

NASA CONTRACTOR REPORT 166309

UH-60A Black Hawk Engineering Simulation Program:
Volume I - Mathematical Model

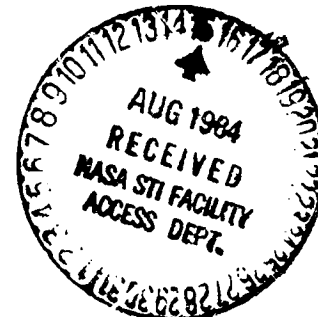
(NASA-CR-166309) UH-60A BLACK HAWK
ENGINEERING SIMULATION PROGRAM. VOLUME 1:
MATHEMATICAL MODEL Final Report (Sikorsky
Aircraft, Stratford, Conn.) 359 p
HC A16/MF A01

884-28806

Unclass
19790

CSCI 01C G3/08

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CONTRACT NAS2-10626
December 1981

NASA

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UH-60A Black Hawk Engineering Simulation Program:
Volume I - Mathematical Model

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Prepared for
Ames Research Center
under Contract NAS2-10626

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SUMMARY

A non-linear mathematical model of the UH-60A BLACK HAWK helicopter has been developed under Contract NAS2-10626. This mathematical model, which is based on the Sikorsky General Helicopter (Gen Hel) Flight Dynamics Simulation, provides the Army with an engineering simulation for Performance and Handling Qualities evaluations. Initially it will be applied in an analysis mode with eventual application to real time pilot-in-the-loop simulation.

This mathematical model is a total systems definition of the BLACK HAWK helicopter represented at a uniform level of sophistication considered necessary for Handling Qualities evaluations. The model is a total force, large angle representation in six rigid body degrees of freedom. Rotor blade flapping, lagging and hub rotational degrees of freedom are also represented. In addition to the basic helicopter modules, supportive modules have been defined for the landing interface, power unit, ground effects and gust penetration. Information defining the cockpit environment relevant to pilot-in-the-loop simulation is presented. This same model was activated on Sikorsky's DEC PDP KL10 computer to generate check cases for use during the validation of the simulation at NASA.

Volume I of this report defines the mathematical model using a modular format. The documentation of each module is self-contained and includes a description, mathematical definition and input for the BLACK HAWK. Volume II provides background and descriptive information supportive to an understanding of the mathematical model.

2.0 INTRODUCTION

This report is Volume I of two volumes, which document the mathematical model of the UH-60A BLACK HAWK helicopter. This work was funded under Contract NAS2-10626 by the U.S. Army Research and Technology Laboratories (AVRADCOM) Ames Research Center.

The objective of this contract was to provide the Army and NASA with a well documented, operational and verified engineering simulation of the BLACK HAWK helicopter. This work, undertaken by Sikorsky, provides the Army with a flying qualities analysis methodology for the BLACK HAWK helicopter which could eventually be extended to a real time pilot-in-loop simulation. The mathematical model provided under this contract is a total system, free flight representation based on the Sikorsky General Helicopter (Gen Hel) Flight Dynamics simulation. It is defined at a uniform level of sophistication currently considered appropriate for handling qualities evaluations. This model is also considered to give representative performance trends, but should not be used to define critical performance characteristics. The modular format presented facilitates the introduction of additional or more sophisticated modules.

Volume I of this report documents the basic BLACK HAWK mathematical model, and in addition, defines supportive routines developed by Sikorsky under this contract. These routines include a landing interface, power unit, ground effects and rotor gust penetration models. Presented in this volume is an Overview of the Simulation Model, Section 3, followed by a Description of the BLACK HAWK helicopter in Section 4. Section 5 contains the documentation of the Simulation modules. Each of the module definitions is segregated with its own Table of Contents. Section 6 contains information defining the BLACK HAWK cockpit relevant to pilot-in-the-loop simulation. A single copy Appendix to Volume I, containing extensive program verification data generated from a similar model on the Sikorsky Simulation Facility, provides NASA and Sikorsky with the necessary information for validating the BLACK HAWK helicopter simulation on the Ames Simulation facility.

Volume II of this report documents the derivation of the equations, the assumptions inherent in the model and provides supportive discussion to aid in the understanding of the mathematical model.

3.0 OVERVIEW OF THE SIMULATION MODEL

The mathematical model of BLACK HAWK provided under this contract is based on the Sikorsky General Helicopter (Gen Hel) Flight Dynamics Simulation. This model is a generalized, modularized analytical representation of a total helicopter system. It normally operates in the time domain and allows the simulation of any steady or maneuvering flight condition which can be experienced by a pilot.

The overall structure of the model is presented on figures 3.1 and 3.2 in functional and block diagram format respectively. The solution in terms of aircraft motion is obtained iteratively by summing the component forces and moments acting at the aircraft's center of gravity and subsequently obtaining the body axes accelerations. Resulting velocities and displacements then condition the environment for the components on the next pass through the program. The datum axes system used are a set of right coordinate body axes with the origin at the fuselage center of gravity. The X axis points towards the nose of the aircraft and is parallel to the center line of the aircraft. All calculations are related to this axis system. The final aircraft motion is transferred into earth axes for simulator, VFR display and instrumentation drives. The mathematical module defining the equations of motion are presented in Section 5.10.

The basic model is a total force, non-linear, large angle representation in six rigid body degrees of freedom. In addition, rotor rigid blade flapping, lagging and hub rotational degrees of freedom are represented. The latter degree of freedom is coupled to the engine and fuel control. Motion in the lag degree of freedom is resisted by a non-linear lag damper model.

The total rotor forces and moments are developed from a combination of the aerodynamic, mass and inertia loads acting on each simulated blade. The rotor aerodynamics are developed using a blade element approach. The angle of attack and dynamic pressure on individual blade segments are determined from the three orthogonal velocity components. These arise as a result of airframe motion, rotor speed, blade motion and downwash resulting from the generation of thrust. In the latter case, which represents the air mass degree of freedom, a uniform downwash is derived from momentum considerations, passed through a first order lag, and then distributed first harmonically as a function of rotor wake skew angle and the aerodynamic hub moment. Finally, blade geometric pitch is summed with the inflow angle of attack to obtain the total angle of attack at the blade segment. The full angle of attack range for blade aerodynamics is represented as a function of Mach number.

Blade inertia, mass and weight effects are fully accounted for and their resulting loads, dependent on blade and aircraft motion, are added to the aerodynamic loads for each blade. This summation gives the shear loads on the blade root hinge pins. Total rotor forces are obtained by summing all the blade hinge pin shears with regard to azimuth. Rotor moments result from the offset of the hinge shears from the center of the shaft. Blade flapping and lagging motion is determined from aerodynamic and inertia moments about the hinge pins. During one pass through the program all segments and all simulated blades are computed. If because of execution time considerations the simulated number of blades are not made equal to the actual number, then they are redistributed in azimuth accordingly. The mathematical module defining the rotor is presented in Section 5.1.

The fuselage is defined by six component aerodynamic characteristics which are loaded from wind tunnel data which have been extended analytically to large angles. The angle of attack at the fuselage is developed from the free stream plus interference effects from the rotor. These interference effects are based on rotor loading and rotor wake skew angle. Local velocity effects are not accounted for. The mathematical module defining the fuselage is presented in Section 5.2.

The aerodynamics of the empennage are treated separately from the forward airframe. This separate formulation allows good definition of non-linear tail characteristics that would otherwise be lost in the simplifications of multivariate total aircraft maps. With this approach, changes to the empennage can be made without reloading basic airframe maps. The angles of attack at the empennage are developed from the free stream velocity, plus rotor wash and airframe wash. Dynamic pressure effects from the airframe are accounted for by factoring the free stream velocity component. By necessity the wash and dynamic pressure effects are averaged over the stabilizing surfaces. The tail rotor is represented by the linearised closed form Bailey theory solution. Terms in tip speed ratio greater than M squared have been eliminated. The airflow encountered by the tail rotor is developed in the same manner as the empennage. An empirical blockage factor, due to the proximity of the vertical tail, is applied to the thrust output. The mathematical modules defining the Empennage and tail rotor are presented in Sections 5.3 and 5.4 respectively.

The flight control system for BLACK HAWK presented in this model covers the primary mechanical flight control system and the Automatic Flight Control System (AFCS). The latter incorporates the Stability Augmentation System (SAS), the Pitch Bias Actuator (PBA), the Flight Path Stabilization (FPS) and the Stabilator mechanization. These automatic control functions collectively enhance the stability and control characteristics of the BLACK HAWK. The analytical definition of the control system given in Section 5.5 incorporates the Sensors, shaping networks, logic switching, authority limits and actuators. Some of these components have wide band-widths which are beyond the frequencies normally associated with piloted simulation. They have been included for completeness and accuracy in analytical evaluations. The model provided represents the control system in a complete manner except for the FPS. In this case only the attitude hold and turn features have been defined.

The engine/fuel control model provided with this simulation is a linearized representation with coefficients which vary as a function of engine operating condition. The model adequately provides for closing the rotor shaft speed loop throughout the normal operating envelope of the helicopter. However, maneuvers which result in significant rotor speed excursions may result in discrepancies in the simulation. This engine module should not be used for engine performance evaluation. The modular formulation does however, facilitate the introduction of a sophisticated model if necessary at a later time. The interface of the engine with the rotor module, shown on Figure 3.3, is via the rotor clutch. This is programmed to disengage the rotor shaft from the engine if rotor required torque drops below zero. Under these circumstances, the engine speed feedback to the fuel control will cause the engine to seek an operating condition dictated by the control. The clutch will reengage when the rotor speed drops below power turbine speed. The engine fuel control equations are presented in Section 5.6.

The landing interface module, Section 5.7, allows for ground contact. It is a generalized representation consisting of a landing gear force reaction model complete with all necessary space/body geometry calculations to track a free helicopter during a landing onto level ground. The landing gear is represented by separate non-linear tire and strut dynamic characteristics. Tire in-ground-plane loads are developed as a non-linear function of the tire deflection and normal load. These forces are adjusted depending on the friction criteria which determines tire skid properties at the ground plane. The strut is simulated by an isentropic air spring and velocity squared damper in parallel. The output loads from the three landing gears are finally transferred to the center of gravity where they are summed with other external forces and moments.

Section 5.8 defines a simplified ground effects model. During the development of this module it was evident that anything more than a simple model based empirically on distance above the ground plane, was beyond the scope of this contract. The model provided, adjusts the downwash at the rotor (and therefore rotor loads) as a function of height above the ground plane and forward speed (in terms of rotor wake skew angle).

The gust penetration routine documented in Section 5.9 provides for a gust front passing across the rotor disc from any direction. Behind the front, gust velocities are varied with distance from the front. The variation can be prescribed by a choice of several discrete velocity profiles or a continuous turbulence form due to Dryden. Each rotor blade element and the aerodynamic centers of fuselage and tail components may be subjected to horizontal and vertical gust velocities whose magnitude is a function of the geometric distance from the traversing gust front. Thus penetration or velocity distributional effects are almost fully accounted for in the rotor simulation and are adequately dealt with in the fuselage/tail model.

GENHEL FLIGHT DYNAMICS SIMULATION

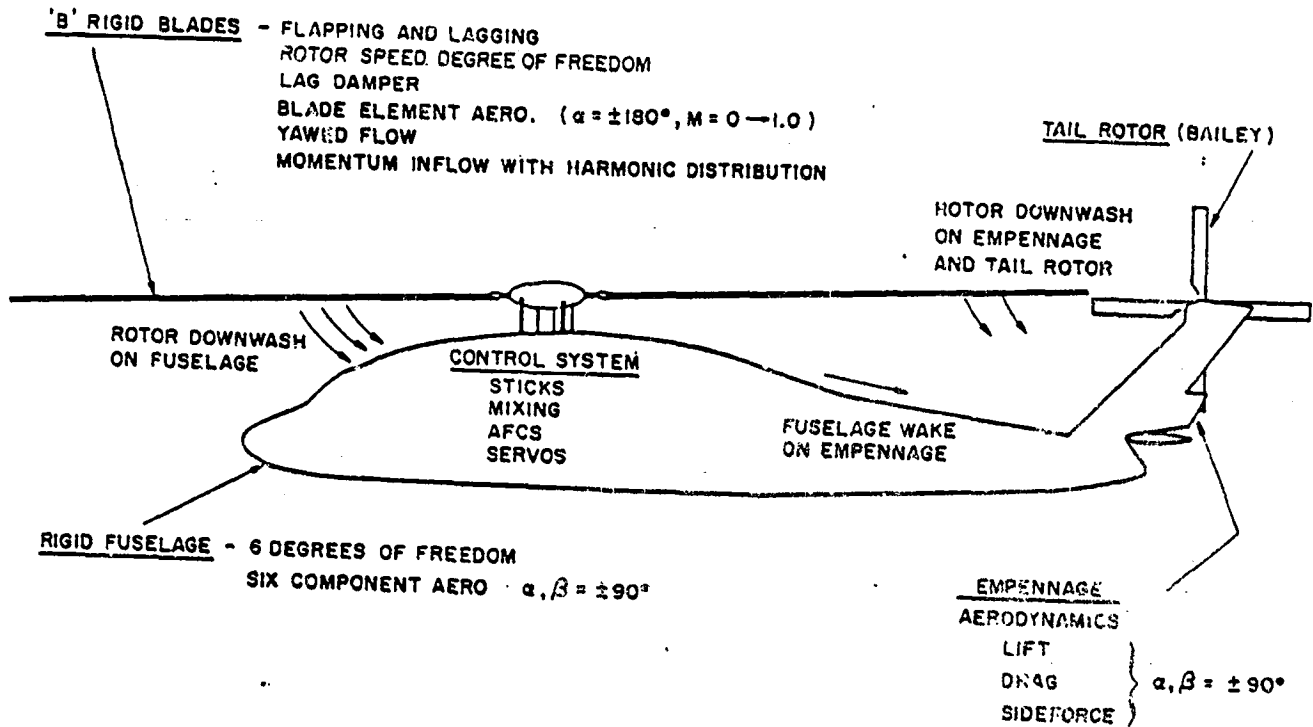


FIGURE 3.1

GEN. HEL. FLIGHT DYNAMICS SIMULATION MODEL

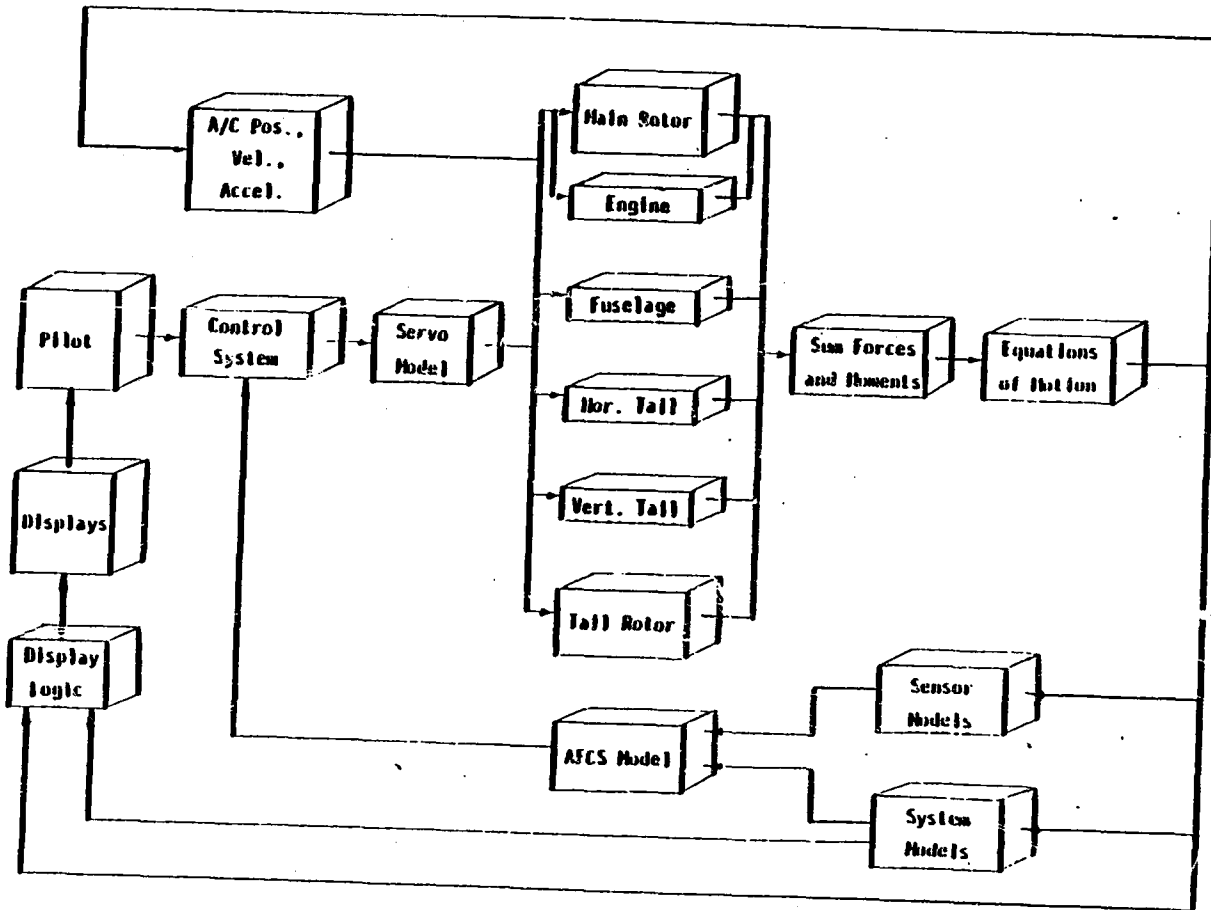


Figure 3.2

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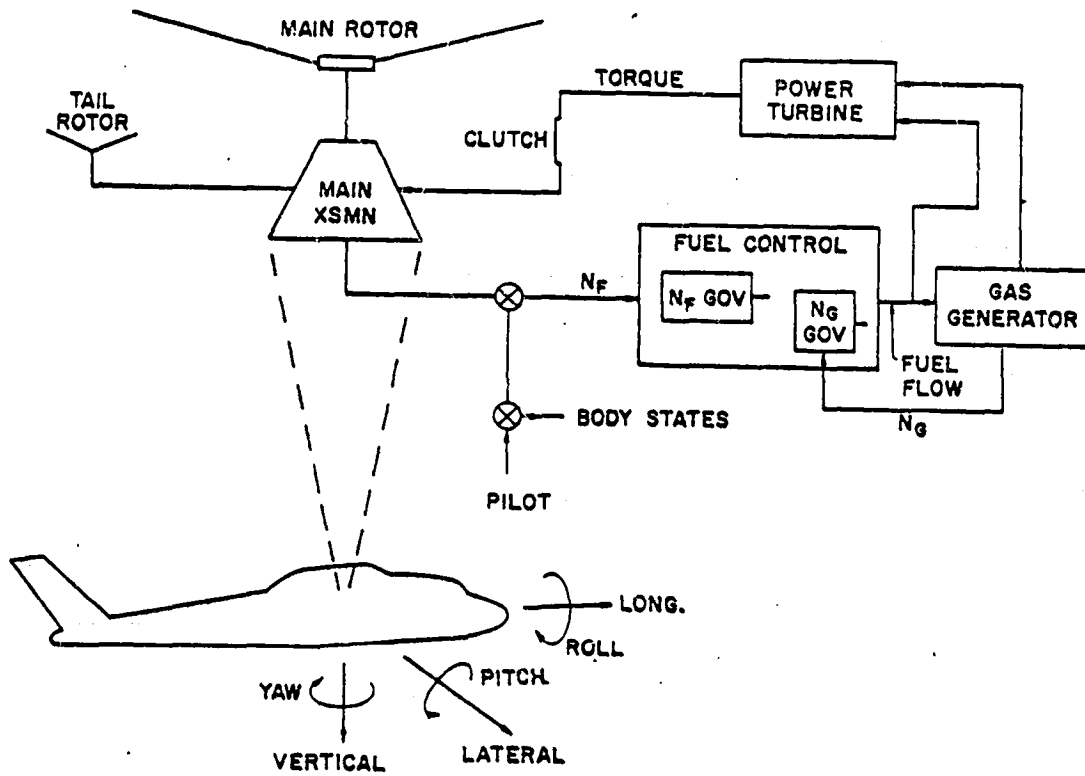


FIGURE 3.3

4.0 Description of the BLACK HAWK Helicopter

The UH-60A BLACK HAWK, shown in the figures 4.1 and 4.2, is a utility transport helicopter developed by Sikorsky for the Army under the UTTAS program. This medium-sized helicopter is designed to carry eleven combat-equipped troops and a crew of three. The Basic Structural Design Gross Weight is 16,825 lbs. with a maximum Alternate Gross Weight of 20,250 pounds. Missions include: Troop Assault, Aeromedical Evacuation, Aerial Recovery and Extended Range Missions. The BLACK HAWK has a maximum level flight speed in excess of 160 knots and a diving speed in excess of 170 knots.

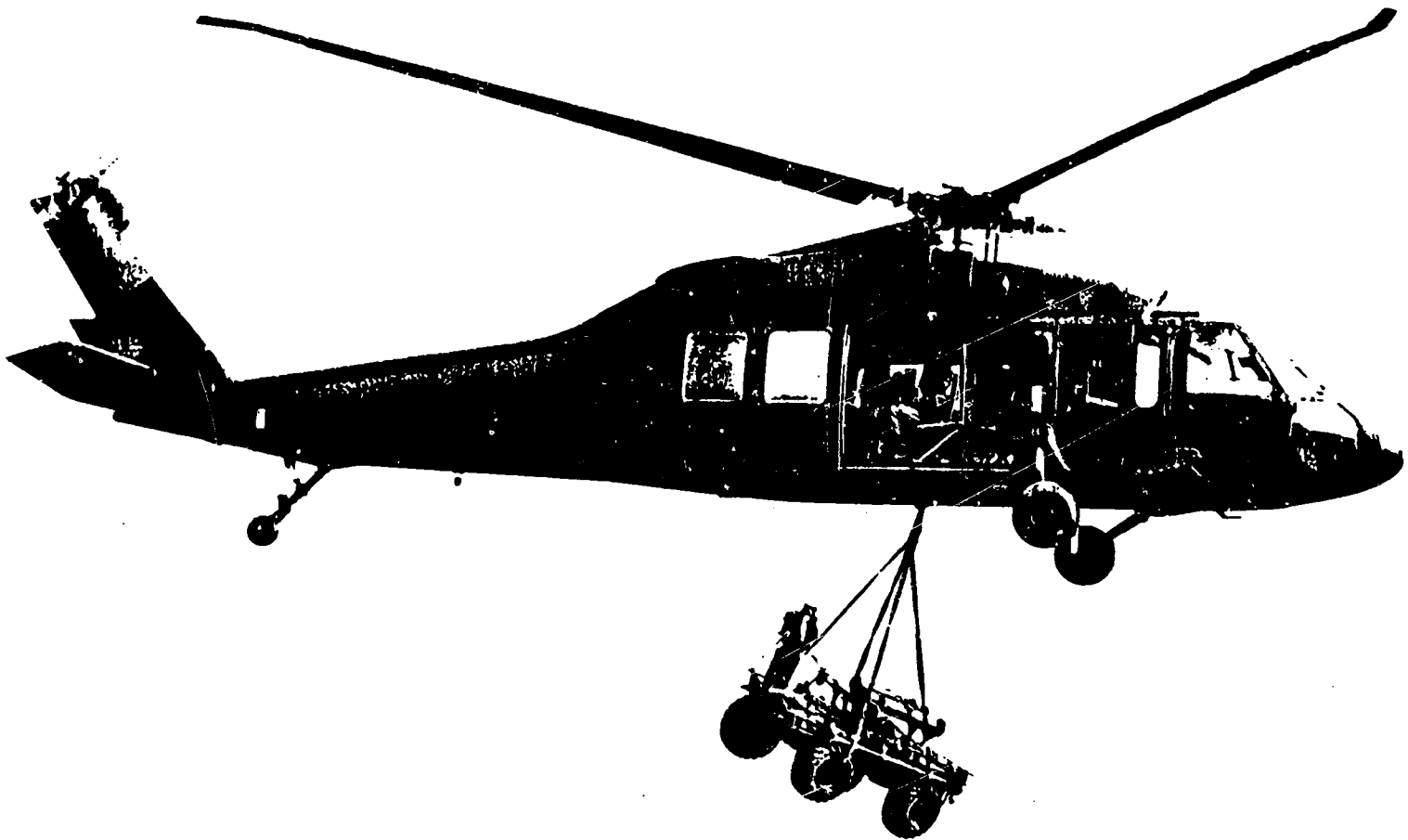
The BLACK HAWK has a single main rotor and a canted tail rotor as shown on the general arrangement drawing, figure 4.3. A list of physical characteristics is given on Table 4.1. The main rotor consists of four fully articulated titanium/fiberglass blades which are retained by a flexible elastomeric bearing in a forged titanium single piece hub and restrained in plane by a conventional hydraulic lag damper. The 11.0 feet diameter four bladed tail rotor is a bearing-less cross-beam arrangement with the shaft tilted 20 degrees upwards. Both rotors have an SC 1095 aerofoil section. The engines mounted on top of the cabin which together provide approximately 2,800 h.p. at normal continuous rating. These engines have Hamilton Standard hydraulic, and General Electric Company electronic fuel control components. The drive system consists of main, intermediate, and tail gear boxes with interconnecting shafts.

The flight control system on the BLACK HAWK is a redundant hydro-electrical-mechanical system. It includes three two stage main rotor servos, a Stability Augmentation System (SAS), a Flight Path Stability System (FPS), and a triple redundant hydraulic supply. The horizontal tail rotates from a positive angle of about 40° in hover up to zero with increasing forward speed.

A summary of the mass properties characteristics is given on Table 4.2. Recommended overall longitudinal center of gravity limits are given on figure 4.4.

A more comprehensive description of the BLACK HAWK helicopter is given in References 4.1.1, 4.1.2, and 4.1.3.

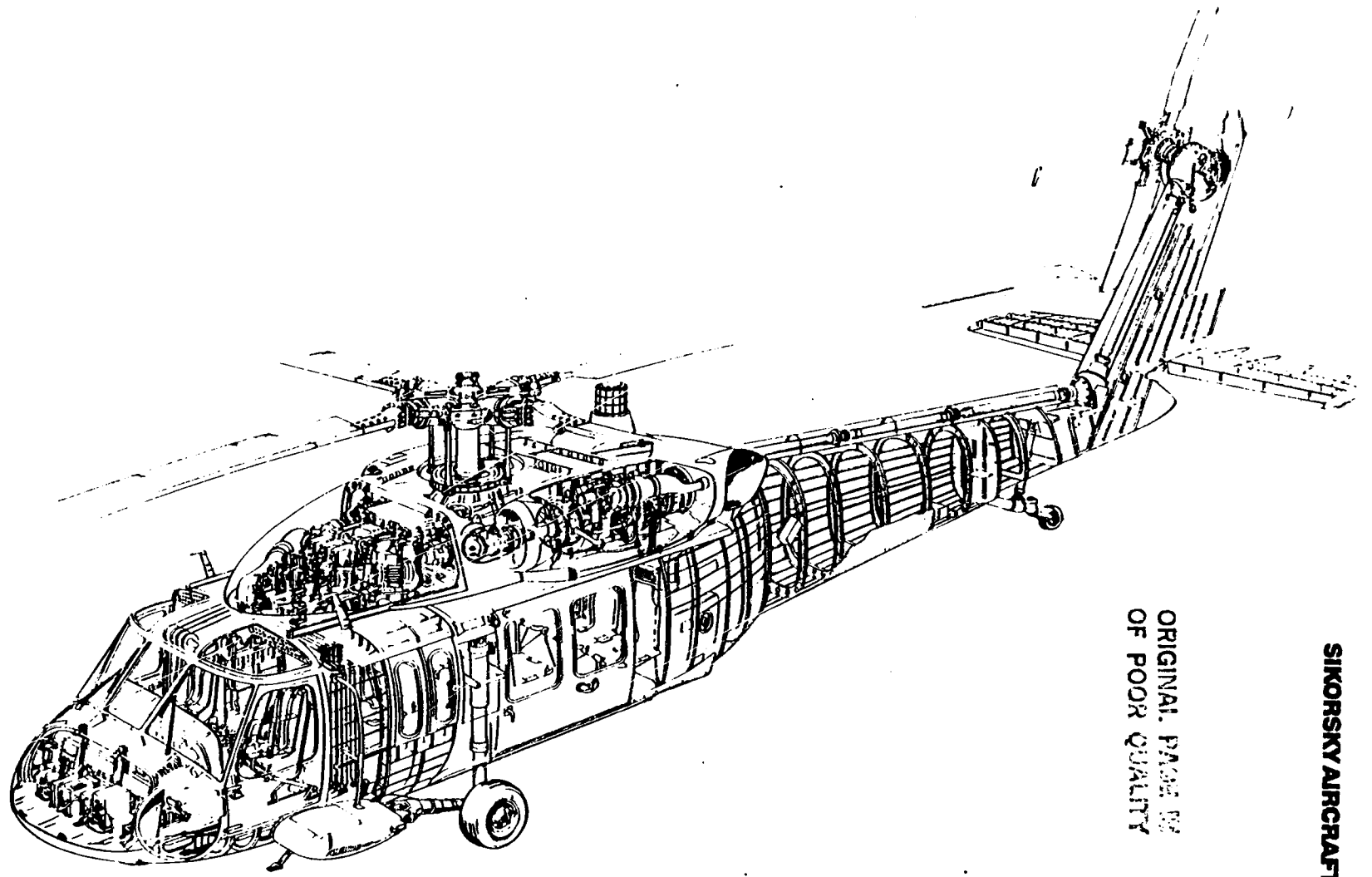
- 4.1 References
 - 4.1.1 Flight International, Week Ending 25 February 1978, Article by Mark Lambert.
 - 4.1.2 U.S. Army UH-60A Helicopter Development Production Program. Prime Item Development Specification (PIDS) DARCOM-CP-2222-S1000D Part I October 15, 1979.
 - 4.1.3 UH-60A Familiarization Training Course Manual - Sikorsky Training Document.



UH-60A BLACK HAWK HELICOPTER

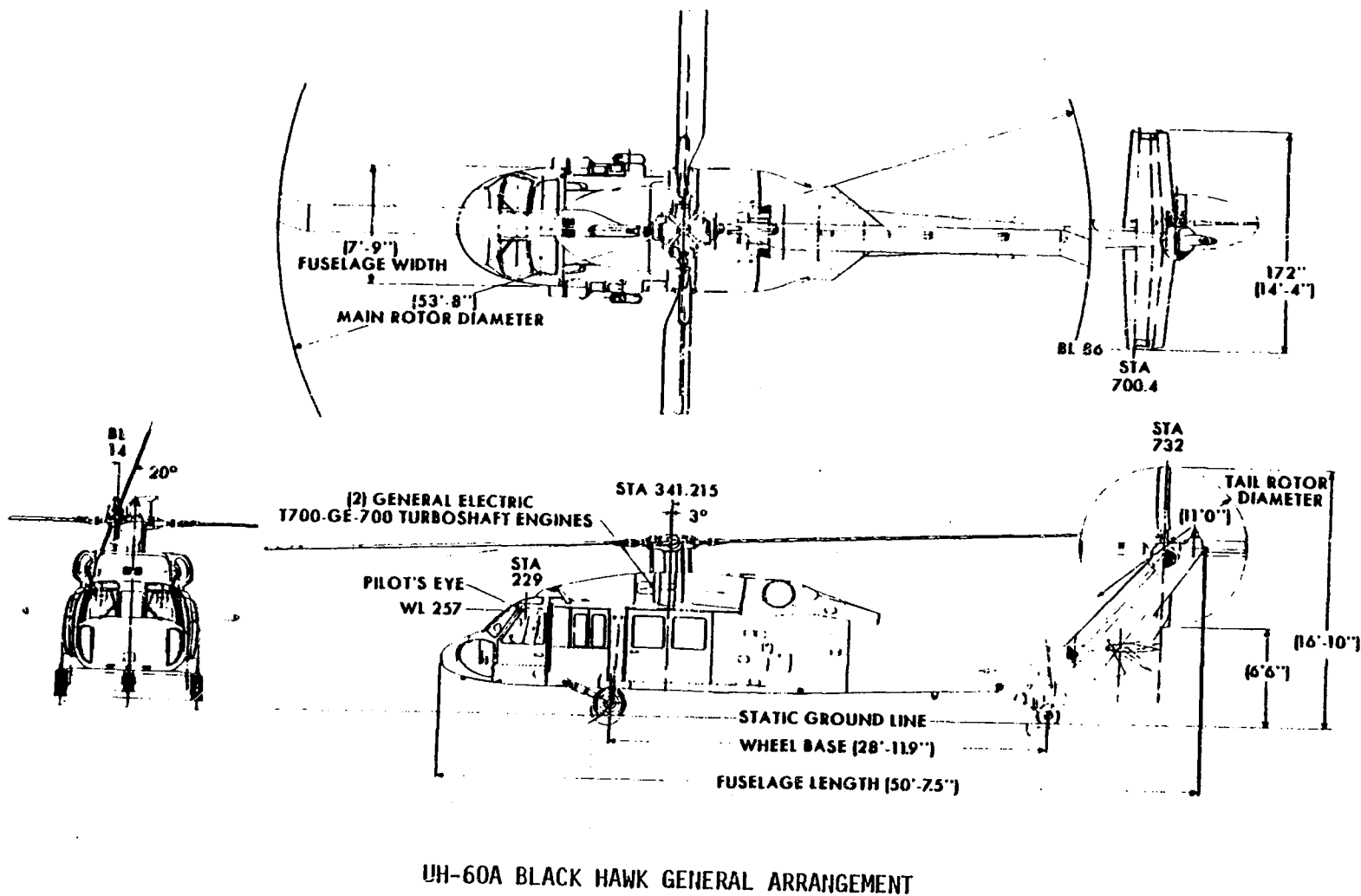
FIGURE 4.1

FIGURE 4.2



UH-60A BLACK HAWK

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UH-60A BLACK HAWK RECOMMENDED LONGITUDINAL CENTER OF GRAVITY LIMITS

(DATA FROM SER-70288 PREPARED UNDER CONTRACT DAAJ01-77-C-0001(P6A))

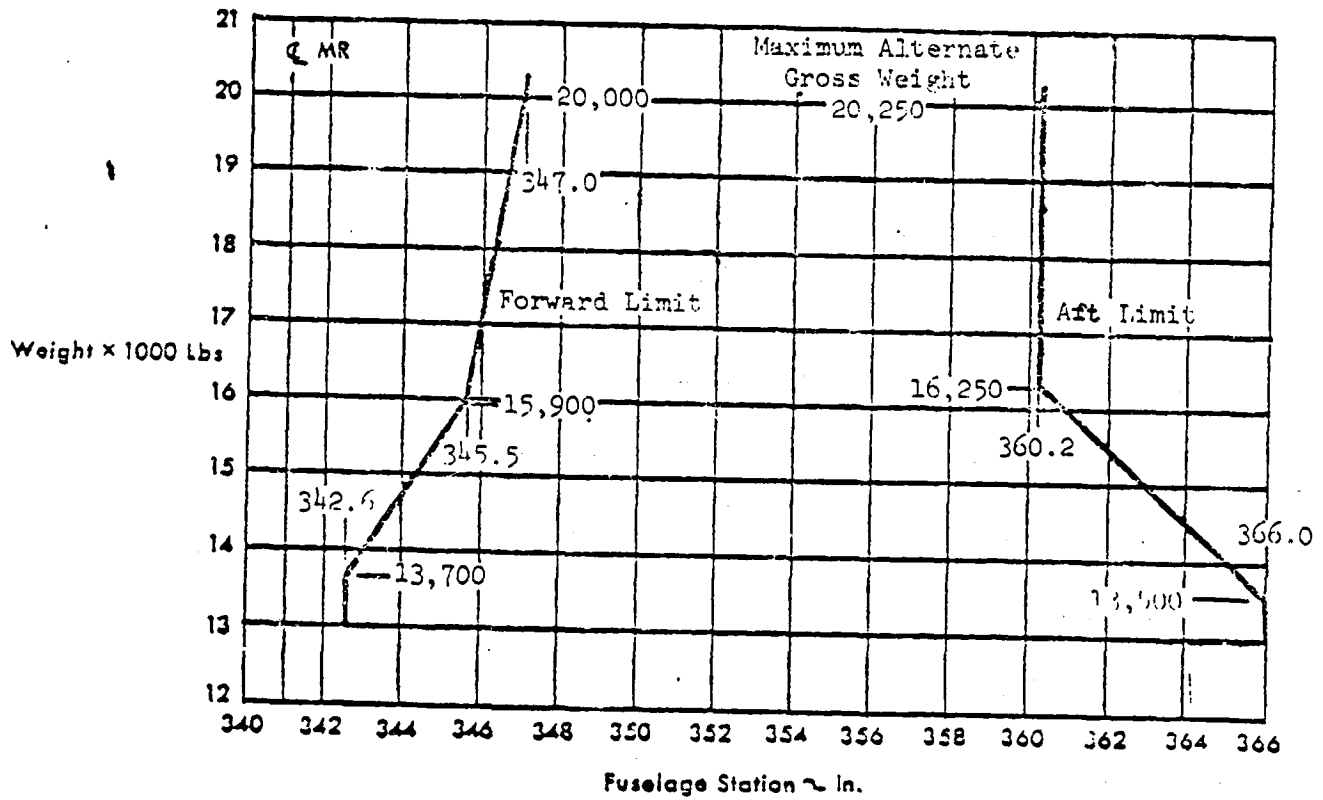


FIGURE 4.4

BLACK HAWK

TABLE 4.1. LIST OF PHYSICAL CHARACTERISTICS

MAIN ROTOR

Diameter	53.67
Blades	4
Chord	1.75 ft.
Airfoil	SC 1095
Blade Area	186.8 ft ²
Solidity	.0826
Tip Sweep	20 Deg.
Twist	-18 Deg.
Shaft Angle	3 Deg.

VERTICAL STABILIZER

Span	8.167 ft.
Area	32.3 ft ²
Root Chord	6 ft.
Tip Chord	2.83 ft.
Sweep ($\frac{1C}{4}$)	41 Deg.
Aspect Ratio	1.92
Airfoil	NACA 0021 (Mod)

TAIL ROTOR

Diameter	11 ft.
Blades	4
Chord	.81 ft.
Airfoil	SC 1095
Blade Area	17.82 ft ²
Solidity	.1875
Twist	-18 Deg.
Cant Angle	20 Deg.

GENERAL

Overall Length	64.83 ft.
Fuselage Length	50.06 ft.
Wheel Tread	8.88 ft.
Wheel Base	28.93 ft.

HORIZONTAL STABILATOR

Span	14.38 ft.
Area	45 ft ²
Root Chord	3.67 ft.
Tip Chord	2.54 ft.
Sweep ($\frac{1C}{4}$)	0 Deg.
Aspect Ratio	4.6
Airfoil	NACA 0014
Incidence/Dihedral	0 Deg.

BLACK HAWK

TABLE 4.2. SUMMARY OF MASS PROPERTIES CHARACTERISTICS

CONDITION	WEIGHT	CENTER OF GRAVITY POSITION		MOMENT OF INERTIA lb-In-Sec ²			
		STA	WL	I _{xx}	I _{yy}	I _{zz}	I _{xz}
Design Mission - Troops	16000.9	358.0	251.0	65550	473626	442646	18886
Aeromedical Mission	15479.3	359.0	251.1	64058	475389	441954	19510
Aerial Recovery Mission	20250.0	359.6	234.7	100200	502116	430804	22130
Extended Range Mission	19193.7	352.5	245.1	74633	502044	461813	28076
Basic Structural Design-Fwd.	16330.9	345.7	248.3	71141	500923	465328	34144
Basic Structural Design-Aft	16330.9	360.2	249.5	68263	465774	432719	18268
Maximum Alternate GW-Fwd	20250.0	347.1	244.4	79532	514803	479012	33850
Maximum Alternate GW-Aft	20250.0	360.2	245.1	77898	482141	447627	18408

(Data from SER-70288 Prepared Under Contract DAAJ01-77-C-0001(P6A))

5.1 MAIN ROTOR MODULE

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5.1 Main Rotor Module
5.1.1 Module Description

The main rotor model is based on a blade element analysis in which total rotor forces and moments are developed from a combination of aerodynamic, mass and inertia loads acting on each simulated blade. The blade segment set up option defined for this Black Hawk model is that of equal annuli area swept by the segment. This technique allows the number of segments to be minimized and distributes the segments towards the higher dynamic pressure areas.

The total forces acting on the blade are derived from the total acceleration and velocity components at the blade together with control inputs. Accelerations develop from body motion and blade motion. Velocity components are made up of body velocities, gust velocities, the rotor's own downwash and blade motion.

Before calculations at the blade segment can be executed several axes transformations must be implemented. Initially body axes angular and translational accelerations and velocities are transferred to the rotor hub and rotated through the shaft inclination angles i_1 and i_2 into rotor shaft axes. These angles with positive rotation of i_1 about the Y_H axis followed by i_2 about the resulting X_1 axis are shown in Figure 1.1.1. The body velocities are non-dimensionalized by rotor tip speed ($S_T R_T$) to conform with usual helicopter analysis practice. Motion accruing from the rotor shaft degree of freedom is derived from the engine module.

The rotor air mass degree of freedom is primarily based on a uniform downwash distribution developed from rotor thrust by application of momentum theory. This uniform downwash, which is passed through a first order lag, is modified to account for the changing distribution with forward speed and aerodynamic pitching and rolling moment loading on the rotor. In the first case the resulting uniform downwash is distributed 1st harmonically around the azimuth as a cosine function depending on the inclination of the rotor wake. The desirability of including this first harmonic distribution, which results in a uniform downwash at hover and a weighted distribution towards the aft of the rotor disc at high speed, is discussed in Reference 1.6.1. Since this effect is really dependent on the resultant velocity vector a lateral velocity term is also added. The desirability of adding a harmonic distribution of downwash depending on aerodynamic rotor moments has not been established for Black Hawk but the necessary equations are incorporated for completeness.

The remaining contribution to the velocities at the blade segment is that due to blade motion. Blade flapping and lagging velocities and angles are obtained by application of a Fourier prediction technique, rather than direct integration of acceleration. This approach is derived in Reference 1.6.2. This method although simple has been shown to be stable and accurate for the frequencies of concern in helicopter stability and control evaluations.

The blade segment total velocity components are developed in three parts. Those independent of segment position, those dependent on segment position and interference effects made up of downwash and gust effects. The velocities at the blade segments are obtained by transforming the fixed shaft vectors into the rotating hub axes system, then transferring to the blade hinge position, transforming into blade span axes through the Euler angles β (flapping) and δ (lagging) and finally transferring to the segment position on the blade. These transformations are illustrated on figure 1.1.2. These total velocity components are subsequently used to calculate the resultant velocity, local Mach number, yawed angle of attack and flow yaw angle. It should be noted that the radial component of velocity is omitted in calculating the Mach Number which is used in the aerodynamic map look-up. Reference 1.6.3, which describes the use of simple sweep theory, indicates that Mach Number should be based on the unyawed component of flow.

The total local segment angle of attack on the blade is made up of the blade local pitch angle and the yawed angle of attack at the segment. The former is made up of the control impressed pitch (collective θ_{CUFF} , cyclic A_{1S} , B_{1S}), pitch/flap coupling (S_2), pitch/lag coupling (α_1), preformed blade twist and a dynamic component of blade twist due to torsional elastic deformation. This empirical dynamic component is prescribed harmonically based on blade loading. The yawed angle of attack is complicated by the requirement to resolve blade pitch into the local stream direction as shown in Figure 1.1.3. The resulting equation assumes the series approximation for the tangent of blade pitch.

The treatment of the blade segment aerodynamic force calculation is completely non-linear. Lift and drag characteristics are provided for the range $-180 \leq \alpha \leq 180$. Bivariate maps as a function of angle of attack and Mach Number are used in the range $-30 \leq \alpha \leq +30$ allowing good definition of blade stall. The complete coverage of angle of attack allows good definition of aerodynamic characteristics on the retreating blade side of the disc. This is important at high advance ratios. The blade segment lift coefficient is determined by applying simple sweep theory to the unyawed blade aerodynamic data. This is rigorously applied in the linear lift range where the entry to the unyawed lift coefficient is transformed by the cosine of the yaw angle (i.e. $\alpha_{\text{TRANS}} = \alpha_Y \cos \gamma$) and the entry Mach Number is a function of the unyawed component of flow. At higher angles of attack some liberties are taken where sweep theory is not valid. These steps are taken to avoid discontinuities in blade lift data as the blade proceeds around the azimuth. Discontinuities can result in an unstable flapping and lagging solution. The application of sweep theory to the determination of drag is not well established. For this model, drag is determined by entering the drag map data with the actual yawed angle of attack. ($\alpha_{\text{TRANS}} = \alpha_Y$). A development of sweep theory can be found in Reference 1.6.3. Sikorsky evaluations of this theory, as applied to rotors, are documented in Reference 1.6.4. The map entry logic is developed in the equations. It should be noted that a tip loss factor is applied to the tip segment. Univariate and bivariate maps of blade lift and drag coefficients as a function of α_{TRANS} and Mach Number are given in Section 5.1.5. The aerodynamic segmental loads are resolved from local wind axes into blade span axes and summed along each blade to obtain root shears at the hinge. These forces are subsequently transformed into rotating shaft axes. It should be noted that β and δ are Euler angles and order of treatment must be observed. The aerodynamic moments used in the flapping and lagging motion equations are calculated at this point in the flow. Also, it is convenient to develop the aerodynamic feedbacks for the rotor downwash calculation at this time.

The blade inplane or lagging motion is restrained by a damper system. In the case of Black Hawk the damper is non-linear and the kinematics of this system are complicated by geometry. A generalized representation is utilized as presented on Figure 1.1.4. Essentially the relative positions of the pickups of the damper on the hub and blade are tracked. From these values an axial velocity is determined. This velocity is used to enter the map data presented in Section 5.1.5. The output of this map, which is an axial force, is multiplied by the instantaneous moment arm to obtain the damper moment about the hinge. Although flapping restraint equations are included they will not be activated for Black Hawk.

The contributions to the lagging and flapping motion about the hinge are the aerodynamic moments, the hinge restraint moments and the inertia moments. The aerodynamic and hinge restraint components have been previously discussed. The inertia components have been explicitly introduced into the flapping and lagging equations of motion. It should be noted that lagging motion takes place in an intermediate set of bladespan axes because of the definition of the Euler angles. Also small terms have been eliminated from these equations. A software provision to inhibit the lagging degree of freedom has been incorporated.

Before the final shear forces and hub moments can be developed it is necessary to calculate the inertia shear loading on the hinge pins. Again small terms have been eliminated. Subsequent to these calculations the three component total shears at the hinge are determined and the total rotor forces in fixed shaft axes at the hub center developed. The rotor hub moments result from a combination of the shear forces at the hinge pins and moments from the blade hinge restraint. An arithmetic manipulation of the equations is introduced on these final equations which allow the simulated blades to be different from the actual number. This artifice is intended for use in piloted simulation where computer execution time is critical. With the lagging degree of freedom operating, the major portion of rotor torque is developed through the lag damper. Therefore, if lagging degree of freedom is selected out, an alternative equation containing the aerodynamic moment must be introduced as specified.

The oscillating nature of the rotor forces and moments make it expedient to filter the outputs under some circumstances. A simple first order filter is used. The final rotor forces and moments are obtained by transforming the filtered shaft axes forces and moments into body axes with the origin at the center of gravity. These are eventually summed with other component outputs to give the total external forces and moments at the center of gravity.

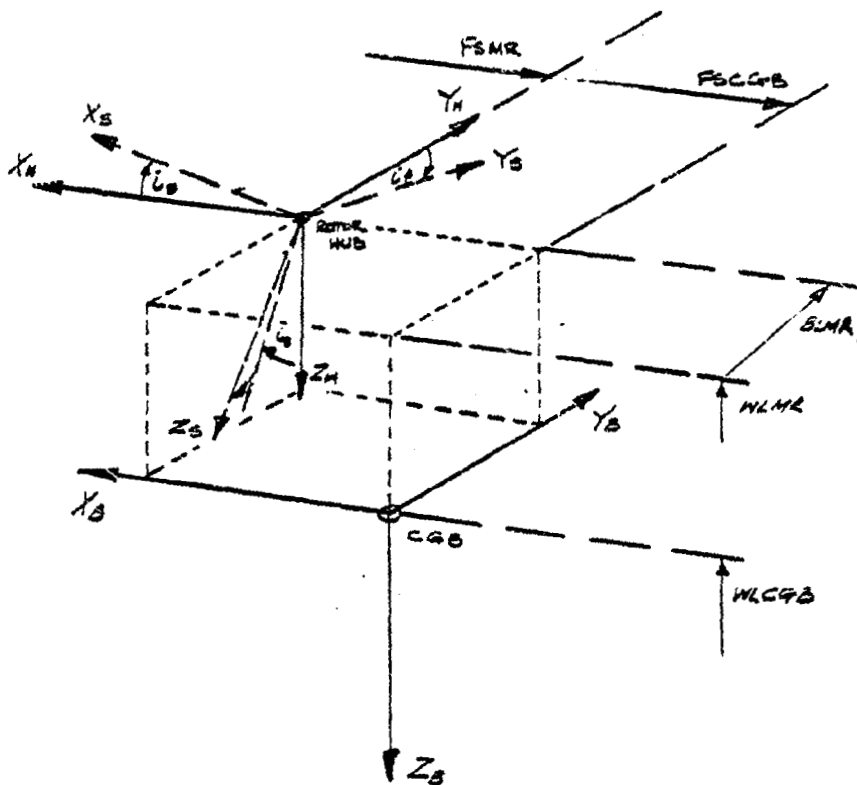
It is necessary to make provision in these final rotor outputs for the option of selecting to run with the engine module in or out. If the engine is selected out, perfect rotor speed governing is assumed and the shaft torque reaction on the airframe is assumed equal to rotor required torque. If the engine module is activated then its output torque is introduced into the airframe.

Finally, the rotor wake skew angle is determined. This is the angle that the center line of the rotor wake makes with the rotor shaft. It is the dependent parameter used to establish the variation of rotor wash on the fuselage, wing and tail.

The sequencing of the program flow in the main rotor is critical and should follow the equation flow in Section 5.1.2.

A block diagram of the Main Rotor Module is given on figure 1.1.5. All input data for the Black Hawk Main Rotor is specified in Section 5.1.5.

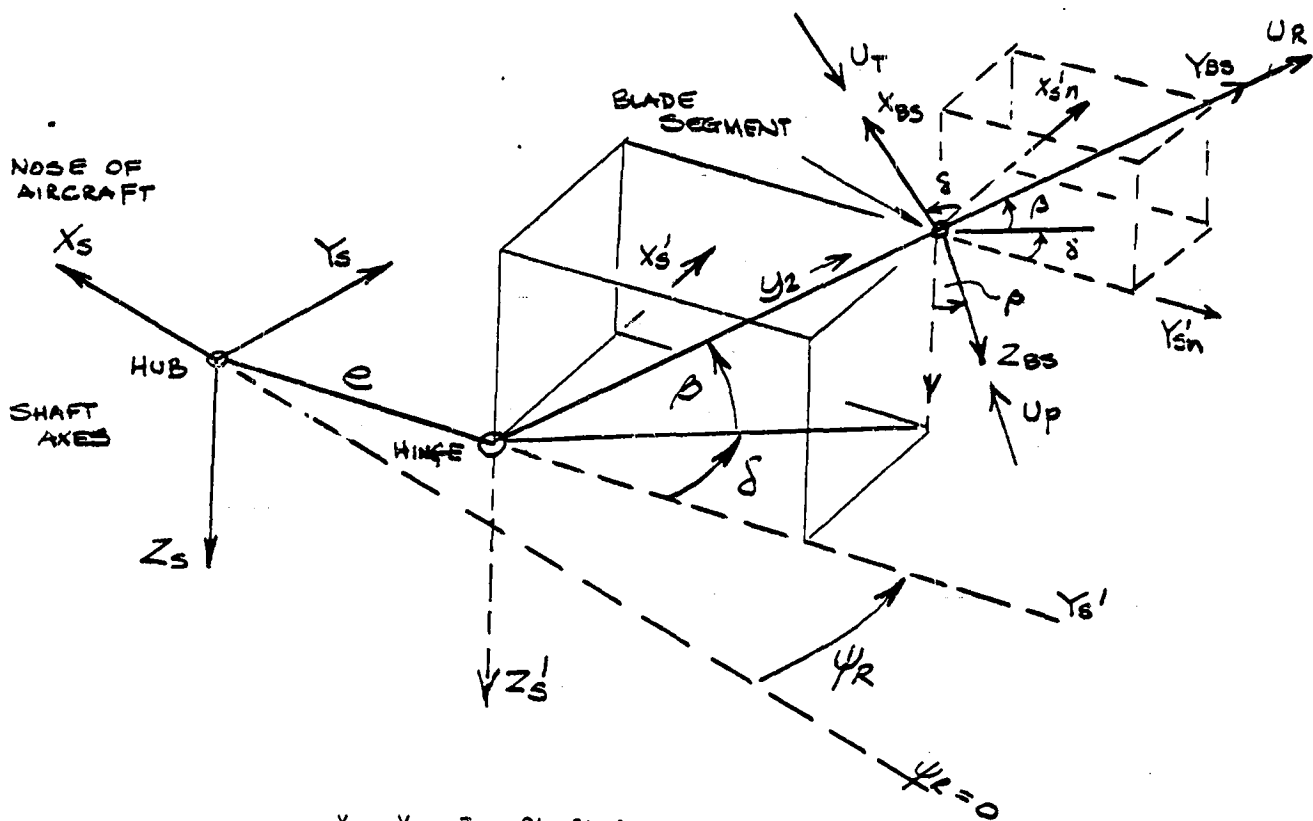
BODY AXES TO SHAFT AXES TRANSFORMATION



X_B, Y_B, Z_B Body Axes System
 X_H, Y_H, Z_H Hub Axes System
 X_S, Y_S, Z_S Shaft Axes System

FIGURE 1.1.1

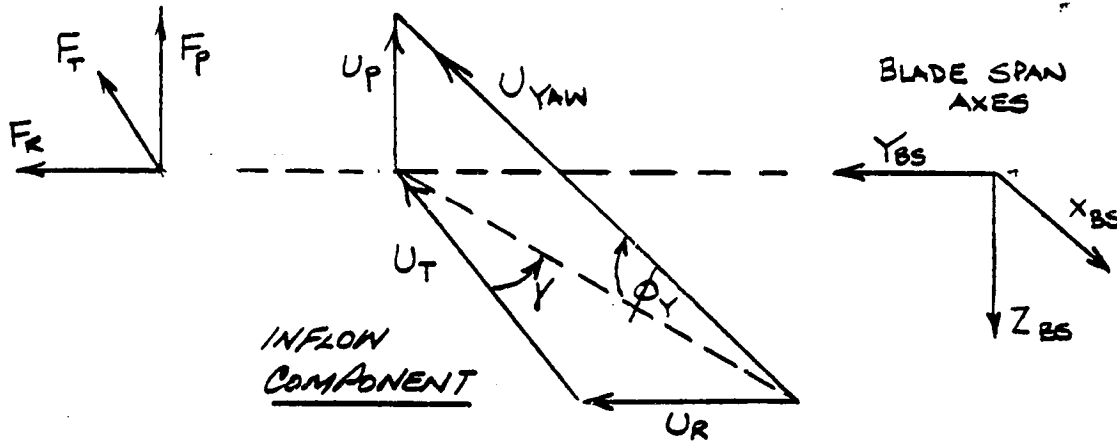
SHAFT AXES TO ROTATING BLADE SPAN AXES
TRANSFORMATION



X_s, Y_s, Z_s Shaft Axes System
 X'_s, Y'_s, Z'_s Rotating Shaft Axes System
 X_{BS}, Y_{BS}, Z_{BS} Blade Span Axes System
 U_T, U_R, U_p Blade Element Velocities Along
 X_{BS}, Y_{BS}, Z_{BS} Respectively
 δ and β are Euler Angles with δ Rotation about
 Z'_s then β Rotation About X_{BS}.

