metals are defined in MIL-STD-889.

3.3.1.1.1 Corrosion

All system parts shall be treated or finished so as to provide protection from corrosion in accordance with MIL-F-7179.

3.3.1.1.2 Surface Roughness

Surface roughness finishes of machined and forged surfaces shall be in accordance with ASA standard B46.1.

3.3.1.1.3 Fatigue

The rigorous operating regime of the HHC actuator demands that critical attention be paid to the prevention of fatigue damage through design and analysis. The beneficial effects afforded by surface treatments shall be fully exploited. Such treatments include shot peening, bearingizing, nitriding carburizing, chrome plating, nickel plating and decarburization. Castings shall be clean, sound and free from blowholes, excess porosity, cracks, small radii, sharp corrosion pits, burrs on holes and deep tool marks. Classification and inspection of ferrous and nonferrous metal castings shall be in accordance with MIL-C-6021. Indentations due to staking to retain rod end bearing shall be avoided.

Attention shall specifically be paid to hydrogen embrittlement resulting from welding, plating or any process during which hydrogen is present, and stress corrosion resulting from improper heat treatment or use of alloys predisposed to stress corrosion. Non-metallic inclusions associated with air melt steel shall of particular concern. Provisions shall be made for preventing loss of bolt torque resulting in increased freeting, loss of preload, loss of friction and loss of bolt-spacer action which reduces the effective section modulus. In general, the prevailing concern during design, analysis and manufacturing shall be insuring the actuator's resistance to static fracture and fatigue crack initiation and propagation.

3.3.1.2 Processes

3.3.1.2.1 Anodizing

Aluminum alloys shall be anodized according to MIL-A-8625, using a Type II coating.

3.3.1.2.2 Chromium Plating

All chromium plating on piston rods or sliding surface shall be in accordance with QQ-C-320, Type II.

3.3.1.2.3 Nickel Coating

Nickel coating shall be applied in accordance with QQ-N-290.

3.3.1.2.4 Sub-Zero Stabilization

Close tolerance sliding steel parts shall be sub-zero stabilized to reduce warpage tendencies.

3.3.1.3 Parts

MS and AN standard parts shall be used when suitable and shall be identified by part number on drawings. Hughes approval shall be required when deviating from standard parts.

3.3.1.3.1 Bearings

Bearings requiring no lubrication shall be utilized where practical. Oil-impregnated sintered steel bearings shall not be used.

3.3.1.3.2 Bearing Installation

Actuator rod end bearings shall not be installed by staking.

3.3.1.3.3 Bolts

Fatigue stresses shall be minimized in structural bolts loaded in tension by prestressing. Aluminum alloy nuts, bolts and screws shall not be used. MS 27575 collar-type self-retaining bolts shall be used in any joint requiring frequent disassembly, that is a single attachment, that serves as an axis of rotation, or that is designed to transmit motion that may result in relative rotation between the components of the joints.

3.3.2 Electromagnetic Radiation

The actuator shall meet the requirements of MIL-STD-461.

3.3.3 Nameplates and Product Marking

Identification and nameplates shall be in accordance with MIL-P-15024 and MIL-P-19692. The plate shall specify as a minimum:

    a. Manufacturer's name

    b. Manufacturer's part number

    c. Hughes part number

    d. Operating pressure

    e. Manufacturer's serial number

3.3.4 Workmanship

Workmanship shall be in accordance with MIL-H-8775 and MIL-C-5503.
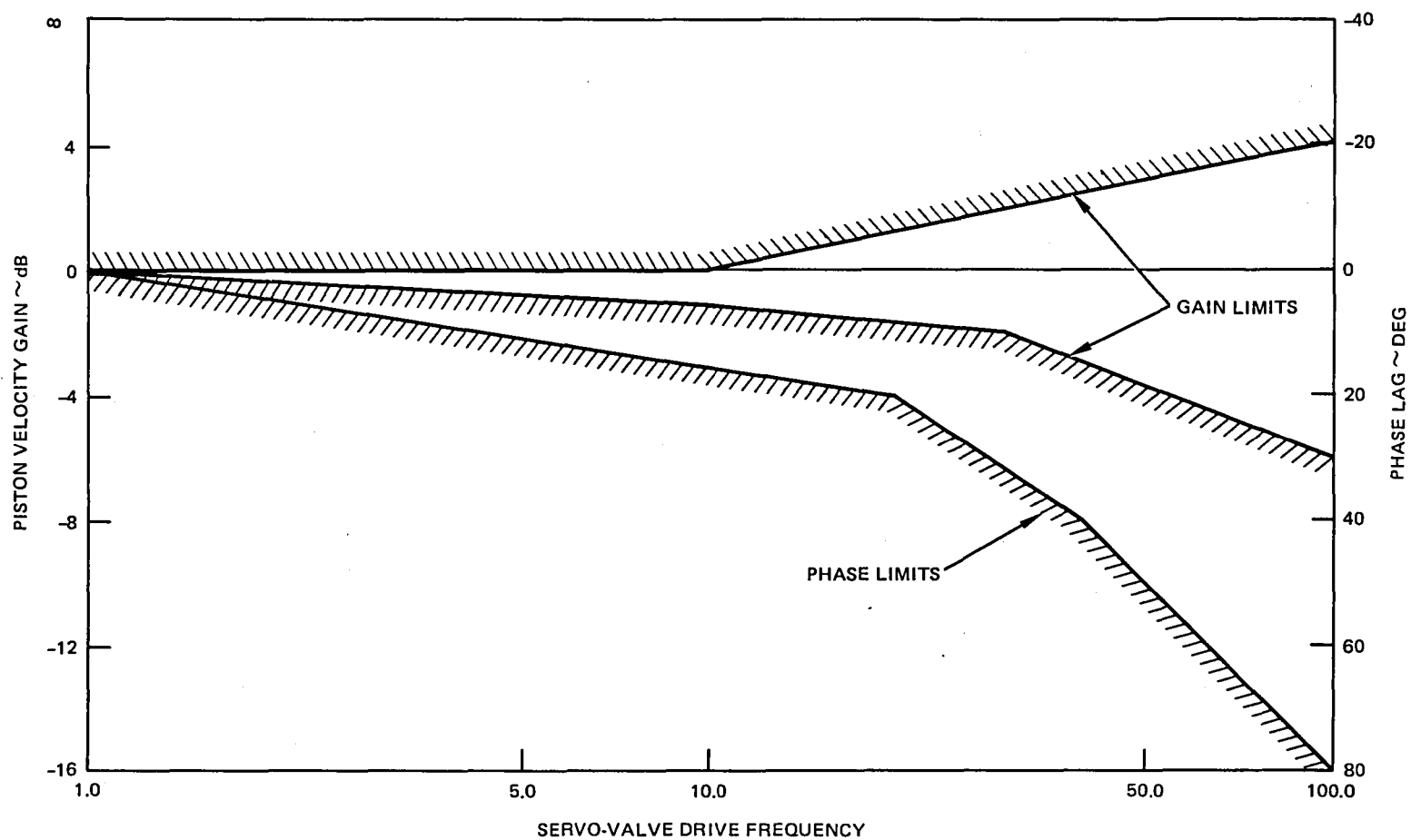
Figure B-1. – Open-loop frequency response requirements.

APPENDIX C

HIGHER HARMONIC CONTROL SYSTEM FAILURE MODES,
EFFECTS and CRITICALITY ANALYSIS (FMECA)

RM LEVEL
FAILURE MODES, EFFECTS AND CRITICAL ANALYSIS (FMECA)