FIGURE 1.1.2

DEFINITION OF YAWED ANGLE OF ATTACK



TOTAL YAWED ANGLE OF ATTACK = $\phi_Y + \theta_Y$

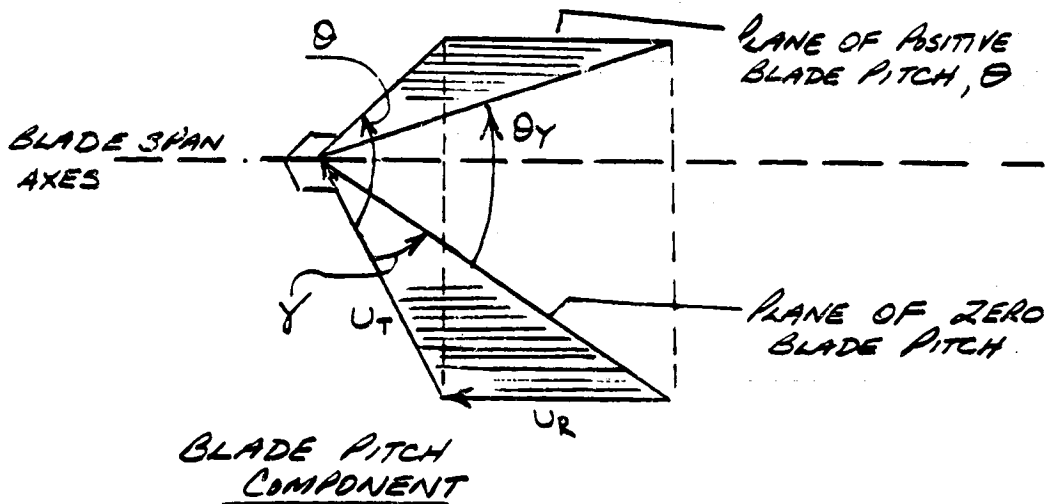
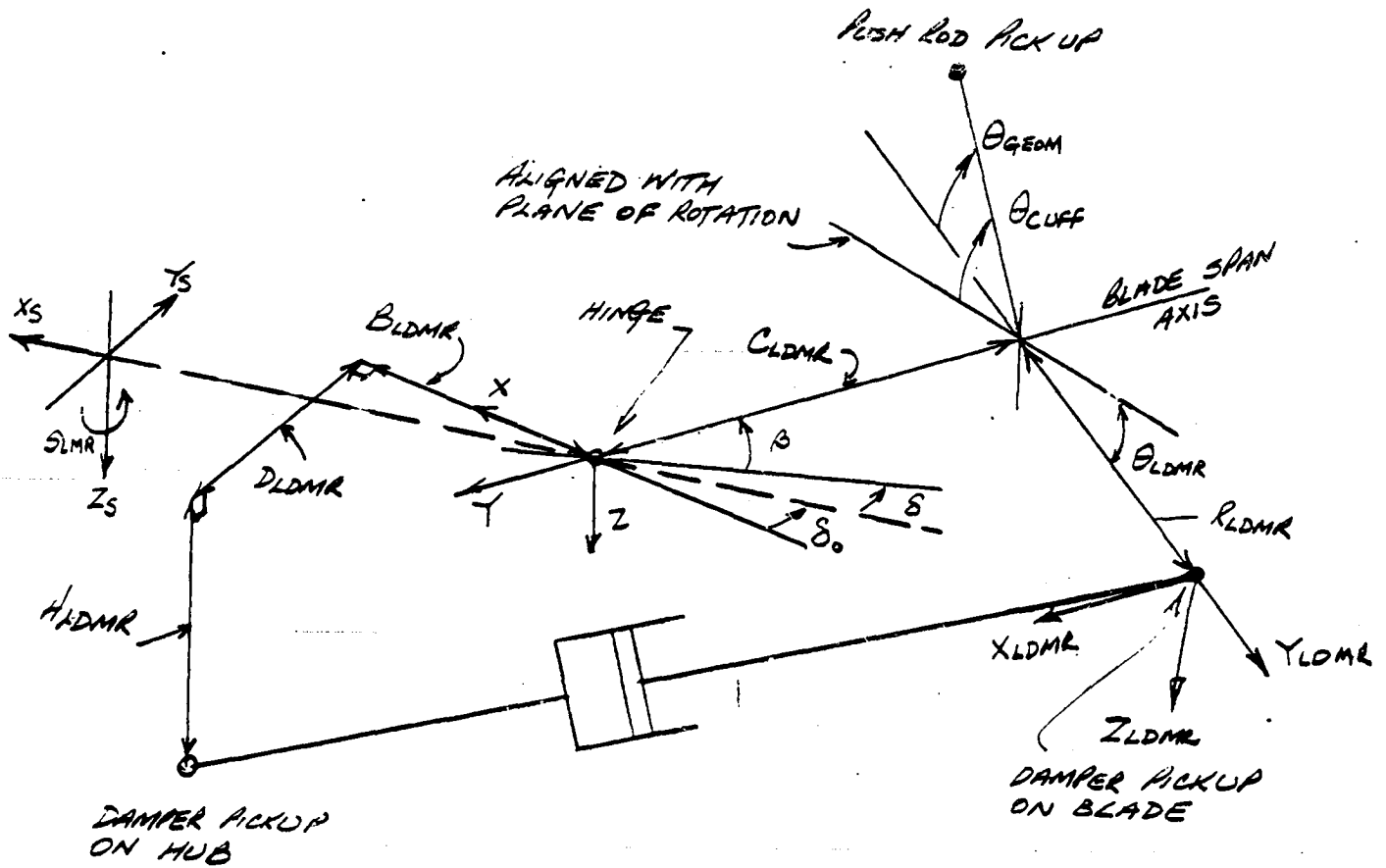


FIGURE 1.1.3

LAG DAMPER KINEMATIC GEOMETRY



$(XYZ)_{LDMR}$ Lag Damper Axes
 X_{LDMR} Aligned with the blade span
 and $(Y,Z)_{LDMR}$ rotated through θ_{LDMR}

FIGURE 1.1.4

MAIN ROTOR FLOW DIAGRAM

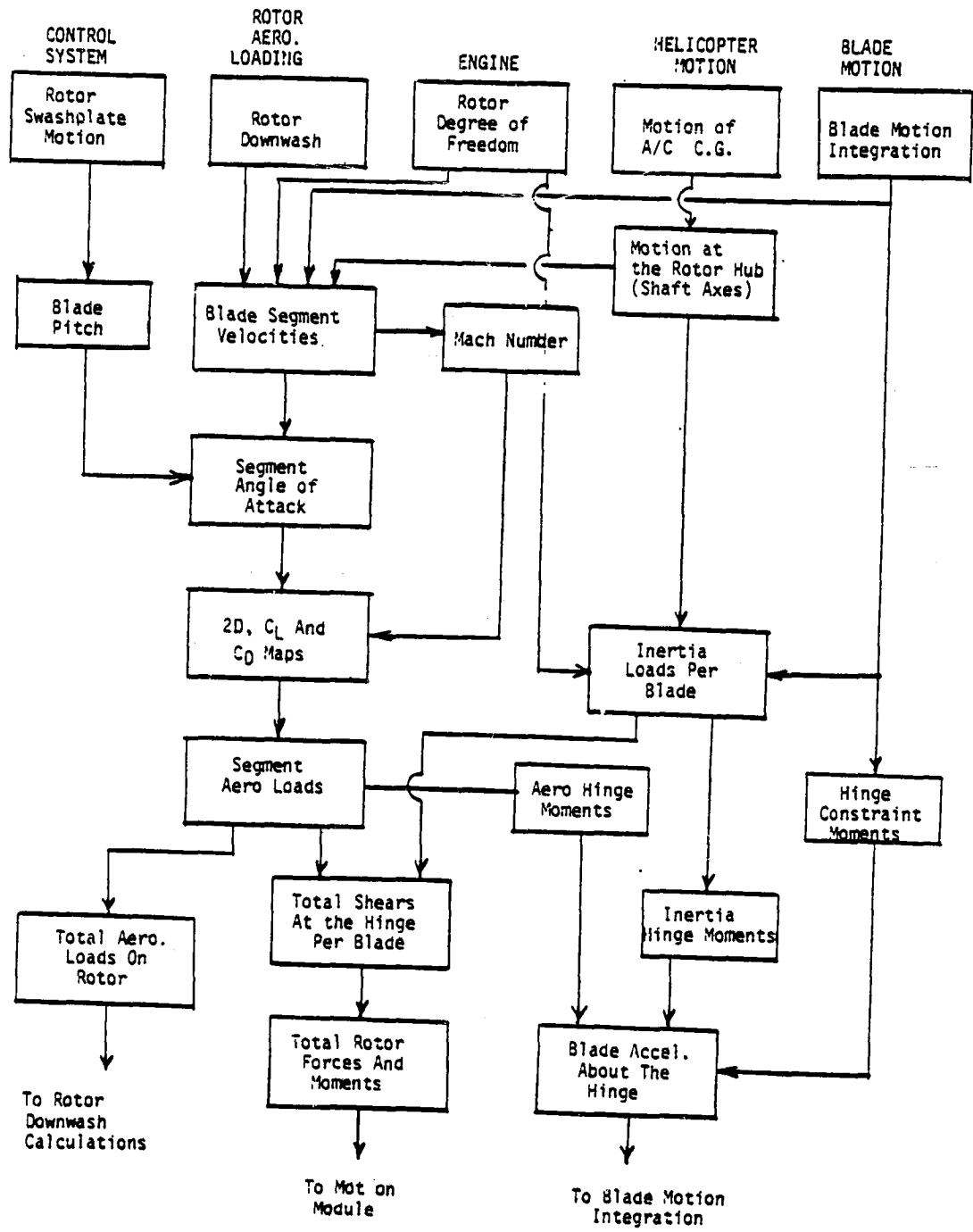
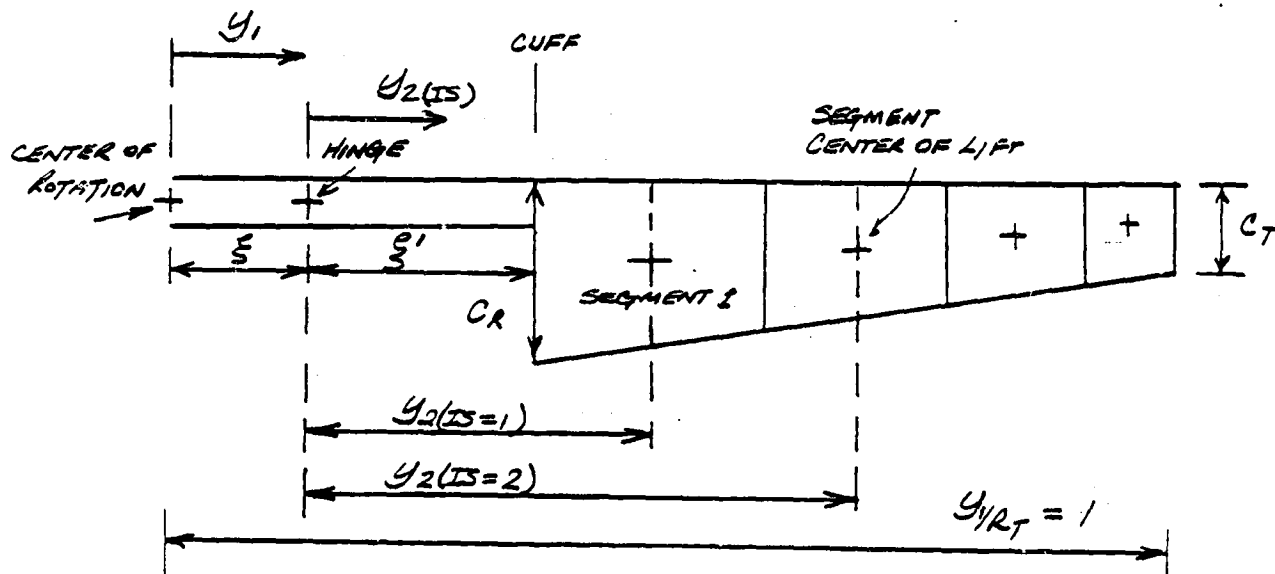


FIGURE 1.1.5

5.1.2 MAIN ROTOR MODULE EQUATIONS

BLADE GEOMETRY

BLADE SEGMENTS BASED ON EQUAL ANNULI AREA



HINGE OFFSET $\xi = e/R_T$ NORMALIZED BY R_T
(KSGMR)

SPAR LENGTH $\xi' = e'/R_T$ NORMALIZED BY R_T

ACTUAL BLADE LENGTH = $1 - \xi - \xi'$ NORMALIZED BY R_T

SEGMENT MIDPOINT FOR FIRST SEGMENT (NORMALIZED) $y_{2,1S} = \left\{ \left[\frac{1 - (\xi + \xi')^2}{2 N_{SS}} \right] + (\xi + \xi')^2 \right\}^{1/2} - \xi$
(KMRBK)

NOTE. THE COMPUTER PROGRAM MNEMONICS FOR THE RHS OF THE EQUATIONS ARE INDICATED THUS (.....).

SUBSEQUENT MIDPOINTS $y_{2IS=2 \dots NSS} = \left\{ \left[\frac{1 - (\xi + \xi')^2}{NSS} \right] + (\xi + y_{2IS-1})^2 \right\}^{1/2} - \xi$

SEGMENT WIDTH $\Delta y_{IS} = [y_{OUTB_{IS}} - y_{INB_{IS}}]$

WHERE $y_{OUTB_{IS}} = \left\{ (y_{2IS} + \xi)^2 + \left(\frac{1 - (\xi + \xi')^2}{2 \cdot NSS} \right) \right\}^{1/2}$

$y_{INB_{IS}} = \left\{ (y_{2IS} + \xi)^2 - \left(\frac{1 - (\xi + \xi')^2}{2 \cdot NSS} \right) \right\}^{1/2}$

SEGMENT MEAN CHORD (FT) $C_{y_{IS}} = \left[\frac{(C_T - C_R) \cdot (y_{OUTB_{IS}} + y_{INB_{IS}} - 2(\xi + \xi'))}{(1 - \xi - \xi') \cdot 2} \right] + C_R$

SEGMENT AREA (FT)² $S_{y_{IS}} = R_T (C_{y_{IS}}) (y_{OUTB_{IS}} - y_{INB_{IS}})$

EFFECT OF ROTOR BLADE WEIGHT ON CG POSITION

$W_{BD} = \text{WEIGHT} - b \cdot W_b$

$F_{CGB} = \frac{(\text{WEIGHT} \cdot F_{SCG} - b \cdot W_b \cdot F_{SMR})}{W_{BD}}$

$W_{LCGB} = \frac{(\text{WEIGHT} \cdot W_{LCG} - b \cdot W_b \cdot W_{LMR})}{W_{BD}}$

$B_{LCGB} = \frac{(\text{WEIGHT} \cdot B_{LCG} - b \cdot W_b \cdot B_{LMR})}{W_{BD}}$

TRANSLATIONAL ACCELERATIONS AT THE ROTOR HUB (BODY AXES)

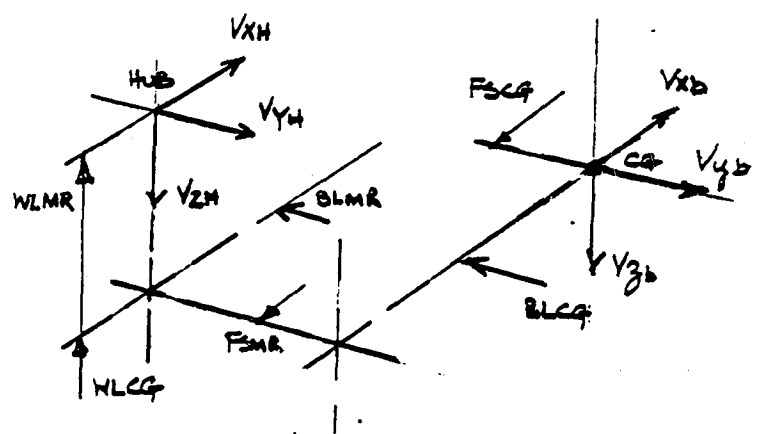
$$\begin{aligned} \dot{V}_{XH} &= \dot{V}_{Xb} - rV_{yb} + qV_{zb} - X_H(q^2 + r^2) + Y_H(pq - \dot{r}) + Z_H(p\dot{r} + \dot{q}) + g_x \\ \dot{V}_{YH} &= \dot{V}_{Yb} - pV_{zb} + rV_{xb} + X_H(pq + \dot{r}) - Y_H(p^2 + r^2) + Z_H(q\dot{r} - \dot{p}) + g_y \\ \dot{V}_{ZH} &= \dot{V}_{Zb} + pV_{yb} - qV_{xb} + X_H(p\dot{r} - \dot{q}) + Y_H(q\dot{r} + \dot{p}) - Z_H(p^2 + q^2) + g_z \end{aligned}$$

WHERE

$$\begin{aligned} g_x &= g \sin \theta_b \\ g_y &= -g \sin \phi_b \cos \theta_b \\ g_z &= -g \cos \phi_b \cos \theta_b \end{aligned}$$

THESE ARE WEIGHT VECTORS ADDED FOR CONVENIENCE AT THIS POINT

$$\begin{aligned} X_H &= (FSCG_b - FMR) / I_2 \\ Y_H &= (BLCG_b - BLMR) / I_2 \\ Z_H &= (WLCG_b - WLMR) / I_2 \end{aligned}$$



TRANSLATIONAL VELOCITIES AT THE ROTOR HUB (BODY AXES)

$$\begin{aligned} M_{XH} &= \frac{1}{\Omega_T R_T} \{ V_{Xb} + V_{Xg} + q \cdot Z_H - r \cdot Y_H \} \\ M_{YH} &= \frac{1}{\Omega_T R_T} \{ V_{Yb} + V_{Yg} + r \cdot X_H - p \cdot Z_H \} \\ M_{ZH} &= \frac{1}{\Omega_T R_T} \{ V_{Zb} + V_{Zg} - q \cdot X_H + p \cdot Y_H \} \end{aligned}$$

BODY TO SHAFT AXES TRANSFORMATION MATRIX

$$[A_{BDSH}] = \begin{bmatrix} \cos \dot{\iota}_\theta & , & 0 & , & -\sin \dot{\iota}_\theta \\ \sin \dot{\iota}_\theta \sin \dot{\iota}_\phi & , & \cos \dot{\iota}_\phi & , & \cos \dot{\iota}_\theta \sin \dot{\iota}_\phi \\ \sin \dot{\iota}_\theta \cos \dot{\iota}_\phi & , & -\sin \dot{\iota}_\phi & , & \cos \dot{\iota}_\theta \cos \dot{\iota}_\phi \end{bmatrix}$$

NOTE $\dot{\iota}_\theta$ AND $\dot{\iota}_\phi$ ARE EULER ANGLES WITH POSITIVE ROTATION OF $\dot{\iota}_\theta$ ABOUT Y_H THE $\dot{\iota}_\phi$ ABOUT A RESULTING X_S

BODY TRANSLATIONAL ACCELERATIONS AT THE HUB
(SHAFT AXES)

$$\begin{bmatrix} \dot{V}_{XS} \\ \dot{V}_{YS} \\ \dot{V}_{ZS} \end{bmatrix} = [A_{BDSH}] \begin{bmatrix} \dot{V}_{XH} \\ \dot{V}_{YH} \\ \dot{V}_{ZH} \end{bmatrix} \begin{matrix} , (V_{XS.MR}) \\ , (V_{YS.MR}) \\ , (V_{ZS.MR}) \end{matrix}$$

BODY ANGULAR ACCELERATIONS AT THE HUB (SHAFT AXES)

$$\begin{bmatrix} \dot{P}_S \\ \dot{Q}_S \\ \dot{R}_S \end{bmatrix} = [A_{BDSH}] \begin{bmatrix} \dot{P}_H \\ \dot{Q}_H \\ \dot{R}_H \end{bmatrix} \begin{matrix} , (P_S.MR) \\ , (Q_S.MR) \\ , (R_S.MR) \end{matrix}$$

BODY TRANSLATIONAL VELOCITIES AT THE HUB (SHAFT AXES)

$$\begin{bmatrix} U_{XS} \\ V_{YS} \\ W_{ZS} \end{bmatrix} = \begin{bmatrix} A_{BDSH} \end{bmatrix} \begin{bmatrix} U_{XH} \\ V_{YH} \\ W_{ZH} \end{bmatrix} \begin{matrix} , (M_{UXSMR}) \\ , (M_{UYSMR}) \\ , (M_{UZSMR}) \end{matrix}$$

BODY ANGULAR VELOCITIES AT THE HUB (SHAFT AXES)

$$\begin{bmatrix} P_S \\ Q_S \\ R_S \end{bmatrix} = \begin{bmatrix} A_{BDSH} \end{bmatrix} \begin{bmatrix} P \\ Q \\ R \end{bmatrix} \begin{matrix} , (P_{SMR}) \\ , (Q_{SMR}) \\ , (R_{SMR}) \end{matrix}$$

ROTOR SHAFT SPEED DEGREE OF FREEDOM

$$OMR.MR = \Omega_T \left(\frac{\dot{\Omega}}{\Omega_T} \right), \left(\frac{\dot{\Omega}}{\Omega_T} \right) \equiv OMR.MR$$

$$OMRMR = \Omega_T \left(\frac{\Omega}{\Omega_T} \right), \left(\frac{\Omega}{\Omega_T} \right) \equiv OMRMR$$

OMR.MR, OMRMR ARE DERIVED FROM THE ENGINE MODULE IF SELECTED. IF NOT SELECTED $\Omega = \Omega_T$

ROTOR AZIMUTH UPDATE CALCULATION

IF DELPSR = 0, $\Delta\psi_R = 57.3 * \Omega * \text{TIME}$

IF DELPSR \neq 0, $\Delta\psi_R = \text{DELPSR}$ TYPICALLY USED IN THE REAL TIME APPLICATION.

$\psi_R = \psi_R + \Delta\psi_R$, IF $\psi_R \leq 180^\circ$, (PSIMR)

$\psi_R = \psi_R + \Delta\psi_R - 360^\circ$, IF $\psi_R > 180^\circ$

SNPSMR_i, CSPSMR_i DERIVED FROM SIN COS (ψ_R) ROUTINE FOR BLADE #1

$SNPSMR_{IB} = \sin \left(\psi_{R, IB-1} + \frac{360}{NBS} \right)$, IB = 2, --- NBS

$CSPSMR_{IB} = \cos \left(\psi_{R, IB-1} + \frac{360}{NBS} \right)$, IB = 2, --- NBS.

FLAPPING RATE AND DISPLACEMENT

$$\dot{\beta}_{IB}^{(t)} = \dot{\beta}_{IB}^{(t-1)} \left(\frac{\sin \Delta\psi_R}{\Omega} \right) + \dot{\beta}_{IB}^{(t-1)} \cos \Delta\psi_R \quad IB = 1, NBS$$

$$\beta_{IB}^{(t)} = \beta_{IB}^{(t-1)} + \dot{\beta}_{IB}^{(t-1)} \frac{\sin \Delta\psi_R}{\Omega} + \left(\frac{1 - \cos \Delta\psi_R}{\Omega^2} \right) \ddot{\beta}_{IB}^{(t-1)} \quad IB = 1, NBS$$

$$ADFMR = \frac{57.3}{b_s} \sum_{IB=1}^{NBS} \beta_{IB}$$

$$AIFMR = -2 * \frac{57.3}{b_s} \sum_{IB=1}^{NBS} \beta_{IB} \cos \psi_{IB}$$

$$BIFMR = -2 * \frac{57.3}{b_s} \sum_{IB=1}^{NBS} \beta_{IB} \sin \psi_{IB}$$

LAGGING RATE AND DISPLACEMENT

$$\dot{\delta}_{IB}^{(t)} = \dot{\delta}_{IB}^{(t-1)} \left(\frac{\sin \Delta\psi_R}{\Omega} \right) + \dot{\delta}_{IB}^{(t-1)} \cos \Delta\psi_R \quad IB = 1, \dots, NBS$$

$$\delta_{IB}^{(t)} = \delta_{IB}^{(t-1)} + \dot{\delta}_{IB}^{(t-1)} \frac{\sin \Delta\psi_R}{\Omega} + \left(\frac{1 - \cos \Delta\psi_R}{\Omega^2} \right) \ddot{\delta}_{IB}^{(t-1)}$$

IB = 1, ... NBS

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OF POOR QUALITY

$$\begin{aligned}ADLMR &= \frac{57.3}{b_s} \sum_{IB=1}^{NBS} \delta_{IB} \\A1LMR &= \frac{2 * 57.3}{b_s} \sum_{IB=1}^{NBS} \delta_{IB} \cos \psi_{IB} \\B1LMR &= \frac{2 * 57.3}{b_s} \sum_{IB=1}^{NBS} \delta_{IB} \sin \psi_{IB}\end{aligned}$$

BLADE FLAPPING AND LAGGING ANGLE COEFFICIENTS

$\left. \begin{array}{l}SNBRMR_{IB} \\CSBRMR_{IB}\end{array} \right\}$ DERIVED FROM $SIN \cos(57.3 \beta_{IB})$ ROUTINE

$\left. \begin{array}{l}SNLQMR_{IB} \\CSBRMR_{IB}\end{array} \right\}$ DERIVED FROM $SIN \cos(57.3 \beta_{IB})$ ROUTINE

$$SNPLMR_{IB} = \sin(\psi_{IB} + \delta_{IB}), \quad IB=1, \dots, NBS$$

$$CSPLMR_{IB} = \cos(\psi_{IB} + \delta_{IB}), \quad IB=1, \dots, NBS$$

MAIN ROTOR AIRMASS DEGREE OF FREEDOM
GLAUERT DOWNWASH FACTORS

$$\begin{matrix} \mu_{TOT} \\ (U_{TOTMR}) \end{matrix} = \left(\mu_{XS}^2 + \mu_{YS}^2 + \lambda_0^2 \frac{2}{(E-1)} \right)^{1/2}$$

$$\begin{matrix} K_{IX} \\ (K_{IXMR}) \end{matrix} = \frac{(\mu_{XS}^2 + \mu_{YS}^2)^{1/2} \mu_{XS}}{\mu_{TOT}^2}$$

$$\begin{matrix} K_{IY} \\ (K_{IYMR}) \end{matrix} = \frac{(\mu_{XS}^2 + \mu_{YS}^2)^{1/2} \mu_{YS}}{\mu_{TOT}^2}$$

UNIFORM PLUS HARMONIC INFLOW

$$\begin{matrix} C_{TA} \\ (C_{THAMR}) \end{matrix} = \frac{T_{HA}}{(\rho \pi S_T^2 R_T^4)}$$

$$\begin{matrix} C_{MHA} \\ (C_{MHAMR}) \end{matrix} = \frac{M_{HA}}{(\rho \pi S_T^2 R_T^5)}$$

$$\begin{matrix} C_{LHA} \\ (C_{LHAMR}) \end{matrix} = \frac{L_{HA}}{(\rho \pi S_T^2 R_T^5)}$$

$$\begin{matrix} D_{wo}(s) = \\ (D_{wshmr}) \end{matrix} = \frac{K_{CT} C_{TA}}{2 \mu_{TOT}} \left\{ \frac{1}{1 + \frac{(T_{DNO})s}{\mu_{TOT}}} \right\}$$

$$\begin{matrix} D_{wc}(s) = \\ (D_{wcmr}) \end{matrix} = \frac{K_{CM} C_{MHA}}{\mu_{TOT}} \left\{ \frac{1}{1 + \frac{(T_{DWC})s}{\mu_{TOT}}} \right\}$$

$$\begin{matrix} D_{ws}(s) = \\ (D_{wsmr}) \end{matrix} = \frac{K_{SM} C_{LHA}}{\mu_{TOT}} \left\{ \frac{1}{1 + \frac{(T_{DWS})s}{\mu_{TOT}}} \right\}$$

NOTE LAG OPERATES ON $\left(\frac{C_{TA}}{\mu_{TOT}}\right)$ ETC.
TO TERM

TOTAL DOWNWASH CONTRIBUTION AT THE ROTOR DISK

$$\begin{aligned} UPDMR_I = & -D_{wo} \cos \beta_{IB} + (D_{wc} - K_{IX} D_{wo}) \cos \beta_{IB} \left\{ \xi \cos \psi_{IB} + \eta_{2n_{IS}} \cos(\psi + \delta)_{IB} \right\} \\ & + (D_{ws} + K_{IX} D_{wo}) \cos \beta_{IB} \left\{ \xi \sin \psi_{IB} + \eta_{2n_{IS}} \sin(\psi + \delta)_{IB} \right\} \end{aligned}$$

$$UTDMR_I = 0$$

$$\begin{aligned} URDMR_I = & -D_{wo} \sin \beta_{IB} + (D_{wc} - K_{IX} D_{wo}) \sin \beta_{IB} \left\{ \xi \cos \psi_{IB} + \eta_{2n_{IS}} \cos(\psi + \delta)_{IB} \right\} \\ & + (D_{ws} + K_{IX} D_{wo}) \sin \beta_{IB} \left\{ \xi \sin \psi_{IB} + \eta_{2n_{IS}} \sin(\psi + \delta)_{IB} \right\} \end{aligned}$$

$$\lambda = \mu_{IS} - D_{wo} + \mu_{GAVMR}$$

$$I_2 = 1 \dots NBS$$

$$I_3 = 1 \dots NSS$$

$$I = 1 \dots (NBS-1) NSS$$

TOTAL BLADE SEGMENT INTERFERENCE VELOCITIES

$$UPIMR_I = UPDMR_I + UPWMR_I + UPGMR_I$$

$$UTIMR_I = UTDMR_I + UTNMR_I + UTGMR_I$$

$$URIMR_I = URDMR_I + URWMR_I + URGMR_I$$

↑ AIRFRAME UPWASH TERMS
NOT CURRENTLY DEFINED FOR
BLACK HAWK.

BLADE SEGMENT VELOCITIES

$$UPAMR_{IB} = -\mu_{25} \sin \beta_{IB} \cos(\psi + \delta)_{IB} + \mu_{15} \sin \beta_{IB} \sin(\psi + \delta)_{IB} + \mu_{25} \cos \beta_{IB} + \frac{e}{S_T} \left\{ \cos \beta_{IB} (q_3 \cos \psi_{IB} + p_3 \sin \psi_{IB}) - \sin \beta_{IB} \sin \delta_{IB} (r_3 - S) \right\}$$

$$JPBMR_{IB} = \left\{ -\dot{\beta}_{IB} + q_3 \cos(\psi + \delta)_{IB} + p_3 \sin(\psi + \delta)_{IB} \right\}$$

$$UP_I = \frac{UPAMR_{IB}}{S_T} + \frac{Y_{2N_{IB}}}{S_T} * UPBMR_{IB} + UPIMR_I$$

(UPMR_I)

$$UTAMR_{IB} = \mu_{15} \sin(\psi + \delta)_{IB} + \mu_{25} \cos(\psi + \delta)_{IB} - \frac{e}{S_T} \cos \delta_{IB} (r_3 - S)$$

$$UTBMR_{IB} = \dot{\delta}_{IB} \cos \beta_{IB} + \sin \beta_{IB} (p_3 \cos(\psi + \delta)_{IB} - q_3 \sin(\psi + \delta)_{IB}) - \cos \beta_{IB} (r_3 - S)$$

$$UT_I = \frac{UTAMR_{IB}}{S_T} + \frac{Y_{2N_{IB}}}{S_T} * UTBMR_{IB} + UTIMR_I$$

(UTMR_I)

$$URAMR_{IB} = \mu_{NS} \cos \beta_{IB} \cos(\psi + \delta)_{IB} - \mu_{YS} \cos \beta_{IB} \sin(\psi + \delta)_{IB} + \mu_{YS} \sin \beta_{IB} + \sum_{S_2} \left\{ \sin \beta_{IB} (q_{YS} \cos \psi_{IB} + p_{YS} \sin \psi_{IB}) + \cos \beta_{IB} \sin \delta_{IB} (1 - S_2) \right\}$$

$$UR_I = URAMR_{IB} + URIMR_I$$

(URMR_I)

WHERE $I_S = 1 \dots NBS$
 $I = 1 \dots NBS + NBS$

RESULTANT VELOCITY AT THE BLADE SEGMENT

$$U_{YI} = (U_{T_I}^2 + U_{P_I}^2 + U_{R_I}^2)^{1/2}$$

(UYAWMR)

MACH NUMBER FOR BLADE MAP LOOK-UP

$$MACHMR_I = (U_{T_I}^2 + U_{P_I}^2)^{1/2} \frac{S_{2T} R_T}{a}$$

YAW ANGLE OF FLOW ON SEGMENT

$$\cos \gamma_I = \frac{|U_{T_I}|}{(U_{T_I}^2 + U_{P_I}^2)^{1/2}}$$

(COSMMR_I)

BLADE SEGMENT GEOMETRIC PITCH ANGLE (BLADE SPAN AXES)

$$\begin{aligned} \theta_{HOAMR}_{IB} = & \theta_{CLIFF} - A_{IS} \cos(\psi_R + \Delta sp)_{IB} - B_{IS} \sin(\psi_R + \Delta sp)_{IB} \\ & - 57.3 \beta_{IB} \tan \delta_3 \\ & + 57.3 \delta_{IB} K_{\alpha 1} + (57.3 \delta_{IB})^2 K_{\alpha 2} \end{aligned}$$

BLADE SEGMENT DYNAMIC TWIST

$$F_{PDYMR}_{IB} = \left[F_{PO}_{IB}^2 + F_{TD}_{IB}^2 \right]^{1/2}$$

$$(F_{POMR}) \quad F_{PO}_{IB} = \frac{1}{b_s} \cdot \left(\frac{b}{b_s} \right) \sum_{IB=1}^{NBS} (F_{PDYMR}_{IB})$$

$$(F_{PCMR}) \quad F_{PC}_{IB} = \frac{2}{b_s} \left(\frac{b}{b_s} \right) \sum (F_{PDYMR}_{IB} \cos(\psi_R + \delta))$$

$$(F_{PSMR}) \quad F_{PS}_{IB} = \frac{2}{b_s} \left(\frac{b}{b_s} \right) \sum (F_{PDYMR}_{IB} \sin(\psi_R + \delta))$$

$$MODESP_{IS} = .28 + .72 \sin [90 \{ \psi_{2IS} + \delta \}]$$

$$\theta_{DYTIP}_{IB} = K_{FPD} F_{PO}_{IB} + K_{FPC} \cdot F_{PC}_{IB} \cos(\psi + \delta) + K_{FPS} \cdot F_{PS}_{IB} \sin(\psi + \delta)$$

$$\theta_{DYMRI} = \theta_{DYTIP}_{IB} \cdot MODESP_{IS}$$

ACTUAL SEGMENT GEOMETRIC BLADE PITCH

$$\theta_I = T_{HOAMR}_{IG} + T_{DYMR}_I + f(X_{SEGMR})$$

$f(X_{SEGMR})$ DEFINES THE PREFORMED BLADE TWIST - TABLE 1.5.1
 $X_{SEGMR} = Y_{2(I)} + \xi$ FIGURE 1.5.1

BLADE SEGMENT ANGLE OF ATTACK

THIS ANGLE OF ATTACK IS RESOLVED IN THE DIRECTION OF THE LOCAL SEGMENT AIRFLOW

$$\alpha_Y = \tan^{-1} \left\{ \frac{[U_{T_I} (\theta_{A_I} + \frac{\theta_{A_I}^3}{3} + \frac{2\theta_{A_I}^5}{15}) + U_{P_I}] / \cos \alpha_I}{[U_{T_I} - U_{P_I} (\theta_{A_I} + \frac{\theta_{A_I}^3}{3} + \frac{2\theta_{A_I}^5}{15}) \cos^2 \alpha_I]} \right\}$$

(AFYNMR)

WHERE $\theta_{A_I} = \frac{\theta_I}{57.3}$ RADS

NOTE: $\tan \theta_{A_I} \approx \theta_{A_I} + \frac{\theta_{A_I}^3}{3} + \frac{2\theta_{A_I}^5}{15}$

TABLE LOOK-UP ALGORITHM FOR SEGMENT AERODYNAMICS

If $\alpha_{yI} \geq 0$

IB = 1...NBS

IS = 1...NSS

I = 1... (NBS-1)NSS

if $0 < \alpha_{yI} / |\cos \delta_I| < ACL1MR$

$$\alpha_{TRANSI} = \alpha_{yI} / |\cos \delta_I|$$

if $ACL1MR < \alpha_{yI} / |\cos \delta_I| \leq (180 / |\cos \delta_I|) - (180 - ACL2MR)$

$$\alpha_{TRANSI} = \frac{(\alpha_{yI} / |\cos \delta_I| - ACL1MR)(ACL2MR - ACL1MR)}{180 / |\cos \delta_I| - 180 + (ACL2MR - ACL1MR)} + ACL1MR$$

if $180 / |\cos \delta_I| - (180 - ACL2MR) < \alpha_{yI} / |\cos \delta_I| < 180 / |\cos \delta_I|$

$$\alpha_{TRANSI} = 180 - (180 - \alpha_{yI}) * |\cos \delta_I|$$

If $\alpha_{yI} < 0$

if $ACL3MR < \alpha_{yI} / |\cos \delta_I| < 0$

$$\alpha_{TRANSI} = \alpha_{yI} / |\cos \delta_I|$$

if $-180 / |\cos \delta_I| + (ACL4MR + 180) \leq \alpha_{yI} / |\cos \delta_I| < ACL3MR$

$$\alpha_{TRANSI} = \frac{(\alpha_{yI} / |\cos \delta_I| - ACL3MR)(ACL4MR - ACL3MR)}{180 - 180 / |\cos \delta_I| + (ACL4MR - ACL3MR)} + ACL3MR$$

if $-180 < \alpha_{yI} < -180 / |\cos \delta_I| + (ACL4MR + 180)$

$$\alpha_{TRANSI} = -180 + (\alpha_{yI} + 180) / |\cos \delta_I|$$

SEGMENT AERODYNAMIC COEFFICIENTS

$$\left. \begin{aligned} C_{LYI} &= f(\alpha_{TRANSI}, MACHMR_I) \\ C_{LYI} &= f(\alpha_{TRANSI}) \end{aligned} \right\} \begin{array}{l} MAP - CLMRMP \\ \text{FIGURE 1.5.2(a),(b)} \\ \text{TABLE 1.5.2} \end{array}$$

(CLMR_I)

$$\left. \begin{aligned} C_{DYI} &= f(\alpha_{YI}, MACHMR_I) \\ C_{DYI} &= f(\alpha_{YI}) \end{aligned} \right\} \begin{array}{l} MAP - CDMRMP \\ \text{FIGURE 1.5.3(a),(b)} \\ \text{TABLE 1.5.3} \end{array}$$

(CDMR_I)

LIFT COEFFICIENT TIP LOSS CORRECTION

$$C_{LYI} = C_{LYI} * (\text{SEGMENT LOSS FACTOR})_I$$

(CLMR_I)

IF THE OUTER END OF THE SEGMENT, $y_{OUTBIS} \leq BTLMR$
(SEGMENT LOSS FACTOR)_I = 1.0

IF THE INNER END OF THE SEGMENT, $y_{INBIS} > BTLMR$
(SEGMENT LOSS FACTOR)_I = 0.0

IF THE INNER END OF THE SEGMENT, $y_{INBIS} \leq BTLMR$
(SEGMENT LOSS FACTOR)_I = $\frac{(BTLMR - y_{INBIS})}{\Delta y_{IS}}$

TOTAL DRAG COEFFICIENT

$$C_{DYI} = C_{DYI} - \Delta C_{DMR}$$

(CDMR_I)

BLADE SEGMENT FORCES (BLADE SPAN AXES)

$$F_{PI} = \frac{1}{2} \rho \Omega_T^2 R_T^3 (C_y \Delta y)_{IS} u_{yI} \left\{ \frac{C_{LYI} U_{TI} + C_{DYI} U_{PI}}{|C_{D\delta I}|} \right\}$$

(FAMR_I)

$$F_{TI} = \frac{1}{2} \rho \Omega_T^2 R_T^3 (C_y \Delta y)_{IS} u_{yI} \left\{ C_{DYI} U_{TI} - C_{LYI} U_{PI} |\cos \gamma_I| \right\}$$

(FTMR_I)

$$F_{RI} = \frac{1}{2} \rho \Omega_T^2 R_T^3 (C_y \Delta y)_{IS} u_{yI} \left\{ C_{DYI} - C_{LYI} \frac{U_{PI}}{U_{TI}} |\cos \gamma_I| \right\} U_{RI}$$

(FRMR_I)

AERODYNAMIC SHEARS PER BLADE (BLADE SPAN AXES)

$$F_{PBIB} = \sum_{IS=1}^{NS} F_{PI}$$

(FABMR)

$$F_{TBIB} = \sum_{IS=1}^{NS} F_{TI}$$

(FTBMR)

$$F_{RBIB} = \sum_{IS=1}^{NS} F_{RI}$$

(FRBMR)

**AERODYNAMIC MOMENTS ABOUT THE HINGE
(BLADE SPAN AXES, FLAPPING MOTION IS IN Y_{IS}-Z_{IS} PLANE)**

FLAPPING

$$M_{FABIB} = R_T \sum_{IS=1}^{NS} y_{2IS} F_{PI}$$

(MFABMR)

LAGGING

$$M_{LABIB} = R_T \sum_{IS=1}^{NS} y_{2IS} F_{TI}$$

(MLABMR)

AERODYNAMIC MOMENTS ABOUT THE HINGE
FIXED SHAFT AXES, FLAPPING COMPONENT ONLY. THESE TERMS
USED AS DRIVER FOR 1ST HARMONIC INFLOW.

$$\begin{matrix} LHA \\ (LHAMR) \end{matrix} = - \frac{b}{b_s} \sum_{IB=1}^{NBS} M_{FAB IB} \sin(\psi + \delta)_{IB}$$

$$\begin{matrix} MHA \\ (MHAMR) \end{matrix} = - \frac{b}{b_s} \sum_{IB=1}^{NBS} M_{FAB IB} \cos(\psi + \delta)_{IB}$$

AERODYNAMIC SHEARS PER BLADE (ROTATING SHAFT AXES)

$$\begin{matrix} F_{XA IB} \\ (FXAMR) \end{matrix} = F_{RB IB} \cos \beta_{IB} \sin \delta_{IB} - F_{TB IB} \cos \delta_{IB} - F_{PB IB} \sin \beta_{IB} \sin \delta_{IB}$$

$$\begin{matrix} F_{YA IB} \\ (FYAMR) \end{matrix} = F_{RB IB} \cos \beta_{IB} \cos \delta_{IB} + F_{TB IB} \sin \delta_{IB} - F_{PB IB} \sin \beta_{IB} \cos \delta_{IB}$$

$$\begin{matrix} F_{ZA IB} \\ (FZAMR) \end{matrix} = - \left[F_{RB IB} \sin \beta_{IB} + F_{PB IB} \cos \beta_{IB} \right]$$

AERODYNAMIC THRUST COMPONENT (SHAFT AXES.)
USED IN UNIFORM DOWNWASH CALCULATION

$$\begin{matrix} THA \\ (THAMR) \end{matrix} = - \frac{b}{b_s} \sum_{IB=1}^{NBS} F_{ZA IB}$$

BLADE LAG DAMPER KINEMATICS

COMPONENTS OF LAG DAMPER DISPLACEMENT

$$X_{LDNR}_{IB} = A_{LDNR} \sin \beta_{IB} + B_{LDNR} \cos(\delta + \delta_0)_{IB} \cos \beta_{IB} + C_{LDNR} + D_{LDNR} \sin(\delta + \delta_0)_{IB} \cos \beta_{IB}$$

$$Y_{LDNR}_{IB} = -R_{LDNR} \cos \theta_{LDNR}_{IB} - B_{LDNR} \sin(\delta + \delta_0)_{IB} + D_{LDNR} \cos(\delta + \delta_0)_{IB}$$

$$Z_{LDNR}_{IB} = A_{LDNR} \cos \beta_{IB} - R_{LDNR} \sin \theta_{LDNR}_{IB} - C_{LDNR} \sin \beta_{IB} \cos(\delta + \delta_0)_{IB} - D_{LDNR} \sin \beta_{IB} \sin(\delta + \delta_0)_{IB}$$

WHERE

$$\theta_{LDNR}_{IB} = \theta_{HOANR}_{IB} - \theta_{GEONR}$$

$$L_{DT}_{IB} = \left(X_{LDNR}_{IB}^2 + Y_{LDNR}_{IB}^2 + Z_{LDNR}_{IB}^2 \right)^{1/2} \text{ INS}$$

(LDNR)

$$\dot{L}_{DT}_{IB} = \left[L_{DT}_{(k)} - L_{DT}_{(k-1)} \right] \frac{1}{\Delta t}$$

(LD.MR)

$$F_{\delta}_{IB} = \frac{\dot{L}_{DT}_{IB}}{|L_{DT}_{IB}|} f(L_{DT}_{IB})$$

(FLD.MR)

MAP-LDNRMP
FIGURE 1.5.4(a), (b)
TABLE 1.5.4

$$M_{LLD}_{IB} = -F_{\delta}_{IB} \left[R_{LDNR} \cos \theta_{LDNR}_{IB} \left\{ \frac{X_{LDNR}_{IB} \cos \beta_{IB}}{L_{DT}} - \frac{Z_{LDNR}_{IB} \sin \beta_{IB}}{L_{DT}} \right\} + \frac{Y_{LDNR}_{IB}}{L_{DT}} \left\{ C_{LDNR} \cos \beta_{IB} + R_{LDNR} \sin \theta_{LDNR}_{IB} \sin \beta_{IB} \right\} \right]_{IB}$$

$$M_{FLD} = -F_{\dot{\delta}} \left[\frac{Z_{LDMR}}{LDT} C_{LDMR} + \frac{X_{LDMR}}{LDT} R_{LDMR} \sin \theta_{LDMR} \right]_{IB}$$

(MFLDMR)

$$M_{FFD} = - \left[K_{\beta} \beta + K_{\dot{\beta}} \dot{\beta} \right]_{IB}$$

(MFFDMR)

$$M_{LFD} = 0$$

(MLFDMR)

$$\Delta L_{HBC} = (M_{FLD} + M_{FFD})_{IB} \sin(\psi + \delta)_{IB}$$

(DLHBMR)

$$\Delta M_{HBC} = (M_{FLD} + M_{FFD})_{IB} \cos(\psi + \delta)_{IB}$$

(DMHBMR)

$$\Delta N_{HBC} = M_{LLD}_{IB}$$

(DNHBMR)

BLADE LAGGING DEGREE OF FREEDOM (ROTATING SHAFT AXES)

$$\ddot{\delta}_{IB} = \frac{M_0}{I_b \cos \beta} \left[\sin \delta \left\{ \dot{V}_{ys} \sin \psi_{IB} - \dot{V}_{xs} \cos \psi_{IB} - e(\dot{\tau}_s - \dot{\Omega})^2 \right\} - \cos \delta \left\{ \dot{V}_{xs} \sin \psi_{IB} + \dot{V}_{ys} \cos \psi_{IB} + e(\dot{\Omega} - \dot{\tau}_s) \right\} \right]$$

(LG...MR)

$$+ \frac{\sin \beta_{IB}}{\cos \beta_{IB}} \left[2 \dot{\beta}_{IB} (\dot{\Omega} + \dot{\delta}_{IB} - \dot{\tau}_s) + \dot{q}_s \sin(\psi + \delta)_{IB} - \dot{p}_s \cos(\psi + \delta)_{IB} \right] + (\dot{\tau}_s - \dot{\Omega})$$

$$+ 2 \dot{\beta}_{IB} \left[\cos \delta \left(\dot{q}_s \sin \psi_{IB} - \dot{p}_s \cos \psi_{IB} \right) + \sin \delta \left(\dot{p}_s \sin \psi_{IB} + \dot{q}_s \cos \psi_{IB} \right) \right]$$

$$+ \left[\frac{M_{LLD} + M_{LFD}}{I_b \cos^2 \beta} \right]_{IB} - \left[\frac{M_{LAB}}{I_b \cos \beta} \right]_{IB}$$

SOFTWARE SWITCH - NO LAG DEGREE OF FREEDOM, $\ddot{\delta} = \dot{\delta} = \delta = 0$

BLADE FLAPPING DEGREE OF FREEDOM (ROTATING SHUNT AXES)

$$\begin{aligned}
 \ddot{\beta}_{10} &= \frac{M_b}{I_b} \left[\cos \beta_{10} \left\{ \dot{V}_{2s} + e \left[2\Omega (p_s \cos \psi_{10} - q_s \sin \psi_{10}) + \dot{p}_s \sin \psi_{10} + \dot{q}_s \cos \psi_{10} \right] \right\} \right. \\
 &\quad \left. + \sin \beta_{10} \cos \delta_{10} \left\{ \dot{V}_{1s} \sin \psi_{10} - \dot{V}_{2s} \cos \psi_{10} - e (\tau_s - \Omega)^2 \right\} \right] \\
 &\quad + \cos^2 \beta_{10} \left[\dot{\delta}_{10} \left\{ \dot{p}_s \sin \psi_{10} + \dot{q}_s \cos \psi_{10} - 2(\dot{\delta}_{10} + \Omega) (q_s \sin \psi_{10} - p_s \cos \psi_{10}) \right\} \right. \\
 &\quad \left. - 2\Omega \sin \delta_{10} (p_s \sin \psi_{10} + q_s \cos \psi_{10}) \right] \\
 &\quad + \cos \beta_{10} \sin \beta_{10} \left[2\dot{\delta}_{10} (\tau_s - \Omega) - (\tau_s - \Omega)^2 \right] \\
 &\quad + \frac{(M_{FAB} + M_{FFD} + M_{FLD})_{10}}{I_b}
 \end{aligned}$$

INERTIA SHEARS AT THE HINGE PER BLADE

$$\begin{aligned}
 F_{XI_{1B}} &= M_b \left[\cos \beta_{1B} \cos \delta_{1B} (\dot{\gamma}_s - \dot{\Omega} - \ddot{\delta}_{1B}) + 2 \sin \beta_{1B} \cos \delta_{1B} \left\{ \dot{\delta}_{1B} \dot{\beta}_{1B} - (\gamma_s - \Omega) \dot{\beta}_{1B} \right\} \right. \\
 (FXIMR) \quad &+ \cos \beta_{1B} \sin \delta_{1B} \left\{ \dot{\delta}_{1B}^2 + \dot{\beta}_{1B}^2 - 2(\gamma_s - \Omega) \dot{\delta}_{1B} + (\gamma_s - \Omega)^2 \right\} \\
 &+ 2 \dot{\beta}_{1B} \cos \beta_{1B} (p_s \cos \psi_{1B} - q_s \sin \psi_{1B}) + \sin \beta_{1B} \sin \delta_{1B} \ddot{\beta}_{1B} \left. \right] \\
 &- \frac{W_b}{g} \left[\dot{V}_{Xs} \sin \psi_{1B} + \dot{V}_{Ys} \cos \psi_{1B} \right]
 \end{aligned}$$

$$\begin{aligned}
 F_{YI_{1B}} &= M_b \left[\cos \beta_{1B} \cos \delta_{1B} \left\{ \dot{\delta}_{1B}^2 + \dot{\beta}_{1B}^2 - 2(\gamma_s - \Omega) \dot{\delta}_{1B} + (\gamma_s - \Omega)^2 \right\} \right. \\
 (FYIMR) \quad &+ \sin \beta_{1B} \cos \delta_{1B} (\ddot{\beta}_{1B}) + \cos \beta_{1B} \sin \delta_{1B} (\ddot{\delta}_{1B}) \\
 &- 2 \dot{\beta}_{1B} \cos \beta_{1B} (p_s \sin \psi_{1B} + q_s \cos \psi_{1B}) + \frac{W_b c}{g M_b} (\gamma_s - \Omega)^2 \left. \right] \\
 &+ \frac{W_b}{g} \left[\dot{V}_{Xs} \cos \psi_{1B} - \dot{V}_{Ys} \sin \psi_{1B} \right]
 \end{aligned}$$

$$\begin{aligned}
 F_{ZI_{1B}} &= M_b \left[\ddot{\beta}_{1B} \cos \beta_{1B} - \dot{\beta}_{1B}^2 \sin \beta_{1B} + \sin \beta_{1B} \cos \delta_{1B} 2 \dot{\beta}_{1B} (p_s \sin \psi_{1B} + q_s \cos \psi_{1B}) \right. \\
 (FZIMR) \quad &+ \cos \beta_{1B} \sin \delta_{1B} \left\{ 2(\Omega + \dot{\delta}_{1B}) (p_s \sin \psi_{1B} + q_s \cos \psi_{1B}) + \dot{q}_s \sin \psi_{1B} - \dot{p}_s \cos \psi_{1B} \right\} \\
 &- \cos \beta_{1B} \cos \delta_{1B} \left\{ 2(\Omega + \dot{\delta}_{1B}) (p_s \cos \psi_{1B} - q_s \sin \psi_{1B}) + \dot{p}_s \sin \psi_{1B} + \dot{q}_s \cos \psi_{1B} \right\} \\
 &- \frac{W_b c}{g M_b} \left\{ 2\Omega (p_s \cos \psi_{1B} - q_s \sin \psi_{1B}) + \dot{p}_s \sin \psi_{1B} + \dot{q}_s \cos \psi_{1B} \right\} \left. \right] \\
 &- \frac{W_b}{g} \left[\dot{V}_{Zs} \right]
 \end{aligned}$$

TOTAL SHEAR FORCE AT THE HINGE (ROTATING AXES)

$$\begin{matrix} F_{XT_{IB}} \\ (F_{XTMR}) \end{matrix} = F_{XA_{IB}} + F_{XI_{IB}}$$

$$\begin{matrix} F_{YT_{IB}} \\ (F_{YTMR}) \end{matrix} = F_{YA_{IB}} + F_{YI_{IB}}$$

$$\begin{matrix} F_{ZT_{IB}} \\ (F_{ZTMR}) \end{matrix} = F_{ZA_{IB}} + F_{ZI_{IB}}$$

TOTAL ROTOR FORCES (FIXED SHAFT AXES)

$$\begin{matrix} T_H \\ (T_{HMR}) \end{matrix} = - \frac{b}{b_S} \sum_{IB=1}^{NBS} F_{ZT_{IB}}$$

$$\begin{matrix} H_H \\ (H_{HMR}) \end{matrix} = \frac{D}{b_S} \sum_{IB=1}^{NBS} (F_{YT} \cos \psi - F_{XT} \sin \psi)_{IB}$$

$$\begin{matrix} J_H \\ (J_{HMR}) \end{matrix} = - \frac{b}{b_S} \sum_{IB=1}^{NBS} (F_{XT} \cos \psi + F_{YT} \sin \psi)_{IB}$$

ROTOR HUB MOMENTS (FIXED SHAFT AXES)

$$\begin{matrix} M_H \\ \text{(MHMR)} \end{matrix} = \frac{b}{b_s} \sum_{IB=1}^{NBS} (e F_{ZT_{IB}} \cos \psi_{IB} + \Delta M_{HB_{IB}})$$

$$\begin{matrix} L_H \\ \text{(LHMR)} \end{matrix} = \frac{b}{b_s} \sum_{IB=1}^{NBS} (e F_{ZT_{IB}} \sin \psi_{IB} + \Delta L_{HB_{IB}})$$

$$\begin{matrix} Q_H \\ \text{(QHMR)} \end{matrix} = -\frac{b}{b_s} \sum_{IB=1}^{NBS} (e F_{XT_{IB}} - \Delta N_{HB_{IB}})$$

IF LAG DEGREE OF FREEDOM IS FIXED

$$\text{SET - } \delta = \dot{\delta} = \ddot{\delta} = 0, \quad M_{LD_{IB}} = M_{FLD_{IB}} = 0$$

$$\begin{matrix} \text{SET } Q_H \\ \text{(QHMR)} \end{matrix} = -\frac{b}{b_s} \sum_{IB=1}^{NBS} (e F_{XT_{IB}} - M_{LAB_{IB}} \cos \beta_{IB})$$

MAIN ROTOR OUTPUT FILTER

$$\begin{bmatrix} H_{HB} \\ J_{HB} \\ T_{HB} \\ L_{HB} \\ M_{HB} \\ Q_{HB} \end{bmatrix} = \frac{1}{(T_{FILMR} S + 1)} \begin{bmatrix} H_H \\ J_H \\ T_H \\ L_H \\ M_H \\ Q_H \end{bmatrix}$$

H FILTER CONSTANT
 $T_{FILMR} < \Delta C$ ELIMINATES
THE FILTER

ROTOR DEGREE OF FREEDOM.

THESE EQUATIONS ARE DEFINED IN THE ENGINE MODULE

ROTOR FORCE AND MOMENT OUTPUT TRANSFORMATION

TRANSFORMATION OF ROTOR FORCES INTO BODY AXES

$$\begin{bmatrix} X_{MR} \\ Y_{MR} \\ Z_{MR} \end{bmatrix} = \begin{bmatrix} A_{SHBD} \\ \cdot \\ \cdot \end{bmatrix} \begin{bmatrix} -H_{HB} \\ -J_{HB} \\ -T_{HB} \end{bmatrix}$$

$$\begin{bmatrix} L_{MR} \\ M_{MR} \\ N_{MR} \end{bmatrix} = \begin{bmatrix} A_{SHBD} \\ \cdot \\ \cdot \end{bmatrix} \begin{bmatrix} L_{HB} \\ M_{HB} \\ Q_E \end{bmatrix} + \begin{bmatrix} Y_H Z_{MR} - Z_H Y_{MR} \\ Z_H X_{MR} - X_H Z_{MR} \\ X_H Y_{MR} - Y_H X_{MR} \end{bmatrix}$$

WHERE $[A_{SHBD}] = [A_{BDSH}]^T$

$Q_E = Q_{HB}$ IF ROTOR DEGREE OF FREEDOM IS NOT
SELECTED

$Q_E = Q_E$ FROM THE ENGINE MODULE
(QHEG)

ROTOR WAKE SKEW ANGLE

$$\gamma_{PMR} = \tan^{-1} \left\{ \frac{\mu_{XS}}{1.71} \right\} + AAIFMR$$

(CHIPMR)

ROTOR HORSEPOWER REQUIRED

$$HPMR = \frac{Q_{HB} * S_{MR}}{550}$$

ROTOR FORCE AND MOMENT COEFFICIENTS

$$\frac{C_T}{\delta}, \frac{C_H}{\delta}, \frac{C_J}{\delta} = \frac{1}{\rho b C_{T0.75} S_T^2 R_T^3} \{ T_{HB}, H_{HB}, J_{HB} \}$$

$$\frac{C_M}{\delta}, \frac{C_L}{\delta}, \frac{C_Q}{\delta} = \frac{1}{\rho b C_{T0.75} S_T^2 R_T^4} \{ M_{HB}, L_{HB}, Q_{HB} \}$$

5.1.3 MAIN ROTOR MODULE INPUT/OUTPUT DATA TRANSFER

INPUT TRANSFER	
*PARAMETER	ORIGIN MODULE
THETA ϕ	FLIGHT CONTROL
ALS	
BIS	
OMR.MR	ENGINE
OMRMR	
QHBEQ	
DWSHMR	GROUND EFFECTS
URGMRI	GUST
UTGMRI	
URGMRI	
UGAVMR	
VXB DOT	MOTION
YB DOT	
VZB DOT	
P DOT	
Q DOT	
R DOT	
VXB	
YB	
YZB	
P	
Q	
R	
THETAB	
PHIB	
PSIB	

OUTPUT TRANSFER	
*PARAMETER	DESTINATION MODULE
FSCQB	FUSELAGE EM, PENNAGE
NLCQB	
DWSHMR	
CHIPMR	
AIFMR	TAIL ROTOR
OMGMR	
QHMR	ENGINE
UTOTMR	GROUND EFFECTS
DWSHMR	
LAMMR	
XMR	MOTION
YMR	
ZMR	
LMR	
MMR	
NMR	

* COMPUTER MNEMONICS

5.1.4 NOTATION FOR THE MAIN ROTOR MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
(IB)	-	INDEX	Indicating 1...NBS blades simulated
(IS)	-	INDEX	Indicating 1...NSS segments/blade simulated
(I)	-	INDEX	Indicating 1...(NBS-1)NSS blade segments
e	OFSTMR	FT	Blade hinge offset from center of rotation
e'	SPRLMR	FT	Spar length exposed
R _T	RMR	FT	Rotor radius
Ω_T	OMGTMR	RADS/SEC	Rotor nominal input rotational speed
$\frac{e}{S}$	KSGMR	ND	Normalized offset
$\frac{e'}{S}$	-	ND	Normalized spar length.
y_2 (IS)	KMRBK1	ND	Distance from hinge to segment midpoint
C_y (IS)	-	FT	Segment chord
C_T	CHDTMR	FT	Blade top chord
C_R	CHDRMR	FT	Blade root chord
S_y (IS)	-	FT ²	Blade segment area
WEIGHT	WEIGHT	LB	Total helicopter weight
b	BMR		Number of rotor blades
w_b	WTBDMR	LB	Weight of one blade
w_{bd}	WTBOD	LB	Weight of helicopter less blades
F_{SCG}	FSCG	INS	Total helicopter c.g. position
w_{LCG}	WLCG	INS	
F_{SCGB}	FSCGB	INS	C.G. position less rotor blades
w_{LCGB}	WLCGB	INS	

5.1.4 (Cont.d) NOTATION FOR THE MAIN ROTOR MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
FSMR	FSMR	INS	Fuselage station for the main rotor
WLMR	WLMR	INS	Waterline station for the main rotor
\dot{V}_{xb}	VXBDOT	FT/SEC ²	Accel. along X-axis
\dot{V}_{yb}	VYBDOT	FT/SEC ²	Accel. along Y-axis
\dot{V}_{zb}	VZBDOT	FT/SEC ²	Accel. along Z-axis
\dot{p}	PDOT	RADS/SEC ²	Angular accel about X-axis
\dot{q}	QDOT	RADS/SEC ²	Angular accel about Y-axis
\dot{r}	RDOT	RADS/SEC ²	Angular accel about Z-axis
V_{xb}	VXB	FT/SEC	Vel. along X-axis
V_{yb}	VYB	FT/SEC	Vel. along Y-axis
V_{zb}	VZB	FT/SEC	Vel. along Z-axis
p	P	RADS/SEC	Angular rate about X-axis
q	Q	RADS/SEC	Angular rate about Y-axis
r	R	RADS/SEC	Angular rate about Z-axis
X_H	-	FT	Longitudinal rotor arm
Y_H	-	FT	Lateral rotor arm
Z_H	-	FT	Vertical rotor arm
θ_b	THETAB	DEG	Pitch attitude
ϕ_b	PHIB	DEG	Roll attitude
ψ_b	PSIB	DEG	Heading
g_x	-	FT/SEC ²	Gravity vectors
g_y	-	FT/SEC ²	
g_z	-	FT/SEC ²	
V_{XG}	V_{XG}	FT/SEC	Point gust velocities. For use when gust penetration is not required.
V_{YG}	V_{YG}	FT/SEC	
V_{ZG}	V_{ZG}	FT/SEC	

5.1.4 (Cont'd) NOTATION FOR THE MAIN ROTOR MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
μ_{XH}	MUXHMR	ND	Hub velocities - normalized
μ_{YH}	MUYHMR	ND	
μ_{ZH}	MUZHMR	ND	
μ_{XS}	MUXSMR	ND	Shaft velocities - normalized
μ_{YS}	MUYSMR	ND	
μ_{ZS}	MUZSMR	ND	
P_S	PSMR	RADS/SEC	Shaft angular rates
q_S	QSMR	RADS/SEC	
r_S	RSMR	RADS/SEC	
\dot{P}_S	PS.MR	RADS/SEC ²	Shaft angular acceleration
\dot{q}_S	QS.MR	RADS/SEC ²	
\dot{r}_S	RS.MR	RADS/SEC ²	
\dot{V}_{XH}	VXH.MR	FT/SEC ²	Hub accelerations
\dot{V}_{YH}	VYH.MR	FT/SEC ²	
\dot{V}_{ZH}	VZH.MR	FT/SEC ²	
\dot{V}_{XS}	VXS.MR	FT/SEC ²	Shaft accelerations
\dot{V}_{YS}	VYS.MR	FT/SEC ²	
\dot{V}_{ZS}	VZS.MR	FT/SEC ²	
$\ddot{\Omega}$	OMG.MR	RADS/SEC ²	Rotor shaft acceleration
Ω	OMG.MR	RADS/SEC	Rotor shaft speed
Ω_T	OMGT.MR	RADS/SEC	Rotor shaft datum speed



5.1.4 (Cont'd) NOTATION FOR THE MAIN ROTOR MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
-	OMR.MR	ND	Rotor shaft acceleration ratio
-	OMRMR	ND	Rotor shaft speed ratio
ψ_R	PSIMR	DEG	Rotor azimuth position
β	BRMR	RADS	Flapping angle
$\dot{\beta}$	BR.MR	RADS/SEC	Flapping rate
$\ddot{\beta}$	BR..MR	RADS/SEC ²	Flapping acceleration
δ	LGMR	RADS	Lagging angle
$\dot{\delta}$	LG.MR	RADS/SEC	Lagging rate
$\ddot{\delta}$	LG..MR	RADS/SEC ²	Lagging acceleration
A_{AOF}	AOFMR	DEG	Steady flapping (coning)
A_{A1F}	A1FMR	DEG	Long. first harmonic flapping
B_{B1F}	B1FMR	DEG	Lateral first harmonic flapping
A_{AOL}	AOLMR	DEG	Steady lagging
A_{A1L}	A1LMR	DEG	Long. first harmonic lagging
B_{B1L}	B1LMR	DEG	Lateral first harmonic lagging
U_{TOT}	UTOTMR	ND	Total velocity component at the rotor
K_{1X}	K1XMR	ND	Longitudinal Glauert inflow factor
K_{1Y}	K1YMR	ND	Lateral Glauert inflow factor
T_{HA}	THAMR	LB	Aerodynamic component of thrust
M_{HA}	MHAMR	FT LB	Aerodynamic component of pitching moment
L_{HA}	LHAMR	FT LB	Aerodynamic component of rolling moment
ρ	RHO	SLUGS/FT ³	Air density
C_{TA}	CTHAMR		Thrust coefficient
C_{MHA}	CMHAMR		Pitching moment coefficient
C_{LHA}	CLHAMR		Rolling moment coefficient

5.1.4 (Cont'd) NOTATION FOR THE MAIN ROTOR MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
K_{CT}	KCTMR	ND	Gain factors on harmonic inflow
K_{CM}	KCMR	ND	
K_{SM}	KSLMR	ND	
T_{DWO}	TDWOMR		Time factors on harmonic inflow
T_{DWC}	TDWCMR		
T_{DWS}	TDWSMR		
D_{WO}	DWSHR	1/RADS	Uniform component of downwash at the rotor disk
D_{WC}	DWCMR	1/RADS	Cosine component of downwash
D_{WS}	DWSMR	1/RADS	Sine component of downwash
U_{PDMR_I}	UPDMR	1/RADS	Total components of downwash in blade span axes
U_{TDMR_I}	UTDMR	1/RADS	
U_{RDMR_I}	URDMR	1/RADS	
λ	LAMBMR	1/RADS	Total normal rotor inflow velocity
M_{gAVMR}	VGAVMR	1/RADS	Average gust velocity-used with gust penetration routine.
U_{PIMR_I}	UPIMR_I	1/RADS	Total segment interference velocities in blades span axes
U_{TIMR_I}	UTIMR_I	1/RADS	
U_{RIMR_I}	URIMR_I	1/RADS	
U_{PGMR_I}	UPGMR_I	1/RADS	Gust penetration components at the blade segment in blade span axes.
U_{TGMR_I}	UTGMR_I	1/RADS	
U_{RGMR_I}	URGMR_I	1/RADS	
U_{PWMR_I}	UPWMR_I		
U_{TWMR_I}	UTWMR_I	1/RADS	Airframe upwash components
U_{RNMRI}	URNMRI		

5.1.4 (Cont'd) NOTATION FOR THE MAIN ROTOR MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
U_{PAMR}	UPAMR	1/RADS	Blade segment total velocity components in blade span axes.
U_{PBMR}	UPBMR	1/RADS	
U_p	UPMR	1/RADS	
U_{TAMR}	UTAMR	1/RADS	
U_{TBMR}	UTBMR	1/RADS	
U_T	UTMR	1/RADS	
U_{RAMR}	URAMR	1/RADS	
U_{RBMR}	URBMR	1/RADS	
U_R	URMR	1/RADS	
U_Y	UYAWMR	1/RADS	
M_{ACHMR}	MACHMR		Blade segment Mach Number.
a	VSOUND	FT/SEC	Speed of sound
$\cos \gamma$	CSGMMR		Cosine of segment flow skew angle
θ_{CUFF}	THETA θ	DEG	Collective blade pitch
A_{1S}	A1S	DEG	Lateral cyclic blade pitch
B_{1S}	B1S	DEG	Longitudinal cyclic blade pitch
Δ_{SP}	DELSMR	DEG	Swashplate phase angle
δ_3	DEL3MR	DEG	Hinge angle-Pitch/Flap coupling
K_1	KAF1MR		Hinge angle coefficients - Pitch/Lag coupling
K_2	KAF2MR		
T_{HOAMR}	TH θ AMR	DEG	Geometric blade pitch angle
F_{PDYMR}	FPDYMR	LB	Resultant force at blade root
b_s	NBSMR		Number of blades simulated
-	NSSMR		Number of segments simulated
F_{po}	FPOMR	LB	Harmonic components of the blade resultant force
F_{pc}	FPCMR	LB	
F_{ps}	FPSMR	LB	

5.1.4 (Cont'd) NOTATION FOR THE MAIN ROTOR MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
MODESP	-		Equivalent blade first torsional mode
K_{FPO}	KFPOMR		Harmonic weighting coefficients for blade torsional wind-up.
K_{FPC}	KFPCMR		
K_{FPS}	KFPSMR		
T_{HDYMR}	THOYMR	DEG	Blade segment torsional deflection
θ_I		DEG	Actual blade segment geometric pitch
α_Y	AFYWMR	DEG	Blade segment angle of attack
A_{CL1MR}	ACL1MR	DEG	Angle of attack break points for lift curve.
A_{CL2MR}	ACL2MR	DEG	
A_{CL3MR}	ACL3MR	DEG	
A_{CL4MR}	ACL4MR	DEG	
α_{TRANS}	AFTFMR	DEG	Transformed angle of attack for map entry
C_{LY}	CLMR	-	Blade segment lift coefficient
C_{DY}	CDMR	-	Blade segment drag coefficient
B_{TLMR}	BTLMR	-	Blade tip lift loss factor
ΔC_{DMR}	DCDMR		Profile drag correction
F_P	FPMR	LB	Segment aero forces
F_T	FTMR	LB	
F_R	FRMR	LB	
F_{pb}	FPBMR	LB	Blade aero forces - blade span axis
F_{Tb}	FTBMR	LB	
F_{Rb}	FRBMR	LB	
F_{XA}	FXAMR	LB	Blade aero forces-shaft rotating axis
F_{YA}	FYAMR	LB	
F_{ZA}	FZAMR	LB	

5.1.4 (Cont'd) NOTATION FOR THE MAIN ROTOR MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
F_{XI}	FXIMR	LB	Blade inertial forces-shaft rotating axis
F_{YI}	FYIMR	LB	
F_{ZI}	FZIMR	LB	Blade total forces-shaft rotating axis
F_{XT}	FXTMR	LB	
F_{YT}	FYTMR	LB	
F_{ZT}	FZTMR	LB	
M_{FAB}	MFABMR	FT LB	Aero moments about hinge-blade span axis
M_{LAB}	MLABMR	FT LB	
T_{HA}	THAMR	LB	Aerodynamic component of thrust force
M_{HA}	MHAMR	FT LB	Aerodynamic component of pitching moment
L_{HA}	LHAMR	FT LB	Aerodynamic component of rolling moment
A_{LDMR}	ALDMR	INS	Input constants defining the geometry of the lag damper kinematics
B_{LDMR}	BLDMR	INS	
C_{LDMR}	CLDMR	INS	
D_{LDMR}	DLDMR	INS	
R_{LDMR}	RLDMR	INS	
θ_{GEOMR}	THLDMR	DEG	
δ_o	LAGOMR	DEG	
X_{LDMR}	XLDMR	INS	Component displacemnet of lag damper (relative pick up points)
Y_{LDMR}	YLDMR	INS	
Z_{LDMR}	ZLDMR	INS	
L_{DT}	LDMR	INS	Axial displacement of lag damper
\dot{L}_{DT}	LD.MR	INS/SEC	Axial rate of lag damper
F_{δ}	FLD.MR	LB	Axial force output from lag damper.
M_{FFD}	MFFDMR	FT LB	Flapping moment due to flap damper
M_{FLD}	MFLDMR	FT LB	Flapping moment due to lag damper
M_{LLD}	MLLDMR	FT LB	Lagging moment due to lag damper
M_{LFD}	MLFDMR	FT LB	Lagging moment due to flap damper

5.1.4 (Cont'd) NOTATION FOR THE MAIN ROTOR MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
ΔL_{HBC}	DLHBMR	FT LB	Delta moments at hub due to blade constraints.
ΔM_{HBC}	DMHBMR	FT LB	
ΔN_{HBC}	DNHBMR	FT LB	
K_{β}	KBETA	FT LB/RADS	Flapping spring stiffness
$K_{\dot{\beta}}$	KBETA.	FT LB/RAD/SEC	Flapping damper rate
M_b	MBMR	SLUGS FT	1st mass moment of blade about the hinge
I_b	IBMR	SLUGS FT ²	Inertia of blade about the hinge
W_b	WTBDMR	LB	Weight of one blade
H_H	HHMR	LB	Total force component outputs from the rotor in shaft axes at the hub
J_H	JHMR	LB	
T_H	THMR	LB	
L_H	LHMR	FT LB	Total moment component outputs from the rotor in shaft axes at the hub
M_H	MHMR	FT LB	
Q_H	QHMR	FT LB	
H_{HB}	HHBMR	LB	Filtered rotor forces and moments in shaft axes at the hub.
J_{HB}	JHBMR	LB	
T_{HB}	THBMR	LB	
L_{HB}	LHBMR	FT LB	
M_{HB}	MHBMR	FT LB	
Q_{HB}	QHBMR	FT LB	
T_{FILMR}	TFILMR	SEC	Rotor force and moment filter time constant
Q_E	QHEG	FT LB	Engine torque - supplied by engine module if selected.

5.1.4 (Cont'd) NOTATION FOR THE MAIN ROTOR MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
X _{MR}	XMR	LB	Rotor forces and moments in body axes at the fuselage c.g.
Y _{MR}	YMR	LB	
Z _{MR}	ZMR	LB	
L _{MR}	LMR	FT LB	
M _{MR}	MMR	FT LB	Rotor shaft axes force and moment coefficients (output only).
N _{MR}	NMR	FT LB	
X _{PMR}	CHIPMR	DEG	
H _{PMR}	H _{PMR}	HP	
C _{T/σ}	CTSGMR		
C _{H/σ}	CHSGMR		
C _{J/σ}	CJSGMR		
C _{M/σ}	CMSGMR		
C _{L/σ}	CLSGMR		
C _{Q/σ}	CQSGMR		

5.1.5 BLACK HAWK MAIN ROTOR INPUT DATA

CONSTANTS

$R_T = 26.83$
 $S_{LT} = 27.0$
 $b = 4$
 $i_p = 0$
 $i_b = -3$
 $\Delta SP = -9.7$
 $C_T = 1.73$
 $C_R = 1.73$
 $C_{TS} = 1.73$
 $e = 1.25$
 $e' = 3.83$
 $W_b = 256.9$
 $M_b = 86.7$
 $I_b = 1512.6$
 $B_{LTMR} = .97$
 $\Delta C_{DMR} = .002$
 $K_\beta = 0$
 $K_{\dot{\beta}} = 0$
 $K_{\dot{\alpha}_1} = 0$
 $K_{\dot{\alpha}_2} = 0$
 $K_{CT} = 1.0$
 $K_{CM} = 0$
 $K_{CL} = 0$
 $T_{DWO} = .01038$
 $T_{DWC} = 0$
 $T_{DWS} = 0$
 $f = .002378$
 $a = 1117.0$

$F_{SMR} = 341.2$
 $N_{LMR} = 315.0$
 $B_{LMR} = 0$
 $A_{LDMR} = .227$
 $B_{LDMR} = 3.242$
 $C_{LDMR} = 12.040$
 $D_{LDMR} = 10.0102$
 $R_{LDMR} = 6.898$
 $\theta_{GEOMR} = +17.481$
 $\delta_0 = 7$
 $T_{FILMR} = .3 \text{ SECS}$
 $K_{FAC} = 0$
 $K_{FAS} = 0$
 $NBS = 4$
 $NSS = 5$
 $\delta_3 = 0$

$$K_{FA0} = -.0003 - (.0000044)(V_{KT} - 100)$$
$$-.00052 \leq K_{FA0} \leq -.0003$$

$WEIGHT = 16638.0$
 $FSCG = 355.9$
 $WLCCG = 248.2$
 $BLCCG = 0$

BLACK HAWK BLADE TWIST INPUT

MAP NAME: TWMAMP
 MAP TYPE: UVA
 INPUT VARIABLE(S): XSEGMR
 OUTPUT VARIABLE: TWSTMR
 PRIMARY MAP:
 0.00 LOWER LIMIT
 1.00 UPPER LIMIT
 0.05 DELTA

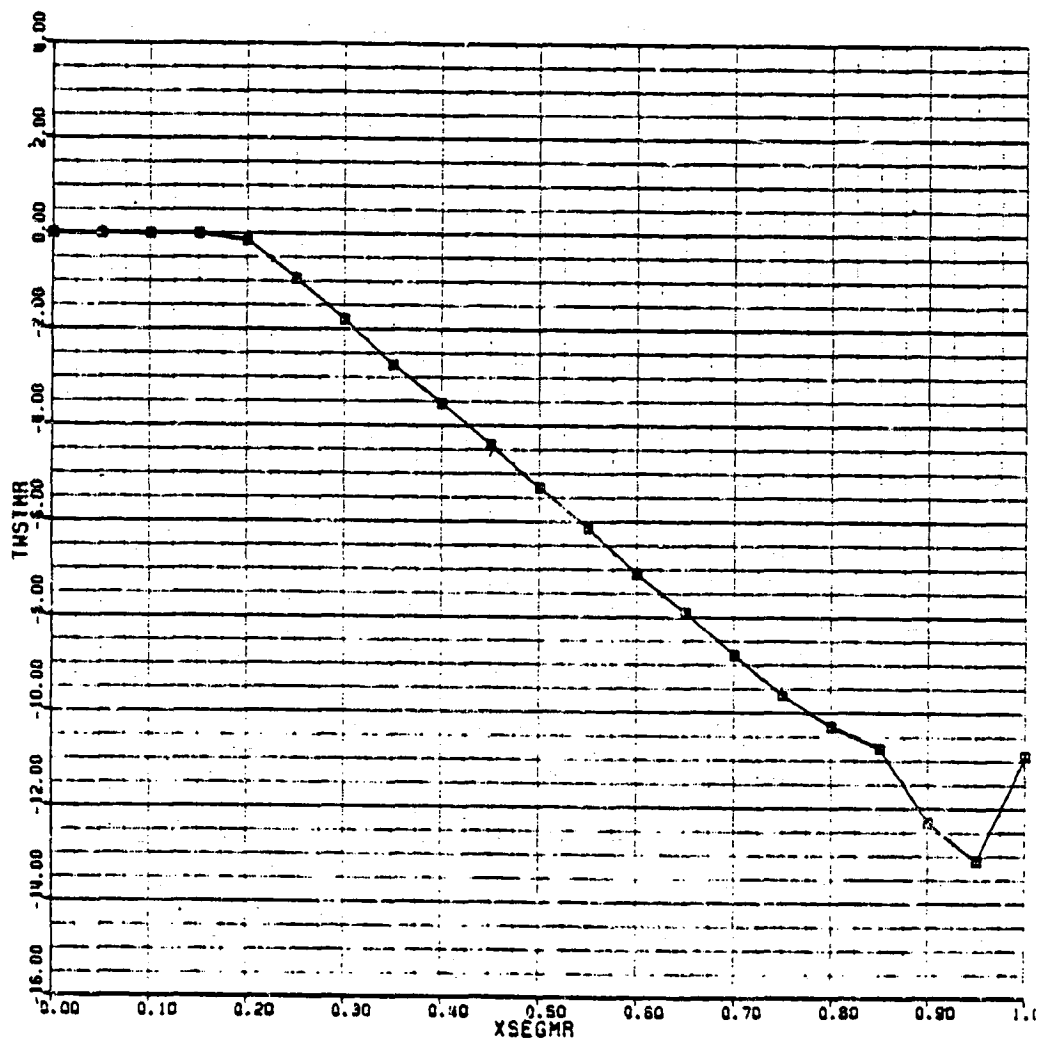


FIGURE 1.5.1



BLACK HAWK MAIN ROTOR BLADE SECTION LIFT COEFFICIENT - AIRFOIL SC 1095

BLACKHAWK - NASA STUDY 26-SEP-80

CLMAMP (1/2)

MAP NAME:		CLMAMP	
MAP TYPE:		BIVUVR	
INPUT VARIABLE(S):	AFTFMR	CLMA	MACHMA
OUTPUT VARIABLE:			
PRIMARY MAP:			
	-32.00	LOWER LIMIT	0.00
	32.00	UPPER LIMIT	1.00
	2.00	DELTA	0.10
SECONDARY MAP:			
	-180.00	LOWER LIMIT	
	180.00	UPPER LIMIT	
	2.00	DELTA	

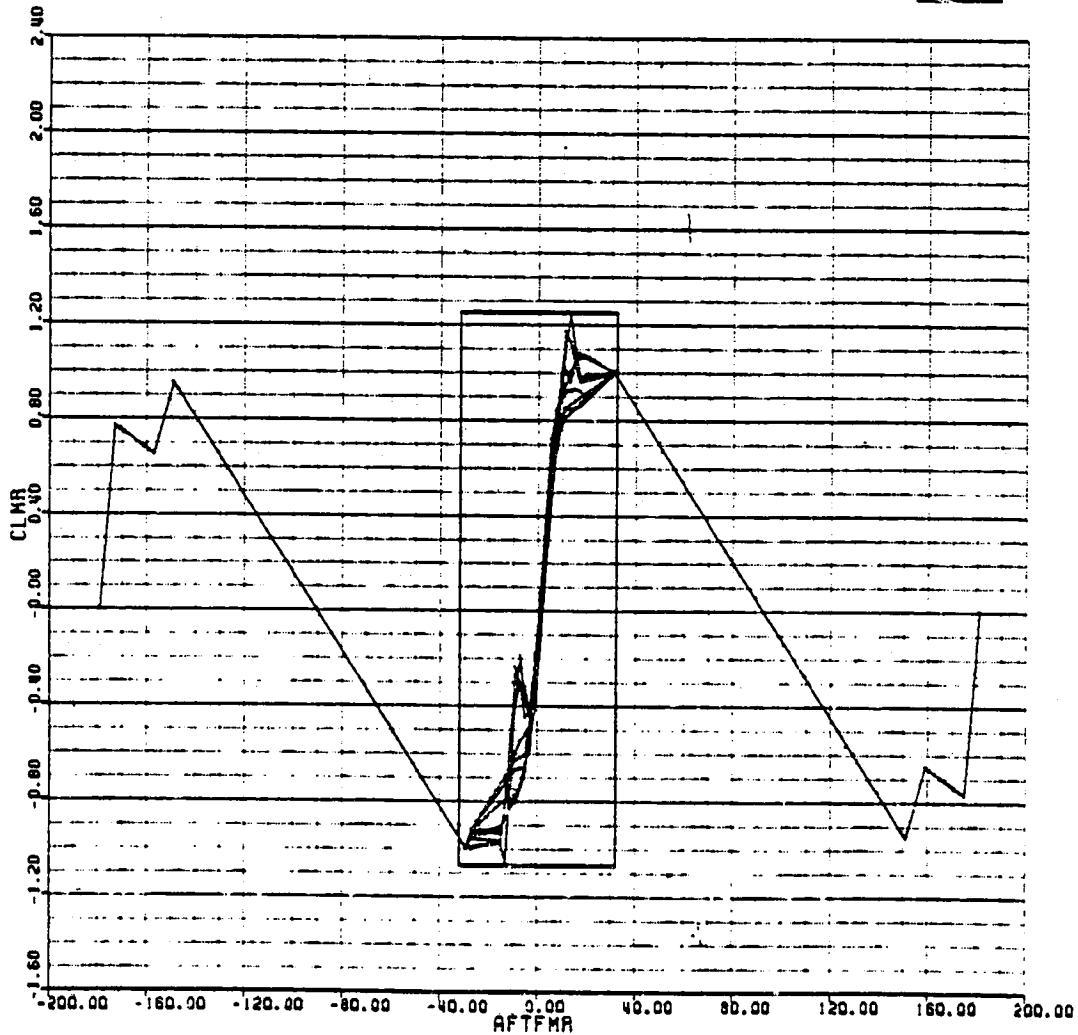
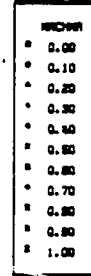


FIGURE 1.5.2(a)

BLACK HAWK MAIN ROTOR BLADE SECTION LIFT COEFFICIENT - AIRFOIL SC 1095 (Cont'd)

BLACKHAWK - NASA STUDY 26-SEP-80

CLMAMP (2/2)

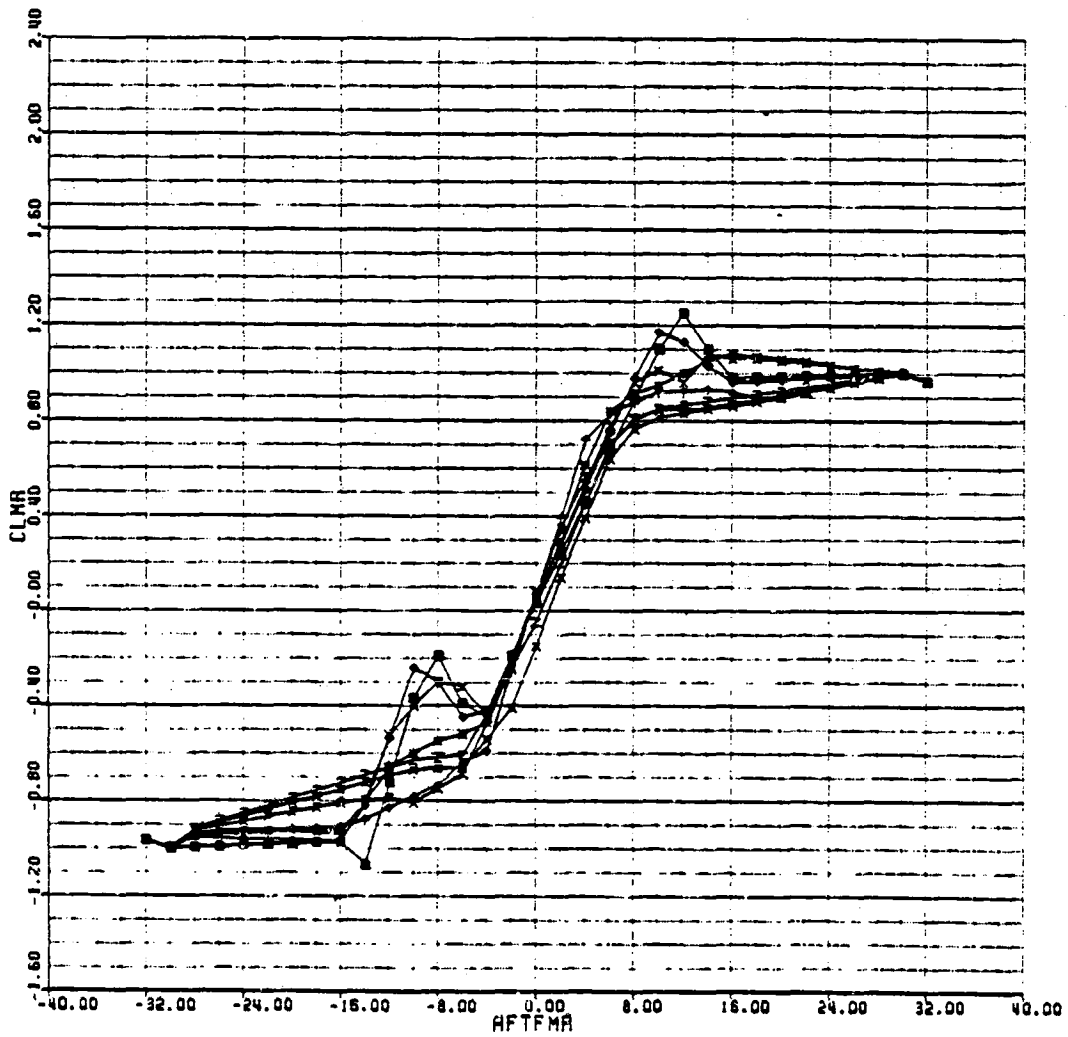


FIGURE 1.5.2(b)

BLACK HAWK MAIN ROTOR BLADE SECTION DRAG COEFFICIENT - AIRFOIL SC 1095

BLACKHAWK - NASA STUDY 26-SEP-80

CDMAMP (1/2)

MAP NAME: CDMAMP
 MAP TYPE: BIVUVR
 INPUT VARIABLE(S): AFWMA MACHMA
 OUTPUT VARIABLE: COMA

PRIMARY MAP:
 -32.00 LOWER LIMIT 0.00
 32.00 UPPER LIMIT 1.00
 2.00 DELTA 0.10

SECONDARY MAP:
 -180.00 LOWER LIMIT
 180.00 UPPER LIMIT
 2.00 DELTA

- 0.05
- 0.10
- 0.20
- 0.30
- 0.40
- 0.50
- 0.60
- 0.70
- 0.80
- 0.90
- 1.00

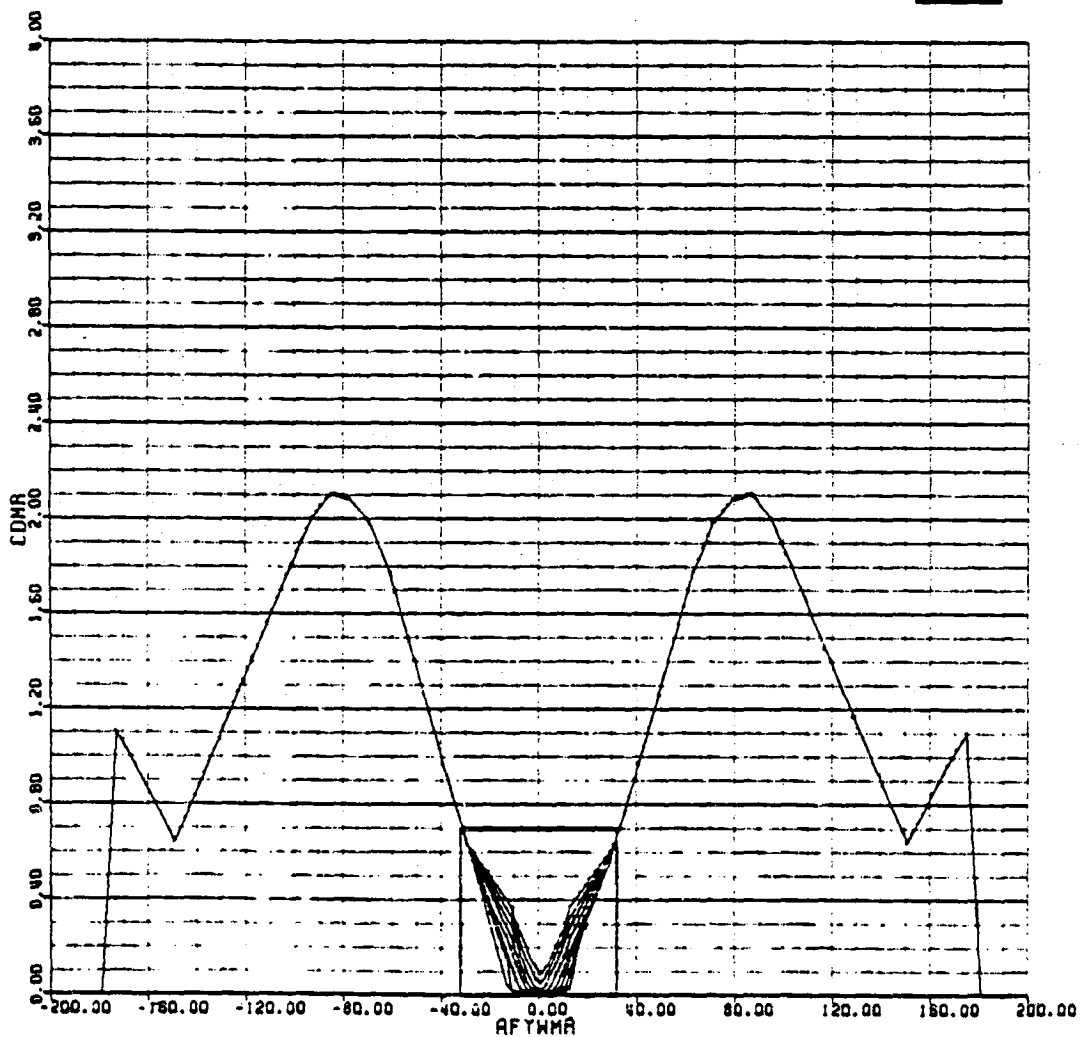


FIGURE 1.5.3(a)

BLACK HAWK MAIN ROTOR BLADE SECTION DRAG COEFFICIENT - AIRFOIL SC 1095 (Cont'd)

BLACKHAWK - NASA STUDY 26-SEP-83

CDMAMP (2/2)

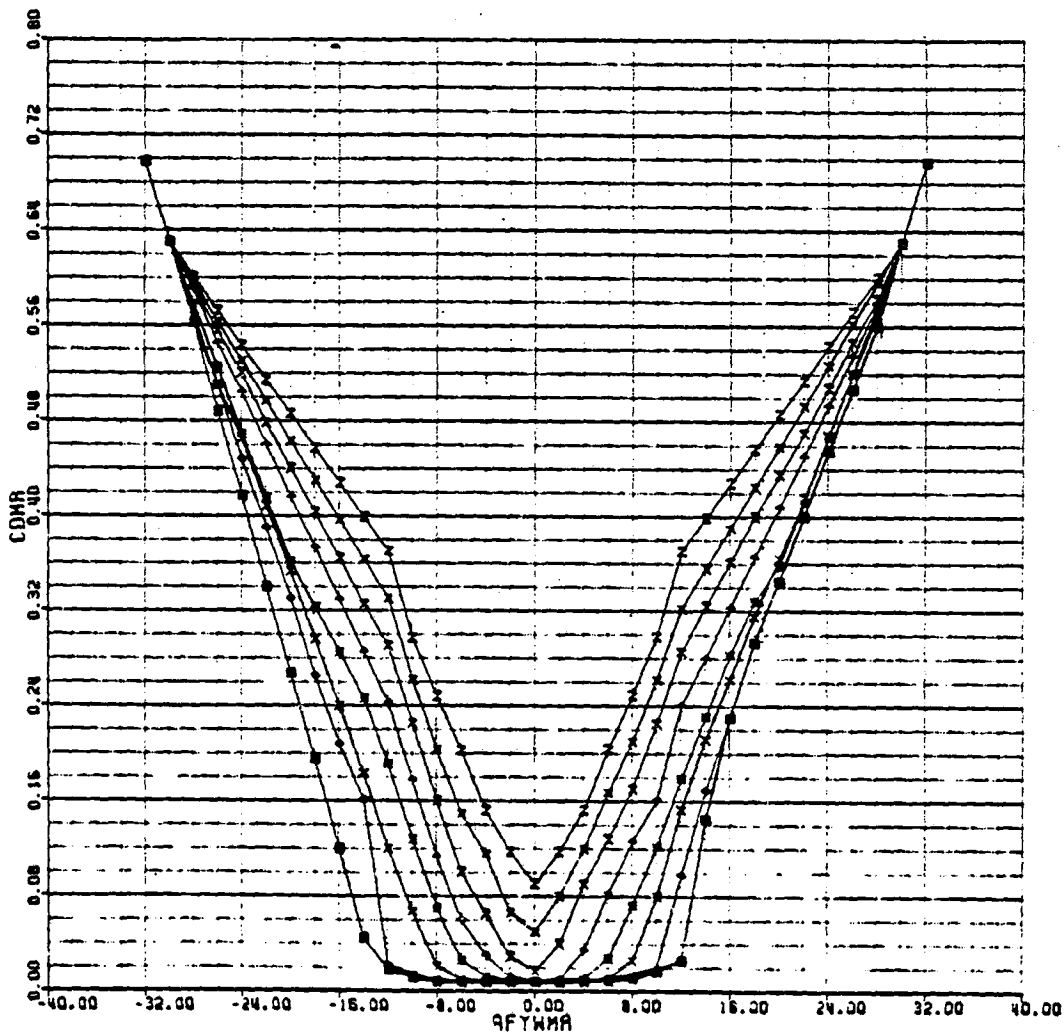


FIGURE 1.5.3(b)

BLACK HAWK MAIN ROTOR BLADE LAG DAMPER FORCE CHARACTERISTICS

BLACKHAWK - NASA STUDY 26-SEP-80

LDMAMP (1/2)

MAP NAME: LDMAMP
 MAP TYPE: UVSUYS
 INPUT VARIABLE(S): LD.MR
 OUTPUT VARIABLE: FLD.MR

PRIMARY MAP: -2.00 LOWER LIMIT
 2.00 UPPER LIMIT
 0.10 DELTA

SECONDARY MAP: -7.00 LOWER LIMIT
 7.00 UPPER LIMIT
 1.00 DELTA

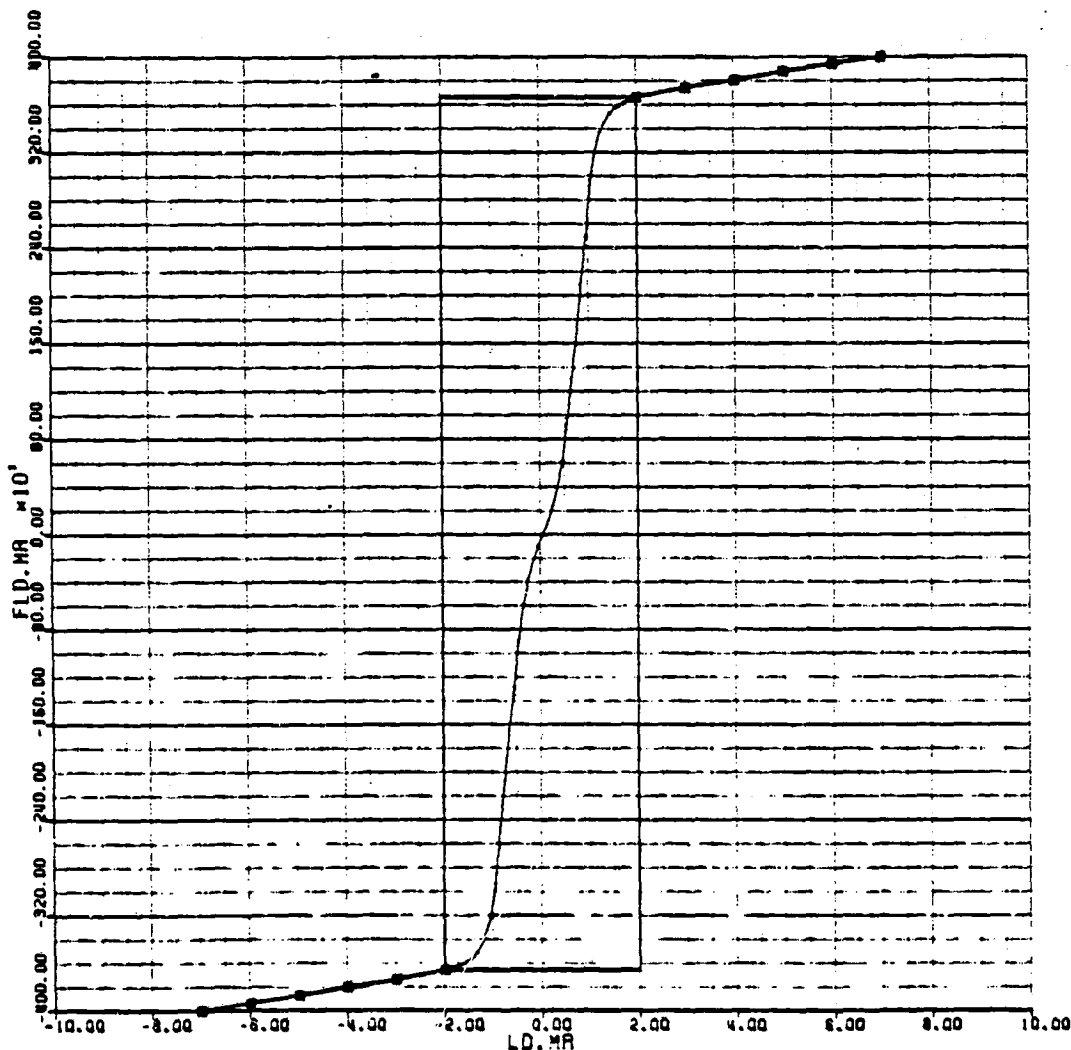


FIGURE 1.5.4(a)

BLACK HAWK MAIN ROTOR BLADE LAG DAMPER FORCE CHARACTERISTICS (Cont'd)

BLACKHAWK - NASA STUDY 26-SEP-80

LODAMP (2/2)

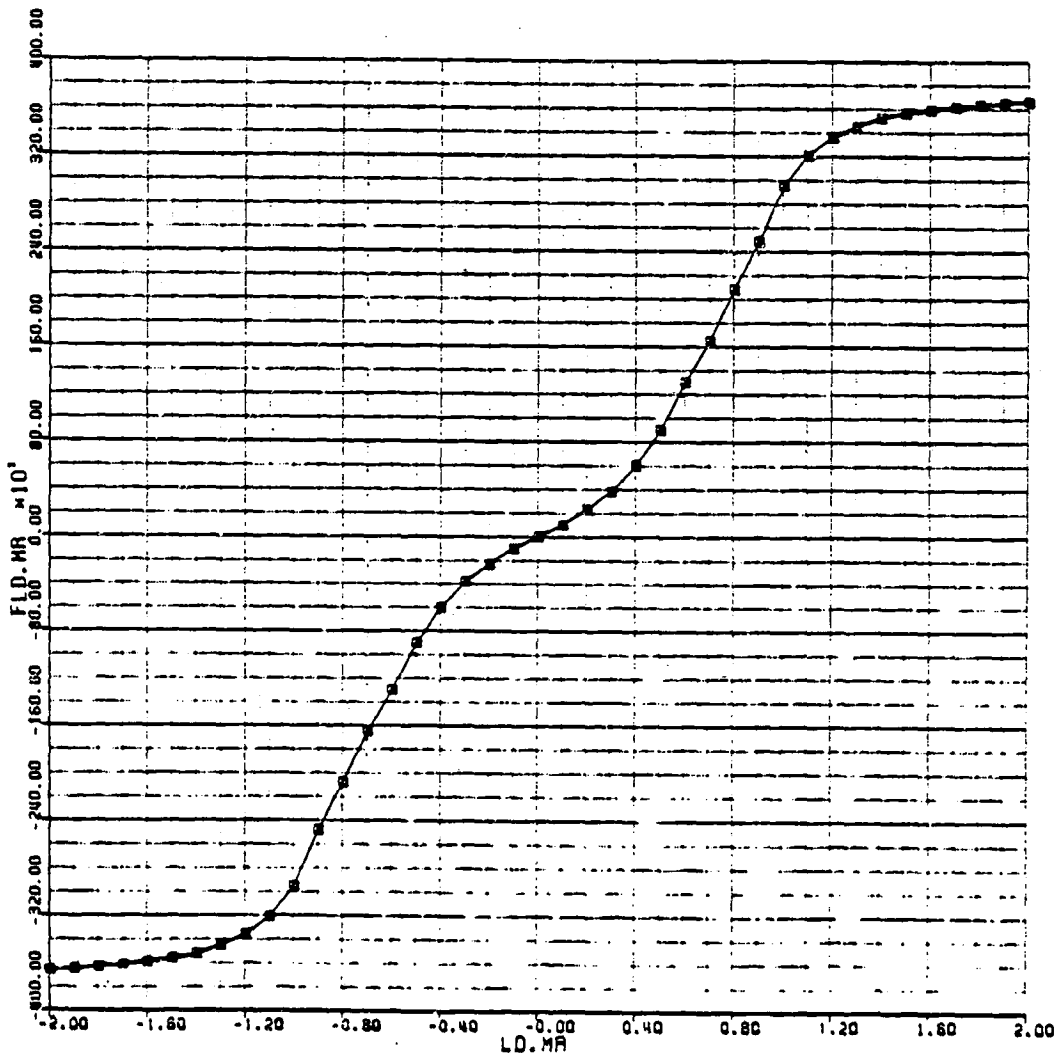


FIGURE 1.5.4(b)

TABLE 1.5.1

BLACK HAWK MAIN ROTOR PRESET BLADE TWIST

TWMRMP: :UVR##						
XSEGMR##						
TWSTMR##						
TWMRLO						
EXP 0.0, 1.0, 0.05						
TWMRLO: EXP 0.0,	0.0,	0.0,	0.0,	0.0,		-0.15
EXP -0.95,	-1.8,	-2.75,	-3.55,			-4.4
EXP -5.3,	-6.15,	-7.1,	-7.9,			-8.8
EXP -9.65,	-10.3,	-10.75,	-12.3,			-13.1
EXP -10.9						

TABLE 1.5.2

BLACK HAWK MAIN ROTOR BLADE SECTION LIFT COEFFICIENT - AIRFOIL SC1095

ACL1MR: 11.8
 ACL2MR: 172.8
 ACL3MR: -5.8
 ACL4MR: -172.8

```

; ** ROTOR COEF. OF LIFT MAP **
CLMRMP: BIVUVR## ;MAP ARGUMENT:LOOK UP ROUTINE
          AFTFMR##(17) ;INPUT VARIABLE #1,
          MACHMR##(A17) ;INPUT VARIABLE #2
          CLMR##(A17) ;OUTPUT VARIABLE
          CLMRLO ;PRIMARY (BASIC) MAP NAME
EXP -32.8,32.8,2.8, 33;LOW. LIM., UP. LIM., DELTA, # ITEMS
EXP 0.8,1.8,.1 ;LOW. LIM., UP. LIM., DELTA
          CLMRHI ;SECONDARY (HIGH ANGLE) MAP NAME
EXP -188.8,188.8,2.8 ;LOW. LIM., UP. LIM., DELTA
    
```

CLMRLO: EXP ; PRIMARY MAP: AFTFMR FROM -32 TO 32,DELTA=2,MACHMR FROM 0 TO 1.
 ; MACH NO. = .0

EXP	-0.9675,	-1.8,	-0.996,	-0.992,	-0.988
EXP	-0.984,	-0.988,	-0.976,	-0.972,	-1.87
EXP	-0.724,	-0.37,	-0.19,	-0.39,	-0.45
EXP	-0.19,	0.83,	0.243,	0.46,	0.67
EXP	0.89,	1.18,	1.25,	1.18,	0.98
EXP	0.9828,	0.9856,	0.9884,	0.9912,	0.9948
EXP	.997,	1.8,	0.9675		

EXP ; MACH NO. = .1

EXP	-0.9675,	-1.8,	-0.996,	-0.992,	-0.988
EXP	-0.984,	-0.988,	-0.976,	-0.972,	-1.87
EXP	-0.724,	-0.37,	-0.19,	-0.39,	-0.45
EXP	-0.19,	0.83,	0.243,	0.46,	0.67
EXP	0.89,	1.18,	1.25,	1.18,	0.98
EXP	0.9828,	0.9856,	0.9884,	0.9912,	0.9948
EXP	.997,	1.8,	0.9675		

EXP ; MACH NO. = .2

EXP	-0.9675,	-1.8,	-0.996,	-0.992,	-0.988
EXP	-0.984,	-0.988,	-0.976,	-0.972,	-1.87
EXP	-0.724,	-0.37,	-0.19,	-0.39,	-0.45
EXP	-0.19,	0.83,	0.243,	0.46,	0.67
EXP	0.89,	1.18,	1.25,	1.18,	0.98
EXP	0.9828,	0.9856,	0.9884,	0.9912,	0.9948
EXP	0.997,	1.8,	0.9675		