SUBSYSTEM: __HHC ELECTRONIC__ RU: __MULTIPLEXER TYPE III A__ PREPARED BY: __R. L. WILKINSON__ SHEET: __1__ OF __5__

S/S LRU NO. __80-33__ RU P/N: __7-211F00002__ RU FAILURE RATE: __.3849/1000 HR__ REV _____ DATE: __3-5-79__

| ITEM | | FAILURE MODE/S | METHOD OF DETECTION | FAILURE EFFECT | | ITEM FAILURE RATE (7) | | Q T Y | T I M E | ɣ | COMMENTS/COMPENSATING PROVISIONS |
REF NO. (1)	PART NO., NOMENCLATURE & FUNCTION (2)	(3)	(4)	RU (5)	SUBSYSTEM (6)	FAILURES 1000 HR	SOURCE	(8)	(9)	(10)	(11)
80-33-7-211 F00 002	MULTIPLEXER REMOTE TERMINAL UNIT (MRTU) III A					-3849	Vendor	1	1.0		
	The MRTU (Sperry SDP-175) performs controller calculations to estimate the current state of the rotor transmitted 4P accelerations in fixed system. Kalman filter-type calculations yield feathering 4P amplitudes and feathering 4P phases required to null out the current acceleration state. Limit testing on loads and control inputs is performed. Limited self-test in the form of keep-alives are also provided by the MRTU.	(a) Failure of MRTU self-test	FD/LS	Unable to verify that MRTU is operating within its specified performance limits	Possible control system abort					.08	
		(b) Loss or partial loss of power to MRTU	FD/LS	MRTU fails to develop DC solutions	Possible control system abort					.08	ECRU disables driving currents, actuators disabled
		(c) Loss of acceleration data interface from ECU	FD/LS	MRTU fails to develop DC solutions	Possible control system abort					.08	ECU disables driving currents, actuators disabled
		(d) Loss of data interface to ECU	FD/LS	ECU detects loss of Keep-Alive	Possible control system abort					.08	ECU disables drive signals
		(e) Controls exceed limits	FD/LS	Keep-Alive disabled	Possible control system abort					.08	ECU disables drive signals
		(f) Loads exceed limits	FD/LS	Keep-Alive disabled	Possible control system abort					.08	ECU disables drive signals
		(g) Loss of ECU self-test	FD/LS	Unable to verify ECU is operating within specified limits	Possible control system abort						
		(h) Loss of RPM data	FD/LS	Unable to compensate for ECU phase shift	Possible degraded performance					.08	MRTU uses last RPM data

FORM 1792

Hughes Helicopters

SUBSYSTEM: HHC ELECTRONIC   RU: MULTIPLEXER TYPE III A   PREPARED BY: R. L. WILKINSON   SHEET: 2   OF 5

S/S LRU NO. 80-33   RU P/N: 7-211F00002   RU FAILURE RATE: .3849/1000 HR   REV_____ DATE: 11-6-78

| ITEM | | FAILURE MODE/S | METHOD OF DETECTION | FAILURE EFFECT | | ITEM FAILURE RATE (7) | | Q T Y | T I M E | Y | COMMENTS/COMPENSATING PROVISIONS |
REF NO. (1)	PART NO., NOMENCLATURE & FUNCTION (2)	(3)	(4)	RU (5)	SUBSYSTEM (6)	FAILURES 1000 HR	SOURCE	(8)	(9)	(10)	(11)
80-33-1	MULTIPLEXER POWER SUPPLY The multiplexer power supply converts +28 VDC to regulated +5 V, +10 V, +15 V, -5 V and -15 V					-02417		1	1		
80-33-1.1	EMI and power filter										
		(a) Degraded filtering capability due to open/shorted capacitor, shorted inductor or shorted diode	ATE	Voltage transients may cause overstress of regulator devices.	Degraded mode operation of multiplexer.					.125	
		(b) Loss of filtering capability due to an open inductor or diode.	ATE	Loss of all regulated DC output voltages.	Multiplexer inoperative					.125	

FORM 1792

Hughes Helicopters

RM LEVEL
FAILURE MODES, EFFECTS AND CRITICAL ANALYSIS (FMECA)

SUBSYSTEM: __HHC ELECTRONIC__   RU: __MULTIPLEXER TYPE III A__   PREPARED BY: __R. L. WILKINSON__   SHEET: __3__ OF __5__

S/S LRU NO. __80-33__   RU P/N: __7-211F00002__   RU FAILURE RATE: __.3849/1000 HR__   REV _____ DATE: __11-6-78__

| ITEM | | FAILURE MODE/S (3) | METHOD OF DETECTION (4) | FAILURE EFFECT | | ITEM FAILURE RATE (7) | | Q T Y (8) | T I M E (9) | γ (10) | COMMENTS/COMPENSATING PROVISIONS (11) |
REF NO. (1)	PART NO., NOMENCLATURE & FUNCTION (2)			RU (5)	SUBSYSTEM (6)	FAILURES 1000 HR	SOURCE				
80-33-1.2	PRE REGULATOR AND DC-DC CONVERTER										
		(a) Regulator output drops to zero due to shorted power transistor.	ATE	Loss of all regulated DC output voltages.	Multiplexer inoperative					.125	
		(b) Open feedback loop due to defective transistor.	ATE	Regulator output goes of regulation.	Degraded mode operation of multiplexer.					.125	

FORM 1792

Hughes Helicopters

SUBSYSTEM: __HHC ELECTRONIC__   RU: __MULTIPLEXER TYPE III A_____   PREPARED BY: __R. L. WILKINSON____   SHEET: __4____   OF __5__