EXP ; MACH NO. = .3

EXP	-0.9675,	-1.8,	-0.996,	-0.992,	-0.988
EXP	-0.984,	-0.988,	-0.976,	-0.972,	-1.87
EXP	-0.724,	-0.37,	-0.19,	-0.39,	-0.45
EXP	-0.19,	0.83,	0.243,	0.46,	0.67
EXP	0.89,	1.18,	1.25,	1.18,	0.98
EXP	0.9828,	0.9856,	0.9884,	0.9912,	0.994
EXP	0.997,	1.8,	0.9675		

TABLE 1.5.2 (Cont'd)

	; MACH NO. = .4				
EXP	-0.9675,	-1.0,	-0.955,	-0.96,	-0.962
EXP	-0.964,	-0.966,	-0.968,	-0.970,	-0.82
EXP	-0.535,	-0.24,	-0.30,	-0.45,	-0.42
EXP	-0.185,	0.05,	0.280,	0.510,	0.75
EXP	0.98,	1.17,	1.13,	1.03,	0.96
EXP	0.9657,	0.9714,	0.9771,	0.9828,	0.9885
EXP	0.9942,	1.0,	0.9675		
	; MACH NO. = .5				
EXP	-0.9675,	-1.0,	-0.94,	-0.93,	-0.92
EXP	-0.925,	-0.93,	-0.935,	-0.94,	-0.80
EXP	-0.525,	-0.4,	-0.3,	-0.32,	-0.44
EXP	-0.195,	0.05,	0.295,	0.53,	0.78
EXP	0.96,	1.01,	0.96,	1.08,	1.06
EXP	1.07,	1.06,	1.05,	1.035,	1.02
EXP	1.01,	1.0,	0.9675		
	; MACH NO. = .6				
EXP	-0.9675,	-1.0,	-0.946,	-0.942,	-0.938
EXP	-0.934,	-0.93,	-0.926,	-0.922,	-0.805
EXP	-0.66,	-0.6,	-0.55,	-0.52,	-0.47
EXP	-0.195,	0.075,	0.34,	0.613,	0.84
EXP	0.916,	0.947,	1.0,	1.054,	1.08
EXP	1.063,	1.053,	1.042,	1.031,	1.02
EXP	1.01,	1.0,	0.9675		
	; MACH NO. = .7				
EXP	-0.9675,	-1.0,	-0.944,	-0.938,	-0.932
EXP	-0.926,	-0.920,	-0.914,	-0.908,	-0.88
EXP	-0.83,	-0.78,	-0.735,	-0.64,	-0.59
EXP	-0.255,	0.07,	0.395,	0.72,	0.83
EXP	0.877,	0.92,	0.923,	0.93,	0.92
EXP	0.895,	0.9,	0.92,	0.94,	0.97
EXP	0.985,	1.0,	0.9675		
	; MACH NO. = .8				
EXP	-0.9675,	-1.0,	-0.93,	-0.91,	-0.89
EXP	-0.87,	-0.85,	-0.83,	-0.81,	-0.80
EXP	-0.79,	-0.81,	-0.75,	-0.69,	-0.47
EXP	-0.25,	0.08,	0.35,	0.56,	0.705
EXP	0.810,	0.845,	0.845,	0.85,	0.86
EXP	0.88,	0.90,	0.92,	0.94,	0.96
EXP	0.98,	1.0,	0.9675		
	; MACH NO. = .9				
EXP	-0.9675,	-1.0,	-0.922,	-0.894,	-0.866
EXP	-0.838,	-0.81,	-0.782,	-0.754,	-0.726
EXP	-0.698,	-0.67,	-0.665,	-0.66,	-0.54
EXP	-0.41,	-0.15,	0.14,	0.39,	0.64
EXP	0.765,	0.81,	0.83,	0.85,	0.87
EXP	0.885,	0.905,	0.925,	0.94,	0.96
EXP	0.98,	1.0,	0.9675		
	; MACH NO. = 1.0				
EXP	-0.9675,	-1.0,	-0.918,	-0.886,	-0.854
EXP	-0.822,	-0.79,	-0.758,	-0.726,	-0.694
EXP	-0.662,	-0.63,	-0.62,	-0.61,	-0.425
EXP	-0.24,	-0.05,	0.2,	0.45,	0.7
EXP	0.8,	0.85,	0.865,	0.88,	0.895
EXP	0.91,	0.925,	0.94,	0.955,	0.97
EXP	0.985,	1.0,	0.9675		

TABLE 1.5.2 (Cont'd)

		HIGH ANGLE MAP: AFTFMR FROM -180 TO 180, DELTA=2				
CLMRHI: EXP	0.0,	.25667,	.5133,	.77,	.755	
EXP	.74,	.725,	.710,	.695,	.68	
EXP	.665,	.65,	.725,	.8,	.875	
EXP	.95,	.9175,	.885,	.8525,	.82	
EXP	.7875,	.755,	.7225,	.69,	.6575	
EXP	.625,	.6925,	.56,	.5275,	.495	
EXP	.4625,	.43,	.3975,	.365,	.3325	
EXP	.3,	.2675,	.235,	.2025,	.17	
EXP	.1375,	.105,	.0725,	.04,	.0075	
EXP	-.025,	-.0575,	-.09,	-.1225,	-.155	
EXP	-.1875,	-.22,	-.2525,	-.285,	-.3175	
EXP	-.35,	-.3825,	-.415,	-.4475,	-.48	
EXP	-.5125,	-.545,	-.5775,	-.61,	-.6425	
EXP	-.675,	-.7075,	-.74,	-.7725,	-.805	
EXP	-.8375,	-.87,	-.9025,	-.935,	-.9675	
EXP	-1.0,	-.996,	-.992,	-.998,	-.984	
EXP	-.98,	-.976,	-.972,	-1.07,	-.724	
EXP	-.37,	-.19,	-.39,	-.45,	-.19	
EXP	.03,	.243,	.46,	.67,	.89	
EXP	1.1,	1.25,	1.1,	.98,	.9828	
EXP	.9856,	.9884,	.9912,	.994,	.997	
EXP	1.0,	.9675,	.935,	.9025,	.87	
EXP	.8375,	.805,	.7725,	.74,	.7075	
EXP	.675,	.6425,	.61,	.5775,	.545	
EXP	.5125,	.48,	.4475,	.415,	.3825	
EXP	.35,	.3175,	.285,	.2525,	.22	
EXP	.1875,	.155,	.1225,	.09,	.0575	
EXP	.025,	-.0075,	-.04,	-.0725,	-.105	
EXP	-.1375,	-.17,	-.2025,	-.235,	-.2675	
EXP	-.3,	-.3325,	-.365,	-.3975,	-.43	
EXP	-.4625,	-.495,	-.5275,	-.56,	-.5925	
EXP	-.625,	-.6575,	-.69,	-.7225,	-.755	
EXP	-.7875,	-.82,	-.8525,	-.885,	-.9175	
EXP	-.95,	-.875,	-.8,	-.725,	-.65	
EXP	-.665,	-.68,	-.695,	-.71,	-.725	
EXP	-.74,	-.755,	-.77,	-.5133,	-.25667	
EXP	0.0					

TABLE 1.5.3

BLACK HAWK MAIN ROTOR BLADE SECTION DRAG COEFFICIENT - AIRFOIL SC1095

```

; ** ROTOR COEF. OF DRAG MAP **
CDMRP: BIVUVR## ; MAP ARGUMENT:LOOK UP ROUTINE
AFYWMR##(17) ; INPUT VARIABLE #1
MACHMR##(A17) ; INPUT VARIABLE #2
CDMR##(A17) ; OUTPUT VARIABLE
CDMRLO ; PRIMARY (BASIC) MAP NAME
EXP -32.0,32.0,2.0,^D33; LOW. LIM., UP. LIM., DELTA, # ITEMS(OCTAL)
EXP 0.0,1.0,.1 ; LOW. LIM., UP. LIM., DELTA
CDMRHI ; SECONDARY (HIGH ANGLE) MAP NAME
EXP -180.0,180.0,2.0 ; LOW. LIM., UP. LIM., DELTA

; PRIMARY MAP: AFYWMR FROM -32 TO 32,DELTA=2,MACHMR FROM 0 TO 1.
; MACH NO.=.0
CDMRLO: EXP 0.6975, 0.63, 0.562, 0.488, 0.417
EXP 0.34, 0.267, 0.195, 0.12, 0.045
EXP 0.018, 0.012, 0.008, 0.00775, 0.0075
EXP 0.0075, 0.0075, 0.008, 0.0085, 0.009
EXP 0.011, 0.017, 0.026, 0.145, 0.23
EXP 0.293, 0.345, 0.4, 0.455, 0.507
EXP 0.56, 0.63, 0.6975

; MACH NO.=.1
EXP 0.6975, 0.63, 0.562, 0.488, 0.417
EXP 0.34, 0.267, 0.195, 0.12, 0.045
EXP 0.018, 0.012, 0.008, 0.00775, 0.0075
EXP 0.0075, 0.0075, 0.008, 0.0085, 0.009
EXP 0.011, 0.017, 0.026, 0.145, 0.23
EXP 0.293, 0.345, 0.4, 0.455, 0.507
EXP 0.56, 0.63, 0.6975

; MACH NO.=.2
EXP 0.6975, 0.63, 0.562, 0.488, 0.417
EXP 0.34, 0.267, 0.195, 0.12, 0.045
EXP 0.018, 0.012, 0.008, 0.00775, 0.0075
EXP 0.0075, 0.0075, 0.008, 0.0085, 0.009
EXP 0.011, 0.017, 0.026, 0.145, 0.23
EXP 0.293, 0.345, 0.4, 0.455, 0.507
EXP 0.56, 0.63, 0.6975

; MACH NO.=.3
EXP 0.6975, 0.63, 0.562, 0.488, 0.417
EXP 0.34, 0.267, 0.195, 0.12, 0.045
EXP 0.018, 0.012, 0.0082, 0.0079, 0.0075
EXP 0.0075, 0.0075, 0.008, 0.0085, 0.009
EXP 0.011, 0.017, 0.026, 0.145, 0.23
EXP 0.293, 0.345, 0.4, 0.455, 0.507
EXP 0.56, 0.63, 0.6975

; MACH NO.=.4
EXP 0.6975, 0.63, 0.57, 0.51, 0.448
EXP 0.39, 0.33, 0.265, 0.208, 0.161
EXP 0.022, 0.013, 0.009, 0.0085, 0.008
EXP 0.008, 0.008, 0.0082, 0.0085, 0.011
EXP 0.014, 0.02, 0.098, 0.169, 0.23
EXP 0.293, 0.345, 0.4, 0.455, 0.507
EXP 0.56, 0.63, 0.6975

```

TABLE 1.5.3 (Cont'd)

: MACH NO. = .5					
EXP	Ø.6975,	Ø.63,	Ø.564,	Ø.51,	Ø.465
EXP	Ø.4Ø8,	Ø.353,	Ø.296,	Ø.24,	Ø.183
EXP	Ø.12,	Ø.Ø67,	Ø.Ø21,	Ø.Ø1,	Ø.ØØ8
EXP	Ø.ØØ75,	Ø.ØØ75,	Ø.ØØ75,	Ø.ØØ8,	Ø.Ø11
EXP	Ø.Ø26,	Ø.Ø8,	Ø.153,	Ø.212,	Ø.262
EXP	Ø.315,	Ø.365,	Ø.416,	Ø.469,	Ø.52
EXP	Ø.569,	Ø.63,	Ø.6975		
: MACH NO. = .6					
EXP	Ø.6975,	Ø.63,	Ø.578,	Ø.525,	Ø.469
EXP	Ø.415,	Ø.361,	Ø.323,	Ø.285,	Ø.246
EXP	Ø.191,	Ø.128,	Ø.Ø7,	Ø.Ø26,	Ø.Ø125
EXP	Ø.ØØ85,	Ø.ØØ8,	Ø.ØØ85,	Ø.Ø11,	Ø.Ø28
EXP	Ø.Ø73,	Ø.122,	Ø.179,	Ø.231,	Ø.283
EXP	Ø.328,	Ø.358,	Ø.412,	Ø.467,	Ø.521
EXP	Ø.576,	Ø.63,	Ø.6975		
: MACH NO. = .7					
EXP	Ø.6975,	Ø.63,	Ø.59,	Ø.545,	Ø.5Ø4
EXP	Ø.46,	Ø.416,	Ø.373,	Ø.329,	Ø.285
EXP	Ø.242,	Ø.177,	Ø.113,	Ø.Ø6,	Ø.Ø3
EXP	Ø.Ø12,	Ø.ØØ8,	Ø.Ø1,	Ø.Ø35,	Ø.Ø82
EXP	Ø.126,	Ø.161,	Ø.24,	Ø.28,	Ø.323
EXP	Ø.365,	Ø.4Ø8,	Ø.451,	Ø.493,	Ø.535
EXP	Ø.578,	Ø.63,	Ø.6975		
: MACH NO. = .8					
EXP	Ø.6975,	Ø.63,	Ø.593,	Ø.555,	Ø.52
EXP	Ø.478,	Ø.44,	Ø.4Ø3,	Ø.364,	Ø.325
EXP	Ø.29,	Ø.225,	Ø.16,	Ø.1,	Ø.Ø65
EXP	Ø.Ø29,	Ø.Ø17,	Ø.Ø4,	Ø.Ø9,	Ø.128
EXP	Ø.17,	Ø.225,	Ø.285,	Ø.324,	Ø.361
EXP	Ø.4,	Ø.435,	Ø.47,	Ø.5Ø8,	Ø.546
EXP	Ø.583,	Ø.63,	Ø.6975		
: MACH NO. = .9					
EXP	Ø.6975,	Ø.63,	Ø.597,	Ø.563,	Ø.53
EXP	Ø.497,	Ø.463,	Ø.43,	Ø.397,	Ø.363
EXP	Ø.33,	Ø.262,	Ø.2Ø3,	Ø.149,	Ø.115
EXP	Ø.Ø56,	Ø.Ø5,	Ø.Ø8,	Ø.12,	Ø.167
EXP	Ø.21,	Ø.262,	Ø.322,	Ø.356,	Ø.39Ø
EXP	Ø.425,	Ø.459,	Ø.493,	Ø.527,	Ø.562
EXP	Ø.596,	Ø.63,	Ø.6975		
: MACH NO. = 1.Ø					
EXP	Ø.6975,	Ø.63,	Ø.6Ø1,	Ø.572,	Ø.543
EXP	Ø.514,	Ø.486,	Ø.457,	Ø.428,	Ø.399
EXP	Ø.37,	Ø.297,	Ø.248,	Ø.2Ø2,	Ø.152
EXP	Ø.117,	Ø.Ø9,	Ø.1175,	Ø.1525,	Ø.2Ø3
EXP	Ø.249,	Ø.298,	Ø.37,	Ø.399,	Ø.428
EXP	Ø.457,	Ø.486,	Ø.514,	Ø.543,	Ø.572
EXP	Ø.6Ø1,	Ø.63,	Ø.6975		

TABLE 1.5.3 (Cont'd)

CDMRHI: EXP ; HIGH ANGLE MAP: AFYWMR FROM -180 TO 180, DELTA=2

EXP	.0,	.367,	.733,	1.1,	1.065
EXP	1.03,	.995,	.96,	.92,	.88
EXP	.84,	.8,	.76,	.72,	.68
EXP	.64,	.6875,	.735,	.7825,	.83
EXP	.87875,	.9275,	.97625,	1.025,	1.07375
EXP	1.1225,	1.17125,	1.22,	1.26875,	1.3175
EXP	1.36625,	1.415,	1.46375,	1.5125,	1.56125
EXP	1.61,	1.66,	1.71,	1.76,	1.81
EXP	1.8575,	1.905,	1.9525,	2.0,	2.02625
EXP	2.0525,	2.07875,	2.105,	2.09875,	2.0925
EXP	2.08625,	2.08,	2.055,	2.03,	2.005
EXP	1.98,	1.92875,	1.8775,	1.82625,	1.775
EXP	1.70125,	1.6275,	1.55375,	1.48,	1.4075
EXP	1.335,	1.2625,	1.19,	1.1175,	1.045
EXP	.9725,	.9,	.8325,	.765,	.6975
EXP	.63,	.562,	.488,	.417,	.34
EXP	.267,	.195,	.12,	.045,	.018
EXP	.012,	.008,	.00775,	.0075,	.0075
EXP	.0075,	.008,	.0085,	.009,	.011
EXP	.017,	.026,	.145,	.23,	.293
EXP	.345,	.4,	.455,	.507,	.56
EXP	.63,	.6975,	.765,	.8325,	.9
EXP	.9725,	1.045,	1.1175,	1.19,	1.2625
EXP	1.335,	1.4075,	1.48,	1.55375,	1.6275
EXP	1.70125,	1.775,	1.82625,	1.8775,	1.92875
EXP	1.98,	2.005,	2.03,	2.055,	2.08
EXP	2.08625,	2.0925,	2.09875,	2.105,	2.07875
EXP	2.0525,	2.02625,	2.0,	1.9525,	1.905
EXP	1.8575,	1.81,	1.76,	1.71,	1.66
EXP	1.61,	1.56125,	1.5125,	1.46375,	1.415
EXP	1.36625,	1.3175,	1.26875,	1.22,	1.17125
EXP	1.1225,	1.07375,	1.025,	.97625,	.9275
EXP	.87875,	.83,	.7825,	.735,	.6875
EXP	.64,	.68,	.72,	.76,	.8
EXP	.84,	.88,	.92,	.96,	.995
EXP	1.03,	1.065,	1.1,	.733,	.367
EXP	.0,				

TABLE 1.5.4

BLACK HAWK MAIN ROTOR BLADE LAG DAMPER FORCE CHARACTERISTICS

```

LDMRMP: : UVSUVS##           ; MAP ARGUMENT: LOOK UP ROUTINE
          LD.MR##(A16)       ; INPUT VARIABLE
          FLD.MR##(A16)     ; OUTPUT VARIABLE
          LDMRLO            ; LOW RANGE MAP NAME
EXP      0.0, 2.0, 0.1     ; LOWER LIMIT, UPPER LIMIT, DELTA
          LDMRHI            ; HIGH RANGE MAP NAME
EXP      2.0, 7.0, 1.0     ; LOWER LIMIT, UPPER LIMIT, DELTA

          ; LOW ANGLE MAP: LD, MR 0 TO 2.0 , DELTA = .1
LDMRLO: EXP      0.0,      100.0,      230.0,      380.0,      600.0
          EXP      900.0,     1300.0,     1650.0,     2000.0,     2400.0
          EXP      2950.0,    3210.0,     3360.0,     3450.0,     3525.0
          EXP      3565.0,    3590.0,     3615.0,     3630.0,     3650.0
          EXP      3660.0

          ; HIGH ANGLE MAP: LD, MR 2.0 TO 7.0 , DELTA=1.0
LDMRHI: EXP      3660.0,    3742.0,     3805.0,     3875.0,     3940.0
          EXP      4000.0
    
```


5.1.6 References

1. Articulated Rotor Blade Flapping Motion at Low Advance Ratios, F. D. Harris, Journal AHS, January, 1972.
2. Generalized Rotor Performance, Tanner, Watson SER-50309, 1964
3. High Speed Aerodynamics and Jet Propulsion, Volume VIII., Aerodynamic Components of Aircraft at High Speed. Princeton University Press.
4. Yawed Blade Element Rotor Performance Method, Paglino, V.M., SER-50620, September, 1969.

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5.2 Fuselage Module

5.2.1 Module Description

The effects of rotor wash on the airframe have been treated in gross terms. No attempt has been made to determine the local flow under the rotor disc and apply it to an element analysis of the fuselage. The implication of the method used is that any variations in local velocity effects have been ignored. It is considered that the technique used provides the essential effects of more interference velocity with increased rotor load, and varies as the rotor wake deflects rearward with increased forward speed.

The angles of attack and sideslip are derived from the body axes components of velocity. These comprise the components of flight path velocity, gust components and rotor downwash. The definition of the angles are those used in the wind tunnel. That is, angle of attack is the geometric angle subtended by the model relative to tunnel axis at zero yaw angle. It does not change with yaw angle. Angle of sideslip, equal to minus yaw, is defined as yaw table angle in the horizontal plane of the tunnel, irrespective of angle of attack. It should be noted that these angles are not Euler angles.

The fuselage aerodynamic characteristics are specific to the Black Hawk helicopter. They are not generalized in any way and are derived directly from wind tunnel tests. The aerodynamic coefficients in terms of ft^2 and ft^3 , for forces and moments respectively, are presented as functions of angle of attack and sideslip in Section 5.2.5. These wind axes forces and moments are subsequently transformed into body axes at the fuselage center of gravity. The data obtained from wind tunnel tests up to post stall conditions must be extended to $+90^\circ$ to cover the low speed flight regimes. Near hover, the most important forces (tail off) are the vertical drag and side force. These can be estimated fairly accurately. Because of the definitions of angle of attack and sideslip the transformation equation gives invalid body axes forces and moments when these angles approach 90° .

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To avoid problems during pilot-in-the-loop simulation, filters are presented which fade out the transformation and introduce fixed body axes parameters, estimated specifically for hover and low speed flight. For open loop analysis the longitudinal degrees-of-freedom are representative. It should be noted however, that inaccuracies will be encountered in pure side flight (i.e., rotor side wash on the fuselage does not exist). A block diagram indicating the fuselage module flow is presented on figure 2.1.1.

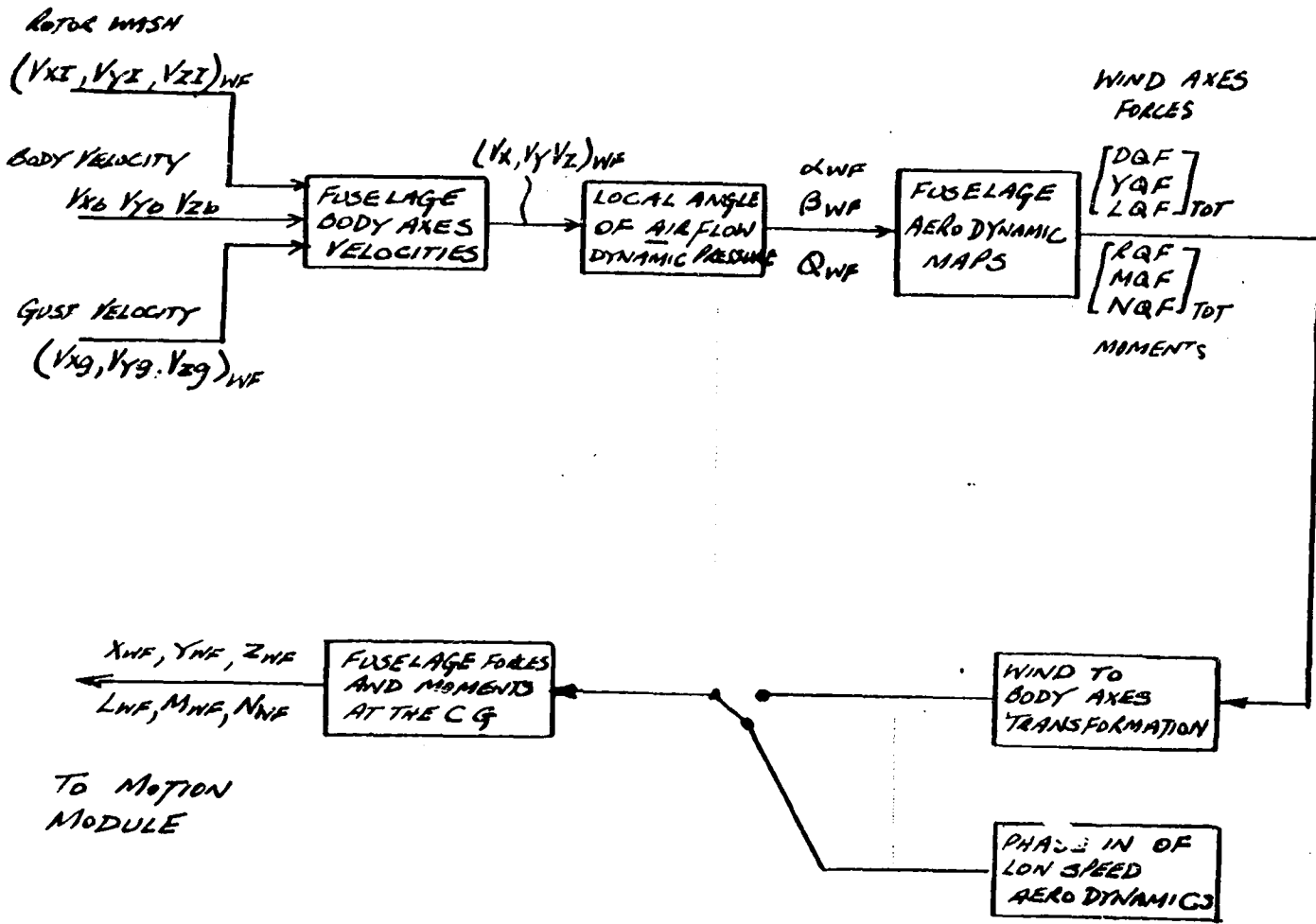


FIGURE 2.1.1

5-2-4
 PAGE

FUSELAGE EQUATION FLOW DIAGRAM

5.2.2. FUSELAGE MODULE EQUATIONS

ROTOR WASH ON THE FUSELAGE

$$V_{XIWF} = E_{XXWF} (D_{NO} \cdot S_{T} \cdot R_T)$$

$$V_{YIWF} = E_{XYWF} (D_{NO} \cdot S_{T} \cdot R_T)$$

$$V_{ZIWF} = -E_{XZWF} (D_{NO} \cdot S_{T} \cdot R_T)$$

WHERE $E_{XXWF} = f(X, A_{AIFMR})$, MAP - TABLE 2.5.1
FIGURE 2.5.1

$E_{XYWF} =$ NO INPUT DATA FOR BLACK HAWK AT THIS TIME

$E_{XZWF} = f(X, A_{AIFMR})$, MAP - TABLE 2.5.2
FIGURE 2.5.2.

FUSELAGE VELOCITY COMPONENTS

$$V_{XWF} = V_{XB} + V_{XGNF} + V_{XIWF}$$

$$V_{YWF} = V_{YB} + V_{YGNF} + V_{YIWF}$$

$$V_{ZWF} = V_{ZB} + V_{ZGNF} + V_{ZIWF}$$

CONTRIBUTIONS FROM ANGULAR RATES DUE
TO MOUNTING POINT OFFSET FROM THE CG ARE
IGNORED

ANGLES OF ATTACK AND SIDESLIP

THE FORMULATION OF THESE EQUATIONS DEPENDS ON THE CONVENTIONAL DEFINITION OF ANGLES IN THE WIND TUNNEL. α_{WF} IS MEASURED RELATIVE TO THE TUNNEL FLOOR, ψ_{WF} IS THE ANGLE OF THE TUNNEL YAW TABLE.

$$\alpha_{WF} = \tan^{-1} \left\{ \frac{V_{ZWF}}{|V_{XWF}|} \right\} \quad \text{ATTACK}$$

(ALFWF)

$$\alpha_{WNF} = \alpha_{WF} + \psi_{WF} \quad \text{NOT APPLICABLE TO BLACK HAWK}$$

(ALFWNF)

$$SNAFWF = \frac{V_{WF}}{(V_{XWF}^2 + V_{ZWF}^2)^{1/2}}$$

$$CSAFWF = \frac{V_{WF}}{(V_{XWF}^2 + V_{ZWF}^2)^{1/2}}$$

$$\beta_{NF} = \tan^{-1} \left\{ \frac{V_{YNF}}{(V_{XNF}^2 + V_{ZNF}^2)^{1/2}} \right\} \quad \text{SIDESLIP}$$

(BETANF)

$$-\psi_{WF} = -\beta_{NF}$$

(PSINF)

DYNAMIC PRESSURE

$$Q_{WF} = \frac{1}{2} \rho (V_{XWF}^2 + V_{YNF}^2 + V_{ZWF}^2)$$

FUSELAGE AERODYNAMIC LOADING COEFFICIENTS

$$D_{RFTOT} = D_{QFMP} + DD_{QFMP}$$

$$Y_{QFTOT} = Y_{QFMP}$$

$$L_{QFTOT} = L_{QFMP} + DL_{QFMP} + DL_{QD}$$

$$R_{QFTOT} = R_{QFMP}$$

$$M_{QFTOT} = M_{QFMP} + DM_{QFMP} + DM_{QD}$$

$$N_{QFTOT} = N_{QFMP} + DN_{QD}$$

WHERE

$$D_{QFMP} = f(\alpha_{WF}) , \text{ TABLE 2.5.3, FIGURE 2.5.3}$$

$$DD_{QFMP} = f(|\psi_{WF}|) , \text{ TABLE 2.5.4, FIGURE 2.5.4}$$

$$Y_{QFMP} = \frac{\psi_{WF}}{|\psi_{WF}|} \cdot f(|\psi_{WF}|) , \text{ TABLE 2.5.5, FIGURE 2.5.5}$$

$$L_{QFMP} = f(\alpha_{WF}) , \text{ TABLE 2.5.6, FIGURE 2.5.6}$$

$$DL_{QFMP} = f(\psi_{WF}) , \text{ TABLE 2.5.7, FIGURE 2.5.7}$$

$$R_{QFMP} = \frac{\psi_{WF}}{|\psi_{WF}|} \cdot f(|\psi_{WF}|) , \text{ TABLE 2.5.8, FIGURE 2.5.8}$$

$$M_{QFMP} = f(\alpha_{WF}) , \text{ TABLE 2.5.9, FIGURE 2.5.9}$$

$$DM_{QFMP} = \frac{\psi_{WF}}{|\psi_{WF}|} \cdot f(|\psi_{WF}|) , \text{ TABLE 2.5.10, FIGURE 2.5.10}$$

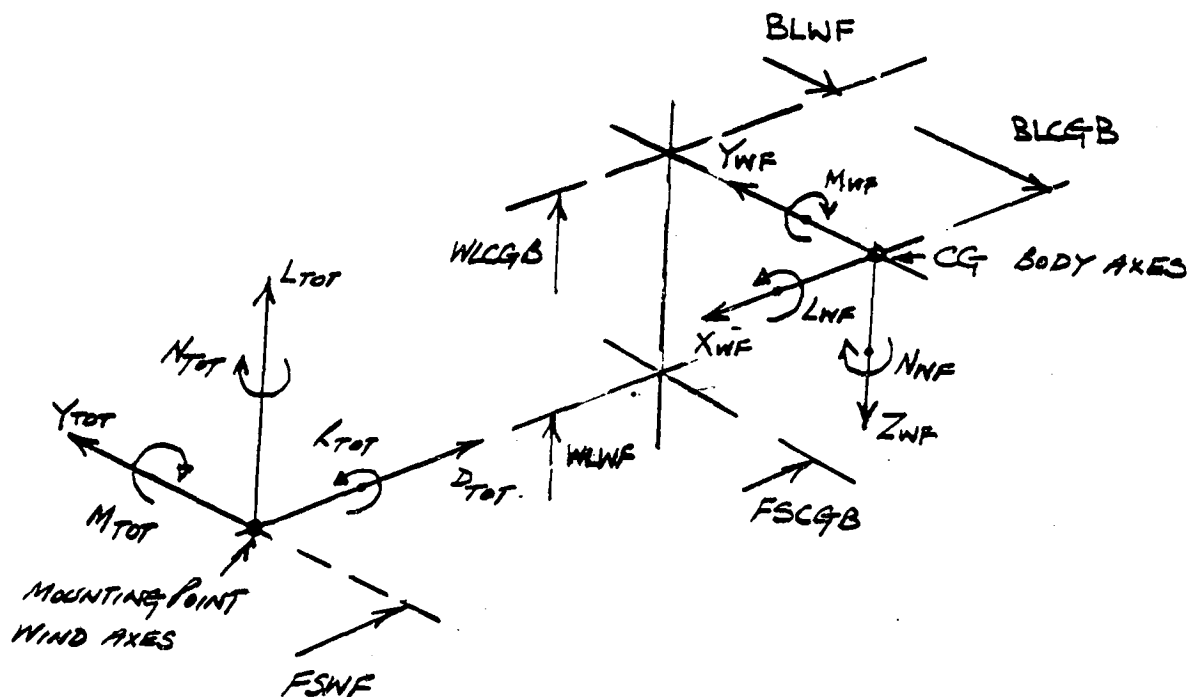
$$N_{QFMP} = f(\psi_{WF}) , \text{ TABLE 2.5.11, FIGURE 2.5.11}$$

G E O M E T R Y

$$F_{WT} = \frac{(F_{SCGB} - F_{SNF})}{12}$$

$$W_{WT} = \frac{(W_{LCGB} - W_{LNF})}{12}$$

$$B_{WT} = \frac{(B_{LCGB} - B_{LNF})}{12}$$



SIDESLIP RESOLUTION COEFFICIENTS

S_{NBTF} } DERIVED FROM $SIN\cos(\beta_{NF})$ ROUTINE
 C_{SBTF} }

TRANSFORMATION OF WIND AXES FORCES AND
MOMENTS INTO BODY AXES

NOTE α_{NF} , β_{NF} ARE NOT EULER ANGLES.

$$\begin{bmatrix} X_{WF} \\ Y_{WF} \\ Z_{WF} \end{bmatrix} = \begin{bmatrix} (\cos\alpha_{NF} \cos\beta_{NF}), (\cos\alpha_{NF} \sin\beta_{NF}), (-\sin\alpha_{NF}) \\ (\sin\beta_{NF}), (-\cos\beta_{NF}), 0 \\ (\sin\alpha_{NF} \cos\beta_{NF}), (\sin\alpha_{NF} \sin\beta_{NF}), (\cos\alpha_{NF}) \end{bmatrix} \begin{bmatrix} -DQ_{TOT} * Q_{WF} \\ -YQ_{TOT} * Q_{WF} \\ -LQ_{TOT} * Q_{WF} \end{bmatrix}$$

$$\begin{bmatrix} L_{WF} \\ M_{WF} \\ N_{WF} \end{bmatrix} = \begin{bmatrix} L_{WFNF} \\ M_{WFNF} \\ N_{WFNF} \end{bmatrix} + \begin{bmatrix} -Y_{WF} \cdot W_{NT} + Z_{WF} \cdot B_{NT} \\ -Z_{WF} \cdot F_{NT} + X_{WF} \cdot W_{NT} \\ + Y_{WF} \cdot F_{NT} - X_{WF} \cdot B_{NT} \end{bmatrix}$$

WHERE

$$\begin{bmatrix} L_{WFNF} \\ M_{WFNF} \\ N_{WFNF} \end{bmatrix} = \begin{bmatrix} \text{SAME MATRIX} \\ \text{AS ABOVE} \end{bmatrix} \begin{bmatrix} RQ_{TOT} * Q_{WF} \\ -MQ_{TOT} * Q_{WF} \\ NQ_{TOT} * Q_{WF} \end{bmatrix}$$

LOW SPEED FILTER OF FUSELAGE AERODYNAMICS

THIS FILTER IS INTENDED FOR USE WITH PILOT-IN-THE-LOOP SIMULATION AND IS AN ATTEMPT TO ELIMINATE PROBLEMS OF HIGH α_{WF} AND β_{WF} OPERATION

* IF $|V_{XWF}| > 25 \text{ FT/SEC}$ $(XYZ LMN)_{WF} = (XYZ LMN)'_{WF}$

IF $|V_{XWF}| \leq 25 \text{ FT/SEC}$.

* $X_{WF} = \left| \frac{V_{XWF}}{25} \right| X_{WF}' - \frac{\alpha_{WF}}{|\alpha_{WF}|} \left\{ 1 - \left| \frac{V_{XWF}}{25} \right| \right\} \cdot X_{LS} q_{WF}$

* $Y_{WF} = \left| \frac{V_{XWF}}{25} \right| Y_{WF}' - \frac{V_{YWF}}{|V_{YWF}|} \left\{ 1 - \left| \frac{V_{XWF}}{25} \right| \right\} \cdot Y_{LS} q_{WF}$

* $Z_{WF} = \left| \frac{V_{XWF}}{25} \right| Z_{WF}' - \frac{\alpha_{WF}}{|\alpha_{WF}|} \left\{ 1 - \left| \frac{V_{XWF}}{25} \right| \right\} Z_{LS} q_{WF}$

* $L_{WF} = \left| \frac{V_{XWF}}{25} \right| L_{WF}' - \frac{V_{YWF}}{|V_{YWF}|} \left\{ 1 - \left| \frac{V_{XWF}}{25} \right| \right\} L_{LS} q_{WF}$

* $M_{WF} = \left| \frac{V_{XWF}}{25} \right| M_{WF}' - \frac{\alpha_{WF}}{|\alpha_{WF}|} \left\{ 1 - \left| \frac{V_{XWF}}{25} \right| \right\} M_{LS} q_{WF}$

* $N_{WF} = \left| \frac{V_{XWF}}{25} \right| N_{WF}' - \frac{V_{YWF}}{|V_{YWF}|} \left\{ 1 - \left| \frac{V_{XWF}}{25} \right| \right\} N_{LS} q_{WF}$

* THESE EQUATIONS ARE NOT BEING EXECUTED ON PDP 10

5.2.3 FUSELAGE MODULE INPUT/OUTPUT DATA TRANSFER

INPUT TRANSFER	
PARAMETER	ORIGIN MODULE
BLCGB FSCGB WLCGB DWSHMR CHIAMR AIFMR OMGTMR RMR	MAIN ROTOR
VXGWF VYGWF VZGWF	GUST
VXB VYB VZB	MOTION

OUTPUT TRANSFER	
PARAMETER	DESTINATION MODULE
ALFNF BETNF	EMPENNAGE
ALFNF BETNF	TAIL ROTOR
XWF YWF ZWF LWF MWF NWF	MOTION



5.2.4

Notation for the Fuselage Module

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
V_{XWF}	VXWF	FT/SEC	Total velocity components at the fuselage center of gravity
V_{YWF}	VYWF	FT/SEC	
V_{ZWF}	VZWF	FT/SEC	
V_{XIWF}	VXIWF	FT/SEC	Rotor wash interference on the fuselage.
V_{YIWF}	VYIWF	FT/SEC	
V_{ZIWF}	VZIWF	FT/SEC	
EK_{XWF}	EKXWF	-	Rotor wash interference factors
EK_{YWF}	EKYWF	-	
EK_{ZWF}	EKYWF	-	
D_{WO}	DWSHMR	-	Main rotor uniform downwash.
Ω_T	OMGTMR	RADS/SEC	Rotor speed
R_T	RMR	FT	Rotor radius
q_{WF}	QWF	LB/FT ²	Dynamic pressure at the body
α_{WF}	ALFWF	DEG	Body axis angle of attack
$1\alpha_{WF1}$	AFABWF	DEG	ABS (ALFWF)
α_{WWF}	ALFWWF	DEG	Wing angle of attack
β_{WF}	BETAWF	DEG	Sideslip angle
ψ_{WF}	PSIWF	DEG	W/T model yaw angle (=-BETAWF)
$1\psi_{WF1}$	PSABWF	DEG	ABS (PSIWF)
-	SNAFWF		SIN(ALFWF)
-	CSAFWF		COS(ALFWF)
-	SNBTWF		SIN(BETAWF)
-	CSBTWF		COS(BETAWF)



5.2.4 (Cont'd)

Notation for the Fuselage Module

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
DQFMP	DQFMP	FT ²	Drag coefficient from angle of attack
YQFMP	YQFMP	FT ²	Side force coefficient from sideslip
LQFMP	LWFMP	FT ²	Lift coefficient from angle of attack
RQFMP	RQFMP	FT ³	Rolling moment coefficient from sideslip
MQFMP	MQFMP	FT ³	Pitching moment coefficient from angle of attack
NQFMP	NQFMP	FT ³	Yawing moment coefficient from sideslip
DDQFMP	DDQFMP	FT ²	Deltat drag coefficient from sideslip
DLQFMP	DLWFMP	FT ²	Delta lift coefficient from sideslip
DMQFMP	DMQFMP	FT ³	Delta pitching moment coefficient from sideslip
DQFTOT	DQFTOT	FT ²	Total components of aerodynamic coefficients at the wind tunnel mounting point in wind axes.
YQFTOT	YQFTOT	FT ²	
LQFTOT	LQFTOT	FT ²	
RQFTOT	RQFTOT	FT ³	
MQFTOT	MQFTOT	FT ³	
NQFTOT	NQFTOT	FT ³	
FSCGB	FSCGB	INS	Fuselage station for the fuselage C.G.
WLCGB	WLCGB	INS	Waterline station for the fuselage C.G.
FSWF	FSWF	INS	Fuselage station for tunnel mounting point.



5.2.4 (Cont'd) Notation for the Fuselage Module

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
WLWF	WLWF	INS	Waterline station for tunnel mounting point.
F _{WT}	-	FT	Fuselage longitudinal mounting point arm.
W _{WT}	-	FT	Fuselage vertical mounting point arm.
B _{WT}	-	FT	Fuselage lateral mounting point arm.
X _{WF} '	XWF'	LB	Fuselage aerodynamic component loads in body axes at the C.G.
Y _{WF} '	YWF'	LB	
Z _{WF} '	ZWF'	LB	
L _{WF} '	LWF'	FT LB	
M _{WF} '	MWF'	FT LB	
N _{WF} '	NWF'	FT LB	
X _{WF}		LB	Fuselage aerodynamic component loads in body axes at the C.G. modified by low speed phasing.
Y _{WF}		LB	
Z _{WF}		LB	
L _{WF}		FT LB	
M _{WF}		FT LB	
N _{WF}		FT LB	
V _{XGWF}	V _{XGWF}	FT/SEC	<i>QUST VELOCITIES AT THE FUSELAGE</i>
V _{YGWF}	V _{YGWF}	FT/SEC	
V _{ZGWF}	V _{ZGWF}	FT/SEC	

5.2.5 BLACK HAWK FUSELAGE INPUT DATA

INPUT CONSTANTS

$$FSWF = 345.5 \text{ ms}$$

$$WLWF = 234.0 \text{ ms}$$

$$BLWF = 0.0 \text{ ms}$$

$$IWF = 0.0$$

TABLE 2.5.1

INPLANE COMPONENT OF ROTOR WASH ON THE FUSELAGE

```

EXWFMP::BIV##           ;MAP ARGUMENT:LOOK UP ROUTINE
EXP  CHIPMR##,AAIFMR## ;INPUT VARIABLE #1, INPUT VARIABLE #2
      EKXWF##          ;OUTPUT VARIABLE
      EXWFLO           ;LOW ANGLE MAP NAME
EXP  0.0,100.0,10.0,13 ;LOW LIM,UPPER LIM,DELTA,#ENTRYS(OCT)-CHIPMR
EXP  -6.0,6.0,6.0      ;LOW LIM,UPPER LIM,DELTA-AAIFMR
    
```

```

; LOW ANGLE MAP CHIPMR 0 TO 100 (DEL=10) AAIFMR -6,0,6
; AAIFMR=-6
EXWFLO:EXP  0.08,      0.18,      0.3,      0.43,      0.55
EXP          0.66,      0.79,      0.9,      1.03,      0.55
            0.0

; AAIFMR=0
EXP          0.0,      0.1,      0.21,     0.32,     0.42
EXP          0.54,     0.66,     0.8,      0.94,     0.5

; AAIFMR=6
EXP          -0.12,    0.02,     0.08,     0.18,     0.28
EXP          0.4,     0.53,     0.67,     0.82,     0.4
            0.0
    
```

TABLE 2.5.2

DOWNWASH FROM THE MAIN ROTOR ONTO THE FUSELAGE

```

EZWFMP::BIV##           ;MAP ARGUMENT:LOOK UP ROUTINE
EXP  CHIPMR##,AAIFMR## ;INPUT VARIABLE #1, INPUT VARIABLE #2
      EKZWF##          ;OUTPUT VARIABLE
      EZWFLO           ;LOW ANGLE MAP NAME
EXP  0.0,100.0,10.0,13 ;LOW LIM,UPPER LIM,DELTA,#ENTRYS(OCT)-CHIPMR
EXP  -6.0,6.0,6.0      ;LOW LIM,UPPER LIM,DELTA-AAIFMR
    
```

```

; LOW ANGLE MAP CHIPMR 0 TO 100 (DEL=10) AAIFMR -6,0,6
; AAIFMR=-6
EZWFLO:EXP  1.11,     1.09,     1.08,     1.065,    1.25
EXP          1.04,     1.02,     1.01,     1.0,      0.88
            0.6

; AAIFMR=0
EXP          1.12,     1.12,     1.12,     1.12,     1.12
EXP          1.12,     1.12,     1.12,     1.11,     0.96
            0.6

; AAIFMR=6
EXP          1.15,     1.15,     1.15,     1.15,     1.16
EXP          1.17,     1.18,     1.22,     1.16,     0.98
            0.6
    
```

TABLE 2.5.3

FUSELAGE DRAG COEFFICIENT DUE TO ANGLE OF ATTACK

```

DQFMP: :UVRUVR## ;MAP ARGUMENT:LOOK UP ROUTINE
        ALFWF## ;INPUT VARIABLE
        DQF## ;OUTPUT VARIABLE
        DQFLO ;LOW ANGLE MAP NAME
EXP -38.8,38.8,5.8 ;LOWER LIMIT,UPPER LIMIT,DELTA-LOW ANGLE
        DQFHI ;HIGH ANGLE MAP NAME
EXP -98.8,98.8,18.8 ;LOWER LIMIT,UPPER LIMIT,DELTA-HIGH ANGLE

; LOW ANGLE MAP: ALFWF -38 TO 38 , DELTA=5
DQFLO: EXP 45.88, 37.38, 31.68, 27.48, 25.86
        EXP 23.58, 23.58, 25.88, 27.58, 31.28
        EXP 36.58, 43.88, 51.88

; HIGH ANGLE MAP: ALFWF -98 TO 98 , DELTA=18
DQFHI: EXP 158.8, 145.8, 133.8, 114.8, 88.8
        EXP 61.8, 45.88, 31.68, 25.86, 23.58
        EXP 27.58, 36.58, 51.88, 66.8, 84.8
        EXP 118.8, 132.8, 145.8, 158.8
    
```

TABLE 2.5.4

INCREMENTAL FUSELAGE DRAG COEFFICIENT DUE TO SIDESLIP

```

DDQFMP: :UVRUVR## ;MAP ARGUMENT:LOOK UP ROUTINE
        PSABWF## ;INPUT VARIABLE
        DDQF## ;OUTPUT VARIABLE
        DDQFLO ;LOW ANGLE MAP NAME
EXP 8.8,38.8,5.8 ;LOWER LIMIT,UPPER LIMIT,DELTA-LOW ANGLE
        DDQFHI ;HIGH ANGLE MAP NAME
EXP 38.8,98.8,18.8 ;LOWER LIMIT,UPPER LIMIT,DELTA-HIGH ANGLE

; LOW ANGLE MAP: PSI(ABS) 8 TO 38, DELTA=5
DDQFLO: EXP 8.8, 1.8, 4.8, 9.8, 16.3
        EXP 28.8, 38.5

; HIGH ANGLE MAP: PSI(ABS) 38 TO 98, DELTA=18
DDQFHI: EXP 38.5, 76.5, 113.8, 141.5, 164.5
        EXP 169.5, 178.5
    
```

TABLE 2.5.5

FUSELAGE SIDE FORCE COEFFICIENT DUE TO SIDESLIP

```

YQFMP: :UVSUVS## ;MAP ARGUMENT:LOOK UP ROUTINE
        PSIW## ;INPUT VARIABLE
        YQF## ;OUTPUT VARIABLE
        YQFLO ;LOW ANGLE MAP NAME
EXP 8.8,38.8,5.8 ;LOWER LIMIT,UPPER LIMIT,DELTA-LOW ANGLE
        YQFHI ;HIGH ANGLE MAP NAME
EXP 38.8,98.8,18.8 ;LOWER LIMIT,UPPER LIMIT,DELTA-HIGH ANGLE

; LOW ANGLE MAP: PSIW 8 TO 38, DELTA=5 Y(PSI)=-Y(-PSI)
YQFLO: EXP 8.8, 11.8, 23.8, 35.8, 58.8
        EXP 65.8, 72.8

; HIGH ANGLE MAP: PSIW 38 TO 98, DELTA=18 Y(PSI)=-Y(-PSI)
YQFHI: EXP 72.8, 92.8, 183.8, 188.8, 84.8
        EXP 64.8, 37.8
    
```

BLACK HAWK (TAIL OFF IRS OFF)

TABLE 2.5.6

FUSELAGE LIFT COEFFICIENT DUE TO ANGLE OF ATTACK

LQFMP:	:UVRUVR##						:MAP ARGUMENT:LOOK UP ROUTINE
	ALFWF##						:INPUT VARIABLE
	LQF##						:OUTPUT VARIABLE
	LQFLO						:LOW ANGLE MAP NAME
EXP	-30.0, 30.0, 5.0						:LOWER LIMIT, UPPER LIMIT, DELTA-LOW ANGLE
	LQFHI						:HIGH ANGLE MAP NAME
EXP	-90.0, 90.0, 10.0						:LOWER LIMIT, UPPER LIMIT, DELTA-HIGH ANGLE
; LOW ANGLE MAP: ALFWF -30 TO 30 , DELTA=5							
LQFLO:	EXP	-70.0,	-52.0,	-35.0,	-25.0,	-13.0	
	EXP	-5.0,	1.0,	10.0,	20.0,	25.0	
	EXP	30.0,	34.0,	37.0			
; HIGH ANGLE MAP: ALFWF -90 TO 90 , DELTA=10							
LQFHI:	EXP	-24.0,	-54.0,	-72.0,	-81.0,	-85.0	
	EXP	-83.0,	-70.0,	-35.0,	-13.0,	1.0	
	EXP	20.0,	30.0,	37.0,	43.0,	48.0	
	EXP	50.0,	48.0,	39.0,	22.0		

TABLE 2.5.7

INCREMENTAL FUSELAGE LIFT COEFFICIENT DUE TO SIDESLIP

DLQFMP:	:UVR##						:MAP ARGUMENT:LOOK UP ROUTINE
	PSIWF##						:INPUT VARIABLE
	DLQF##						:OUTPUT VARIABLE
	DLQFLO						:LOW ANGLE MAP NAME
EXP	-30.0, 30.0, 5.0						:LOWER LIMIT, UPPER LIMIT, DELTA-LOW ANGLE
; LOW ANGLE MAP: PSIWF -30 TO 30, DELTA=5							
DLQFLO:	EXP	30.0,	20.0,	12.0,	7.0,	3.0	
	EXP	2.0,	0.0,	2.0,	5.0,	10.0	
	EXP	15.0,	22.0,	30.0			

TABLE 2.5.8

FUSELAGE ROLLING MOMENT COEFFICIENT DUE TO SIDESLIP

RQFMP:	:UVSUVS##						:MAP ARGUMENT:LOOK UP ROUTINE
	PSIWF##						:INPUT VARIABLE
	RQF##						:OUTPUT VARIABLE
	RQFLO						:LOW ANGLE MAP NAME
EXP	0.0, 30.0, 5.0						:LOWER LIMIT, UPPER LIMIT, DELTA-LOW ANGLE
	RQFHI						:HIGH ANGLE MAP NAME
EXP	30.0, 90.0, 10.0						:LOWER LIMIT, UPPER LIMIT, DELTA-HIGH ANGLE
; LOW ANGLE MAP: PSIWF 0 TO 30, DELTA=5 R(PHI)=-R(-PHI)							
RQFLO:	EXP	0.0,	0.0,	0.0,	-30.0,	-75.0	
	EXP	-120.0,	-110.0				
; HIGH ANGLE MAP: PSIWF 30 TO 90, DELTA=10 R(PHI)=-R(-PHI)							
RQFHI:	EXP	-110.0,	-100.0,	-103.0,	-101.0,	-100.0	
	EXP	-100.0,	-100.0				

TABLE 2.5.9

FUSELAGE PITCHING MOMENT COEFFICIENT DUE TO ANGLE OF ATTACK

```

MQFMP: :UVRUVR## ;MAP ARGUMENT:LOOK UP ROUTINE
        ALFWF## ;INPUT VARIABLE
        MQF## ;OUTPUT VARIABLE
        MQFLO ;LOW ANGLE MAP NAME
EXP -30.0,30.0,5.0 ;LOWER LIMIT,UPPER LIMIT,DELTA-LOW ANGLE
        MQFHI ;HIGH ANGLE MAP NAME
EXP -90.0,90.0,10.0 ;LOWER LIMIT,UPPER LIMIT,DELTA-HIGH ANGLE

MQFLO: EXP ; LOW ANGLE MAP: ALFWF -30 TO 30 , DELTA=5
        EXP -740.0, -700.0, -630.0, -520.0, -380.0
        EXP -230.0, -90.0, 10.0, 100.0, 290.0
        EXP 450.0, 600.0, 750.0

MQFHI: EXP ; HIGH ANGLE MAP: ALFWF -90 TO 90 , DELTA=10
        EXP -200.0, -470.0, -645.0, -730.0, -760.0
        EXP -760.0, -740.0, -630.0, -380.0, -90.0
        EXP 100.0, 450.0, 750.0, 810.0, 825.0
        EXP 700.0, 650.0, 470.0, 200.0
    
```

TABLE 2.5.10

INCREMENTAL FUSELAGE PITCHING MOMENT DUE TO SIDESLIP

```

DMQFMP: :UVRUVR## ; ** BLACK HAWK FUSELAGE DEL PITCH MOM VS PSIWF(ABS)
        PSABWF## ;MAP ARGUMENT:LOOK UP ROUTINE
        DMQF## ;INPUT VARIABLE
        DMQFLO ;OUTPUT VARIABLE
        DMQFHI ;LOW RANGE MAP NAME
EXP 0.0,30.0,5.0 ;LOWER LIMIT,UPPER LIMIT,DELTA-LOW ANGLE MAP

DMQFLO: EXP ; LOW ANGLE MAP: PSI(ABS) 0 TO 30, DELTA=5
        EXP 0.0, 10.0, 20.0, 50.0, 90.0
        EXP 130.0, 180.0
    
```

TABLE 2.5.11

FUSELAGE YAWING MOMENT COEFFICIENT DUE TO SIDESLIP

```

NQFMP: :UVRUVR## ;MAP ARGUMENT:LOOK UP ROUTINE
        PSIWF## ;INPUT VARIABLE
        NQF## ;OUTPUT VARIABLE
        NQFLO ;LOW ANGLE MAP NAME
EXP -30.0,30.0,5.0 ;LOWER LIMIT,UPPER LIMIT,DELTA-LOW ANGLE
        NQFHI ;HIGH ANGLE MAP NAME
EXP -90.0,90.0,10.0 ;LOWER LIMIT,UPPER LIMIT,DELTA-HIGH ANGLE

NQFLO: EXP ; LOW ANGLE MAP: PSIWF -30 TO 30, DELTA=5
        EXP -140.0, -190.0, -240.0, -220.0, -180.0
        EXP -100.0, 0.0, 100.0, 180.0, 220.0
        EXP 240.0, 190.0, 140.0

NQFHI: EXP ; HIGH ANGLE MAP: PSIWF -90 TO 90, DELTA=10
        EXP 440.0, 392.0, 332.0, 259.0, 160.0
        EXP 40.0, -140.0, -240.0, -180.0, 0.0
        EXP 180.0, 240.0, 140.0, 59.0, -30.0
        EXP -125.0, -220.0, -320.0, -420.0
    
```

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INPLANE COMPONENT OF ROTOR WASH ON THE FUSELAGE

MAP NAME: EXWFMP
MAP TYPE: BIV
INPUT VARIABLE(S): CHIPMA AAIFMA
OUTPUT VARIABLE: EKXWF
PRIMARY MAP:
0.00
100.00
10.00
LOWER LIMIT: -6.00
UPPER LIMIT: 6.00
DELTA: 6.00

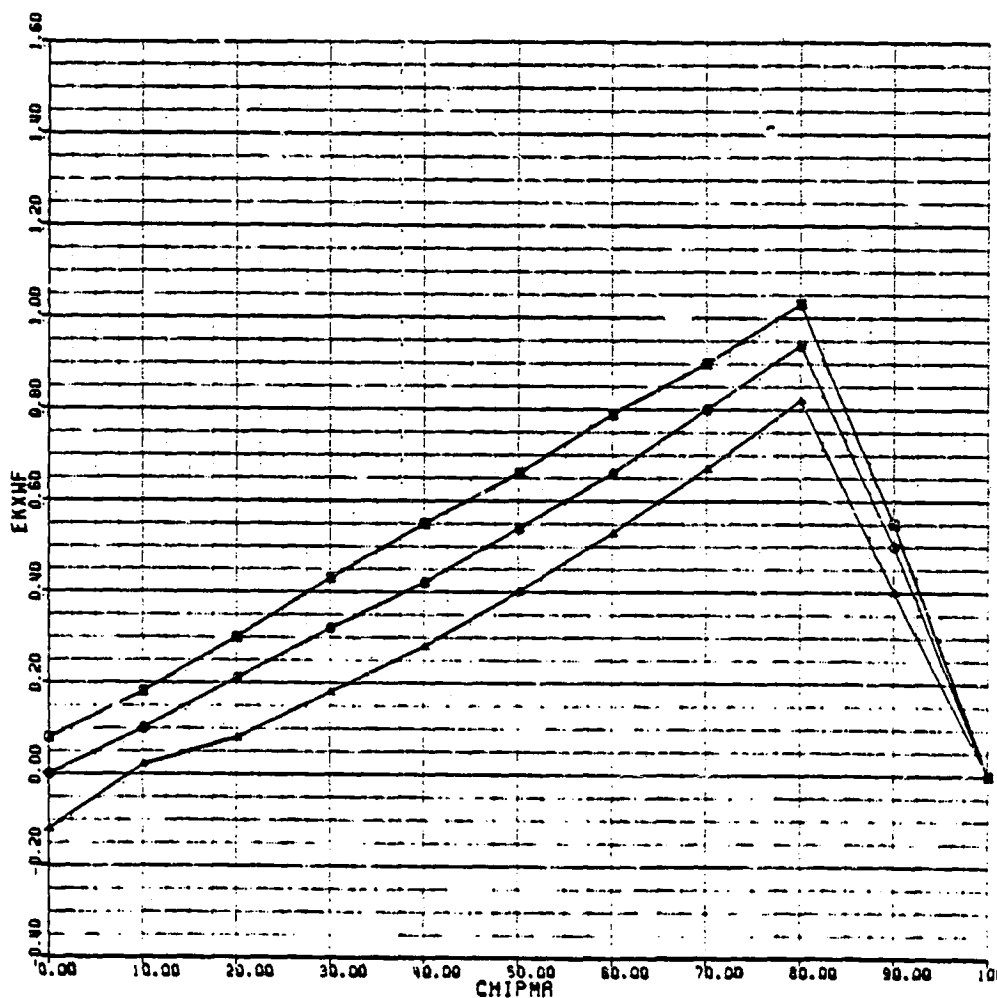
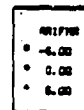


FIGURE 2.5.1

DOWNWASH FROM THE MAIN ROTOR ONTO THE FUSELAGE

MAP NAME: EZWFMP
 MAP TYPE: 81V
 INPUT VARIABLE(S): CHIPMA RA1FMA
 OUTPUT VARIABLE: EKZWF
 PRIMARY MAP:
 0.00 LOWER LIMIT -6.00
 100.00 UPPER LIMIT 6.00
 10.00 DELTA 6.00

RA1FMA
 • -6.00
 • 0.00
 • 6.00

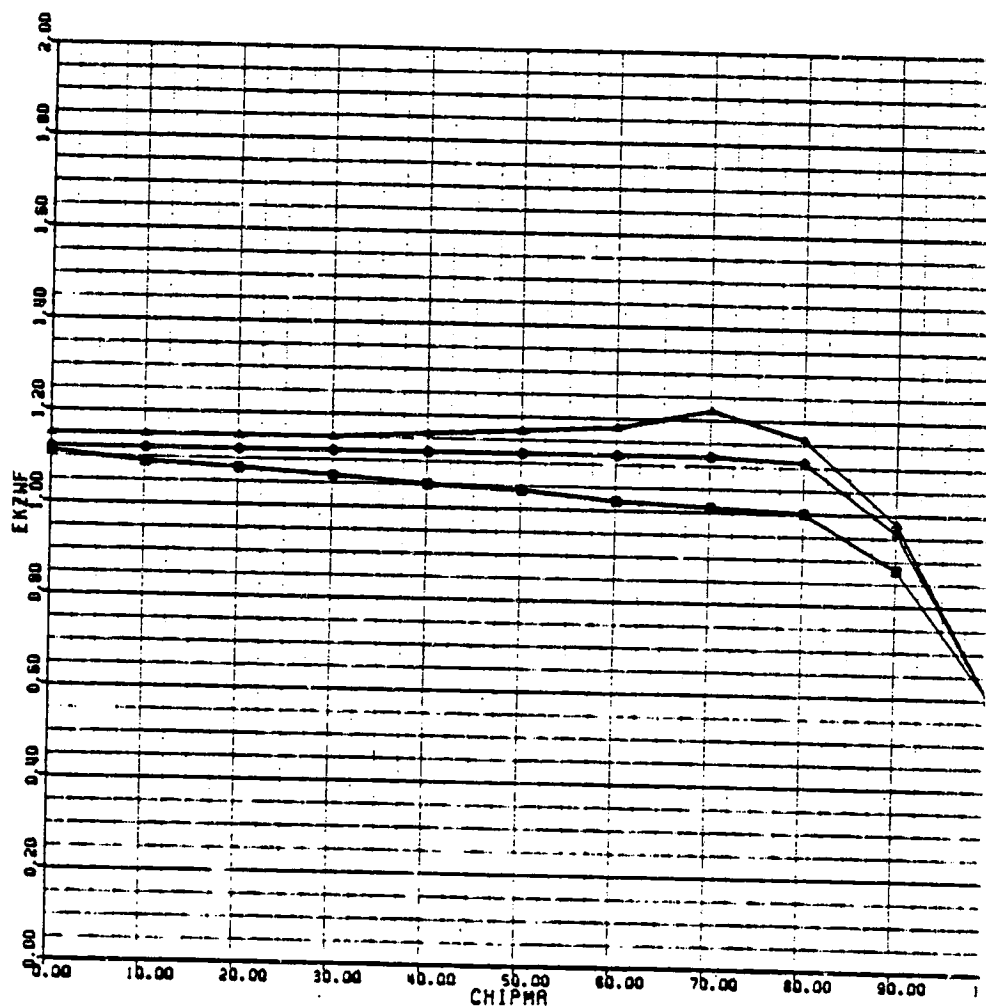


FIGURE 2.5.2

FUSELAGE DRAG COEFFICIENT DUE TO ANGLE OF ATTACK

BLACKHAWK - NASA STUDY 23-SEP-80

QQFMP (1/2)

MAP NAME: QQFMP
 MAP TYPE: UYRUYR
 INPUT VARIABLE(S): ALFWF
 OUTPUT VARIABLE: QQF

PRIMARY MAP:
 -30.00 LOWER LIMIT
 30.00 UPPER LIMIT
 5.00 DELTA

SECONDARY MAP:
 -90.00 LOWER LIMIT
 90.00 UPPER LIMIT
 10.00 DELTA

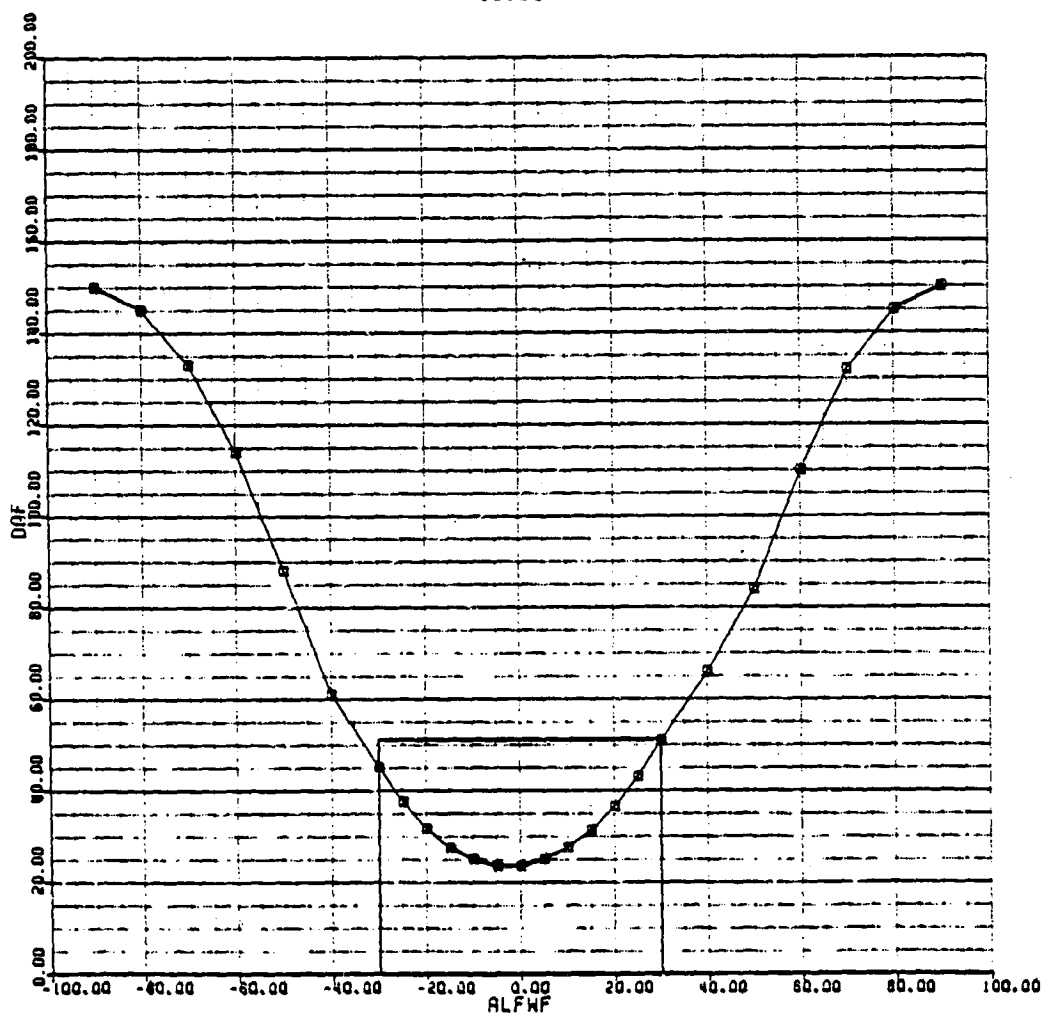


FIGURE 2.5.3(a)

FUSELAGE DRAG COEFFICIENT DUE TO ANGLE OF ATTACK (cont'd)

BLACKHAWK - NASA STUDY 23-SEP-80

DDFMP (2/2)

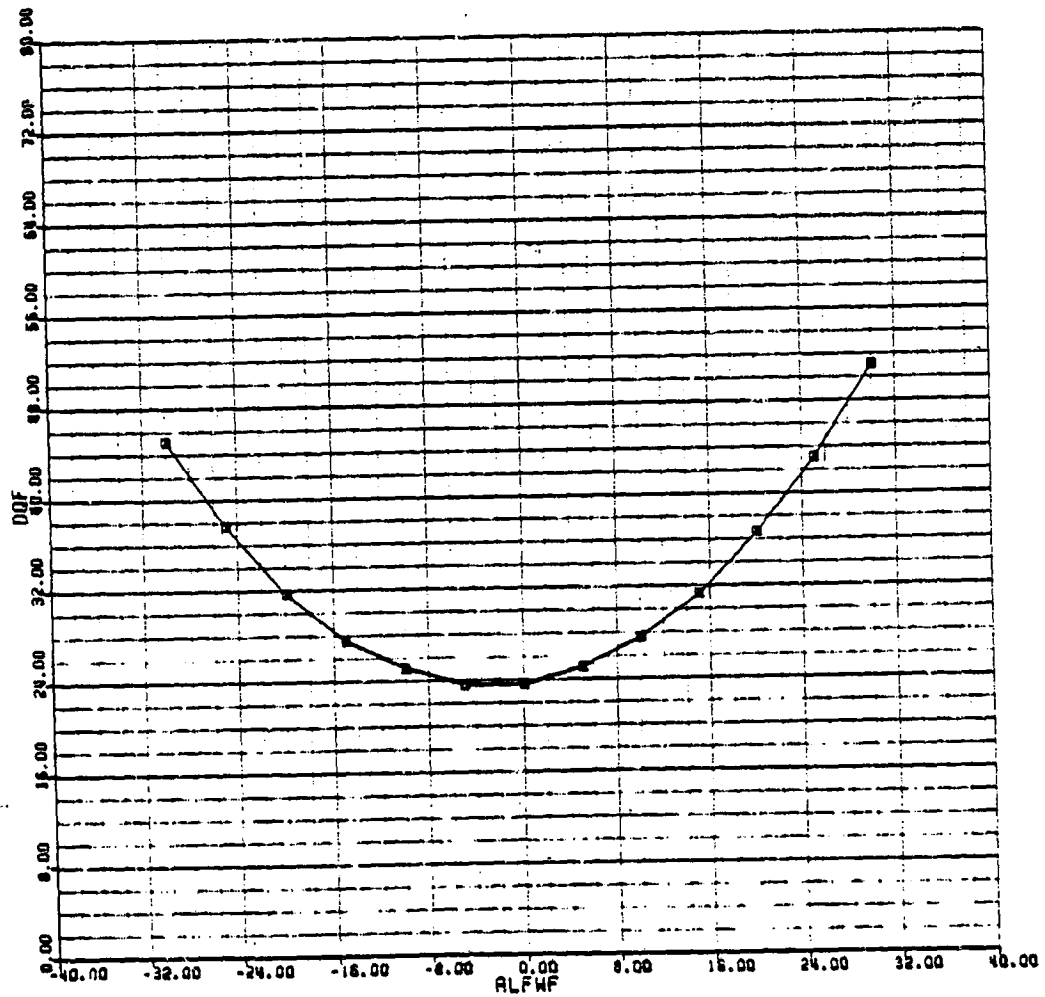


FIGURE 2.5.3(b)

INCREMENTAL FUSELAGE DRAG COEFFICIENT DUE TO SIDESLIP

BLACKHAWK - NASA STUDY 23-SEP-80

DDQFMP (1/2)

MAP NAME: DDQFMP
MAP TYPE: UVAVVA
INPUT VARIABLE(S): PSABWF
OUTPUT VARIABLE: DDQF

PRIMARY MAP:
0.00 LOWER LIMIT
30.00 UPPER LIMIT
5.00 DELTA

SECONDARY MAP:
30.00 LOWER LIMIT
90.00 UPPER LIMIT
10.00 DELTA

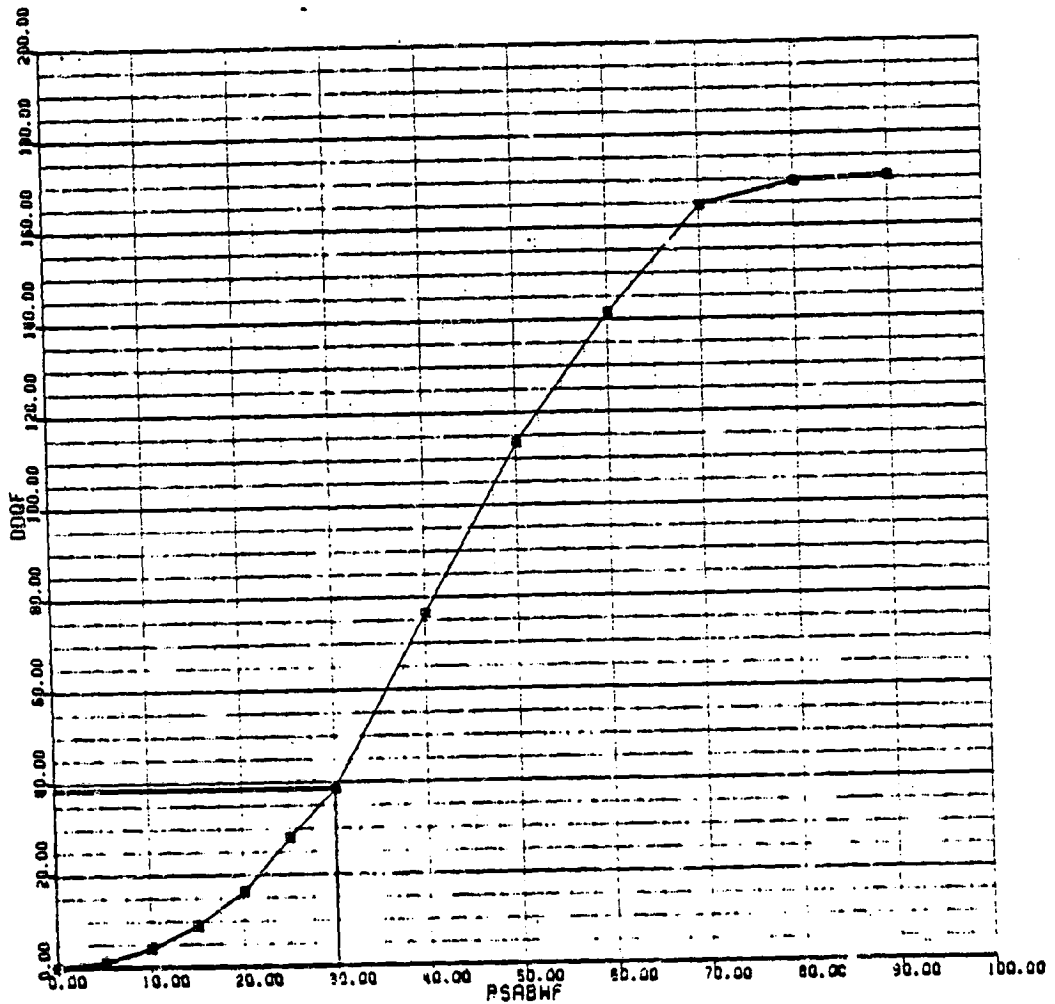


FIGURE 2.5.4(a)

INCREMENTAL FUSELAGE DRAG COEFFICIENT DUE TO SIDESLIP (Cont'd)

BLACKHAWK - NASA STUDY 23-SEP-80

DDGFMP (2/2)

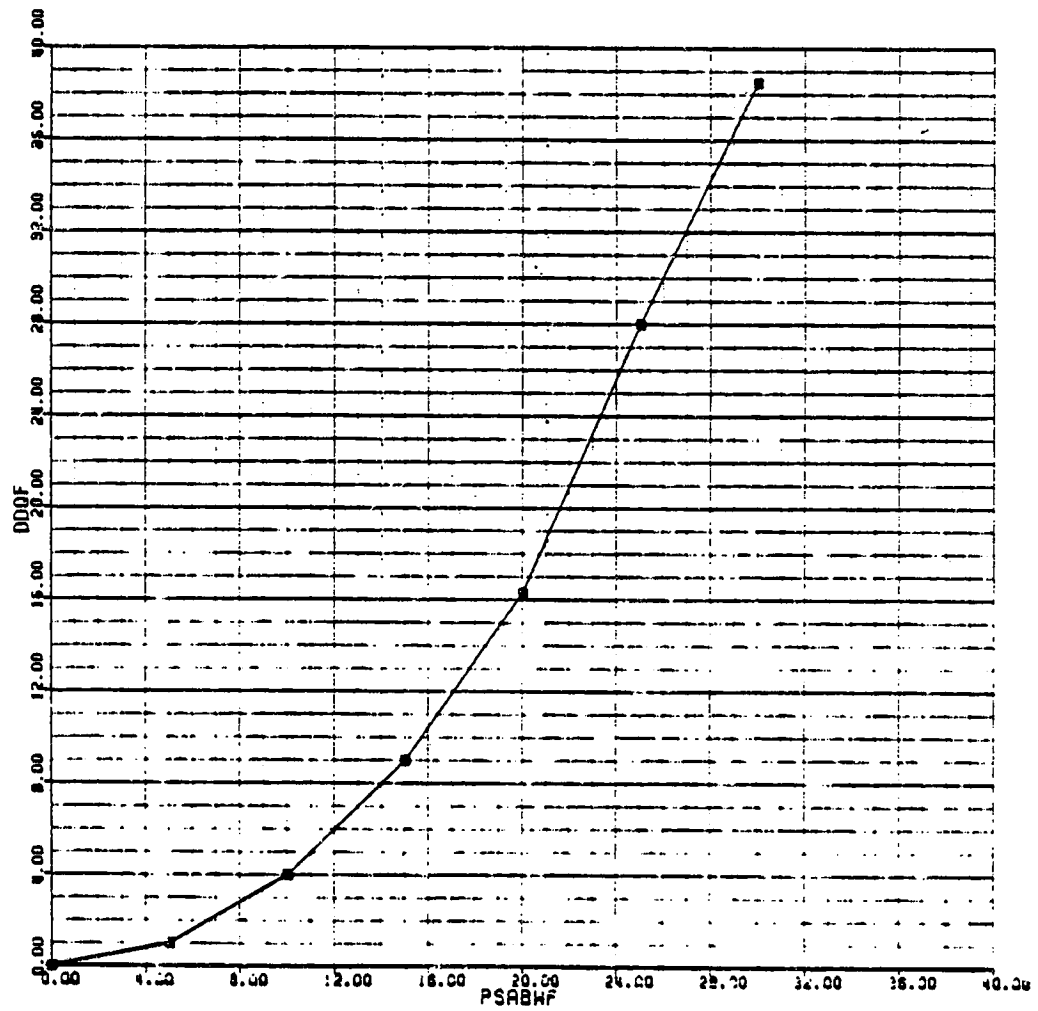


FIGURE 2.5.4(b)

FUSELAGE SIDEFORCE COEFFICIENT DUE TO SIDESLIP

BLACKHAWK - NASA STUDY 23-SEP-80

YQFMP (1/2)

MAP NAME: YQFMP
MAP TYPE: UVSUYS
INPUT VARIABLE(S): PSIH_F
OUTPUT VARIABLE: YQF

PRIMARY MAP:
-30.00 LOWER LIMIT
30.00 UPPER LIMIT
5.00 DELTA

SECONDARY MAP:
-90.00 LOWER LIMIT
90.00 UPPER LIMIT
10.00 DELTA

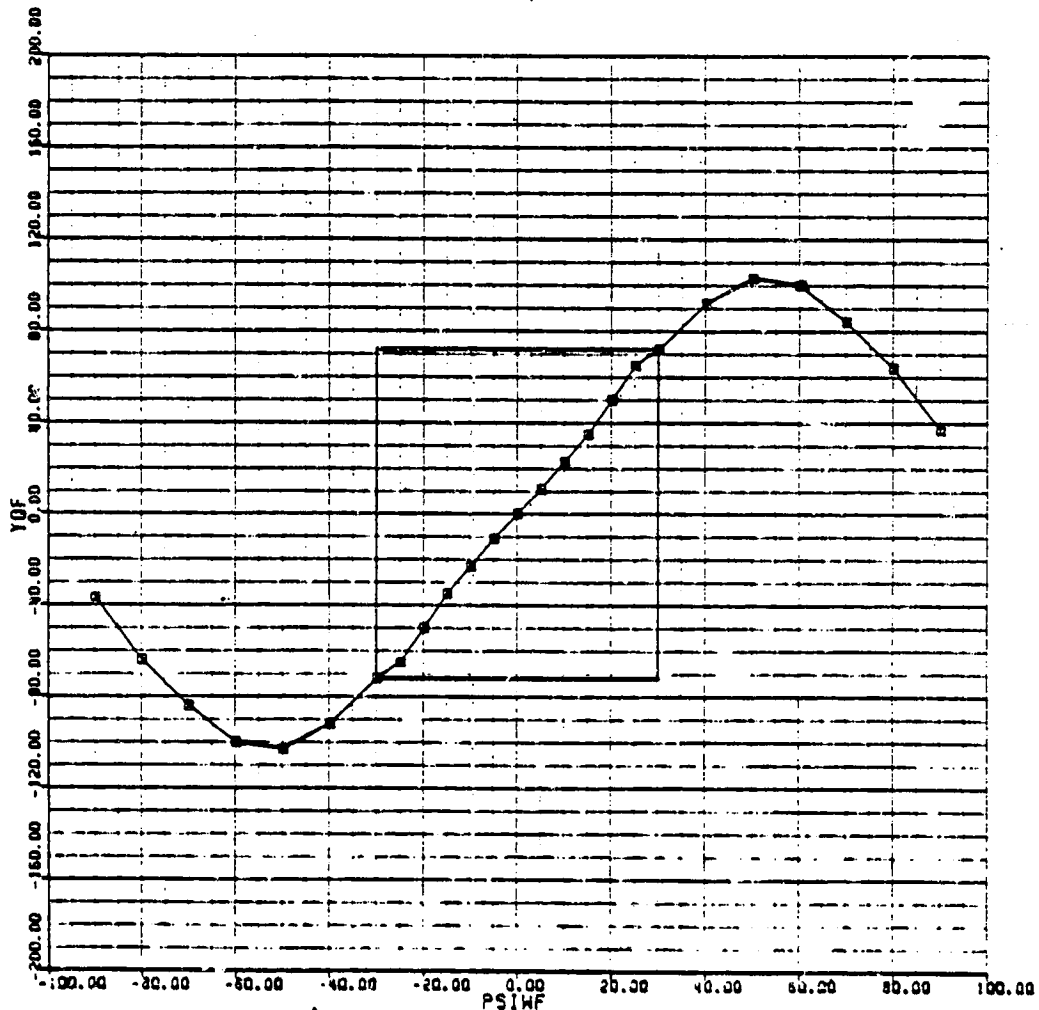


FIGURE 2.5.5(a)

FUSELAGE SIDEFORCE COEFFICIENT DUE TO SIDESLIP (cont'd)

BLACKHAWK - NASA STUDY 23-SEP-80

YQFMP (2/2)

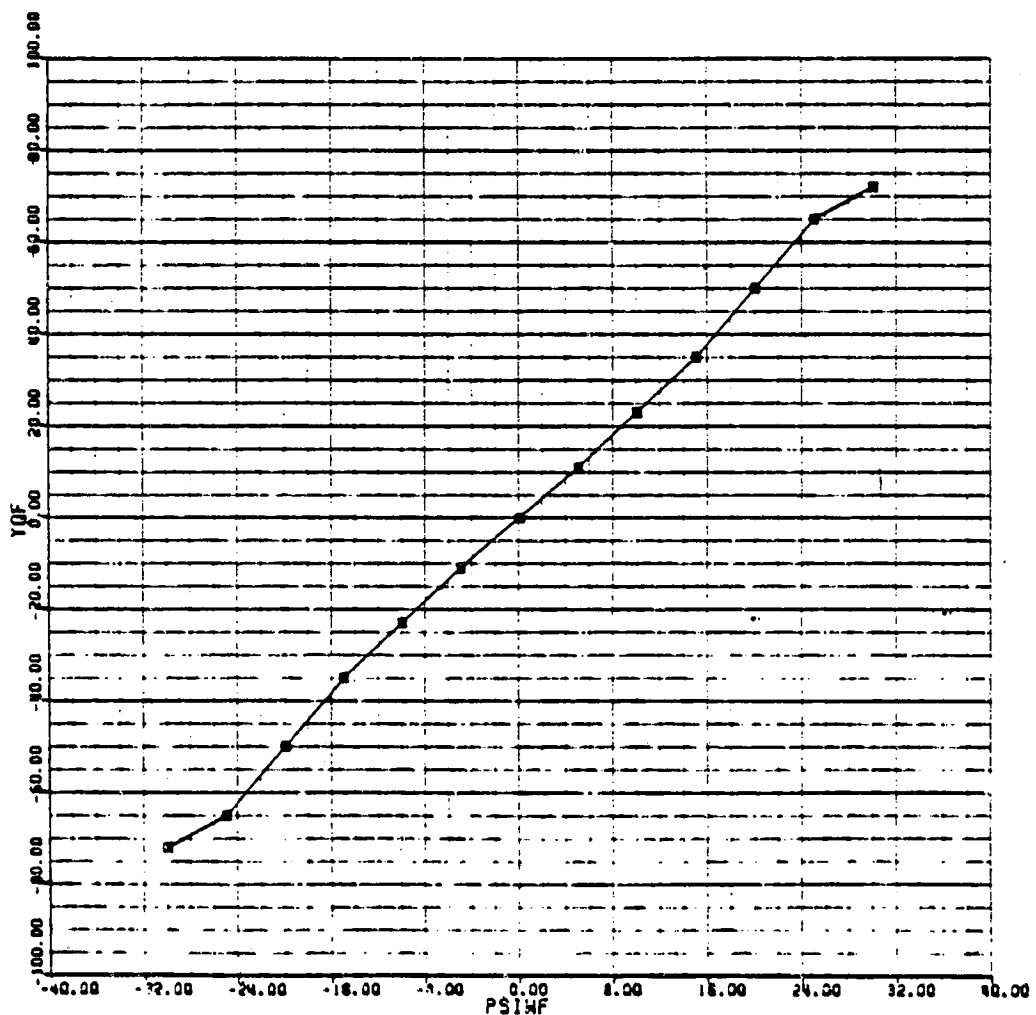


FIGURE 2.5.5(b)

FUSELAGE LIFT COEFFICIENT DUE TO ANGLE OF ATTACK

BLACKHAWK - NASA STUDY 23-SEP-80

LQFMP (1/2)

MAP NAME: LQFMP
MAP TYPE: UVAVUR
INPUT VARIABLE(S): ALFWF
OUTPUT VARIABLE: LQF

PRIMARY MAP: -30.00 LOWER LIMIT
30.00 UPPER LIMIT
5.00 DELTA

SECONDARY MAP: -90.00 LOWER LIMIT
90.00 UPPER LIMIT
10.00 DELTA

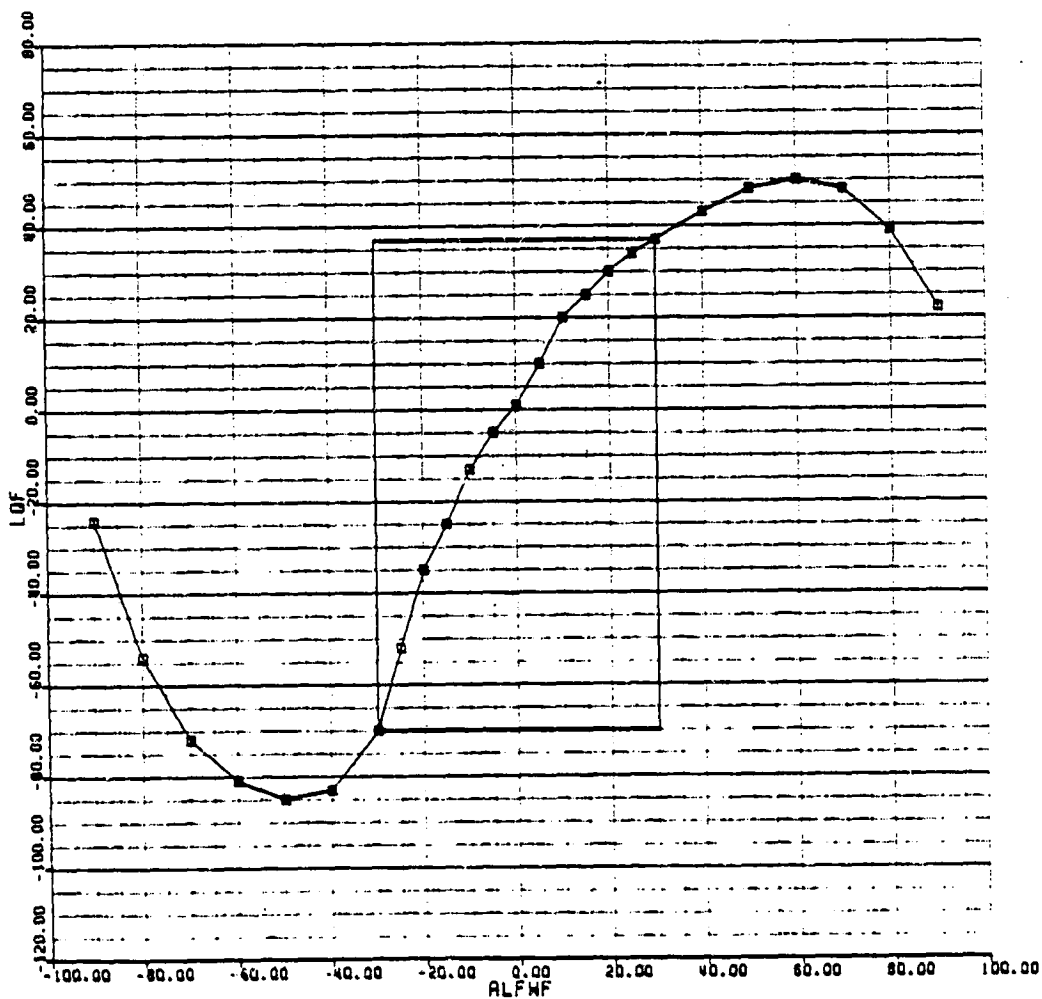


FIGURE 2.5.6(a)

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FUSELAGE LIFT COEFFICIENT DUE TO ANGLE OF ATTACK (Cont'd)

BLACKHAWK - NASA STUDY 23-SEP-80

LQFMP (2/2)

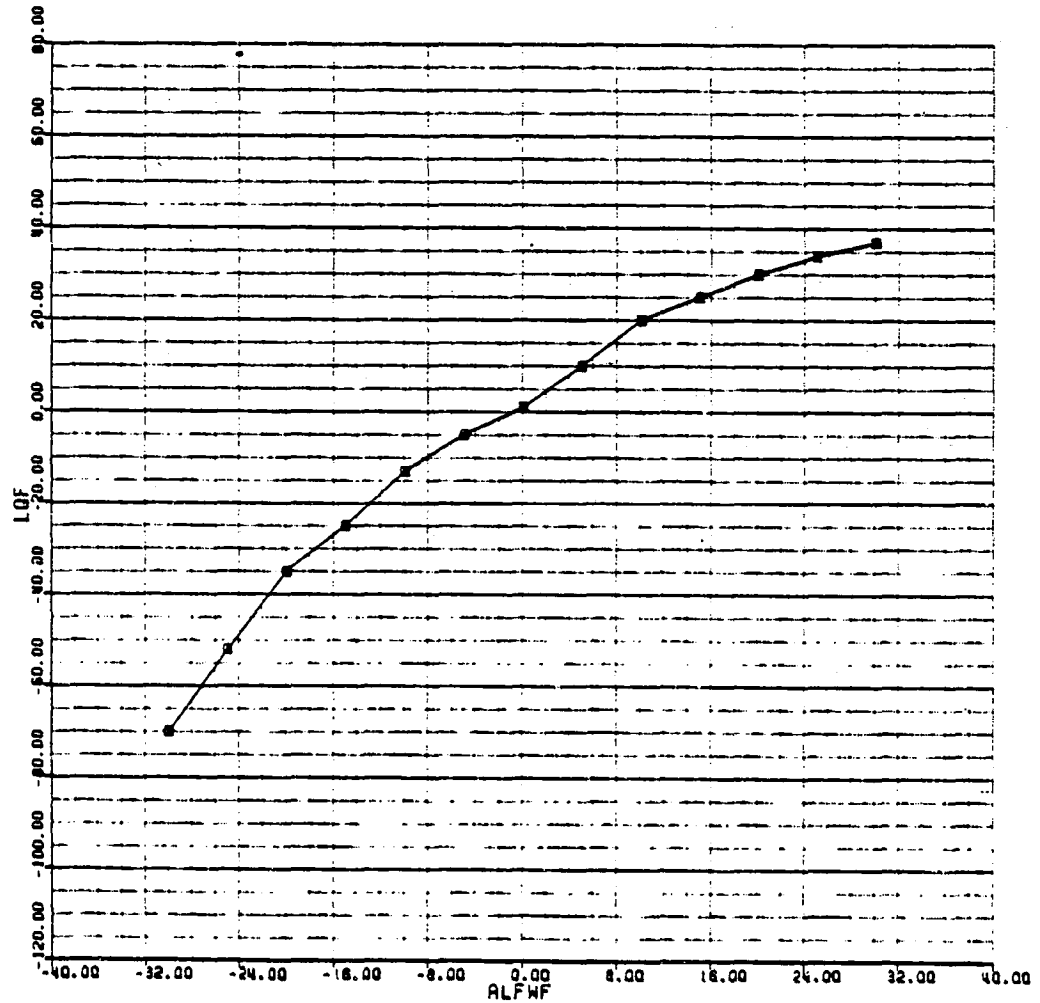


FIGURE 2.5.6(b)

SER 70452

INCREMENTAL FUSELAGE LIFT COEFFICIENT DUE TO SIDESLIP

MAP NAME: OLQFMP
 MAP TYPE: UVA
 INPUT VARIABLE(S): PSINF
 OUTPUT VARIABLE: OLQF
 PRIMARY MAP:
 -30.00 LOWER LIMIT
 30.00 UPPER LIMIT
 5.00 CURVE

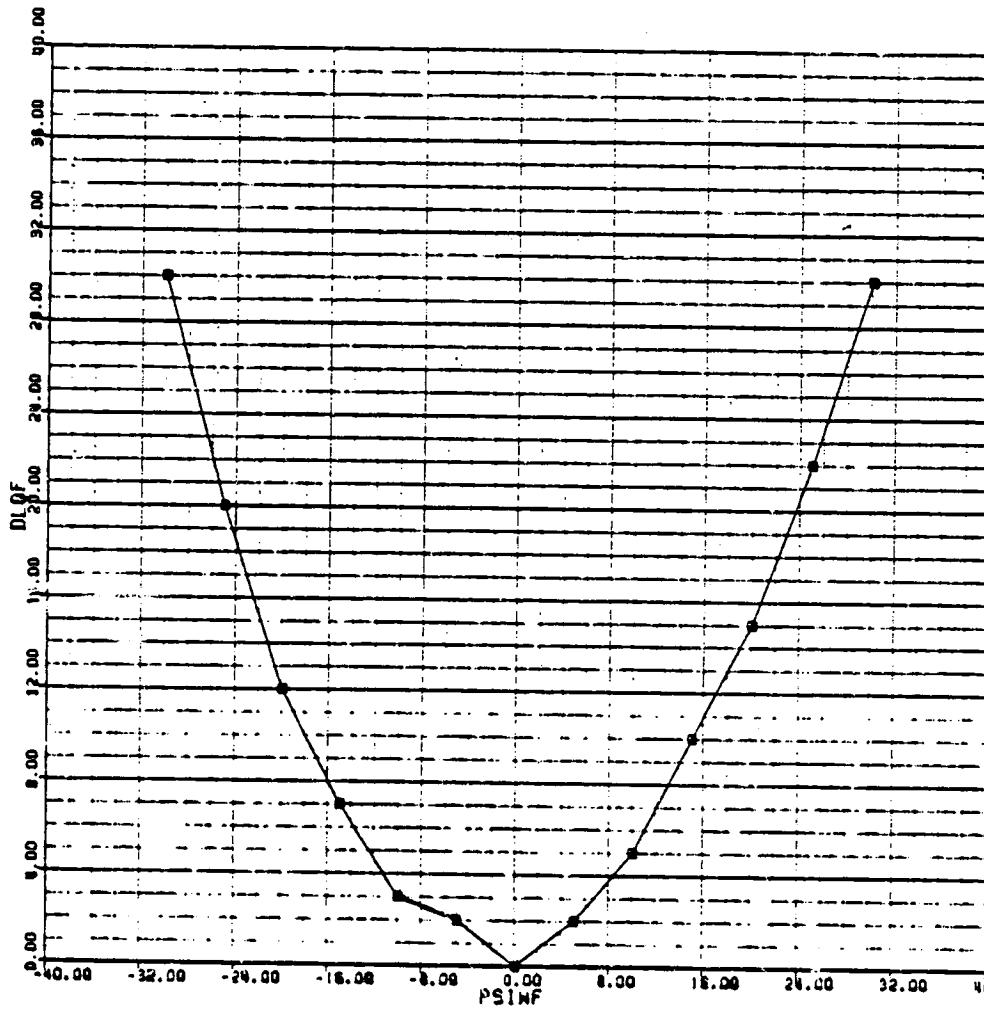


FIGURE 2.5.7

FUSELAGE ROLLING MOMENT COEFFICIENT DUE TO SIDESLIP

BLACKHAWK - NASA STUDY 23-SEP-80

RQFMP (1/2)

MAP NAME: RQFMP
 MAP TYPE: UVSUYS
 INPUT VARIABLE(S): PSINF
 OUTPUT VARIABLE: RQF

PRIMARY MAP:
 -30.00 UPPER LIMIT
 30.00 LOWER LIMIT
 5.00 CENTER

SECONDARY MAP:
 -90.00 UPPER LIMIT
 90.00 LOWER LIMIT
 10.00 CENTER

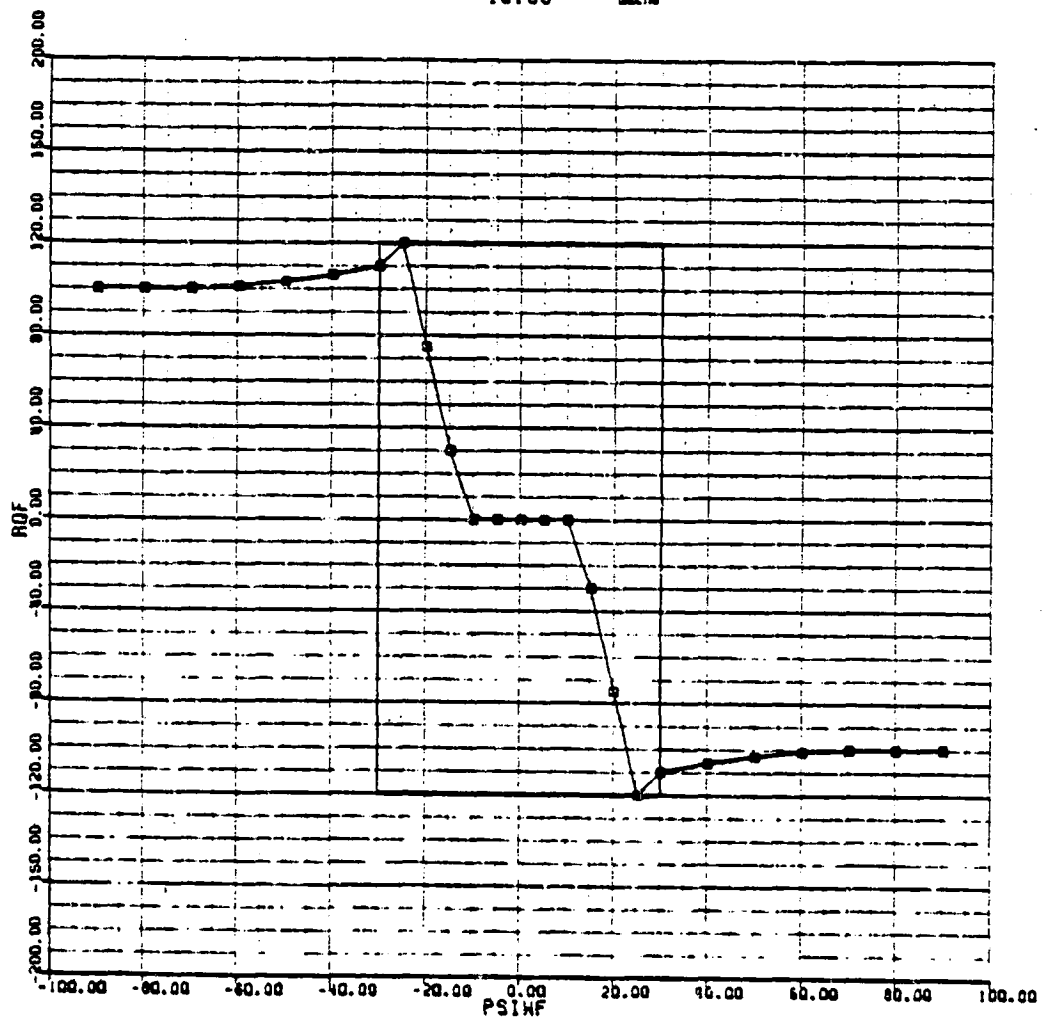


FIGURE 2.5.8(a)

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FUSELAGE ROLLING MOMENT COEFFICIENT DUE TO SIDESLIP (Cont'd)

BLACKHAWK - NASA STUDY 23-SEP-80

RQFMP (2/2)

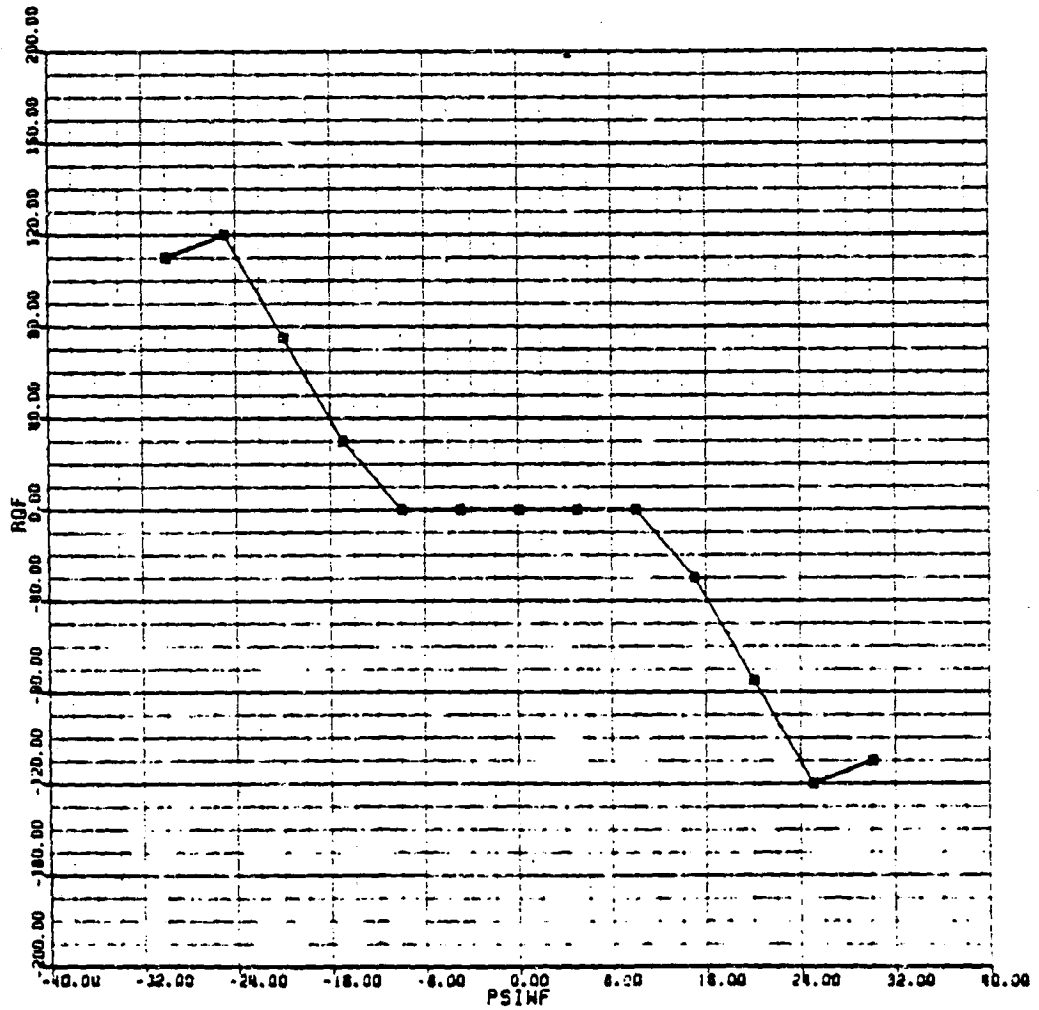


FIGURE 2.5.8(b)

FUSELAGE PITCHING MOMENT COEFFICIENT DUE TO ANGLE OF ATTACK

BLACKHAWK - NASA STUDY 23-SEP-60

MOFMP (1/2)

MAP NAME: MOFMP
 MAP TYPE: UVAUVA
 INPUT VARIABLE(S): ALFWF
 OUTPUT VARIABLE: MOF

PRIMARY MAP: -30.00 LOWER LIMIT
 30.00 UPPER LIMIT
 5.00 DELTA

SECONDARY MAP: -30.00 LOWER LIMIT
 90.00 UPPER LIMIT
 10.00 DELTA

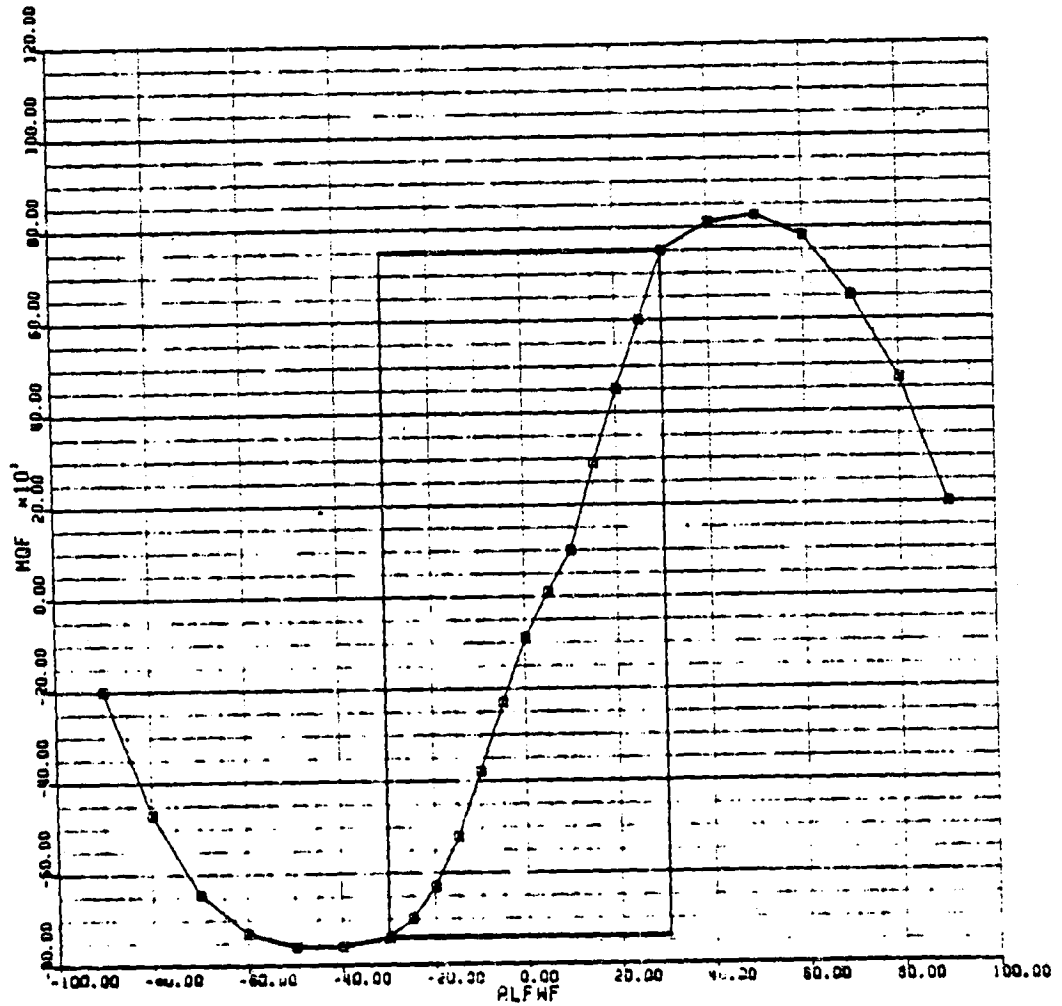


FIGURE 2.5.9(a)

FUSELAGE PITCHING MOMENT COEFFICIENT DUE TO ANGLE OF ATTACK (Cont'd)

BLACKHAWK - NASA STUDY 23-SEP-80

MQFNP (2/2)

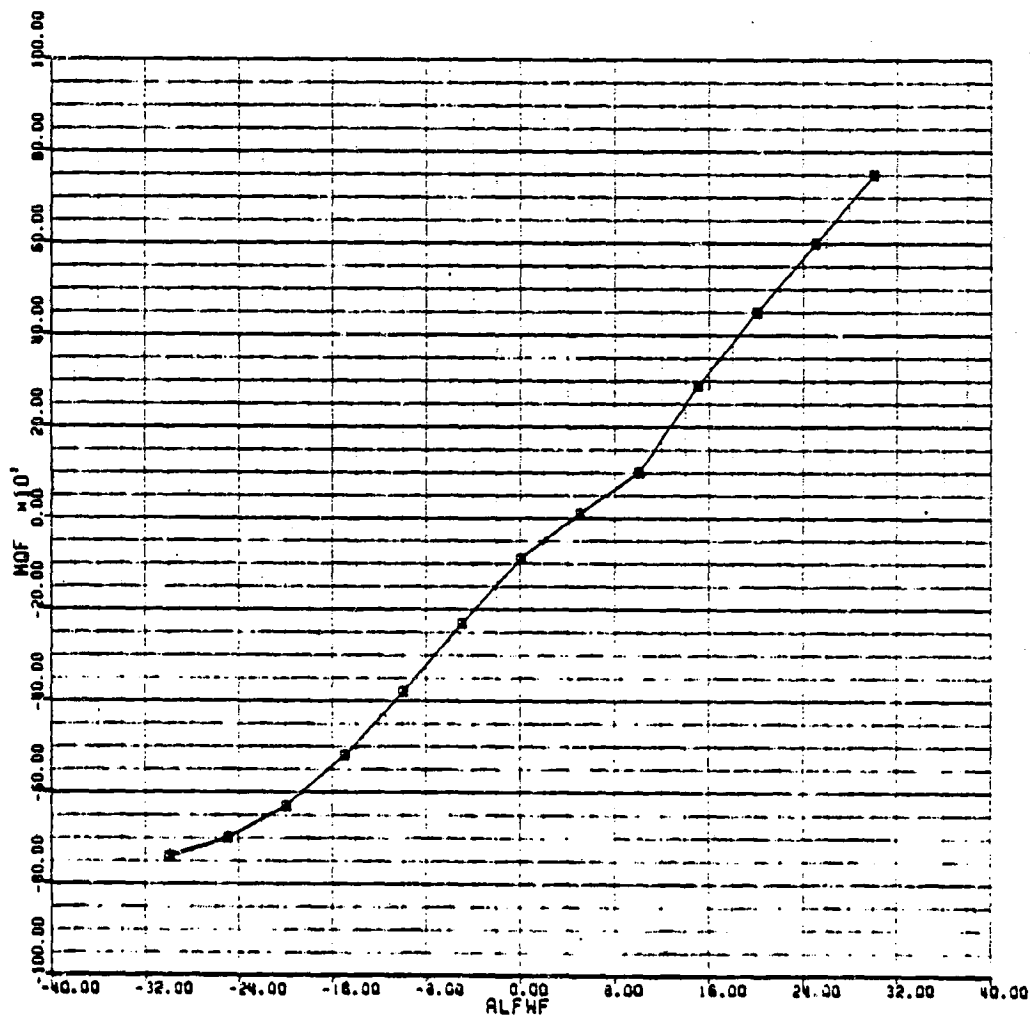


FIGURE 2.5.9(b)