S/S LRU NO. __80-33_____   RU P/N: __7-211F00002_____   RU FAILURE RATE: __.3849/1000 HR_____   REV_____ DATE: __11-6-78__

| ITEM | | FAILURE MODE/S | METHOD OF DETECTION | FAILURE EFFECT | | ITEM FAILURE RATE (7) | | Q T Y | T I M E | ɣ | COMMENTS/COMPENSATING PROVISIONS |
REF NO. (1)	PART NO., NOMENCLATURE & FUNCTION (2)	(3)	(4)	RU (5)	SUBSYSTEM (6)	FAILURES 1000 HR	SOURCE	(8)	(9)	(10)	(11)
80-33-1.3	SQUARE WAVE TRANSFORMER CONVERTER										·
		(a) Open/short in DC-DC converter transformer.	ATE	Loss of all power supply regulated DC outputs.	Multiplexer inoperative.					.05	
		(b) Unbalanced DC-DC converter output due to open/shorted transistor.	ATE	Power supply shuts itself off automatically.	Multiplexer inoperative.					.05	
		(c) Regulated DC outputs shorted to ground via a rectifier diode.	ATE	Loss of all power supply regulated DC outputs.	Multiplexer inoperative.					.05	
		(d) Degraded output filtering for regulated DC voltages due to shorted inductor or open capacitor.	ATE	Transients and/or switching noise on regulated DC output voltages from chopper or regulator.	Degraded or noisy multiplexer operation.					.05	
		(e) Loss of output filter capability for regulated DC voltage due to open inductor or shorted capacitor.	ATE	Loss of one or more of the power supply regulated DC output voltages.	Multiplexer operation degraded or multiplexer inoperative.					.05	

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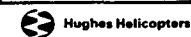
Hughes Helicopters

C-5

RM LEVEL
FAILURE MODES, EFFECTS AND CRITICAL ANALYSIS (FMECA)

SUBSYSTEM: __HHC ELECTRONIC__   RU: __MULTIPLE . TYPE III A__   PREPARED BY: __R. L. WILKINSON__   SHEET: __5__   OF __5__

S/S LRU NO. __80-33__   RU P/N: __7-211F00002__   RU FAILURE RATE: __.3849/1000 HR__   REV _____   DATE: __11-6-78__

| ITEM | | FAILURE MODE/S | METHOD OF DETECTION | FAILURE EFFECT | | ITEM FAILURE RATE (7) | | Q T Y | T I M E | Y | COMMENTS/COMPENSATING PROVISIONS |
REF NO. (1)	PART NO., NOMENCLATURE & FUNCTION (2)	(3)	(4)	RU (5)	SUBSYSTEM (6)	FAILURES 1000 HR	SOURCE	(8)	(9)	(10)	(11)
80-33-1.4	REGULATOR DC-DC CONVERTER										
		(a) Loss of power valid output signal due to defective IC.	ATE	Power supply unable to indicate that it is working properly.	Multiplexer inputs/ outputs not available to interfacing sub-systems.					.125	
		(b) Loss of power supply shut down signals due to defective IC.	ATE	Power supply unable to shut itself off should a power supply malfunction occur.	Multiplexer operates in a degraded mode or is inoperative.					.125	

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SUBSYSTEM: __HYDRAULIC__ RU: _____ PREPARED BY: __A. J. JENDRO__ SHEET: __1__ OF __2__

S/S LRU NO. _____ RU P/N: _____ RU FAILURE RATE: _____ REV __G*__ DATE: __5-25-79__

7-211800021 REV B, LH SIDE FS 165 TO FS 230 HYDRAULI SYSTEM INSTL.