INCREMENTAL FUSELAGE PITCHING MOMENT DUE TO SIDESLIP

MAP NAME: DMQFMP
 MAP TYPE: UVA
 INPUT VARIABLE(S): PSABWF
 OUTPUT VARIABLE: DMQF
 PRIMARY MAP:
 0.00 LOWER LIMIT
 30.00 UPPER LIMIT
 5.00 DELTA

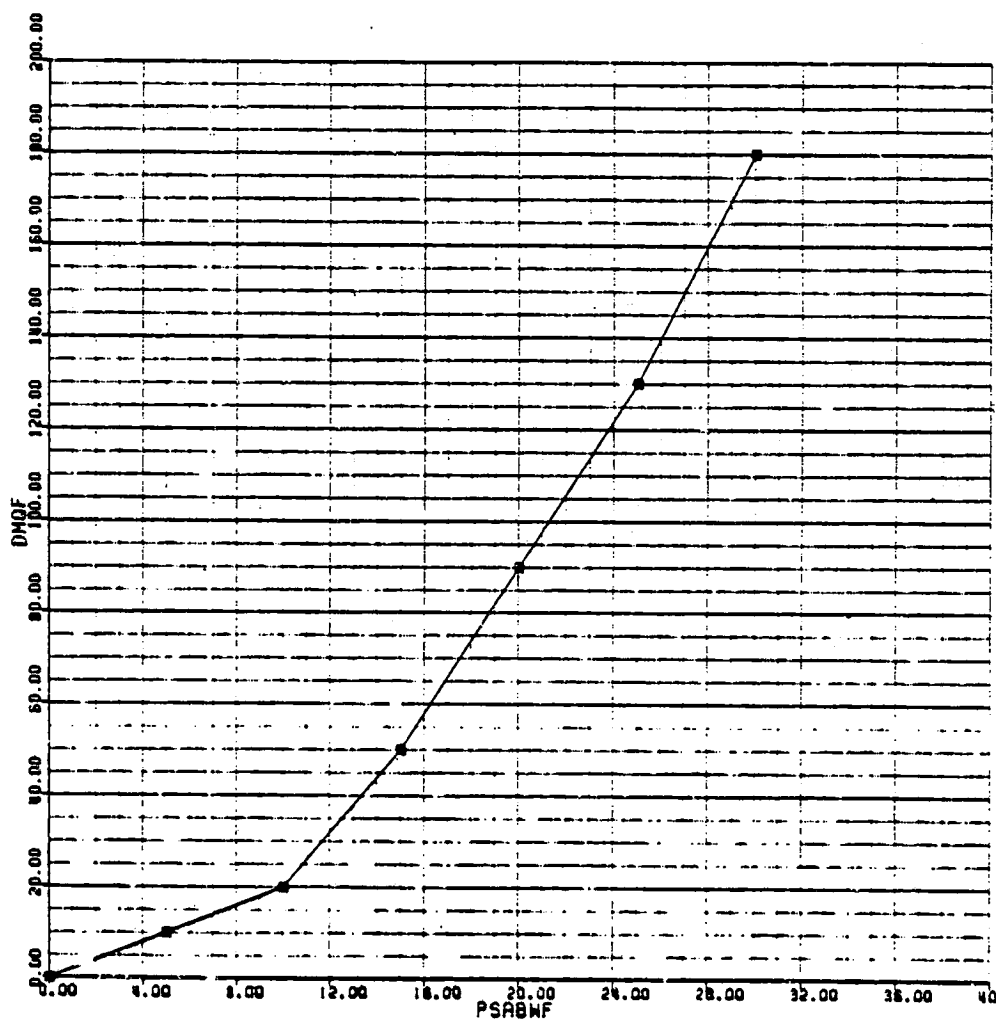


FIGURE 2.5.10

FUSELAGE YAWING MOMENT COEFFICIENT DUE TO SIDESLIP

BLACKHAWK - NASA STUDY 23-SEP-80

NQFMP (1/2)

MAP NAME: NQFMP
 MAP TYPE: UVAUVA
 INPUT VARIABLE(S): PSIMF NQF
 OUTPUT VARIABLE: NQF
 PRIMARY MAP: -30.00 LOWER LIMIT
 30.00 UPPER LIMIT
 5.00 CENTER
 SECONDARY MAP: -90.00 LOWER LIMIT
 90.00 UPPER LIMIT
 10.00 CENTER

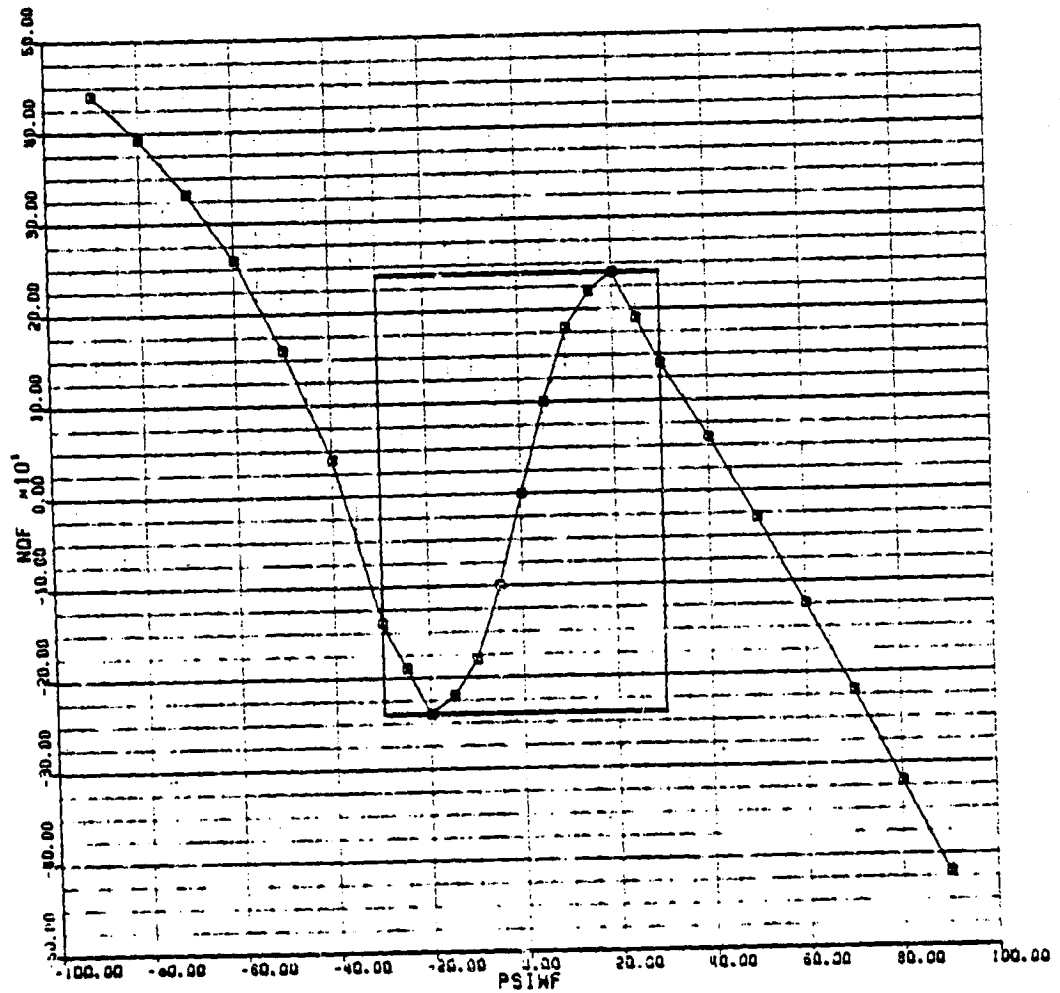


FIGURE 2.5.11(a)

FUSELAGE YAWING MOMENT COEFFICIENT DUE TO SIDESLIP (Cont'd)

BLACKHAWK - NASA STUDY 23-SEP-80

NOFMP (2/2)

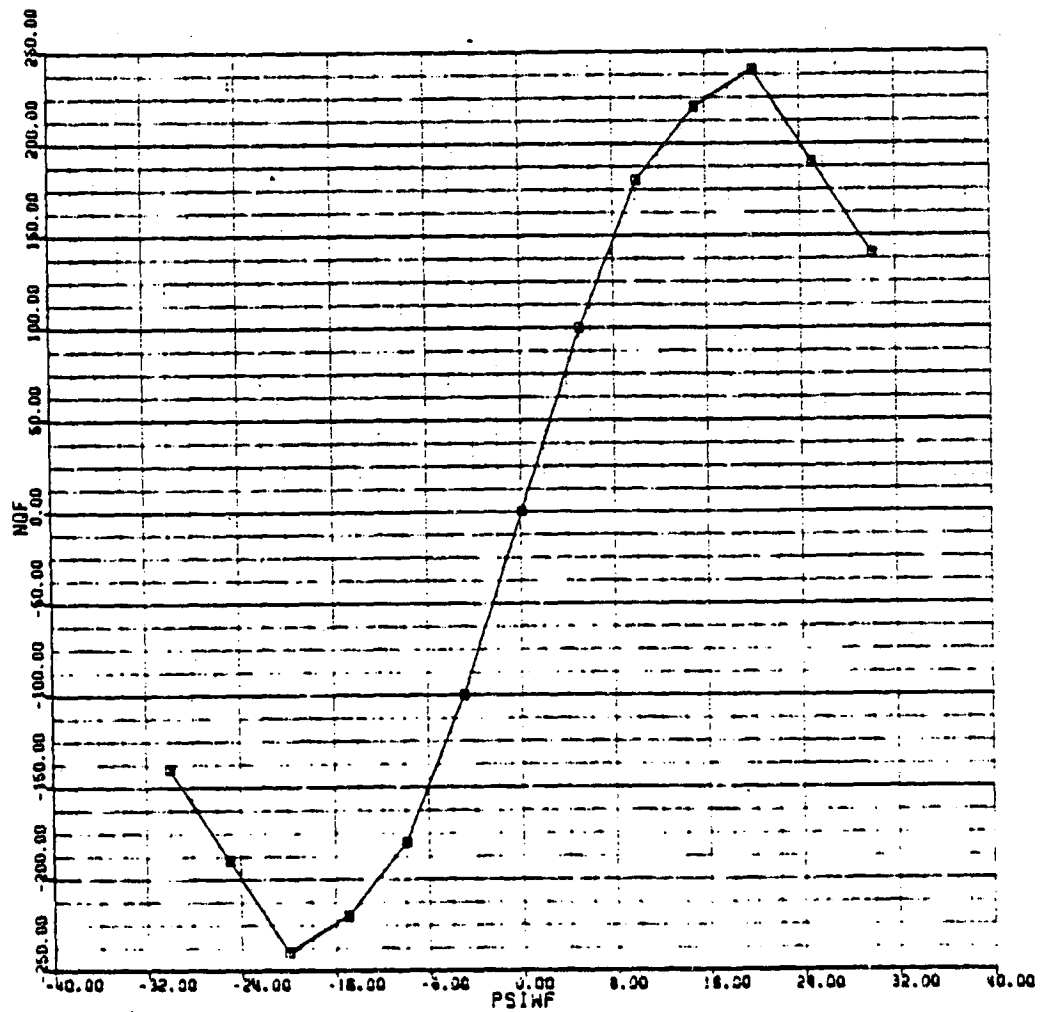


FIGURE 2.5.11(b)

5.3 EMPENNAGE MODULE

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5.3 Empennage Module

5.3.1 Module Description

This module calculates the aerodynamic forces on the horizontal and vertical tail surfaces resulting from the local airflow. The velocities are derived at, and the aerodynamic forces assumed to act at, the panel center of pressure. Aerodynamic forces are developed in the local flow wind axes system and subsequently transferred to the body axes system at the fuselage CG as defined on Figure 3.1.1. The overall module equation flow is shown in block diagram form on Figure 3.1.2.

The tail can experience aerodynamic interference effects from many sources. Components of flow from the main rotor and fuselage are defined at present in this module. However, the equations are formulated to allow easy insertion of other components. Three components of rotor wash are developed as a function of rotor wake skew angle and blade longitudinal flapping. Tail dynamic pressure blockage and downwash from the forebody are developed as a function of angle of attack and sideslip. Small downwash/sidewash angles are assumed and the velocity delayed to account for the time taken by the airflow to reach the tail.

The total velocity components for the tail are made up of contributions from the basic body axes translational and angular velocities, gust effects, rotor wash, fuselage downwash and sidewash. Dynamic pressure loss is introduced by factoring the components of the free stream flow. The actual total dynamic pressure at the tail is calculated from the resultant velocity vector. This allows a more representative definition of dynamic pressure at low speeds where downwash from the rotor predominates the flow at the tail. The lift and drag forces at the tail are obtained from isolated tail data and are a function of tail total angle of attack. In the case of the vertical tail, angle of attack has the same connotation as sideslip. The lift and drag forces in local wind axes are resolved into body axes at the tail. (Moments from the tail about its own axis are not accounted for.) Finally, the component forces at the empennage are transferred to the CG together with the corresponding moments.

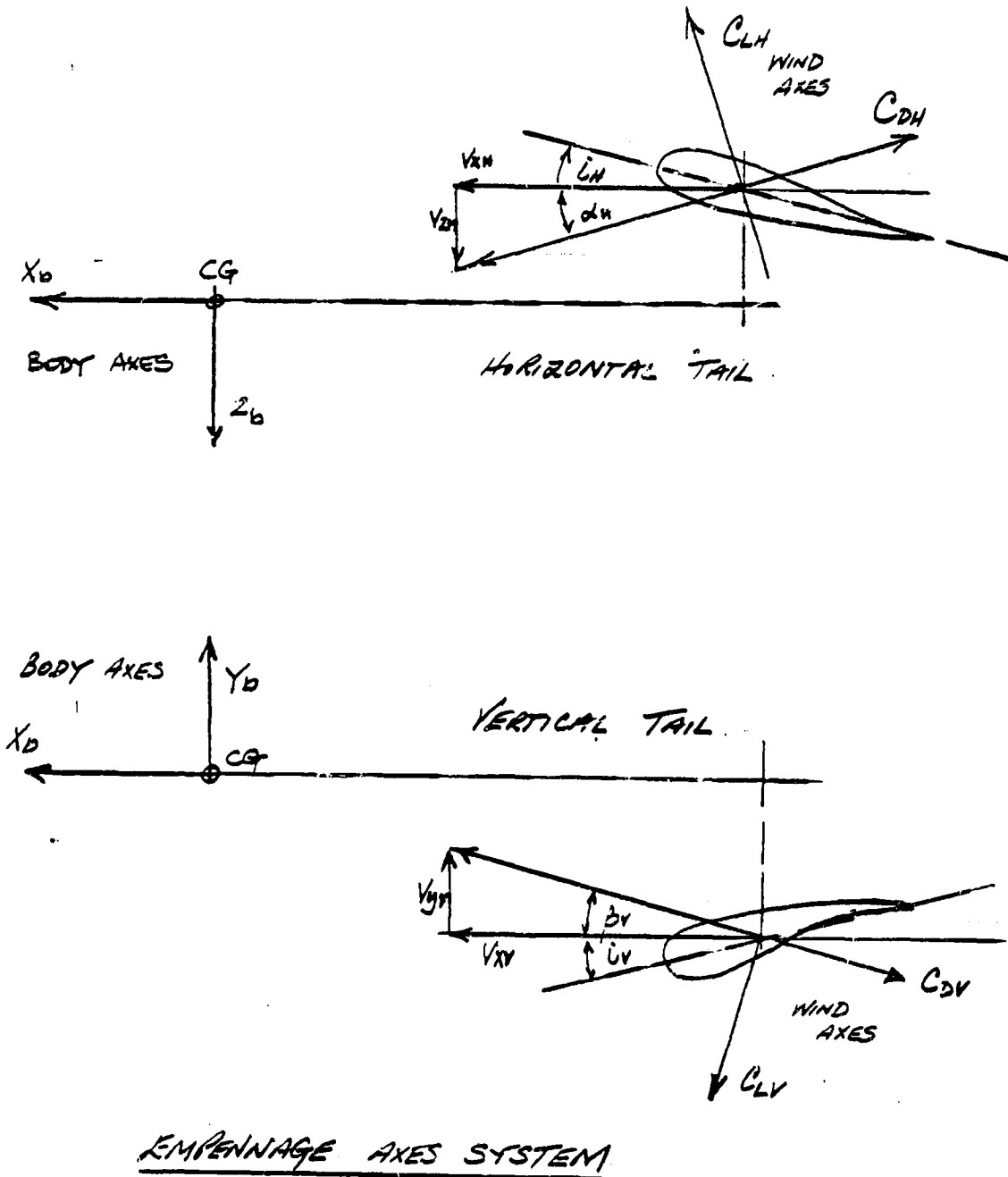
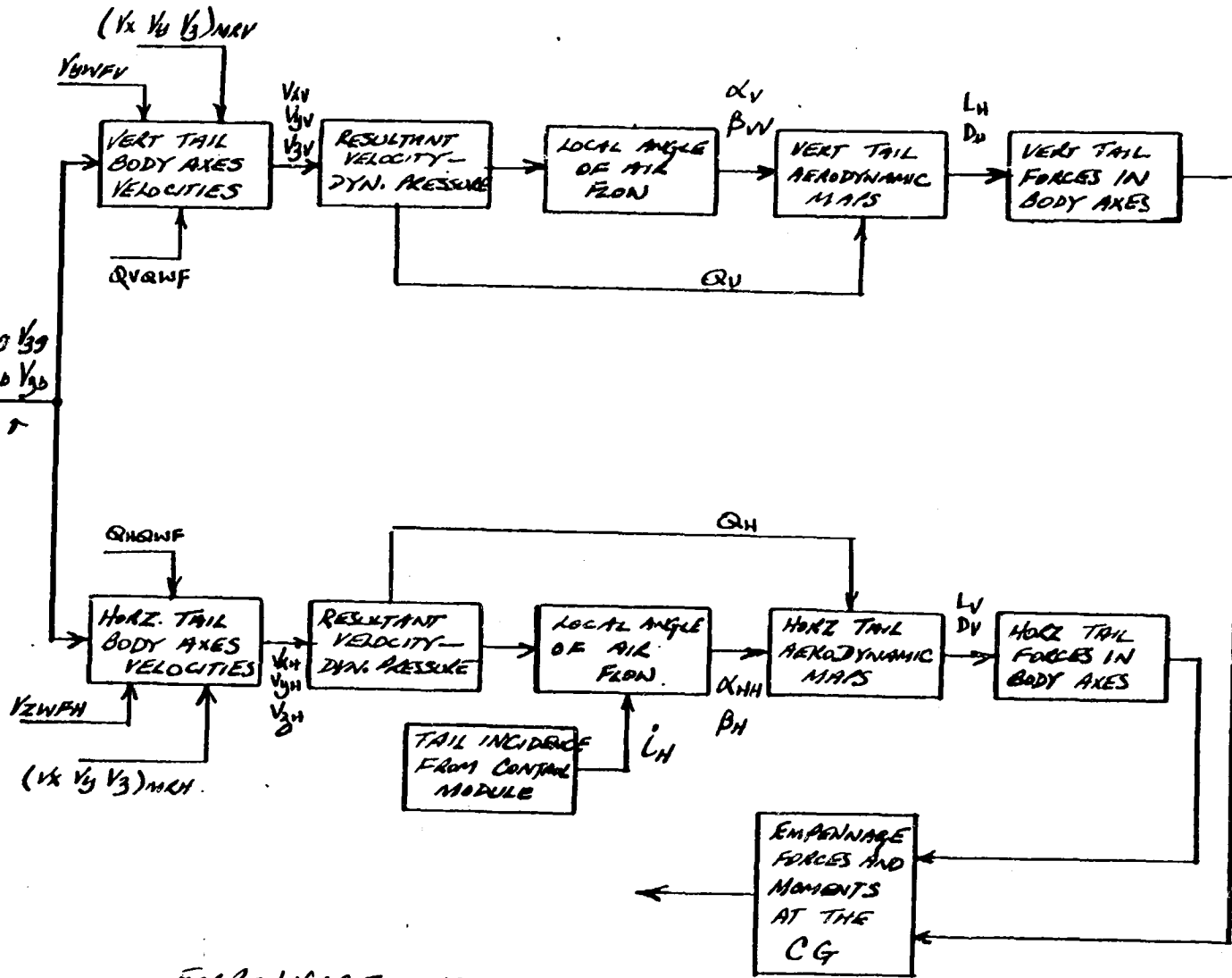


Figure 3.1.1



$V_{xV} V_{yV} V_{zV}$
 $V_{xH} V_{yH} V_{zH}$
 p q r

Figure 3.1.2

5.3-4
 PAGE

EMPAENNAGE EQUATION FLOW

5.3.2 EMPENNAGE MODULE EQUATIONS

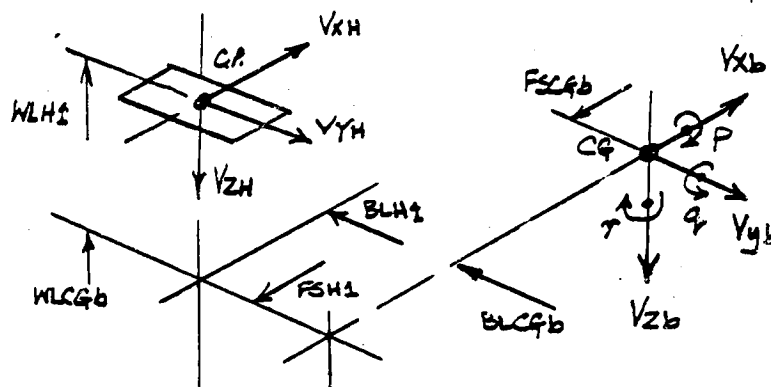
HORIZONTAL TAIL PANEL

GOMETRY

$$F_{HT1} = \frac{(F_{SH1} - F_{SCGb})}{12}$$

$$W_{HT1} = \frac{(W_{LH1} - W_{LCGb})}{12}$$

$$B_{HT1} = \frac{(B_{LH1} - B_{LCGb})}{12}$$



INTERFERENCE VELOCITIES

- FROM THE MAIN ROTOR -

$$E_{X_{H1}} = f(\gamma_{PMR}, \alpha_{IPMR}) \quad , \quad \text{MAP-EXH1MP} \\ \text{TABLE 3.5.1} \\ \text{FIGURE 3.5.1}$$

$$V_{X_{MRH1}} = D_{NSMR} \cdot S_{T,R_T} \cdot E_{X_{H1}}$$

$$E_{K_{Y_{H1}}} = f(\gamma_{PMR}, \alpha_{IPMR}) \quad , \quad \text{NO INPUT FOR BLACK HANK AT THIS TIME}$$

$$V_{Y_{MRH1}} = D_{NSMR} \cdot S_{T,R_T} \cdot E_{K_{Y_{H1}}}$$

$$E_{K_{Z_{H1}}} = f(\gamma_{PMR}, \alpha_{IPMR}) \quad , \quad \text{MAP-EZH1MP} \\ \text{TABLE 3.5.2} \\ \text{FIGURE 3.5.2}$$

$$V_{Z_{MRH1}} = -D_{NSMR} \cdot S_{T,R_T} \cdot E_{K_{Z_{H1}}}$$

- FROM THE FUSELAGE

DYNAMIC PRESSURE RATIO

$$Q_{H1}Q_{WF} = f(\alpha_{WF})$$

, MAP-QH1MP

TABLE 3.5.3

FIGURE 3.5.3

$$K_{QH1} = (Q_{H1}Q_{WF})^{1/2}$$

DOWNWASH COMPONENT

$$E_{PSH1} = f(\alpha_{WF})$$

, MAP-EPH1MP

TABLE 3.5.4

FIGURE 3.5.4

$$V_{ZWFH1} = - \frac{V_{XB}}{57.3} \cdot E_{PSH1} \cdot K_{QH1}$$

$$E_{PSH1} \text{ DELAYED BY } \left(\frac{F_{SH1} - F_{SCGb}}{12 \cdot V_{XB}} \right) \text{ SECS}$$

TOTAL INTERFERENCE VELOCITIES

$$V_{XIH1} = V_{XMRH1}$$

$$V_{YIH1} = V_{YMRH1}$$

$$V_{ZIH1} = V_{ZMRH1} + V_{ZWFH1}$$

LOCAL VELOCITY AT THE HORIZONTAL TAIL PANEL

$$V_{XH1} = (V_{xb} + V_{xgH1}) K_{QH1} - q_{NHT1} + \tau_{BHT1} + V_{xIH1}$$

$$V_{YH1} = (V_{yb} + V_{ygH1}) K_{QH1} + \phi_{WHT1} - \tau_{FHT1} + V_{yIH1}$$

$$V_{ZH1} = (V_{zb} + V_{zgH1}) K_{QH1} + q_{FHT1} - \phi_{BHT1} + V_{zIH1}$$

DYNAMIC PRESSURE AT THE HORIZONTAL TAIL

$$Q_{H1} = \frac{1}{2} \rho (V_{XH1}^2 + V_{YH1}^2 + V_{ZH1}^2)$$

LOCAL ANGLE OF THE FLOW AT THE HORIZONTAL TAIL

$$\sin \alpha_{H1} = \frac{V_{ZH1}}{(V_{XH1}^2 + V_{ZH1}^2)^{1/2}}$$

$$\cos \alpha_{H1} = \frac{V_{XH1}}{(V_{XH1}^2 + V_{ZH1}^2)^{1/2}}$$

$$\alpha_{H1} = \tan^{-1} \left\{ \frac{V_{ZH1}}{|V_{XH1}|} \right\}$$

$$\alpha_{HH1} = \alpha_{H1} + \alpha_{H1}$$

$$\beta_{H1} = \tan^{-1} \left\{ \frac{V_{YH1}}{(V_{XH1}^2 + V_{ZH1}^2)^{1/2}} \right\}$$

$\left. \begin{matrix} \sin \beta_{H2} \\ \cos \beta_{H2} \end{matrix} \right\} \text{ DERIVED FROM SIN COS } (\beta_{H2}) \text{ ROUTINE}$

HORIZONTAL TAIL LIFT AND DRAG COEFFICIENTS

$C_{LH2} = f(\alpha_{HH2})$, MAP - CLH2MP, TABLE 3.5.5 FIGURE 3.5.5

$C_{DH2} = f(|\alpha_{HH2}|)$, MAP - CDH2MP, TABLE 3.5.6, FIGURE 3.5.6

BODY AXES FORCES AND MOMENTS ABOUT THE CG

$$\begin{bmatrix} X_{H2} \\ Y_{H2} \\ Z_{H2} \end{bmatrix} = \begin{bmatrix} (\cos \alpha_{H2} \cos \beta_{H2}) & (\cos \alpha_{H2} \sin \beta_{H2}) & (-\sin \alpha_{H2}) \\ (\sin \beta_{H2}) & (-\cos \beta_{H2}) & 0 \\ (\sin \alpha_{H2} \cos \beta_{H2}) & (\sin \alpha_{H2} \sin \beta_{H2}) & (\cos \alpha_{H2}) \end{bmatrix} \begin{bmatrix} -Q_{H2} \cdot S_{AH2} \cdot C_{DH2} \\ 0 \\ -Q_{H2} \cdot S_{AH2} \cdot C_{LH2} \end{bmatrix}$$

$L_{H2} = Y_{H2} (W_{HT2}) - Z_{H2} (B_{HT2})$

$M_{H2} = -X_{H2} (W_{HT2}) + Z_{H2} (F_{HT2})$

$N_{H2} = -Y_{H2} (F_{HT2}) + X_{H2} (B_{HT2})$

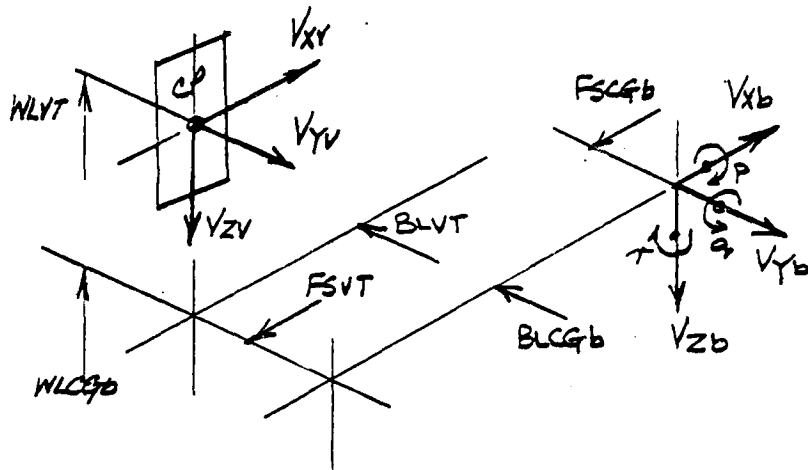
VERTICAL TAIL PANEL

GEOMETRY

$$F_{YT1} = \frac{(F_{SVT1} - F_{SCGb})}{12}$$

$$W_{YT1} = \frac{(W_{LVT1} - W_{LCGb})}{12}$$

$$B_{YT1} = \frac{(B_{LVT1} - B_{LCGb})}{12}$$



INTERFERENCE VELOCITIES

- FROM THE MAIN ROTOR

$$E_{XXV1} = f(\gamma_{PMR}, \alpha_{1FMR})$$

$$V_{XMRV1} = D_{NSMR} S_{T R_T} E_{XXV1}$$

$$E_{KYY1} = f(\gamma_{PMR}, \alpha_{1FMR})$$

$$V_{YMRV1} = D_{NSMR} S_{T R_T} E_{KYY1}$$

$$E_{KZZ1} = f(\gamma_{PMR}, \alpha_{1FMR})$$

$$V_{ZMRV1} = D_{NSMR} S_{T R_T} E_{KZZ1}$$

, SAME AS HORIZONTAL TAIL FOR BLACK HAWK

, MAP-EXHIMP
TABLE 3.5.1
FIGURE 3.5.1

, NO INPUT FOR BLACK HAWK AT THIS TIME

, MAP-EZHIMP
TABLE 3.5.2
FIGURE 3.5.2

- FROM THE FUSELAGE

DYNAMIC PRESSURE RATIO

$$Q_{VI}Q_{WF} = f(\psi_{WF})$$

$$K_{QV1} = (Q_{VI}Q_{WF})^{1/2}$$

, MAP - QV1MP.
TABLE 3.5.7
FIGURE 3.5.7

SIDENASH COMPONENT

$$S_{IGV1} = f(\psi_{WF})$$

$$V_{YWFV1} = - \frac{V_{XB}}{57.3} \cdot S_{IGV1} \cdot K_{QV1}$$

, MAP - SGV1MP
TABLE 3.5.8
FIGURE 3.5.8

$$S_{IGV1} \text{ DELAYED BY } \left(\frac{F_{SV1} - F_{SCG6}}{12 \cdot V_{XB}} \right) \text{ SECS}$$

TOTAL INTERFERENCE VELOCITIES

$$V_{XIV1} = V_{XMRV1}$$

$$V_{YIV1} = V_{YMRV1} + V_{YWFV1}$$

$$V_{ZIV1} = V_{ZMRV1}$$

LOCAL VELOCITIES AT THE VERTICAL TAIL PANEL

$$V_{XV2} = (V_{xb} + V_{xgv2}) K_{QV2} - q (W_{VT2}) + T (B_{VT2}) + V_{XIV2}$$

$$V_{YV2} = (V_{yb} + V_{ygv2}) K_{QV2} - T (F_{VT2}) + p (W_{VT2}) + V_{YIV2}$$

$$V_{ZV2} = (V_{zb} + V_{zgv2}) K_{QV2} + q (F_{VT2}) - p (B_{VT2}) + V_{ZIV2}$$

DYNAMIC PRESSURE AT THE VERTICAL TAIL

$$Q_{V2} = \frac{1}{2} \rho (V_{XV2}^2 + V_{YV2}^2 + V_{ZV2}^2)$$

LOCAL ANGLES OF THE FLOW AT THE VERTICAL TAIL

$$\sin \alpha_{V2} = \frac{V_{ZV2}}{(V_{XV2}^2 + V_{ZV2}^2)^{1/2}}$$

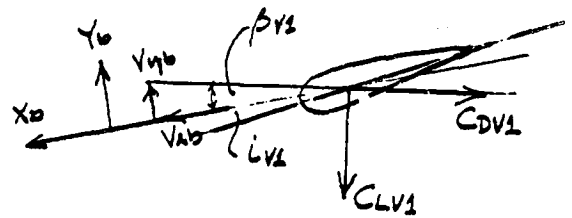
$$\cos \alpha_{V2} = \frac{V_{XV2}}{(V_{XV2}^2 + V_{ZV2}^2)^{1/2}}$$

$$\alpha_{V2} = \tan^{-1} \left\{ \frac{V_{ZV2}}{|V_{XV2}|} \right\}$$

$$\beta_{V2} = \tan^{-1} \left\{ \frac{V_{YV2}}{(V_{XV2}^2 + V_{ZV2}^2)^{1/2}} \right\}$$

$$\beta_{TV2} = \alpha_{V2} + \beta_{V2}$$

$\left. \begin{matrix} \sin \beta_{V2} \\ \cos \beta_{V2} \end{matrix} \right\}$ DERIVED FROM SINCOS (β_{V2}) ROUTINE



VERTICAL TAIL LIFT AND DRAG COEFFICIENTS

$$C_{LV2} = f(\beta_{V2}) \quad , \quad \text{MAP-CLV2MP, TABLE 3.5.9, FIGURE 3.5.9}$$

$$C_{DV2} = f^*(\beta_{V2}) \quad , \quad \text{MAP-CDV2MP, TABLE 3.5.10, FIGURE 3.5.10}$$

BODY AXES FORCES AND MOMENTS ABOUT THE CG

$$\begin{bmatrix} X_{V2} \\ Y_{V2} \\ Z_{V2} \end{bmatrix} = \begin{bmatrix} (\cos \alpha_{V2} \cos \beta_{V2}) & (\cos \alpha_{V2} \sin \beta_{V2}) & (-\sin \alpha_{V2}) \\ (\sin \beta_{V2}) & (-\cos \beta_{V2}) & 0 \\ (\sin \alpha_{V2} \cos \beta_{V2}) & (\sin \alpha_{V2} \sin \beta_{V2}) & (\cos \alpha_{V2}) \end{bmatrix} \begin{bmatrix} -C_{DV2} \cdot S_{AV2} \cdot Q_{V2} \\ C_{LV2} \cdot S_{AV2} \cdot Q_{V2} \\ 0 \end{bmatrix}$$

$$L_{V2} = Y_{V2}(W_{VT2}) - Z_{V2}(B_{VT2})$$

$$M_{V2} = Z_{V2}(F_{VT2}) - X_{V2}(W_{VT2})$$

$$N_{V2} = -Y_{V2}(F_{VT2}) + X_{V2}(B_{VT2})$$

$$X_T = \sum_{i=1}^n (X_{Hi} + X_{Vi})$$

$$Y_T = \sum_{i=1}^n (Y_{Hi} + Y_{Vi})$$

$$Z_T = \sum_{i=1}^n (Z_{Hi} + Z_{Vi})$$

$$L_T = \sum_{i=1}^n (L_{Hi} + L_{Vi})$$

$$M_T = \sum_{i=1}^n (M_{Hi} + M_{Vi})$$

$$N_T = \sum_{i=1}^n (N_{Hi} + N_{Vi})$$

5.3.3 EMPENNAGE MODULE INPUT/OUTPUT DATA TRANSFER

INPUT TRANSFER	
PARAMETER	ORIGIN MODULE
FSCGB	MAIN ROTOR
WLCGB	
BLCGB	
DWSHMR	
CHIPMR	
ALFMR	
OMGTMR	
RMR	
ALFWF	FUSELAGE
BETAWF	
IHT	FLIGHT CONTROL
IVT	
VXGH1	GUST
VYGH1	
VZGH1	
VXGV1	
VYGV1	
VZGV1	
VXB	MOTION
YYB	
YZB	
P	
Q	
R	

OUTPUT TRANSFER	
PARAMETER	DESTINATION MODULE
XT	MOTION
YT	
ZT	
LT	
MT	
NT	

5.3.4

NOTATION FOR THE EMPENNAGE MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
FSH1	FSH1	INS	Fuselage station for horizontal tail C.P.
FSCGb	FSCGB	INS	Fuselage station for CG
F _{HT} 1	KH	FT	
WLH1	WLH1	INS	Waterline station for horizontal tail C.P.
WLCGb	WLCGB	INS	Waterline station for CG
W _{HT} 1	KH+1	FT	
B _{LH} 1	B _{LH} 1	INS	Buttline station for Horizontal tail C.P.
BLCGb	BLCGB	INS	Buttline station for CG
B _{HT}	KH+2	FT	
E _{KXH} 1	EKXH1	ND	Rotor wash factors
E _{KTH} 1	EKYH1	ND	
E _{KZH} 1	EKZH1	ND	
χ PMR	CH1PMR	DEG	Rotor Wake Skew angle
a ₁ FMR	AALFMR	DEG	Rotor longitudinal flapping
D _{WSHMR}	DWSHMR	ND	Rotor uniform downwash
Ω T	OMGTMR	RADS/SEC	Trim rotor speed
R _T	RMR	FT	Rotor radius
V _{XMRH} 1	VXMRH1	FT/SEC	Rotor interference velocity at the Horizontal tail.
V _{YMRH} 1	VYMRH1	FT/SEC	
V _{ZMRH} 1	VXMRH1	FT/SEC	
Q _{H1QWF}	QH1QWF	ND	Horizontal tail dynamic pressure ratio
α WF	ALFWF	DEG	Fuselage angle of attack.
ψ WF	PSIWF	DEG	Fuselage heading
β WF	BETAWF	DEG	Fuselage sideslip.

5.3.4 (Continued)

NOTATION FOR THE EMPENNAGE MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
K_{QH1}	KQH1	ND	
E_{PSH1}	EPSH1	DEG	Downwash angle
V_{Xb}	VXB	FT/SEC	Fuselage X axis velocity
V_{ZWFH1}	VZWFH1	FT/SEC	Fuse/Tail downwash velocity
V_{XIH1}	VXIH1	FT/SEC	Horizontal tail total interference velocity
V_{YIH1}	VYIH1	FT/SEC	
V_{ZIH1}	VZIH1	FT/SEC	
P	P	RADS/SEC	Body axes angular rates.
q	q	RADS/SEC	
r	r	RADS/SEC	
V_{XH1}	VXH1	FT/SEC	Total velocity at the horizontal tail.
V_{YH1}	VYH1	FT/SEC	
V_{ZH1}	VZH1	FT/SEC	
V_{XGH1}	VXGH1	FT/SEC	Gust velocity at the horizontal tail.
V_{YGH1}	VYGH1	FT/SEC	
V_{ZGH1}	VZGH1	FT/SEC	
ρ	RHO	SLUG/FT ³	Air density.
Q_{H1}	QH1	LB/FT ²	Dynamic pressure at the horizontal tail
α_{H1}	ALFH1	DEG	Body axis angle of attack
α_{HH1}	ALFHH1	DEG	Total tail angle of attack
β_{H1}	BETH1	DEG	Sideslip angle

5.3.4 (Continued)

NOTATION FOR THE EMPENNAGE MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
V_{XMRV1}	VXMRV1	FT/SEC	Rotor interference velocities
V_{YMRV1}	VYMRV1	FT/SEC	
V_{ZMRV1}	VZMRV1	FT/SEC	
S_{IGV1}	SIGV1	DEG	Fuselage sidewash angle
V_{YWFV1}	VYWFV1	FT/SEC	Fuselage sidewash velocity
V_{XV1}	VXV1	FT/SEC	Total velocity at the vertical tail.
V_{YV1}	VYV1	FT/SEC	
V_{ZV1}	VZV1	FT/SEC	
V_{XGV1}	VXGV1	FT/SEC	Gust velocity at the vertical tail.
V_{YGV1}	VYGV1	FT/SEC	
V_{ZGV1}	VZGV1	FT/SEC	
V_{XIV1}	VXIV1	FT/SEC	Inteference velocity at the vertical tail.
V_{YIV1}	VYIV1	FT/SEC	
V_{ZIV1}	VZIV1	FT/SEC	
Q_{V1}	QV1	LB/FT ²	Dynamic pressure at the vertical tail.
α_{V1}	AFVF1	DEG	Angle of attack
β_{V1}	BETV1	DEG	Sideslip
β_{VV1}	BETVV1	DEG	Total sideslip angle
-	SNAFV1	-	SIN(ALFV1)
-	CSAFV1	-	COS(ALFV1)
-	SNBTV1	-	SIN(BETV1)
-	CSBTV1	-	COS(BETV1)

5.3.4 (Continued)

NOTATION FOR THE EMPENNAGE MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
-	SNAFHL	-	SIN(ALFH1)
-	CSAFH1	-	COS(ALFH1)
-	SNBTH1	-	SIN(BETH1)
-	SCRTH1	-	COS(BETH1)
C_{LH1}	CLH1	ND	Coef of lift
C_{DH1}	CDH1	ND	Coef of drag.
X_{H1}	XH1	LB	Horizontal tail forces and moments
Y_{H1}	YH1	LB	
Z_{H1}	ZH1	LB	
L_{H1}	LH1	FT LB	
M_{H1}	MH1	FT LB	
N_{H1}	NH1	FT LB	
FSV1	FSVT1	INS	Fuselage Station for the vertical tail C.P.
FVT1	KV	FT	Waterline Station for the vertical tail C.P.
WLV1	WLVT1	INS	Buttline Station for the vertical tail C.P.
BLV1	BLVT1	INS	Buttline Station for the vertical tail C.P.
B_{VT1}	KV+2	FT.	Buttline Station for the vertical tail C.P.
E_{KXV1}	EKXV1	-	Rotor wash factors
E_{KYV1}	EKYV1	-	
E_{KZV1}	EKZV1	-	

5.3.4 (Continued)

NOTATION FOR THE EMPENNAGE MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
Q_{V1QWF}	QV1QWF	-	Dynamic pressure ratio, ratio
K_{QV1}	KWV1	-	SQRT (dynamic pressure ratio)
C_{LV1}	CLV1	ND	Coef of lift
C_{DV1}	CDV1	ND	Coef of drag
X_{V1}	XV1	LB	Vertical tail forces and moments
Y_{V1}	YV1	LB	
Z_{V1}	ZV1	LB	
L_{V1}	LV1	FT LB	
M_{V1}	MV1	FT LB	
N_{V1}	NV1	FT LB	
X_T	-	LB	Total Empennage forces and moments
Y_T	-	LB	
Z_T	-	LB	
L_T	-	FT LB	
M_T	-	FT LB	
N_T	-	FT LB	

5.3.5 BLACK HAWK EMPENNAGE INPUT DATA

INPUT CONSTANTS

$$FSH1 = 700.1 \text{ ms}$$

$$NLH1 = 244.0 \text{ ms}$$

$$BLH1 = 0$$

$$SAH1 = 45.0$$

$$IH1 = \text{CALCULATED IN THE CONTROL SYSTEM}$$

$$FSV1 = 695.0$$

$$NLV1 = 273.0$$

$$BLV1 = 0.0$$

$$SAV1 = 32.3$$

$$IV1 = 0.0$$

BLACK HAWK

TABLE 3.5.1

MAIN ROTOR INPLANE WASH AT THE HORIZONTAL TAIL

```
EXHIMP::BIV## ;MAP ARGUMENT:LOOK UP ROUTINE
EXP CHIPMR##,AAIFMR## ;INPUT VARIABLE #1, INPUT VARIABLE #2
EKXHI## ;OUTPUT VARIABLE
EXHILO ;LOW ANGLE MAP NAME
EXP 0.0,100.0,10.0,13 ;LOW LIM,UPPER LIM,DELTA,#ENTRYS(OCT)-CHIPMR
EXP -6.0,6.0,6.0 ;LOW LIM,UPPER LIM,DELTA-AAIFMR

; LOW ANGLE MAP CHIPMR 0 TO 100 (DEL=10) AAIFMR -6,0.6
; AAIFMR=-6
```

EXHILO:EXP	0.0,	-0.2,	0.05,	0.3,	0.54
EXP	0.0,	1.04,	1.3,	1.55,	0.8
	0.0				
	; AAIFMR=0				
EXP	-0.4,	-0.6,	-0.2,	0.12,	0.36
EXP	0.6,	0.83,	1.06,	1.3,	0.66
	0.0				
	; AAIFMR=6				
EXP	-0.56,	-0.8,	-0.74,	-0.32,	0.04
EXP	0.32,	0.6,	0.86,	1.12,	0.54
	0.0				

TABLE 3.5.2

MAIN ROTOR DOWNWASH AT THE HORIZONTAL TAIL

```
EZHIMP::BIV## ;MAP ARGUMENT:LOOK UP ROUTINE
EXP CHIPMR##,AAIFMR## ;INPUT VARIABLE #1, INPUT VARIABLE #2
EKZHI## ;OUTPUT VARIABLE
EZHILO ;LOW ANGLE MAP NAME
EXP 0.0,100.0,10.0,13 ;LOW LIM,UPPER LIM,DELTA,#ENTRYS(OCT)-CHIPMR
EXP -6.0,6.0,6.0 ;LOW LIM,UPPER LIM,DELTA-AAIFMR
```

	; LOW ANGLE MAP CHIPMR 0 TO 100 (DEL=10) AAIFMR -6,0.6				
	; AAIFMR=-6				
EZHILO:EXP	-0.13,	0.8,	1.8,	1.82,	1.86
EXP	1.88,	1.91,	1.94,	1.69,	1.42
	1.14				
	; AAIFMR=0				
EXP	0.4,	0.94,	1.84,	1.91,	1.98
EXP	2.04,	2.08,	2.14,	1.89,	1.62
	1.35				
	; AAIFMR=6				
EXP	0.78,	1.36,	1.91,	1.98,	2.06
EXP	2.14,	2.21,	2.28,	2.16,	1.96
	1.56				

TABLE 3.5.3

DYNAMIC PRESSURE LOSS AT THE HORIZONTAL TAIL

```

QHIMP::UVR##          ;MAP ARGUMENT:LOOK UP ROUTINE
ALFWF##              ;INPUT VARIABLE
QHIQWF##             ;OUTPUT VARIABLE
QHILO                 ;LOW ANGLE MAP NAME
EXP -38.8,38.8,5.8   ;LOWER LIMIT, UPPER LIMIT, DELTA

; LOW ANGLE MAP ALFHH1 -38 TO 38 DELTA=5
QHILO: EXP 1.8,      1.8,      8.95,    8.76,      8.76
EXP 8.76,          8.73,      8.76,    8.76,      8.82
EXP 8.91,          1.8,      1.8,      8.76,      8.82
    
```

TABLE 3.5.4

FUSELAGE DOWNWASH AT THE HORIZONTAL TAIL

```

EPHIMP::UVRUVR##     ;MAP ARGUMENT:LOOK UP ROUTINE
ALFWF##              ;INPUT VARIABLE
EPSHI##              ;OUTPUT VARIABLE
EPHILO                ;LOW ANGLE MAP NAME
EXP -38.8,38.8,5.8   ;LOWER LIMIT, UPPER LIMIT, DELTA-LOW ANGLE
EPHIHI                ;HIGH ANGLE MAP NAME
EXP -98.8,98.8,18.8 ;LOWER LIMIT, UPPER LIMIT, DELTA-HIGH ANGLE

; LOW ANGLE MAP ALFHH1 -38 TO 38 DELTA=5
EPHILO: EXP 1.8,      1.4,      1.1,      8.8,      8.55
EXP 8.5,      8.45,      8.4,      8.38,      8.33
EXP 8.19,     -8.12,     -8.4,      8.38,      8.33

; HIGH ANGLE MAP ALFHH1 -98 TO 98 DELTA=18
EPHIHI: EXP 8.8,      8.25,      8.7,      1.2,      1.6
EXP 1.9,      1.8,      1.1,      8.55,      8.45
EXP 8.38,     8.19,     -8.4,     -8.7,     -8.75
EXP -8.65,    -8.45,    -8.15,     8.8,      8.8
    
```

TABLE 3.5.5

HORIZONTAL TAIL LIFT COEFFICIENT DUE TO ANGLE OF ATTACK

```

; S=45 FT**2 ,ASPECT RATIO=4.6 ,.8814 AIRFOIL
CLHIMP::UVSUVS##     ;MAP ARGUMENT:LOOK UP ROUTINE
ALFHH1##              ;INPUT VARIABLE
CLHI##                ;OUTPUT VARIABLE
CLHILO                 ;LOW ANGLE MAP NAME
EXP 8.8,38.8,5.8     ;LOWER LIMIT, UPPER LIMIT, DELTA-LOW ANGLE
CLHIHI                 ;HIGH ANGLE MAP NAME
EXP 38.8,98.8,18.8  ;LOWER LIMIT, UPPER LIMIT, DELTA-HIGH ANGLE

; LOW ANGLE MAP ALFHH1 8 TO 38, DELTA=5 CL(ALF)=-CL(-ALF)
CLHILO: EXP 8.8,      8.356,      8.71,      1.83,      8.95
EXP 8.795,          8.745,          8.71,      1.83,      8.95

; HIGH ANGLE MAP ALFHH1 38 TO 98, DELTA=18 CL(ALF)=-CL(-ALF)
CLHIHI: EXP 8.745,     8.847,      8.847,      8.745,      8.558
EXP 8.294,          8.8,          8.847,      8.745,      8.558
    
```

TABLE 3.5.6

HORIZONTAL TAIL DRAG COEFFICIENT DUE TO ANGLE OF ATTACK

```

CDHIMP::UVRUVR## ;MAP ARGUMENT:LOOK UP ROUTINE
          AFAHHI## ;INPUT VARIABLE
          CDHI##   ;OUTPUT VARIABLE
          CDHILO   ;LOW ANGLE MAP NAME
EXP 0.0,30.0,5.0 ;LOWER LIMIT,UPPER LIMIT,DELTA
          CDHIHI   ;HIGH ANGLE MAP NAME
EXP 30.0,90.0,10.0 ;LOWER LIMIT,UPPER LIMIT,DELTA

          ; LOW ANGLE MAP ALFHHI 0 TO 30,DELTA=5 CD(ALF)=CD(-ALF)
CDHILO: EXP 0.01, 0.022, 0.04, 0.19, 0.36
          EXP 0.37, 0.43

          ; HIGH ANGLE MAP ALFHHI 30 TO 90,DELTA=10 CD(ALF)=CD(-ALF)
CDHIHI: EXP 0.43, 0.531, 0.702, 0.888, 1.05
          EXP 1.161, 1.2
    
```

TABLE 3.5.7

DYNAMIC PRESSURE LOSS AT THE VERTICAL TAIL

```

QVIMP:: UVR## ;MAP ARGUMENT:LOOK UP ROUTINE
          PSABWF## ;INPUT VARIABLE
          QP3QWF## ;OUTPUT VARIABLE
          QP3LO   ;LOW ANGLE MAP NAME
EXP 0.0,30.0,5.0,7 ;LOWER LIMIT, UPPER LIMIT, DELTA, NO OF ENTRYS(PSABWF)

          ; LOW ANGLE MAP PSI(ABS) 0 TO 30 DELTA=5
EXP 0.62, 0.64, 0.66, 0.72, 0.79
EXP 0.88, 1.00
    
```

TABLE 3.5.8

FUSELAGE SIDEWASH AT THE VERTICAL TAIL

```

SGVIMP::UVSUVS## ;MAP ARGUMENT:LOOK UP ROUTINE
          PSIW##   ;INPUT VARIABLE
          SIGP3##  ;OUTPUT VARIABLE
          SGP3LO   ;LOW ANGLE MAP NAME
EXP 0.0,30.0,5.0 ;LOWER LIMIT, UPPER LIMIT, DELTA-LOW ANGLE
          SGP3HI   ;HIGH ANGLE MAP NAME
EXP 0.0,90.0,30.0 ;LOWER LIMIT, UPPER LIMIT, DELTA-HIGH ANGLE

          ; LOW ANGLE MAP PSIW 0 TO 30 DELTA=5
EXP 0.0, -0.1, -0.6, 0.8, 1.4
EXP 0.6, 0.2

          ; HIGH ANGLE MAP PSIW 0 TO 90 DELTA=30
EXP 0.0, 0.2, 0.0, 0.0
    
```

TABLE 3.5.9

VERTICAL TAIL LIFT COEFFICIENT DUE TO SIDESLIP

```

; S=32.3 FT**2 ,ASPECT RATIO =1.92 ,S821 MOD AIRFOIL
CLVIMP::UVRUVR## ;MAP ARGUMENT:LOOK UP ROUTINE
      BETVV1## ;INPUT VARIABLE
      CLV1## ;OUTPUT VARIABLE
      CLV1L# ;LOW ANGLE MAP NAME
EXP -38.8,38.8,5.8 ;LOWER LIMIT,UPPER LIMIT,DELTA-LOW ANGLE
      CLV1HI ;HIGH ANGLE MAP NAME
EXP -98.8,98.8,18.8 ;LOWER LIMIT,UPPER LIMIT,DELTA-HIGH ANGLE

; LOW ANGLE MAP BETVV1 -38 TO 38,DELTA=5
CLV1L# :EXP -1.88, -1.88, -8.93, -8.73, -8.5
EXP -8.28, -8.86, 8.16, 8.38, 8.61
EXP 8.82, 8.89, 8.89

; HIGH ANGLE MAP BETVV1 -98 TO 98,DELTA=18
CLV1HI :EXP -8.8, -8.12, -8.28, -8.46, -8.66
EXP -8.88, -1.88, -8.93, -8.5, -8.86
EXP 8.38, 8.82, 8.89, 8.8, 8.63
EXP 8.48, 8.32, 8.17, 8.8