ITEM		FAILURE MODE/S	METHOD OF DETECTION	FAILURE EFFECT		ITEM FAILURE RATE (7)		Q T Y	T I M E	ɣ	COMMENTS/COMPENSATING PROVISIONS
REF NO. (1)	PART NO., NOMENCLATURE & FUNCTION (2)	(3)	(4)	RU (5)	SUBSYSTEM (6)	FAILURES 1000 HR	SOURCE	(8)	(9)	(10)	(11)
20-6 7-2118 10011	PUMP, AXIAL PISTON, VARIABLE DELIVERY (PRIMARY) (Vendor P/N APO 5VC-69)					.3571	VEN	1	1.0		
	A variable displacement hydraulic pump for the primary system, provides hydraulic fluid and pressure necessary to actuate the flight controls.	(a) Complete loss of hydraulic power due to failure of drive coupling group, thrust bearing, creep plates, cam, pistons, heads, cylinder block, piston return group cover, screws.	PLT caution panels lights (hyd. 1) will come on. Also hyd. press. gage 1 on PLT instrument panel will indicate the primary hydraulic pressure.	Loss of hydraulic pressure.	Loss of function. Primary hyd. system inoperative.					.60	
		(b) Reduced hydraulic power due to failure of compensator group static seals.	PLT caution panels lights (hyd. 1) will come on. Also hyd. press. gage 1 on PLT instrument panel will indicate pressure drop.	Degraded hydraulic pressure.	Degraded and possible loss of primary hyd. system function.					.10	
		(c) Instability due to failure of compensator group static seals.	Hyd. press. gage 1 on PLT instrument panel will indicate erratic reading or press. drop. If extent of instability expends, PLT caution panels lights (hyd. 1) will come on.	Degraded hydraulic pressure.	Degradation and possible loss of primary hyd. system function will depend on the extent of pressure instability and time involved.					.08	
											*Rev G - Revised part name and P/N. Was LRU 20-12.

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RM Level
Failure Modes, Effects and Critical Analysis (FMECA)

SUBSYSTEM: HYDRAULICS   RU: _____   PREPARED BY: A. J. JENDRO   SHEET: 2   OF 2

S/S LRU NO. _____   RU P/N: _____   RU FAILURE RATE: _____   REV G*   DATE: 5-25-79

7-211800021 REV B, LH SIDE FS 165 TO FS 230 HYDRAULIC SYSTEM INSTL

ITEM REF NO. (1)	PART NO., NOMENCLATURE & FUNCTION (2)	FAILURE MODE/S (3)	METHOD OF DETECTION (4)	FAILURE EFFECT RU (5)	FAILURE EFFECT SUBSYSTEM (6)	ITEM FAILURE RATE (7) FAILURES 1000 HR	SOURCE	QTY (8)	TIME (9)	γ θ (10)	COMMENTS/COMPENSATING PROVISIONS (11)
20-6 7-2118 10011	PUMP, AXIAL PISTON VARIABLE DELIVERY (PRIMARY) (Vendor P/N APO 5VC-69)										
		(d) Leakage due to failure of housing, shaft seal, static seals.	Hyd. press. gage 1 on PLT instrument panel will indicate pressure drop. If extent of leak expends, PLT caution panels lights' (hyd. 1) will come on.	Degraded hydraulic pressure.	Degradation and possible loss of primary hyd. system function will depend on the extent of leakage and time involved.					.20	
		(e) Structural failure.	Hyd. press. gage 1 on PLT instrument panel will indicate pressure drop. If extent of leak expends, PLT caution panels lights (hyd. 1) will come on.	Loss of hydraulic power, primary system.	Mission abort. Primary hyd. system inoperative.					.02	
		(f) Other unscheduled maintenance actions.	Inspection.	None	No effect on mission.	.3285				1.0	*Rev G - Revised part name and P/N. Was LRU 20-12.

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SUBSYSTEM: __HYDRAULIC__   RU: _____   PREPARED BY: __A. J. JENDRO__   SHEET: __1__   OF __2__