```

TABLE 3.5.10

VERTICAL TAIL DRAG COEFFICIENT DUE TO SIDESLIP

```

CDVIMP::UVRUVR## ;MAP ARGUMENT:LOOK UP ROUTINE
      BETVV1## ;INPUT VARIABLE
      CDV1## ;OUTPUT VARIABLE
      CDV1LO ;LOW ANGLE MAP NAME
EXP -38.8,38.8,5.8 ;LOWER LIMIT,UPPER LIMIT,DELTA-LOW ANGLE
      CDV1HI ;HIGH ANGLE MAP NAME
EXP -98.8,98.8,18.8 ;LOWER LIMIT,UPPER LIMIT,DELTA-HIGH ANGLE

; LOW ANGLE MAP BETVV1 -38 TO 38,DELTA=5
CDV1LO :EXP 8.36, 8.265, 8.174, 8.118, 8.866
EXP 8.833, 8.818, 8.821, 8.844, 8.892
EXP 8.162, 8.248, 8.355

; HIGH ANGLE MAP BETVV1 -98 TO 98,DELTA=5
CDV1HI :EXP 1.1, 1.825, 8.965, 8.875, 8.745
EXP 8.575, 8.36, 8.174, 8.866, 8.818
EXP 8.844, 8.162, 8.355, 8.58, 8.75
EXP 8.875, 8.965, 1.82, 1.88

```

MAIN ROTOR INPLANE WASH AT THE HORIZONTAL TAIL

MAP NAME: EXHIMP
 MAP TYPE: BIY
 INPUT VARIABLE(S): CHIPMA RAIFMA
 OUTPUT VARIABLE: EXXH1
 PRIMARY MAP:
 0.00 LOWER LIMIT -6.00
 100.00 UPPER LIMIT 6.00
 10.00 SLOPE 6.00

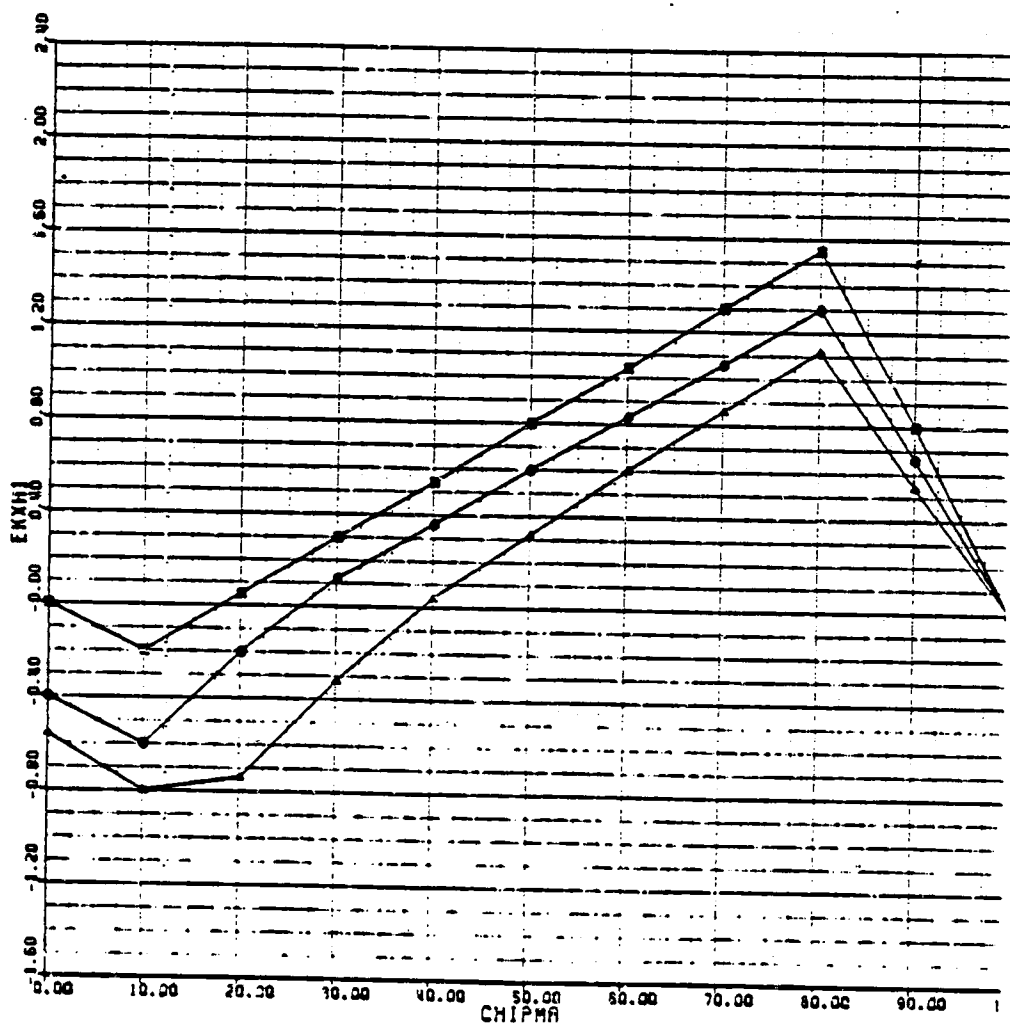
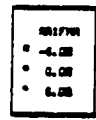


FIGURE 3.5.1

MAIN ROTOR DOWNWASH AT THE HORIZONTAL TAIL

MAP NAME: EZHIMP
 MAP TYPE: 81V
 INPUT VARIABLE(S): CHIPMA AA1FMA
 OUTPUT VARIABLE: EKZH1
 PRIMARY MAP:
 0.00 LOWER LIMIT -6.00
 100.00 UPPER LIMIT 6.00
 10.00 CURVE 6.00

AA1FMA
 • -6.00
 • 0.00
 • 6.00

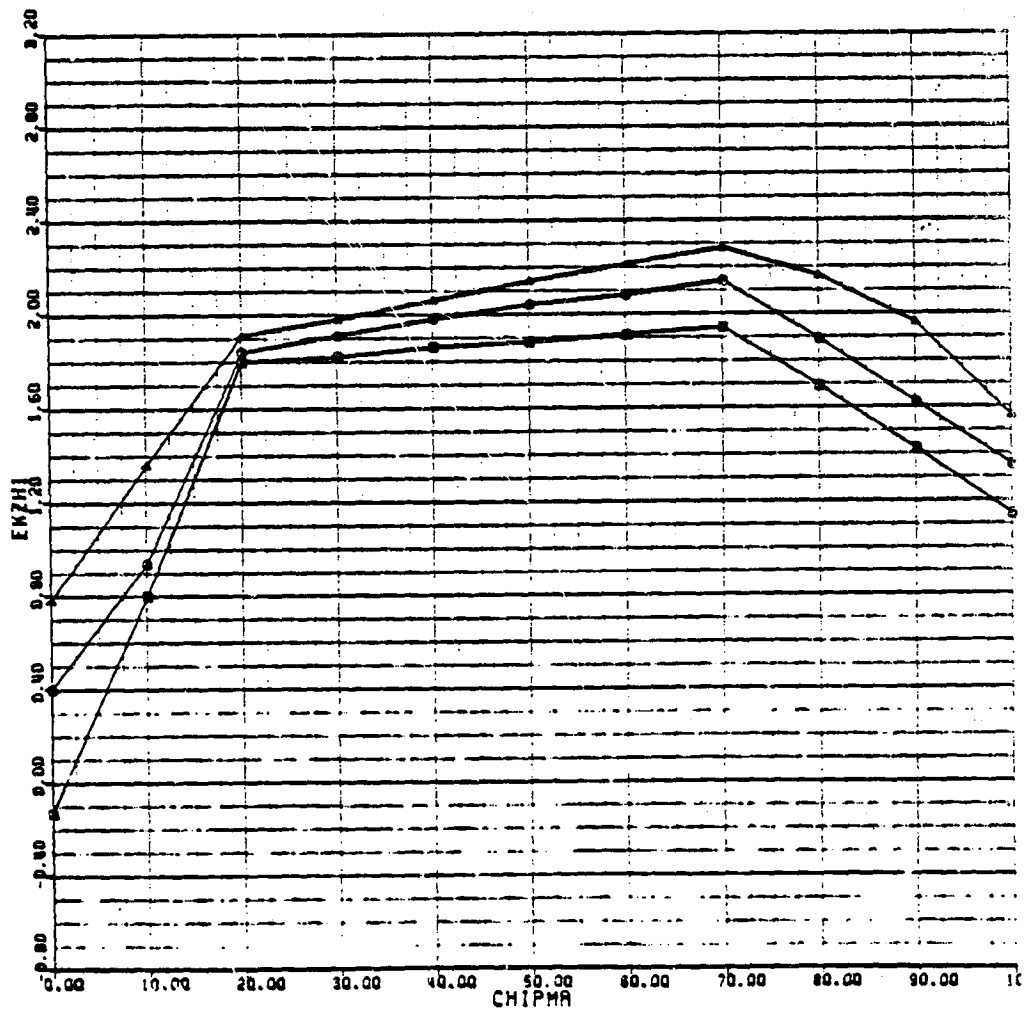


FIGURE 3.5.2

DYNAMIC PRESSURE LOSS AT THE HORIZONTAL TAIL

MAP NAME: QHIMP
 MAP TYPE: QVA
 INPUT VARIABLE(S): ALFWF
 OUTPUT VARIABLE: QH1QWF
 PRIMARY MAP:
 -30.00
 30.00
 5.00

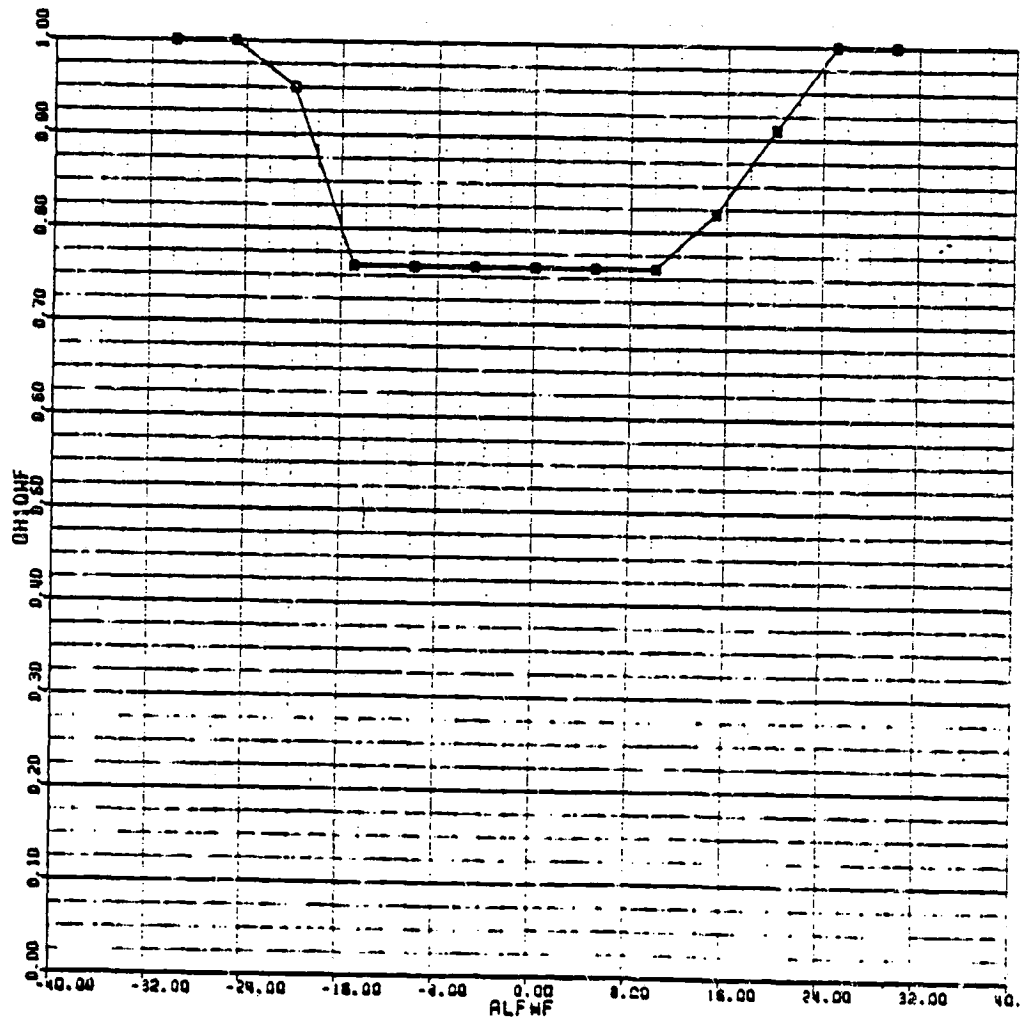


FIGURE 3.5.3

FUSELAGE DOWNWASH AT THE HORIZONTAL TAIL

BLACKHAWK - NASA STUDY 23-SEP-80

EPHIMP (1/2)

MAP NAME: EPHIMP
 MAP TYPE: UVAUVA
 INPUT VARIABLE(S): ALFWF
 OUTPUT VARIABLE: EPSH1

PRIMARY MAP:
 -30.00 LOWER LIMIT
 30.00 UPPER LIMIT
 5.00 DELTA

SECONDARY MAP:
 -90.00 LOWER LIMIT
 90.00 UPPER LIMIT
 10.00 DELTA

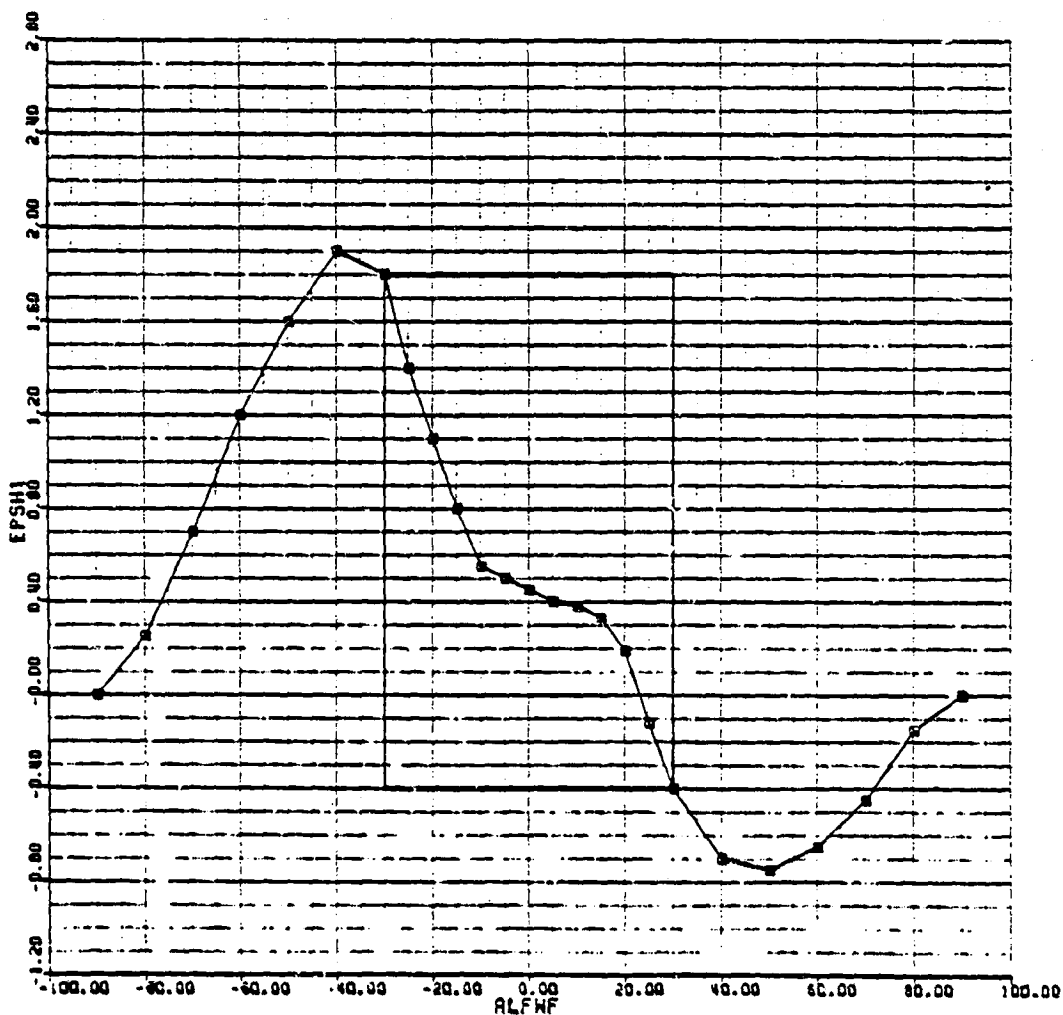


FIGURE 3.5.4(a)

FUSELAGE DOWNWASH AT THE HORIZONTAL TAIL (Cont'd)

BLACKHAWK - NASA STUDY 23-SEP-80

EPH1MP (2/2)

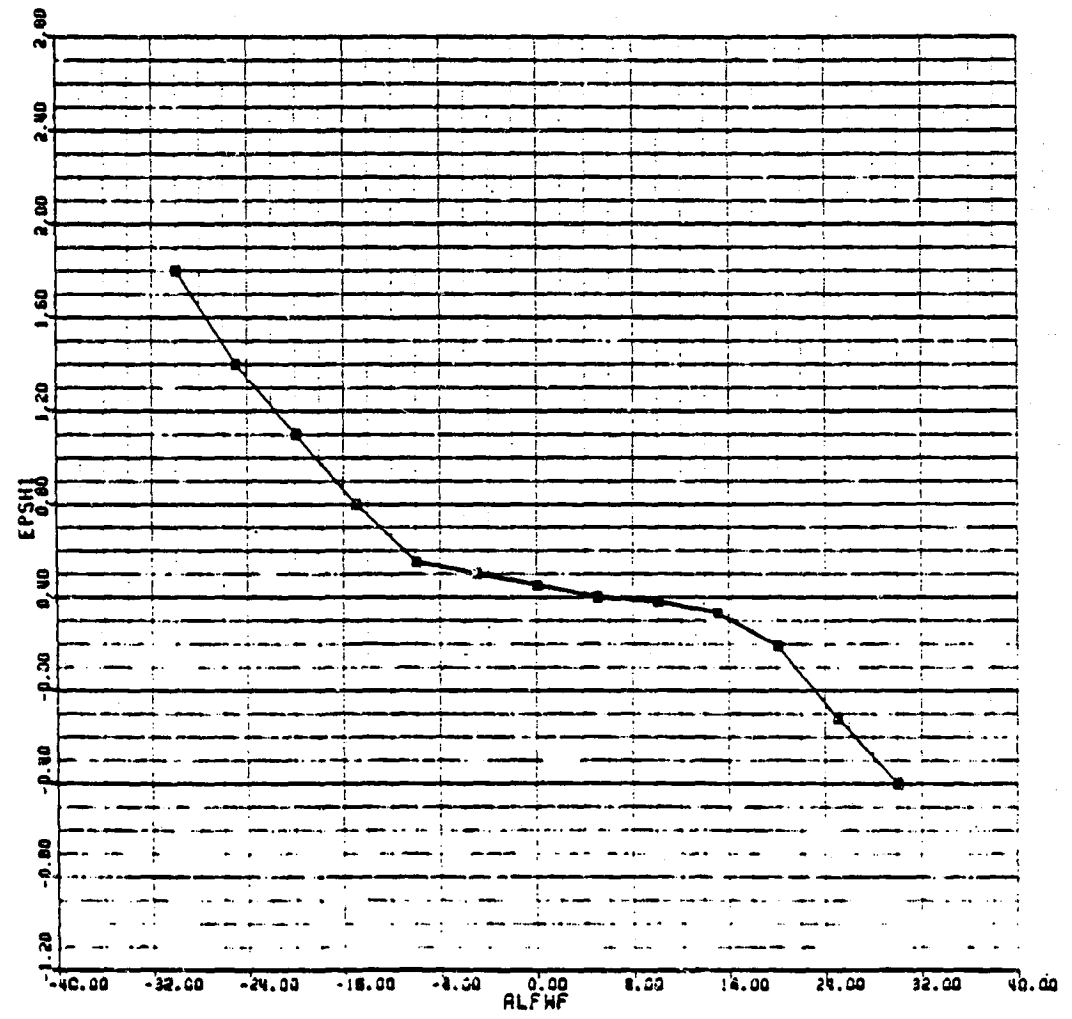


FIGURE 3.5.4(b)

HORIZONTAL TAIL LIFT COEFFICIENT DUE TO ANGLE OF ATTACK

BLACKHAWK - NASA STUDY 29-SEP-80

CLHIMP (1/2)

MAP NAME: CLHIMP
 MAP TYPE: UVSUVS
 INPUT VARIABLE(S): ALFHH1
 OUTPUT VARIABLE: CLH1

PRIMARY MAP:
 -30.00 LOWER LIMIT
 30.00 UPPER LIMIT
 5.00 DELTA

SECONDARY MAP:
 -90.00 LOWER LIMIT
 90.00 UPPER LIMIT
 10.00 DELTA

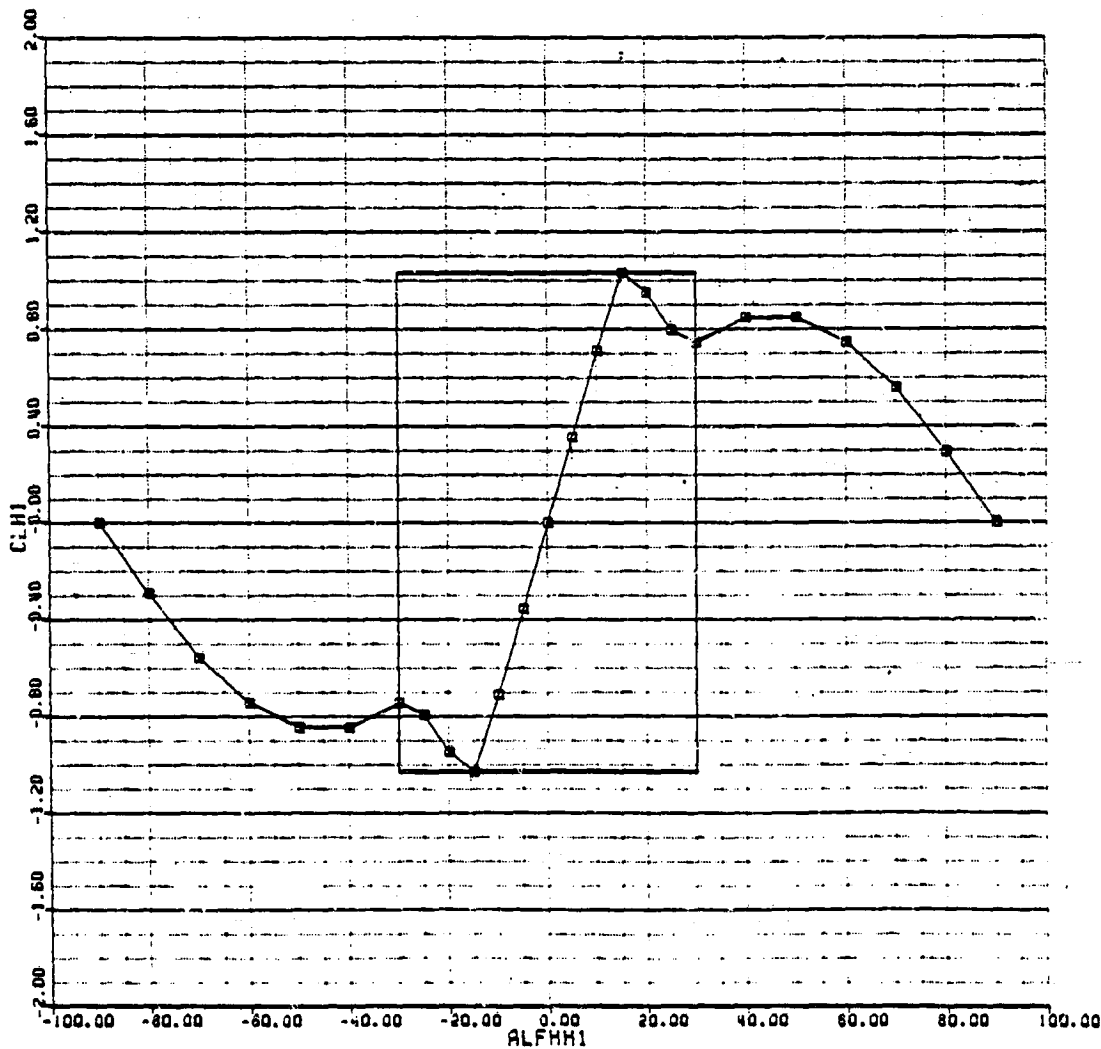


FIGURE 3.5.5(a)

HORIZONTAL TAIL LIFT COEFFICIENT DUE TO ANGLE OF ATTACK (Cont'd)

BLACKHAWK - NASA STUDY 29-SEP-80

CLH1MP (2/2)

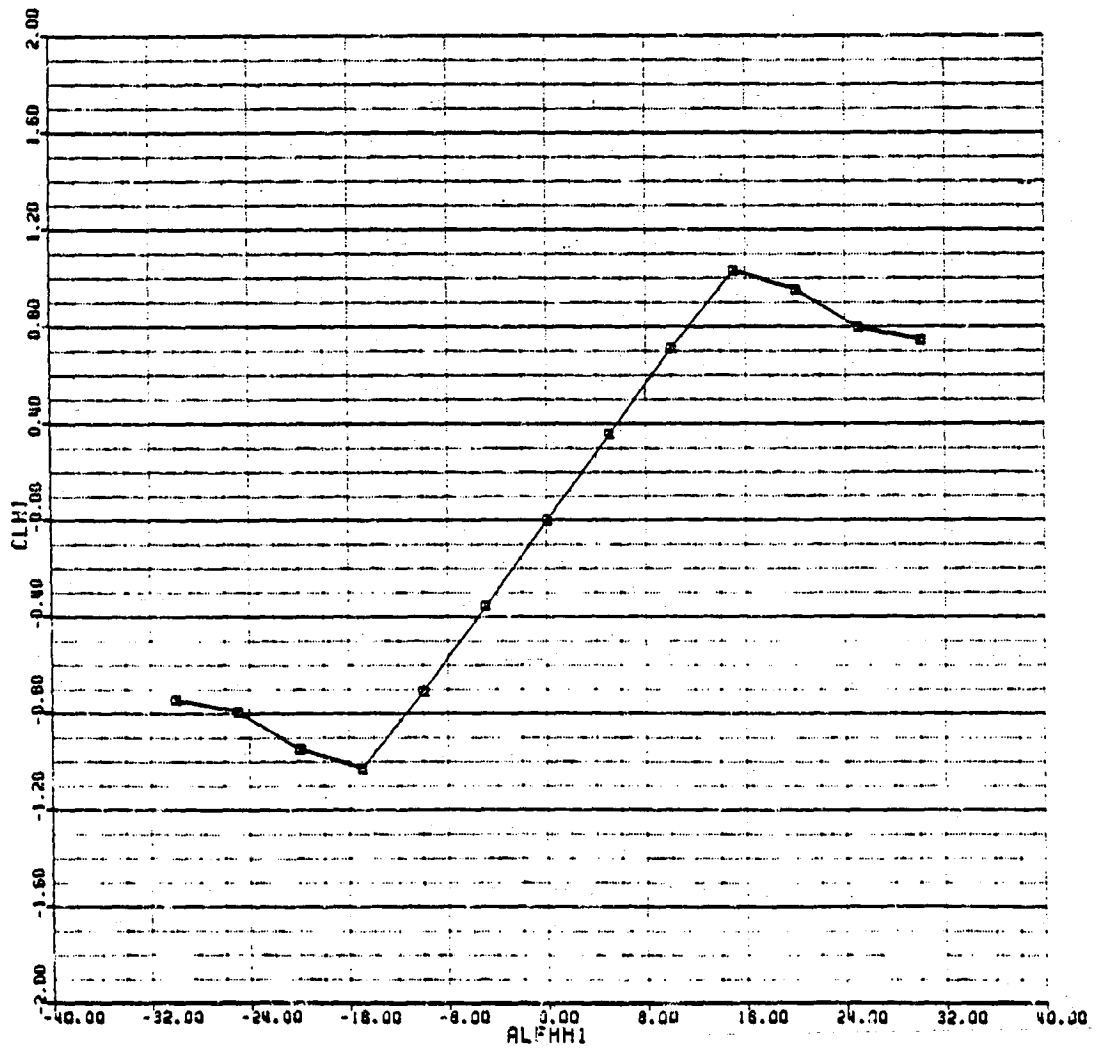


FIGURE 3.5.5(b)
5.3-30
Page

HORIZONTAL TAIL DRAG COEFFICIENT DUE TO ANGLE OF ATTACK

BLACKHAWK - NASA STUDY 23-SEP-80

CDHIMP (1/2)

MAP NAME:		CDHIMP
MAP TYPE:		UVAUVA
INPUT VARIABLE(S):	AFHH1	
OUTPUT VARIABLE:		CDH1
PRIMARY MAP:		
	0.00	LOWER LIMIT
	30.00	UPPER LIMIT
	5.00	DELTA
SECONDARY MAP:		
	30.00	LOWER LIMIT
	90.00	UPPER LIMIT
	10.00	DELTA

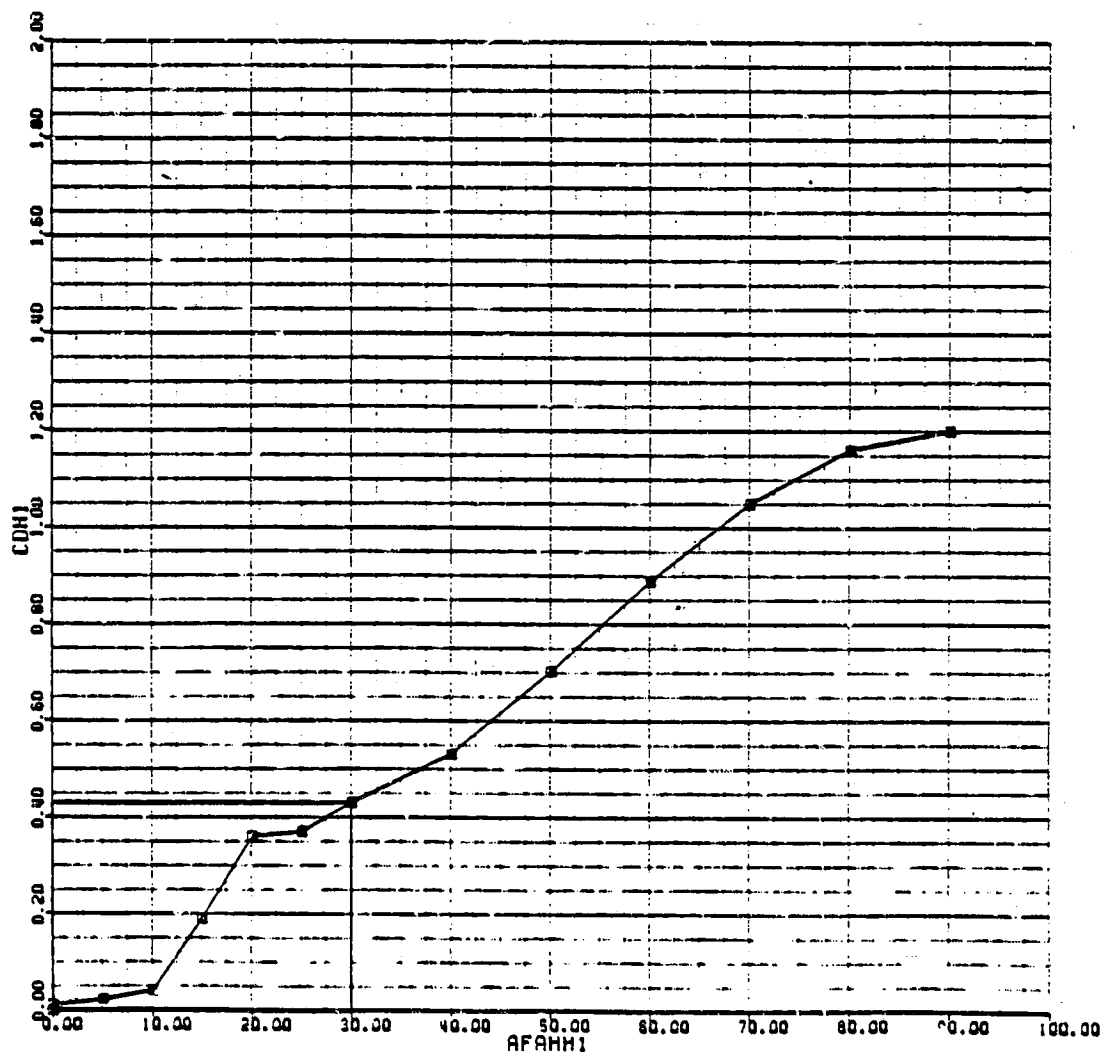


FIGURE 3.5.6(a)

HORIZONTAL TAIL DRAG COEFFICIENT DUE TO ANGLE OF ATTACK (Cont'd)

BLACKHAWK - NASA STUDY 23-SEP-80

COMIMP (2/2)

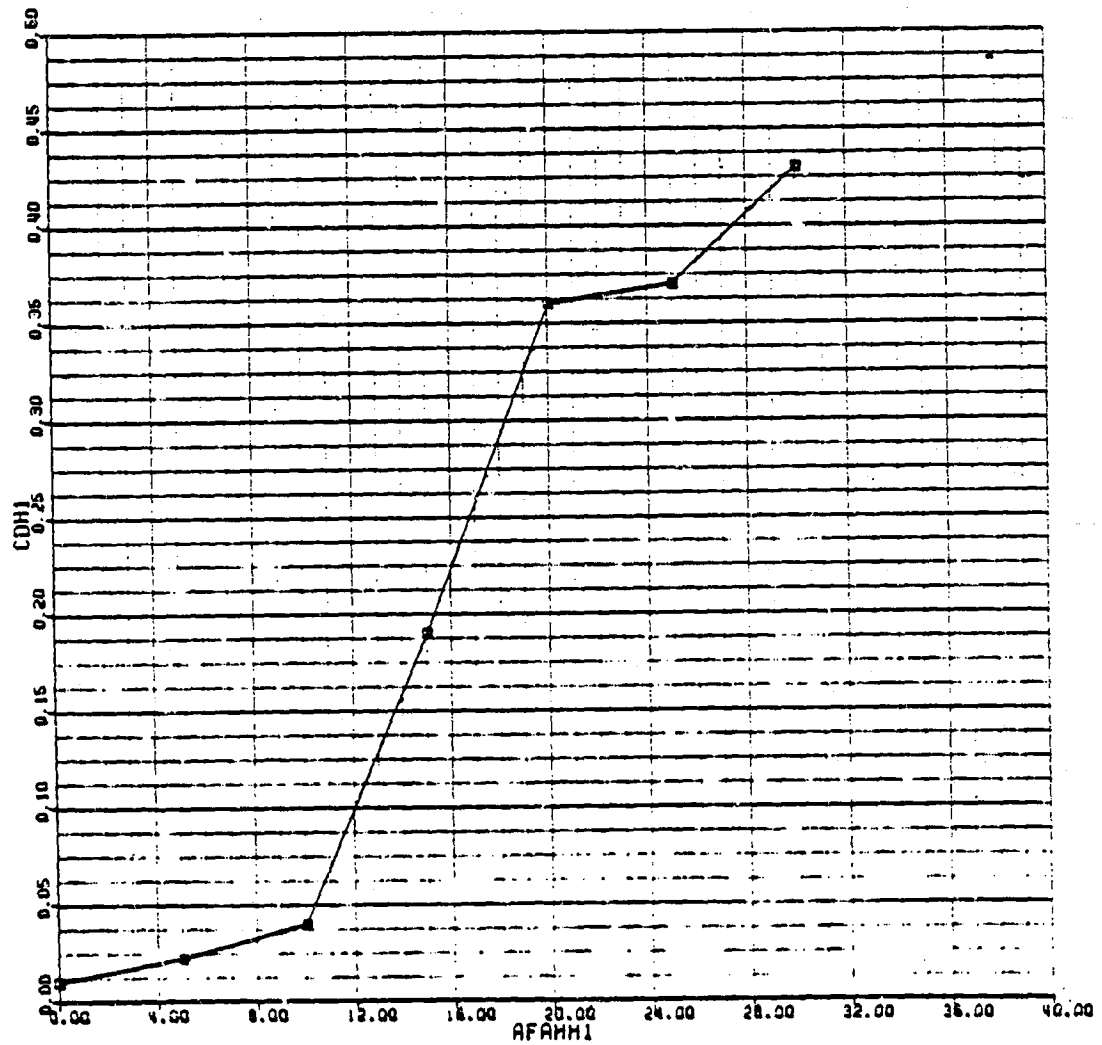


FIGURE 3.5.6(b)

DYNAMIC PRESSURE LOSS AT THE VERTICAL TAIL

MAP NAME: QV1MP
 MAP TYPE: UVA
 INPUT VARIABLE(S): PSABWF
 OUTPUT VARIABLE: QV1QWF
 PRIMARY MAP:
 0.00 LOWER LIMIT
 30.00 UPPER LIMIT
 5.00 DELTA

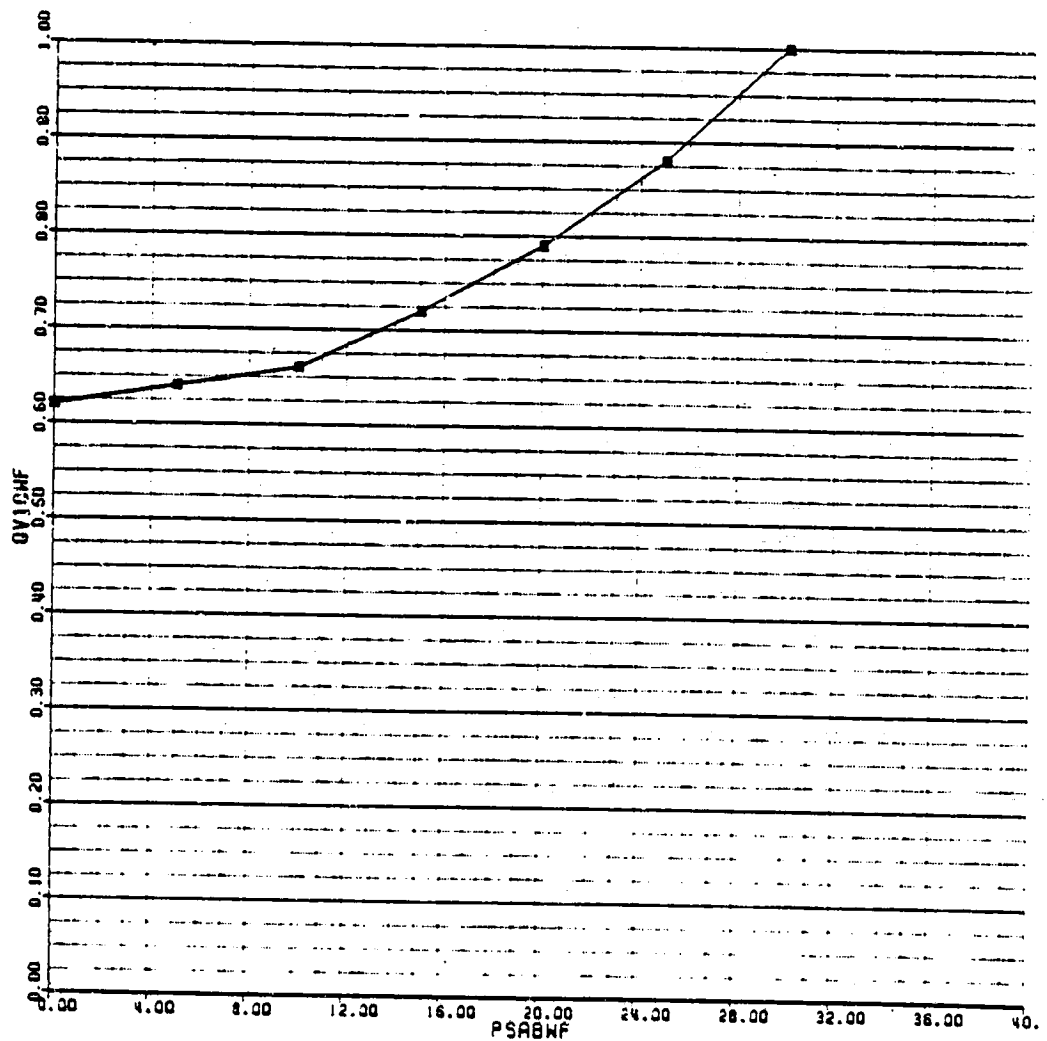


FIGURE 3.5.7

FUSELAGE SIDEWASH AT THE VERTICAL TAIL

BLACKHAWK - NASA STUDY 25-SEP-80

SGV1MP (1/2)

MAP NAME: SGV1MP
 MAP TYPE: UVAUVA
 INPUT VARIABLE(S): PSIXF
 OUTPUT VARIABLE: SIGV1

PRIMARY MAP:
 -30.00 LOWER LIMIT
 30.00 UPPER LIMIT
 5.00 DELTA

SECONDARY MAP:
 -90.00 LOWER LIMIT
 90.00 UPPER LIMIT
 30.00 DELTA

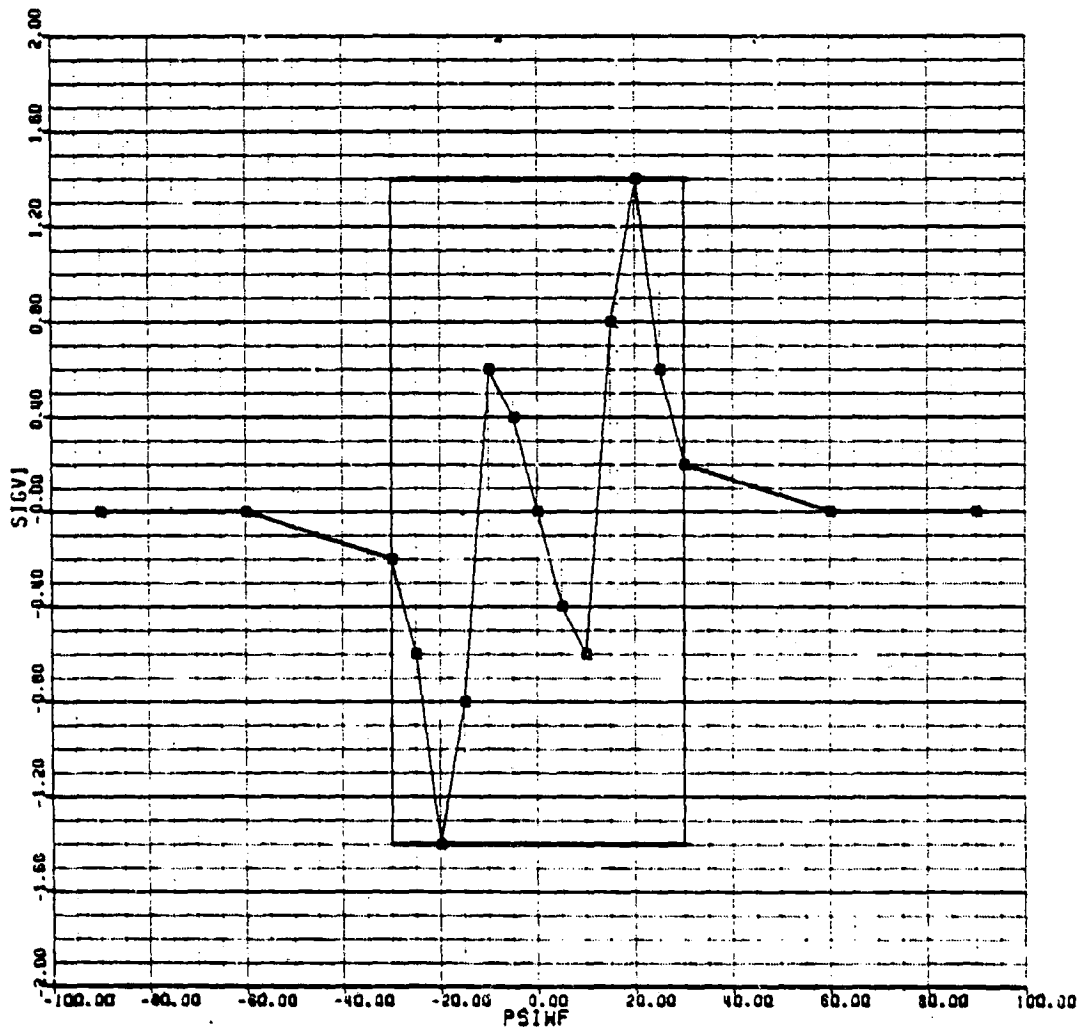


FIGURE 3.5.8(a)

FUSELAGE SIDEWASH AT THE VERTICAL TAIL (Cont'd)

BLACKHAWK - NASA STUDY 25-SEP-90

SGVIMP (2/2)

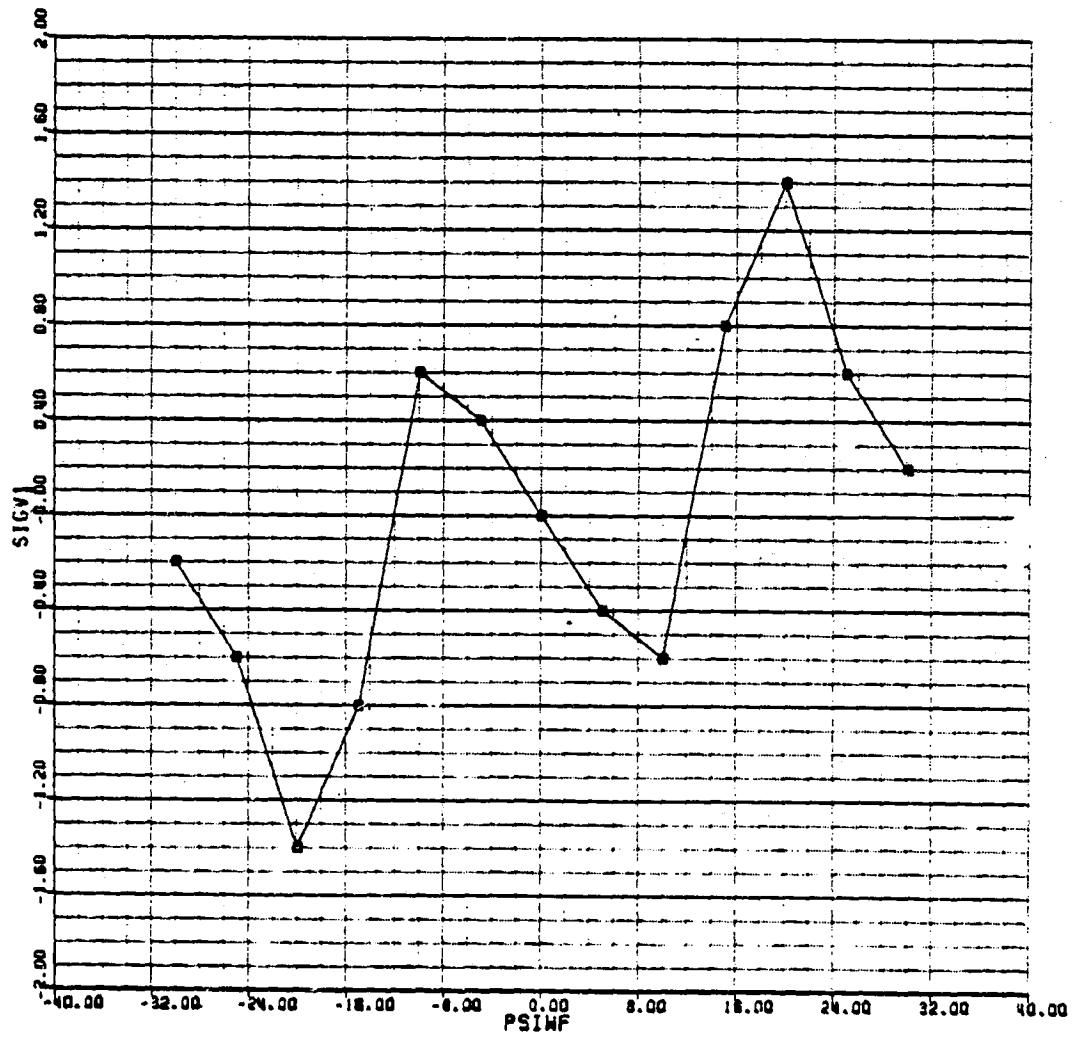


FIGURE 3.5.8(b)

VERTICAL TAIL LIFT COEFFICIENT DUE TO SIDESLIP

BLACKHAWK - NASA STUDY 25-SEP-80

CLV1MP (1/2)

MAP NAME: CLV1MP
 MAP TYPE: UVRUVR
 INPUT VARIABLE(S): BETVV1
 OUTPUT VARIABLE: CLV1

PRIMARY MAP:
 -30.00 LOWER LIMIT
 30.00 UPPER LIMIT
 5.00 DELTA

SECONDARY MAP:
 -90.00 LOWER LIMIT
 90.00 UPPER LIMIT
 10.00 DELTA

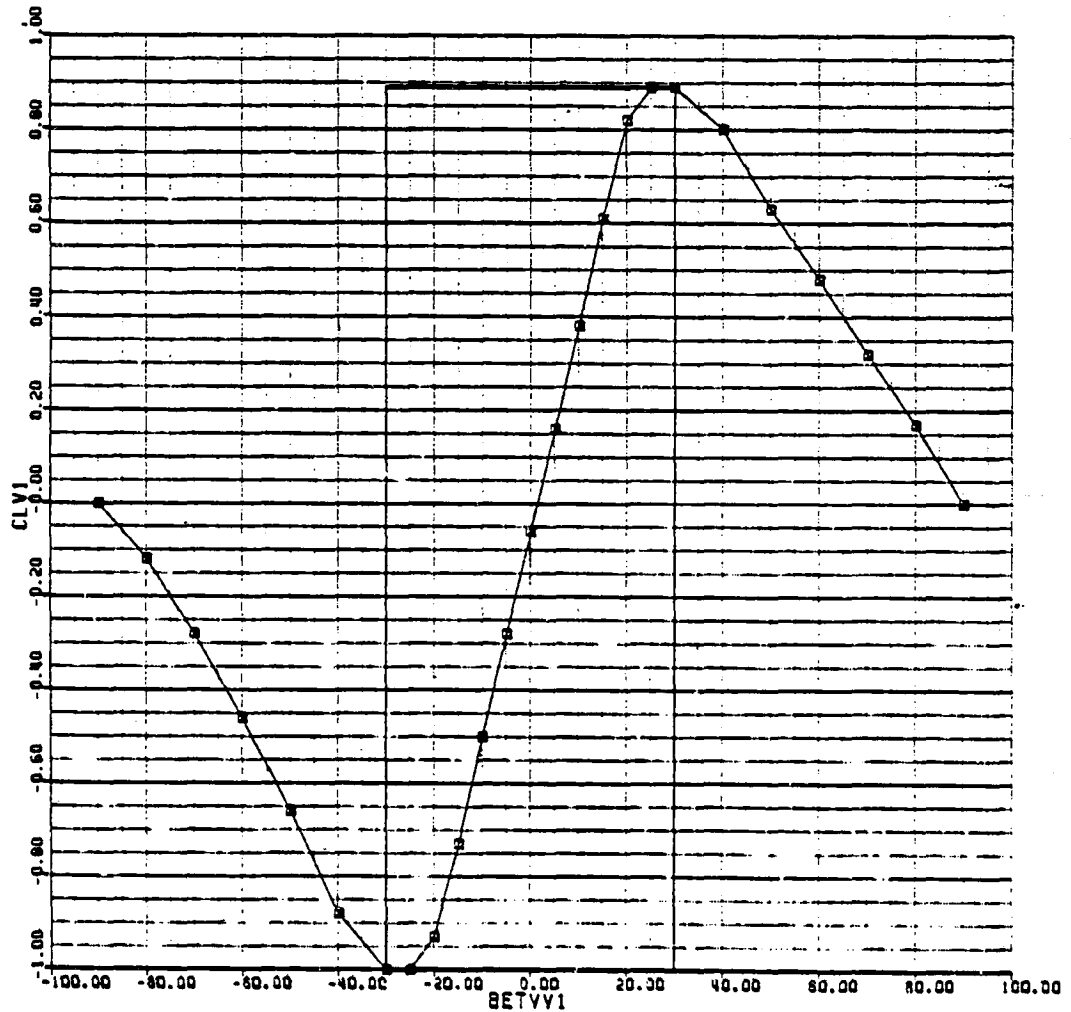


FIGURE 3.5.9(a)

VERTICAL TAIL LIFT COEFFICIENT DUE TO SIDESLIP (Cont'd)

BLACKHAWK - NASA STUDY 23-SEP-80

CLYIMP (2/2)

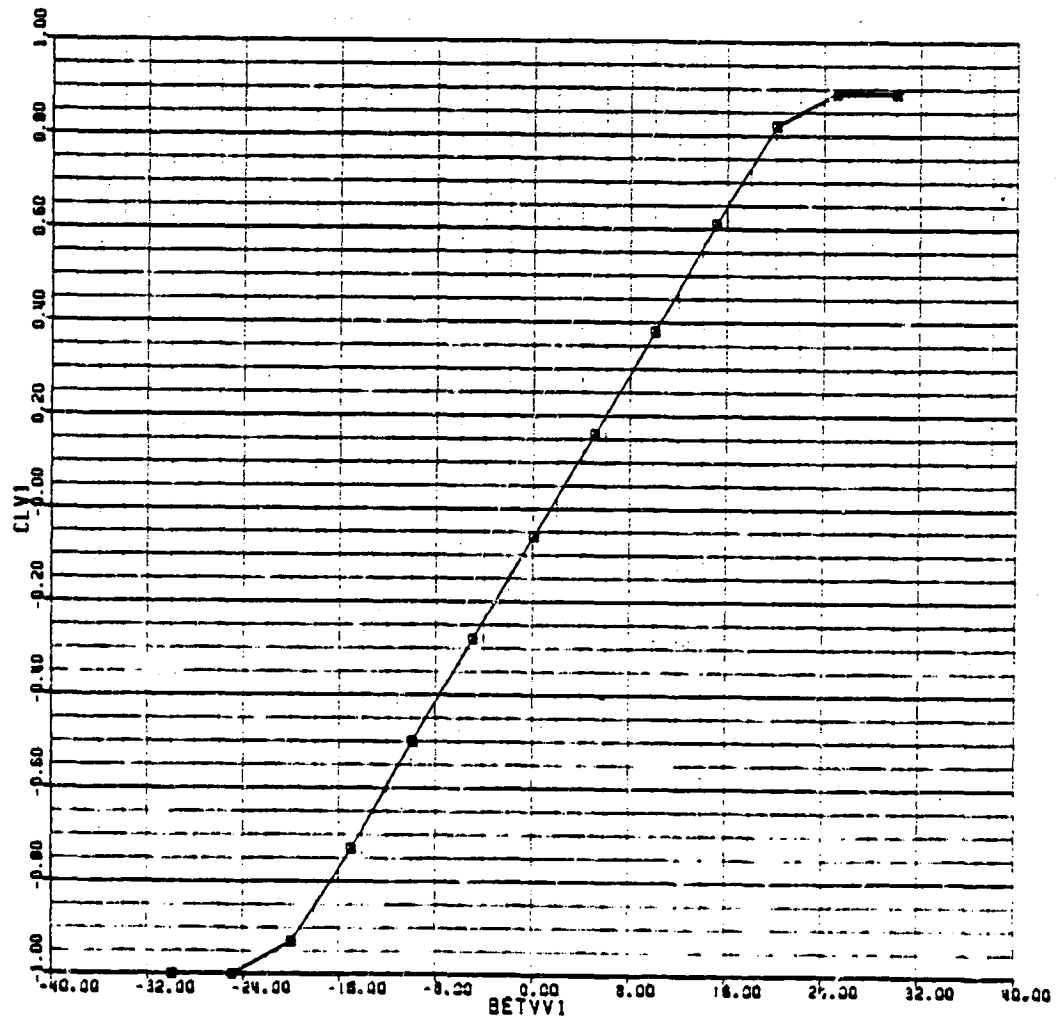


FIGURE 3.5.9(b)