S/S LRU NO. _____   RU P/N: _____   RU FAILURE RATE: _____   REV __G*__   DATE: _5-25-7?_

7-211800021 REV B, LH SIDE FS 165 TO FS 230 HYDRAULIC SYSTEM INSTL

ITEM		FAILURE MODE/S	METHOD OF DETECTION	FAILURE EFFECT		ITEM FAILURE RATE (7)		Q T Y	T I M E	γ	COMMENTS/COMPENSATING PROVISIONS
REF NO. (1)	PART NO., NOMENCLATURE & FUNCTION (2)	(3)	(4)	RU (5)	SUBSYSTEM (6)	FAILURES 1000 HR	SOURCE	(8)	(9)	(10)	(11)
20-7 7-2118 10012	MANIFOLD ASSY, PRIMARY (Vendor P/N 266500-1001)					.1902	BE	1	1.0		
	Provides hydraulic fluid storage for pump suction, and routes the hyd. fluid to the HHC flight control system. It provides ports (G.S.E.) to route charging and filling hyd. fluid to the primary reservoir, and isolates these ports (G.S.E.) when not in use.	(a) External leakage due to damaged/overaged seals.	PLT caution panel lights (hyd. 1) will illuminate. Also hyd. pressure gage 1 on PLT instrument panel will indicate pressure drop. Inspection.	Ability to maintain system fluid is lost.	Loss of primary system function.					.20	
	The major subassemblies and components comprising the primary manifold assy are:	(b) Pressure switch inoperative.	Undetected. Inspection.	Loss of one system pressure indicating function, only.	No effect on mission unless low pressure condition exists.					.20	
	Manifold (1) Reservoir, Primary (1) Relief Valve, L.P. (1) Low Level Switch (1) Pressure Switch (1) Filter (2)	(c) Pressure transducer inoperative.	Hydraulic pressure gage 1 on PLT instrument panel will indicate zero. Inspection.	Loss of system pressure indicating function.	No effect on mission unless low pressure condition exists. Otherwise loss of system pressure indication only.					.52	
	P Indicator, switch, & Thermal Lock (2) Filter Bypass Valve (1) Relief Valve, H.P. (1) Accumulator Check Valve (1) Pressure Transducer (1) G.S.E. Check Valve (1) G.S.E. Isolation Check Valve (1)	(d) Low level switch defective/inoperative.	Undetected. Inspection.	Indicating function of reservoir levels is lost.	No effect on mission unless low level hyd. fluid condition in the reservoir exists. Otherwise loss of indication only.					.04	If the low level switch failed at the same time with sys. press. drop, PLT caution panel lights (hyd. 2) will illuminate. *Rev G - Deleted reference to primary accumulator & other LRU #'s in function. Was LRU 20-06.

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RM LEVEL
FAILURE MODES, EFFECTS AND CRITICAL ANALYSIS (FMECA)

SUBSYSTEM: __HYDRAULIC__   RU: _____   PREPARED BY: __A. J. JENDRO__   SHEET: __2__ OF __2__

S/S LRU NO. _____   RU P/N: _____   RU FAILURE RATE: _____   REV __G*__ DATE: _5-25-79_

7-211800021 REV B, LH SIDE FS 165 TO FS 230 HYDRAULIC SYSTEM INSTL

| ITEM | | FAILURE MODE/S | METHOD OF DETECTION | FAILURE EFFECT | | ITEM FAILURE RATE (7) | | Q T Y | T I M E | γ | COMMENTS/COMPENSATING PROVISIONS |
REF NO. (1)	PART NO., NOMENCLATURE & FUNCTION (2)	(3)	(4)	RU (5)	SUBSYSTEM (6)	FAILURES 1000 HR	SOURCE	(8)	(9)	(10)	(11)
20-7 7-2118 10012	MANIFOLD ASSY, PRIMARY (Vendor P/N 266500-1001) (cont'd)										
		(e) G.S.E. isolation check valve fails to isolate G.S.E. ports.	Inspection.	Loss of function. Fails to isolate G.S.E. ports.	No effect on mission unless leakage at G.S.E. ports.					.04	
		(f) Other unscheduled maintenance actions.	Inspection.	None	None	.1750				1.0	*Rev G - Was LRU 20-6.

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		6. Performing Organization Code
7. Author(s) R. W. Powers		8. Performing Organization Report No. HH 80-415
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9. Performing Organization Name and Address Hughes Helicopters Division of Summa Corporation Culver City, California 90230		
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12. Sponsoring Agency Name and Address Structures Laboratory, USARTL (AVRADCOM) Hampton, Virginia 23665		13. Type of Report and Period Covered Contractor Report
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15. Supplementary Notes

Langley Technical Monitor: John H. Cline
Final Report

16. Abstract

The objective of the work performed was to demonstrate the design feasibility of incorporating an active higher harmonic control (HHC) system on an OH-6A rotorcraft. The introduction of continuously-modulated low amplitude 4P feathering has shown substantial potential for reducing rotor-transmitted oscillatory loads. The design implementation of such a system on a baseline OH-6A required generation of a hydraulic power system, control actuator placement and design integration of an electronic subsystem comprised of an electronic control unit (ECU) and digital microcomputer. Various placements of the HHC actuators in the primary control system were evaluated. Assembly drawings of the actuator concepts and control rigging were prepared and are presented. This study confirms the advantages of generating both hydraulic power and 4P control motions in the nonrotating system.

17. Key Words (Suggested by Author(s)) Higher Harmonic Control, Active Controls, Rotor Vibration Suppression	18. Distribution Statement FEDD Distribution		
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