VERTICAL TAIL DRAG COEFFICIENT DUE TO SIDESLIP

BLACKHAWK - NASA STUDY 23-SEP-80

CDVIMP (1/2)

MAP NAME: CDVIMP
MAP TYPE: UVRUVR
INPUT VARIABLE(S): BETVVI
OUTPUT VARIABLE: CDVI

PRIMARY MAP:
-30.00 LOWER LIMIT
30.00 UPPER LIMIT
5.00 DELTA

SECONDARY MAP:
-90.00 LOWER LIMIT
90.00 UPPER LIMIT
10.00 DELTA

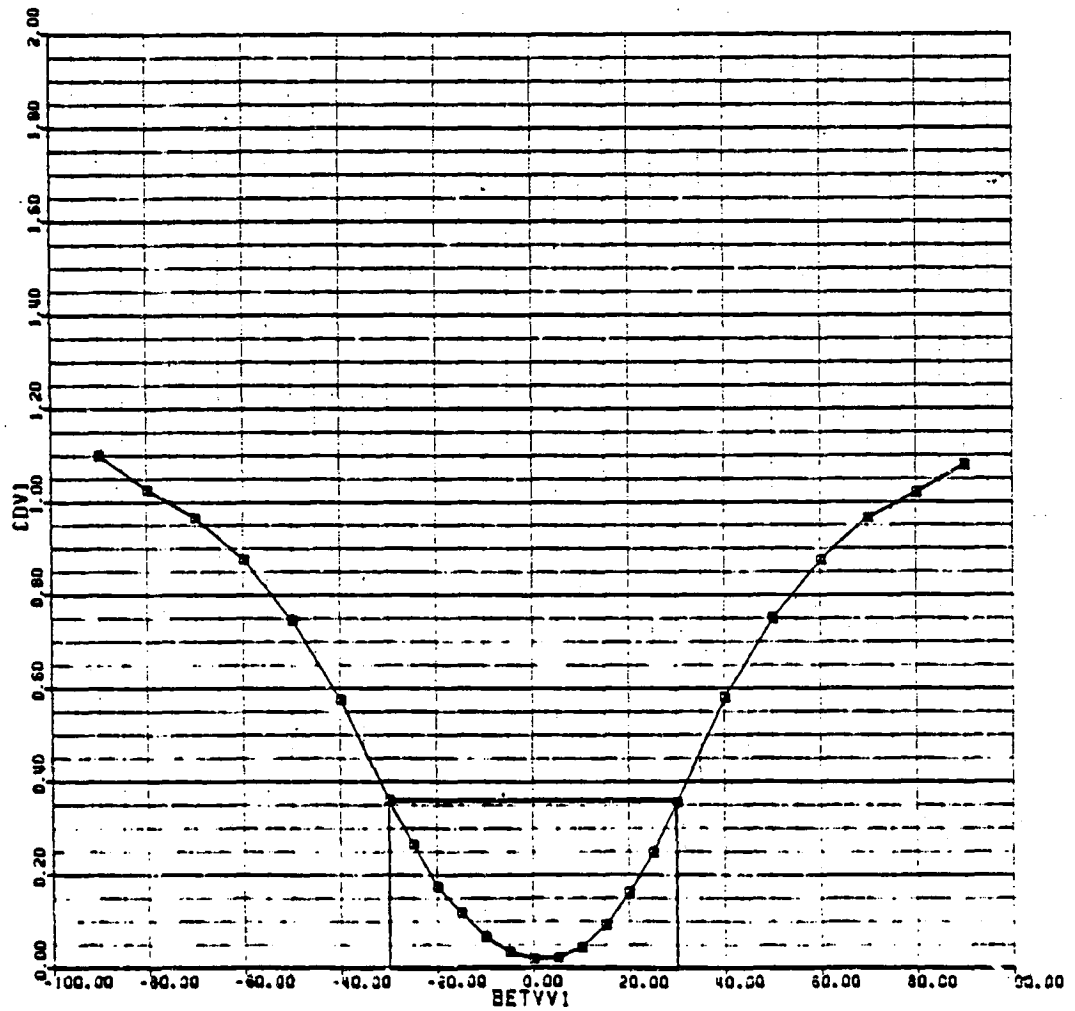


FIGURE 3.5.10(a)

VERTICAL TAIL DRAG COEFFICIENT DUE TO SIDESLIP (Cont'd)

BLACKHAWK - NASA STUDY 25-SEP-80

CDV1MP (2/2)

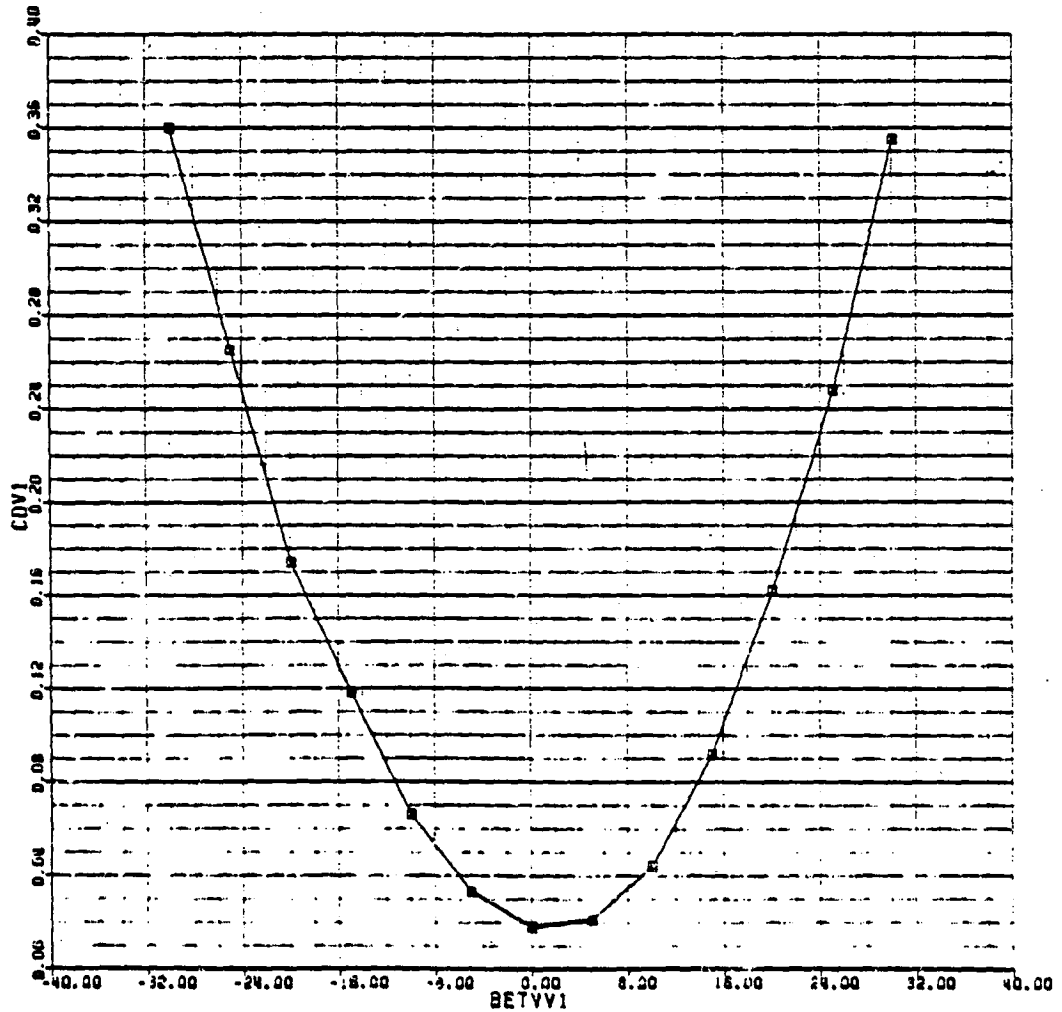


FIGURE 3.5.10(b)

5.4	TAIL ROTOR MODULE	
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5.4.1	Module Description	5.4-2
	FIGURES	
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5.4.2	Module Equations	5.4-4
5.4.3	Module Input/Output Definition	5.4-9
5.4.4	Nomenclature	5.4-10
5.4.5	Black Hawk Tail Rotor Input Data	5.4-14
5.4.6	References	5.4-15

5.4 Tail Rotor Module

5.4.1 Module Description

This module calculates the forces and moments at the center of gravity which are generated by a canted tail rotor. This rotor is represented basically by a simplified, closed form, Bailey Solution as developed in Reference 4.6.1. All terms in tip speed ratio (μ) greater than squared have been eliminated. In order to obtain the actual tail rotor collective pitch value, (θ_{TR}), the Bailey equations have been modified to account for δ_3 (pitch-flap coupling). This reduces the blade pitch which is impressed by the control system.

The airflow impinging on the tail rotor is developed from the free stream, rotor wash and fuselage sidewash, together with body angular rate effects. The total components of velocity are resolved through the cant angle into the tail rotor shaft axes system.

The Bailey theory equation is normally presented as the thrust coefficient in terms of the 't' coefficients. It should be noted that the equations have been manipulated to obtain an expression for downwash. This was found to be necessary to obtain an unconditionally stable solution. It is important that program flow follows the equation flow for a stable tail rotor solution (Figure 4.1.1).

A blockage factor K_{TBLK} is applied to the final thrust output to account for the proximity of the vertical tail. This correction is empirical and based on flight test data of other helicopters.

This simplified tail rotor model only calculates thrust. No account of H force is included in the final tail rotor force outputs. The tail rotor thrust is finally resolved into force and moments in body axes, at the center of gravity.

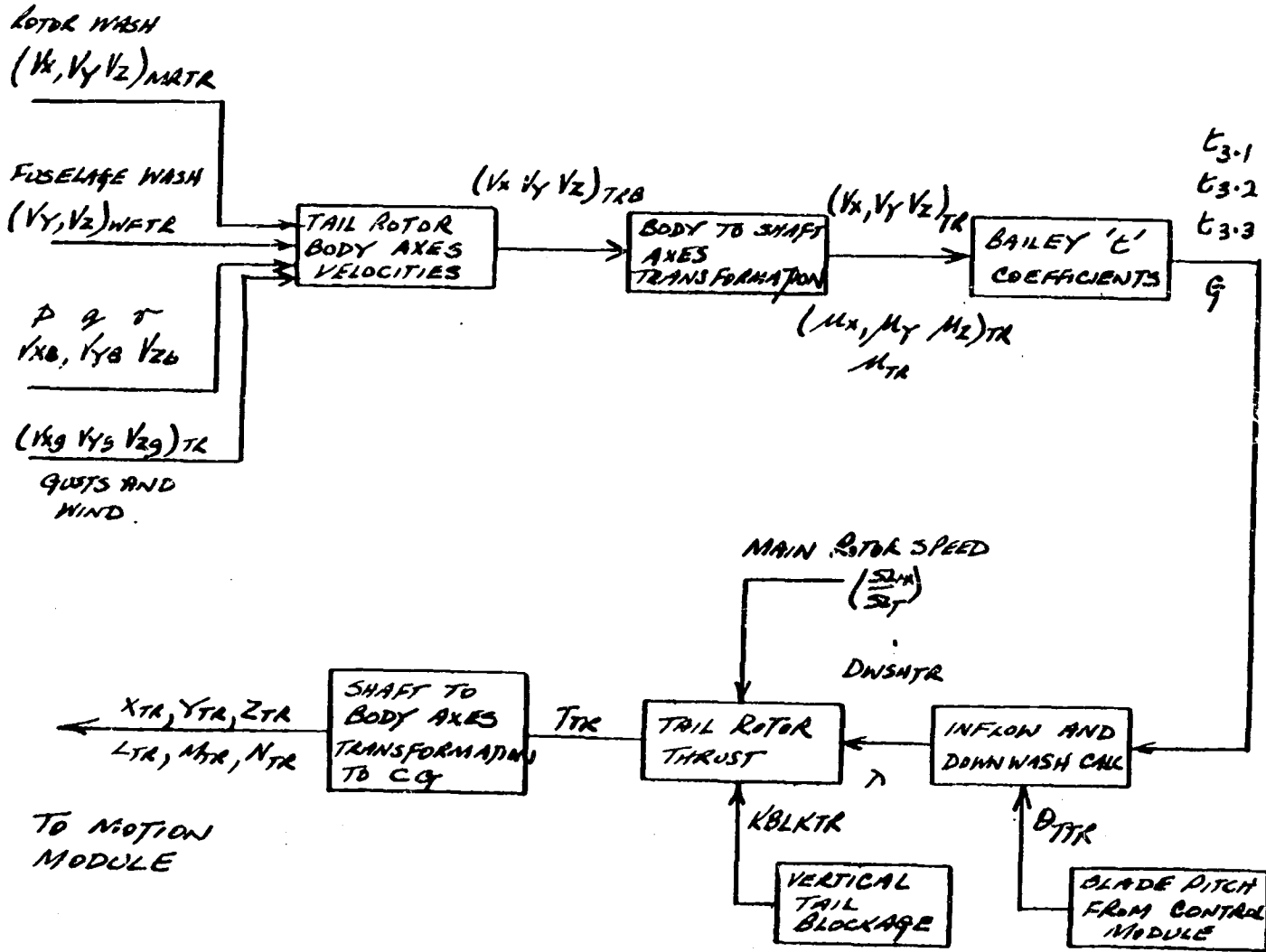
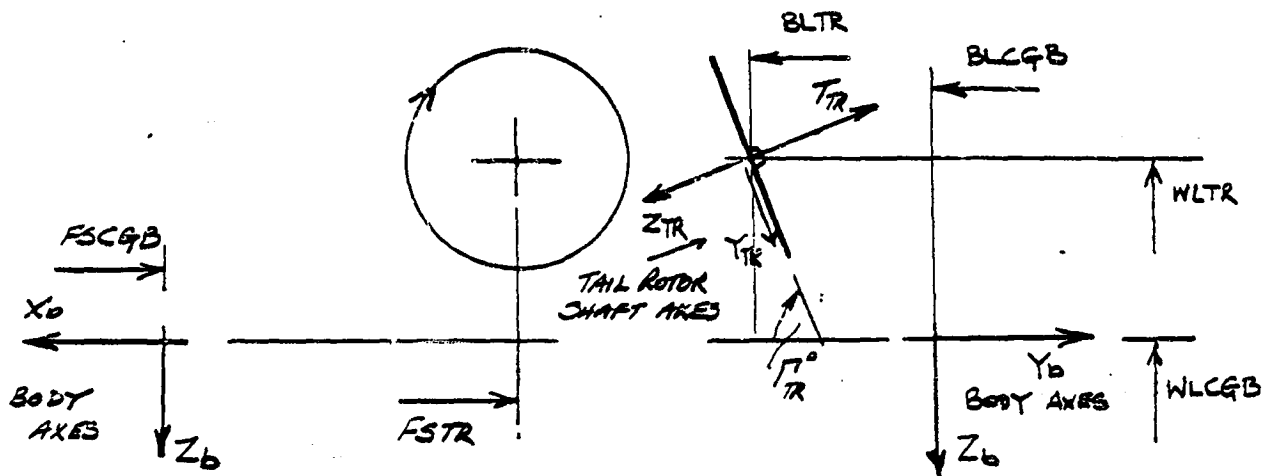


FIGURE 4.1.1

TAIL ROTOR EQUATION FLOW DIAGRAM

5.4.2 TAIL ROTOR MODULE EQUATIONS

GEOMETRY



$$F_{TR} = \frac{(F_{STR} - F_{SCGB})}{12}$$

$$W_{TR} = \frac{(N_{LTR} - W_{LCGB})}{12}$$

$$B_{TR} = \frac{(B_{LTR} - B_{LCGB})}{12}$$

INTERFERENCE VELOCITIES

- FROM THE MAIN ROTOR

$$E_{XTR} = f(\tau_{MR}, \alpha_{MR})$$

$$V_{XMR} = D_{WSPMR} \cdot \Omega_{TR} \cdot E_{XTR}$$

, SAME AS HORIZONTAL TAIL
FOR BLACKHAWK

, MAP-EXH2MP
TABLE 3.5.1
FIGURE 3.5.1

$$E_{KYTR} = f(\alpha_{PMR}, \alpha_{1PMR}) \quad \text{NO INPUT}$$

$$Y_{YMTR} = D_{NSMTR} \cdot S_T R_T E_{KYTR}$$

$$E_{KZTR} = f(\alpha_{PMR}, \alpha_{1PMR}) \quad , \quad \text{MAP-EZH1MP}$$

$$Y_{ZMTR} = -D_{NSMTR} \cdot S_T R_T E_{KZTR}$$

TABLE 3.5.2
FIGURE 3.5.2

- FROM THE FUSELAGE

DYNAMIC PRESSURE RATIO

$$Q_{TRQWF} = f(\psi_{NF}) \quad , \quad \text{MAP-QV1MP}$$

$$K_{QTR} = (Q_{TRQWF})^{1/2} \quad \text{TABLE 3.5.7}$$

FIGURE 3.5.7

SIDENASH COMPONENT

$$S_{IGTR} = f(\psi_{NF}) \quad , \quad \text{MAP-SGV1MP}$$

$$Y_{YNFTR} = - \frac{V_{XB}}{57.3} S_{IGTR} K_{QTR}$$

TABLE 3.5.8
FIGURE 3.5.8

$$S_{IGTR} \text{ DELAYED BY } \left(\frac{F_{STR} - F_{SCG6}}{12 \cdot V_{XB}} \right), \text{ SEC.}$$

DOWNWASH COMPONENT

$$E_{PSTR} = f(\alpha_{NF})$$

$$Y_{ZWNFTR} = - \frac{V_{XB}}{57.3} E_{PSTR} K_{QTR} \quad , \quad \text{MAP-EPH1MP}$$

TABLE 3.5.4
FIGURE 3.5.4

$$E_{PSTR} \text{ DELAYED BY } \left(\frac{F_{STR} - F_{SCG6}}{12 \cdot V_{XB}} \right), \text{ SEC.}$$

TOTAL INTERFERENCE VELOCITIES

$$V_{XITR} = V_{XMRTR}$$

$$V_{YITR} = V_{YMRTR} + V_{YNFTR}$$

$$V_{ZITR} = V_{ZMRTR} + V_{ZNFTR}$$

TAIL ROTOR VELOCITIES IN THE BODY AXES

$$V_{XTRB} = (V_{XB} + V_{XGTR}) K_{QTR} - q(W_{TR}) + r(B_{TR}) + V_{XITR}$$

$$V_{YTRB} = (V_{YB} + V_{YGTR}) K_{QTR} - r(F_{TR}) + p(W_{TR}) + V_{YITR}$$

$$V_{ZTRB} = (V_{ZB} + V_{ZGTR}) K_{QTR} + q(F_{TR}) - p(B_{TR}) + V_{ZITR}$$

TAIL ROTOR VELOCITIES IN THE SHAFT AXES

$$V_{XTR} = V_{XTRB}$$

$$V_{YTR} = V_{YTRB} \cos \beta_{TR} + V_{ZTRB} \sin \beta_{TR}$$

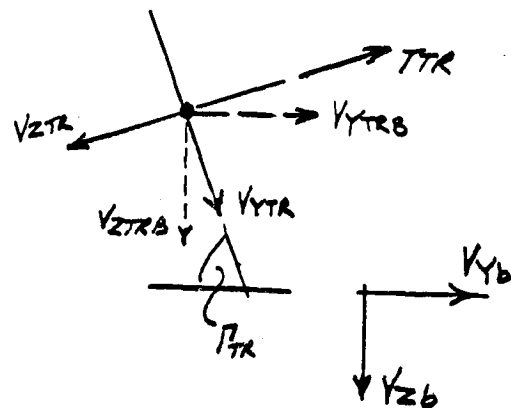
$$V_{ZTR} = -V_{YTRB} \sin \beta_{TR} + V_{ZTRB} \cos \beta_{TR}$$

$$M_{XTR} = V_{XTR} / (S_{2R})_{TR}$$

$$M_{YTR} = V_{YTR} / (S_{2R})_{TR}$$

$$M_{ZTR} = V_{ZTR} / (S_{2R})_{TR}$$

$$M_{TR} = (M_{XTR}^2 + M_{YTR}^2)^{1/2}$$



BAILEY COEFFICIENTS

$$e_{3.1} = \frac{B^2}{2} + \frac{\mu_{TR}^2}{4}$$

$$e_{3.2} = \frac{B^3}{3} + \frac{B}{2} \mu_{TR}^2$$

$$e_{3.3} = \frac{B^4}{4} + \frac{B^2}{4} \mu_{TR}^2$$

$$G = \frac{A_{TR}}{2} \left[\frac{bc}{TR} \right]_{TR}$$

TAIL ROTOR BLADE PITCH

$$Q_{TR} = \frac{1}{57.3} \left[Q_{TR} - \frac{T_{TR}}{(L-1)} \left(\frac{\partial a_0}{\partial TR} \right) \tan \delta_3 + B_{IASTR} \right]$$

TAIL ROTOR INFLOW

$$D_{NSHTR}(L) = \frac{G \left[\mu_{2TR}(e_{3.1}) + Q_{TR}(e_{3.2}) + \frac{T_{NSTR}}{57.3}(e_{3.3}) \right]}{2 \left(\mu_{TR}^2 + \lambda_{TR}^2 \right)^{1/2} + G(e_{3.1})}$$

$$\lambda_{TR}(L) = \mu_{2TR} - D_{NSHTR}(L)$$

ROTOR THRUST

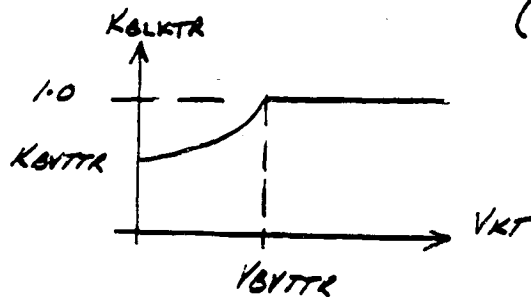
$$T_{TR} = 2 \pi R_{TR}^4 D_{VSHTR} \left(\mu_{TR}^2 + \gamma_{TR}^2 \right)^{1/2} \sigma_{TR}^2 \left(\frac{\sigma_{MR}}{\sigma_T} \right)^2 K_{BLKTR}$$

WHERE K_{BLKTR} IS THE VERTICAL TAIL BLOCKAGE FACTOR

IF $V_{KT} \geq V_{BVTR}$, SET $K_{BLKTR} = 1.0$

IF $V_{KT} < V_{BVTR}$

$$\text{SET } K_{BLKTR} = (1 - K_{BVTR}) \frac{V_{KT}^2}{(V_{BVTR})^2} + K_{BVTR}$$



TAIL ROTOR FORCES AND MOMENTS AT THE CG IN BODY AXES

$$X_{TR} = - (C_{DTR}) \frac{1}{2} \rho (V_{XTR})^2$$

$$Y_{TR} = T_{TR} \sin \beta_{TR}$$

$$Z_{TR} = - T_{TR} \cos \beta_{TR}$$

$$L_{TR} = Y_{TR} (W_{TR}) - Z_{TR} (B_{TR})$$

$$M_{TR} = Z_{TR} (F_{TR}) - X_{TR} (W_{TR})$$

$$N_{TR} = X_{TR} (B_{TR}) - Y_{TR} (F_{TR})$$

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5.4.3 TAIL ROTOR MODULE INPUT/OUTPUT DATA TRANSFER

INPUT TRANSFER	
PARAMETER	ORIGIN MODULE
FSCGB	MAIN ROTOR
YLCGB	
DNSHMR	
CHIPMR	
AIFMR	
OMGTMR	
RMR	
OMRMR	
ALWF	FUSELAGE
BETWF	
THETTR	FLIGHT CONTROL
VXGTR	GUST
VYGTR	
VZGTR	
VXB	MOTION
VYB	
VZB	
P	
Q	
R	
VKT	

OUTPUT TRANSFER	
PARAMETER	DESTINATION MODULE
XTR	MOTION
YTR	
ZTR	
LTR	
MTR	
NTR	

5.4.4

NOTATION FOR THE TAIL ROTOR MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
F _{STR}	FSTR	INS	Fuselage station for tail rotor
F _{SCGB}	FSCGB	INS	Fuselage station for CG
F _{TR}	KTR	FT	Tail rotor longitudinal arm
W _{LTR}	WLTR	INS	Waterline station for tail rotor
W _{LCGB}	WLCGB	INS	Waterline station for CG
W _{TR}	KTR+1	FT	Tail rotor vertical arm
B _{LTR}	BLTR	INS	Buttline station for tail rotor
B _{LCGB}	BLCGB	INS	Buttline station for the CG
B _{TR}	KTR+2	FT	Tail rotor lateral arm
χ _{PMR}	CHIPMR	DEG	Rotor Wake Skew Ang
a _{1FMR}	AA1FMR	DEG	Longitudinal main rotor flapping
D _{WSHMR}	DWSHMR	ND	Uniform downwash at the main rotor
Ω _T	OMGTMR	RADS/SEC	Trimmed rotor speed
R _T	RMR	FT	Main rotor radius
E _{KXTR}	EKXTR	ND	Main rotor wash factors
E _{KYTR}	EKYTR	ND	
E _{KZTR}	EKZTR	ND	
V _{XMTR}	VXMTR	FT/SEC	Main rotor wash at the tail rotor
V _{YMRTR}	VYMRTR	FT/SEC	
V _{ZMRTR}	VZMRTR	FT/SEC	
α _{WF}	ALFAWF	DEG	Fuselage angle of attack
ψ _{WF}	PSIWF	DEG	Fuselage yaw attitude
Q _{TRQWF}	QTRQWF	-	Dynamic Pressure ratio at the tail rotor
K _{QTR}	KQTR	-	----
S _{IGTR}	SIGTR	DEG	Fuselage sidewash at the tail rotor



5.4.4 (Cont'd)

NOTATION FOR THE TAIL ROTOR MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
V_{xb}	VXB	FT/SEC	Body axes velocities
V_{yb}	VYB	FT/SEC	
V_{zb}	VZB	FT/SEC	
V_{YWFTR}	VYWFTR	FT/SEC	Fuselage sidewash velocity
V_{XITR}	VXITR	FT/SEC	Total interference velocities at the tail rotor
V_{YITR}	VYITR	FT/SEC	
V_{ZITR}	VZITR	FT/SEC	
V_{XGTR}	VXGTR	FT/SEC	Body axes gust velocities
V_{YGTR}	VYGTR	FT/SEC	
V_{ZGTR}	VZGTR	FT/SEC	
P	P	RADS/SEC	Body axes angular rates
q	q	RADS/SEC	
r	r	RADS/SEC	
V_{XTRB}	VXTRB	FT/SEC	Total velocities at the tail rotor in body axes.
V_{YTRB}	VYTRB	FT/SEC	
V_{ZTRB}	VZTRB	FT/SEC	
V_{XTR}	VXTR	FT/SEC	Total velocities at the tail rotor in shaft axes.
V_{YTR}	VYTR	FT/SEC	
V_{ZTR}	VZTR	FT/SEC	
β_{TR}	GAMTR	DEG	Tail rotor cant angle
Ω_{TR}	OMEGTR	RADS/SEC	Tail rotor trim speed
R_{TR}	RTR	FT	Tail rotor radius
μ_{XTR}	MUXTR	ND	Shaft axes velocities normalized by rotor tip speed.
μ_{YTR}	MUYTR	ND	
μ_{ZTR}	MUZTR	ND	
μ_{TR}	MUTR	ND	

5.4.4 (Cont'd)

NOTATION FOR THE TAIL ROTOR MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
$t_{3.1}$			Bailey Coefficients
$t_{3.2}$			
$t_{3.3}$			
B	BTLTR	-	Blade tip loss factor
G			Constant
A_{TR}	ATR	1/RADS	Blade section lift curve Slope (2D)
b_{TR}	BTR	-	Actual number of blades on the tail rotor
C_{TR}	CHRDTR	FT	Blade Chord for the Tail rotor
θ_{TTR}	THETTR	DEG	Tail rotor commanded blade pitch
T_{TR}	TTR	LB	Tail Rotor Thrust
ao/T_{TR}	DELTR	-	Rate of change of coning with thrust
δ_3	DEL3TR	DEG	Flapping hinge offset angle
B_{IASTR}	BIASTR	DEG	Blade pitch correction to linear twist
θ_{TR}	-	DEG	Actual Blade pitch
D_{WSHTR}	DWSHTR	-	Uniform downwash at the tail rotor disc
T_{WSTTR}	TWSTTR	DEG/R _{TR}	Linear blade twist
λ_{TR}	LAMBTR	-	Tail rotor inflow
Ω_{MR}	OMGMR	RADS/SEC	Actual rotor speed
K_{BLKTR}	KBLKTR	-	Tail rotor blockage from vertical tail
V_{BVTTR}	VBVTTR		Airspeed breakpoint for blockage factor
C_{DTR}	CDTR	FT ²	Tail rotor drag
ρ	RHO	SLUGS/FT ³	Air density
X_{TR}	XTR	LB	Tail rotor forces at the CG in body axes
Y_{TR}	YTR	LB	
Z_{TR}	ZTR	LB	



5.4.4 (Cont'd)

NOTATION FOR THE TAIL ROTOR MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
L _{TR}	LTR	FT LB	Tail rotor moments at the CG in body axes
M _{TR}	MTR	FT LB	
N _{TR}	NTR	FT LB	
V _{KT}	VKT	KNOTS	Flight Path airspeed

5.4.5 BLACK HAWK TAIL ROTOR INPUT DATA

INPUT CONSTANTS

F_{STR}	=	732.0	INS
W_{LTR}	=	324.7	INS
B_{LTR}	=	-14.0	INS
R_{TR}	=	5.5	FT
S_{TR}	=	124.62	RADS/SEC
b_{TR}	=	4	-
T_{NSTTR}	=	-18	DEG/R
Γ_{TR}	=	70.0	DEG
δ_{STR}	=	35.0	DEG
C_{TR}	=	.81	FT
A_{TR}	=	5.73	1/RAD
B	=	.92	-
C_{DTR}	=	0	-
$(\dot{\theta}_{TR})$	=	.001455	DEG/16
K_{BYTR}	=	.796	
V_{BYTR}	=	30	KNOTS
B_{IASTR}	=	6.0	DEG

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5.4.6 References

1. Simplified Theoretical Method of Determining the Characteristics of a Lifting Rotor in Forward Flight, Bailey, NACA Report 716

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5.5 FLIGHT CONTROL SYSTEM MODULE

5.5.1 Flight Control Simulation Module Description

The simulation of the flight control system is presented entirely in terms of block diagrams indicating the transfer function between signals. The Background information presented in Section 3.5 of Volume II complements this description of the control system simulation. An overview of the control system simulation nomenclature, for the various elements, is presented on Figure 5.1.1. The block diagrams, subsequently presented, are aligned with the flow of this figure. It will be noted in studying this section, that the bandwidth of some control system components is wide. This leads to time constants of some of the transfer functions being small. For completeness these components have been retained in the model, thus making it necessary to provide a test for cycle time in the corresponding algorithms to ensure unity gain if cycle time is large relative to the function's time constant.

- (a) Sensors - The transfer functions for the sensors are presented in Figure 5.1.2.
- (b) Stability Augmentation System (SAS) - The simulation definitions for pitch, roll and yaw SAS channels are shown on Figures 5.1.3 (a), (b), and (c) respectively. Each figure incorporates the representation of the digital and analog SAS channels. In general, the helicopter motion sensed by the gyroscopes is passed through signal conditioning filters before being shaped by the SAS networks. The signal is then processed through a washout, if required, and the 2:1 gain change switch. The switch definition is - ON/ON, both channels working at gain 1.0, ON/OFF, digital channel only working at gain 2.0, OFF/ON, analog channel only working at gain 2.0. Finally, the signal is restricted in amplitude by a 5% authority limit. In the case of the digital channel, the signal is passed through a zero order hold to account for update delays. A switch in the yaw channel inhibits the lagged rate term at speeds above 60 knots and introduces lagged lateral acceleration for aid in turn coordination.

The following logic applies for all control system channels:

In the IC Mode:

1. Outputs of all synchronizers are zero.
2. Outputs of all integraters are zero or have defined initial values.

In the Compute Mode: The logic is defined on the block diagrams.

- (c) Pitch Bias Actuator (PBA) - The PBA representation is presented on Figure 5.1.4. The input signals are derived as indicated. It should be noted that the pitch rate signal is picked off

downstream of the signal conditioning. The three input signals are limited, passed through a gain and summed to obtain the signal output to the PBA actuator. The actuator travel is restricted by a + 3%/sec rate limit and a - 15% authority limit. Because the PBA algorithm is computed by the digital computer, the simulation includes a zero order hold.

- (d) Flight Path Stabilization (FPS) - The simulation representation of the pitch, roll, and yaw Flight Path Stabilization channels are presented on Figures 5.1.5 (a), (b) and (c) respectively. The synchronization elements in the model must track during the simulation trimming phase and in the IC mode. During the compute mode, they must track with the FPS off. The FPS shaping networks are relatively straightforward, however, a degree of complication is introduced by the automatic turn coordination capability. On the helicopter the turn switch is enabled by several channels of complex logic. For this simulation it is recommended that the switch be enabled by one path only.

FPS ON, VXBIKT > 60 kts, plus trim release pressed, plus $\theta_b > 2^\circ$

Exit is by feet on the pedals and trim release pressed. For analysis purposes the turn switch can be enabled as required. Since the FPS is computed in the digital computer, the final output to the trim actuator is passed through a zero order hold. The trim actuator has 100% authority within certain control force constraints and is rate limited at 10%/sec.

- (e) Mechanical Control System - In addition to its function as a mixing unit for control coupling, this area in the simulation is used to bring together the various elements of the control system computed upstream. The representations of the four primary controls, main rotor collective blade pitch, lateral cyclic blade pitch, longitudinal cyclic blade pitch and tail rotor collective pitch are shown on Figures 5.1.6 (a), (b), (c) and (d) respectively. The longitudinal cyclic pitch channel will be used for purposes of discussion. The outputs from the digital and analog SAS channels are summed and processed through the SAS actuator dynamics, resulting in actuator travel in inches (XBILS). This is summed with the PBA (XBBAS), the FPS (XBOLS) and the pilot control stick input (XB+2). The trimming algorithm input (XB+1) is added in the simulation at this point for cases where trim is accomplished using the control sticks. The resultant linkage motion is converted into equivalent degrees of longitudinal cyclic before being summed with coupled motion from other controls in the mixing unit. The output from the mixing unit BISMIX is modified by the primary servo dynamics giving the final longitudinal cyclic impressed on the main rotor. Additional inputs are added at this point for use in analysis. Travel limits for the rotor head are given on page 5.5.20.

- (f) Stabilator - The representation of the analog network controlling the stabilator is presented on figure 5.1.7. The network provides for the feedback of velocity, collective stick, lateral acceleration and pitch rate scheduled as a function of forward speed. The output from the network is passed through the dynamics of the limited rate tail servo. For the simulation provision is made for an alternative input (STBSET) for use in analysis.

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OVERVIEW OF THE CONTROL SYSTEM SIMULATION

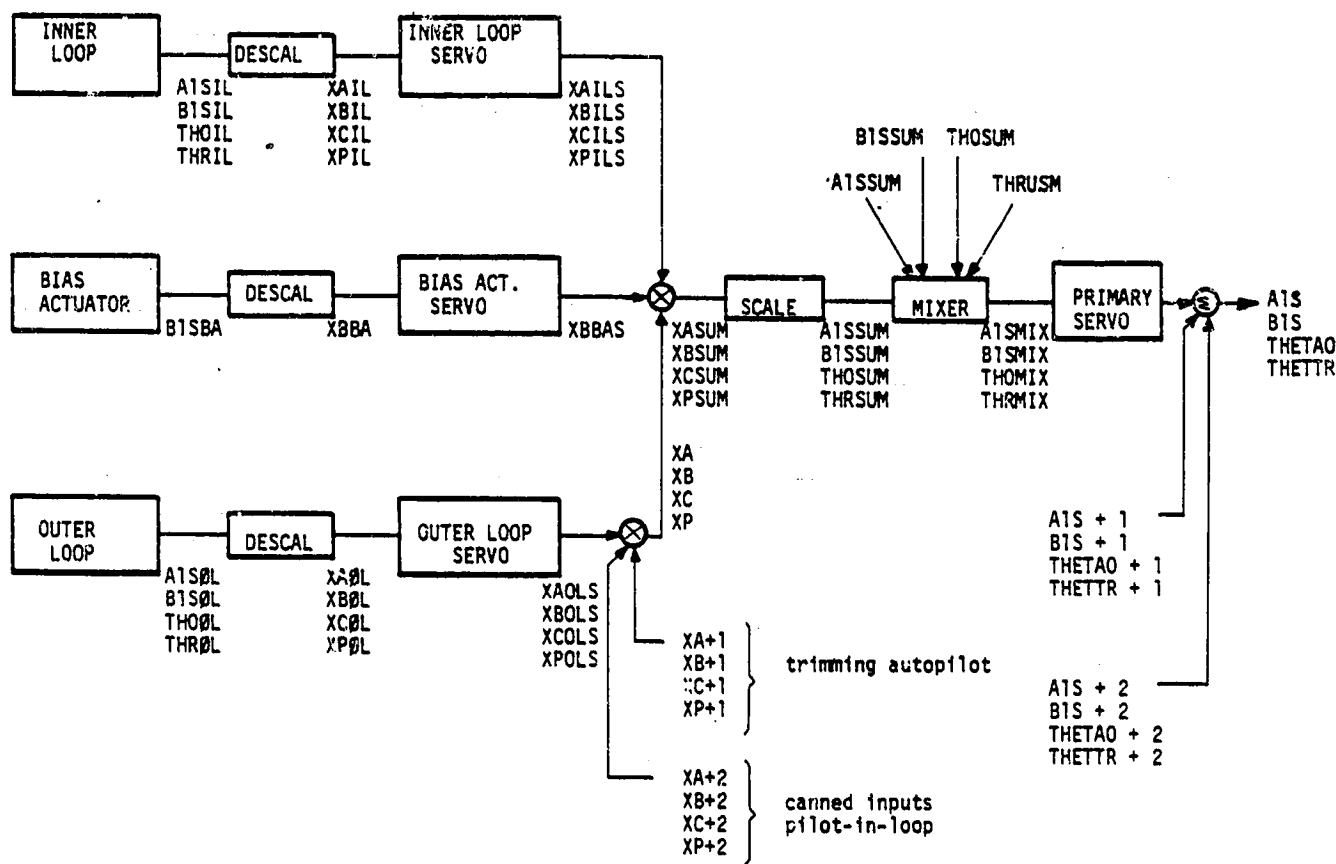


FIGURE 5.1.1

BLACK HAWK HELICOPTER MOTION SENSORS

ATTITUDE GYROS

PITCH THETAB, DEG

ROLL PHIB, DEG

YAW PSIB, DEG

AIR SPEED VxBIKT, KNOTS

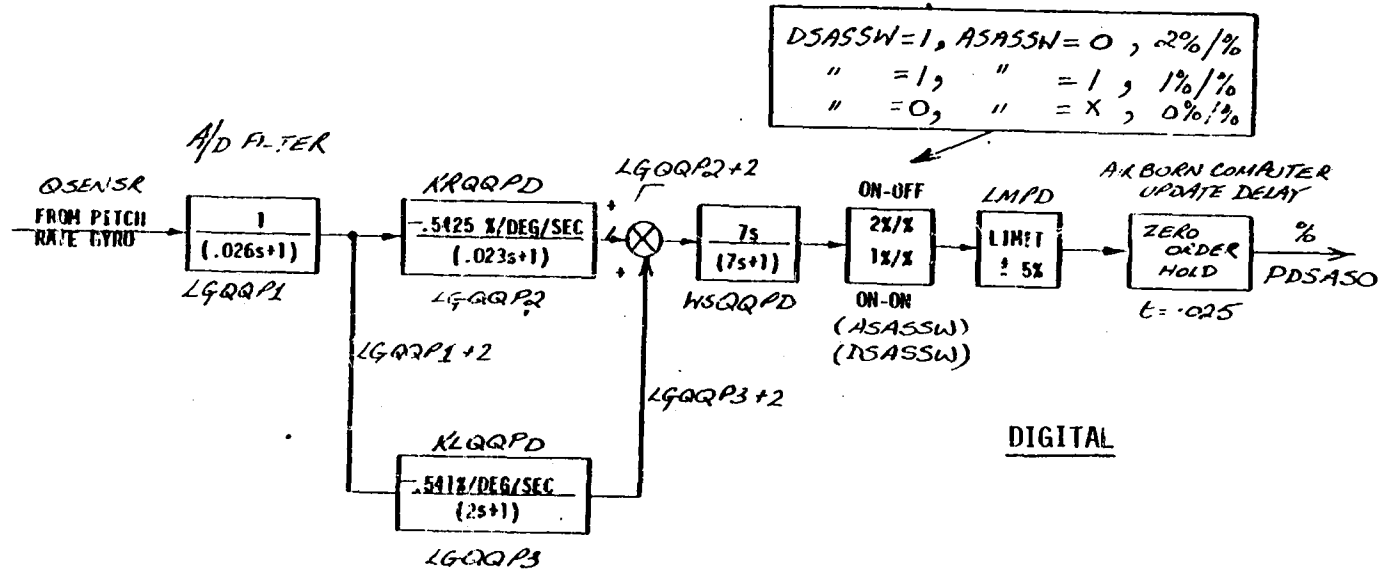
RATE GYROS

$$\begin{matrix} \text{PITCH} \\ \text{ROLL} \\ \text{YAW} \end{matrix} \begin{matrix} ; \\ ; \\ ; \end{matrix} \begin{matrix} \text{Q SENS R} \\ \text{P SENS R} \\ \text{R SENS R} \end{matrix} = \frac{\begin{matrix} \text{Q DEG} \\ \text{P DEG} \\ \text{R DEG} \end{matrix}}{(.00014S^2 + .016S + 1)}$$

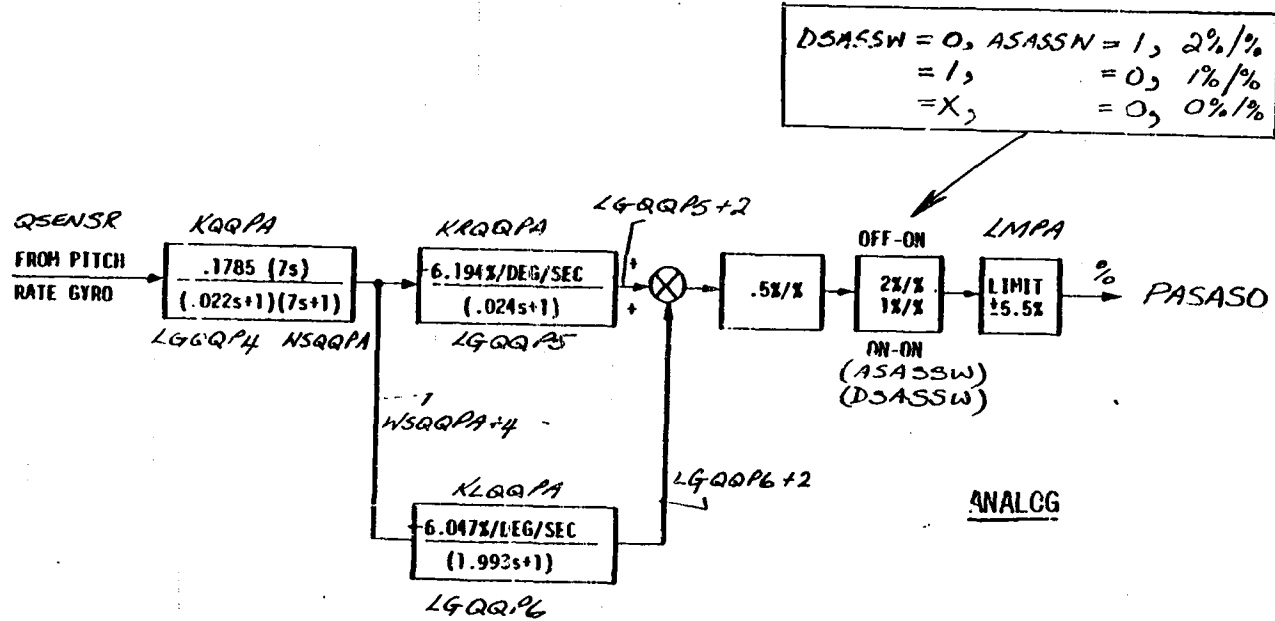
LATERAL ACCELERATION

$$Y. SENS = \frac{AYPS1}{(.0254S^2 + .233S + 1)}$$

FIGURE 5.1.2



DIGITAL

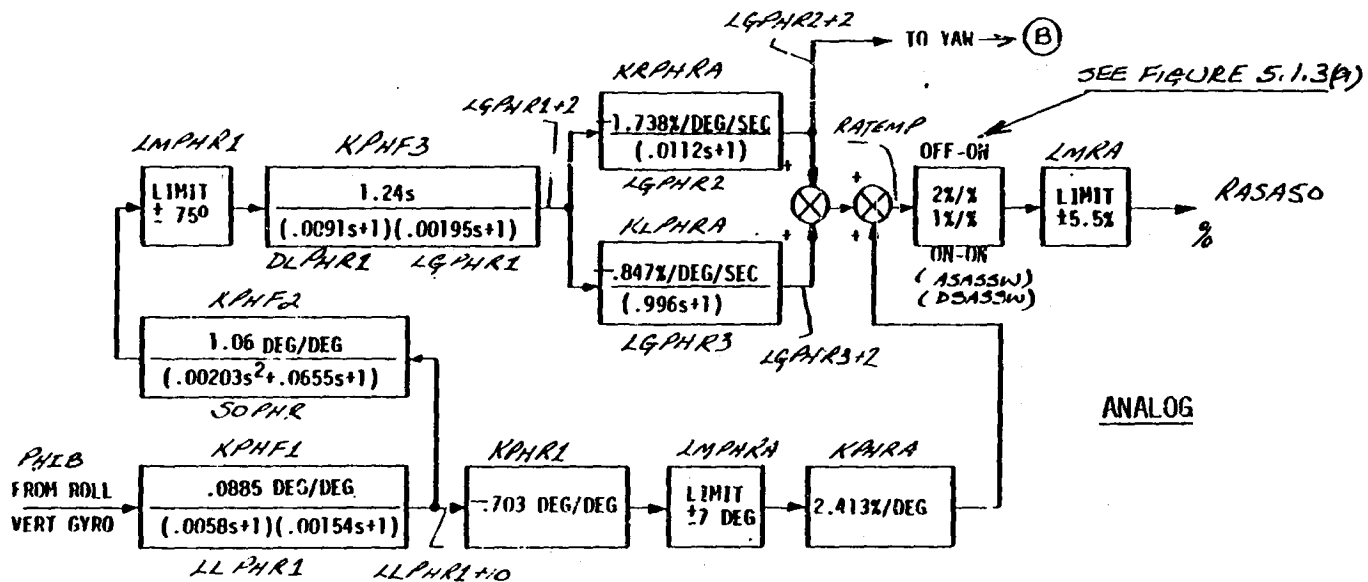
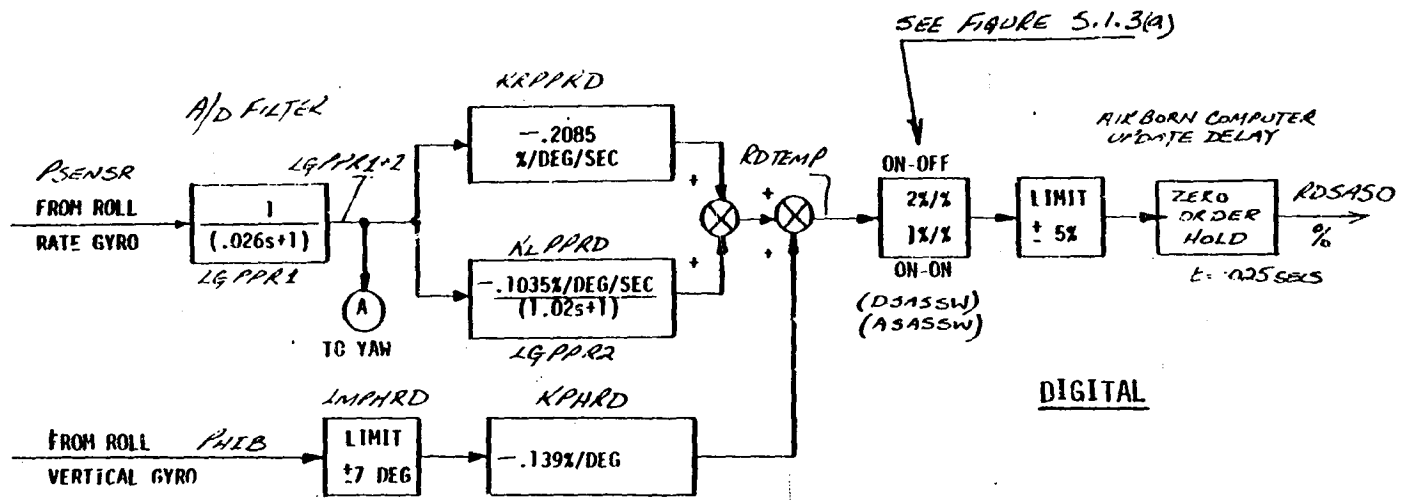


ANALOG

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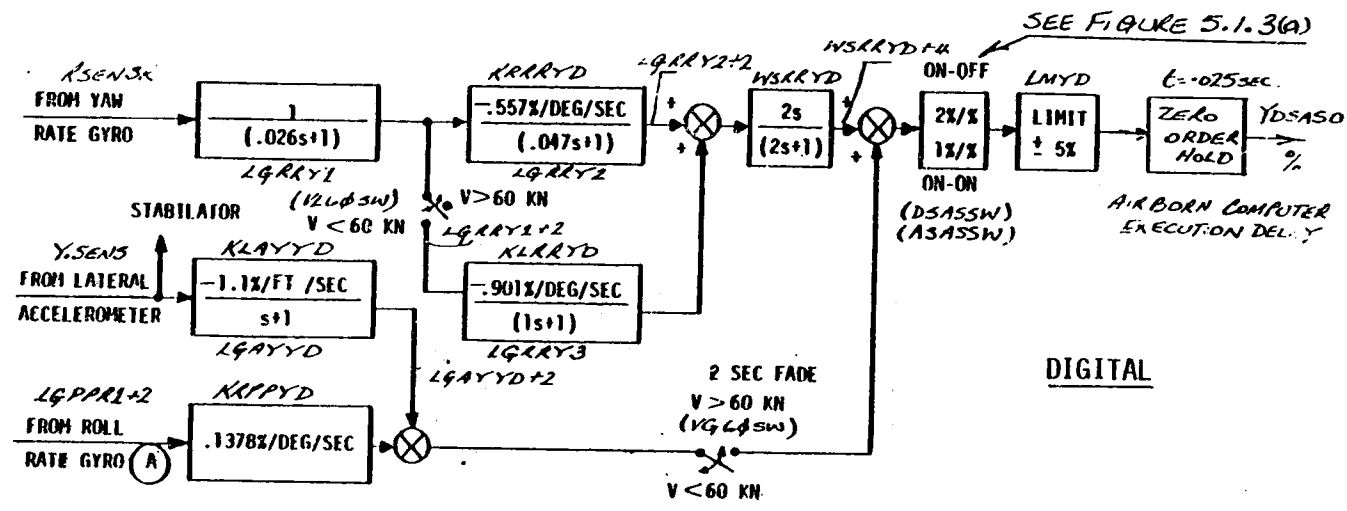
FIGURE 5.1.3(a)

BLACK HAWK PITCH SAS CHANNELS



BLACK HAWK ROLL SAS CHANNELS

FIGURE 5.1.3(b)

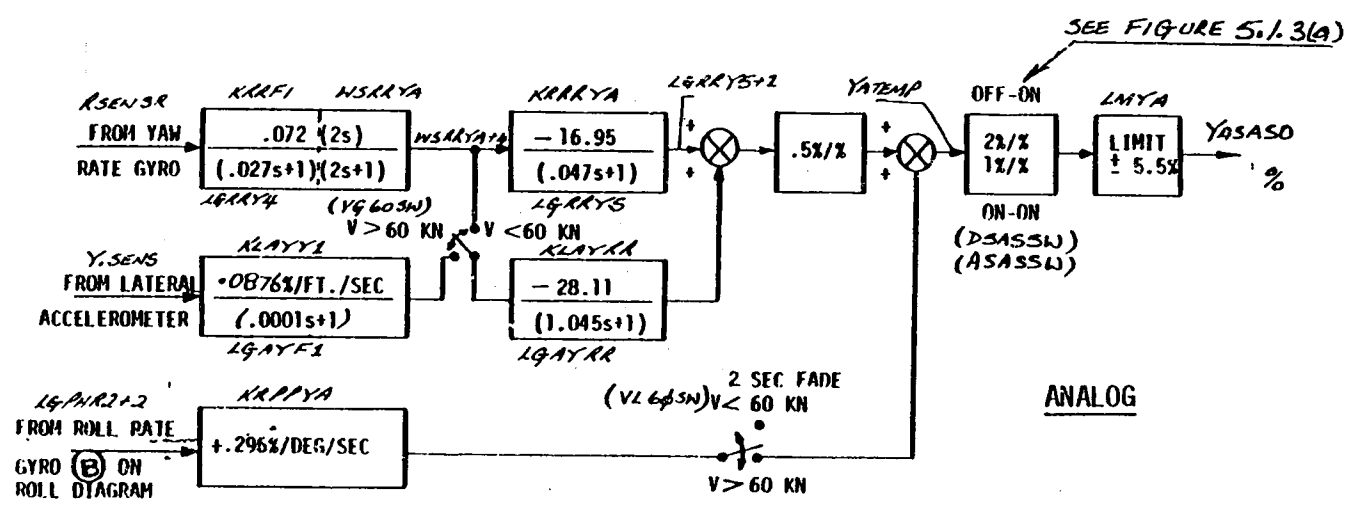


SEE FIGURE 5.1.3(A)

DIGITAL

FIGURE 5.1.3(c)

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SEE FIGURE 5.1.3(A)

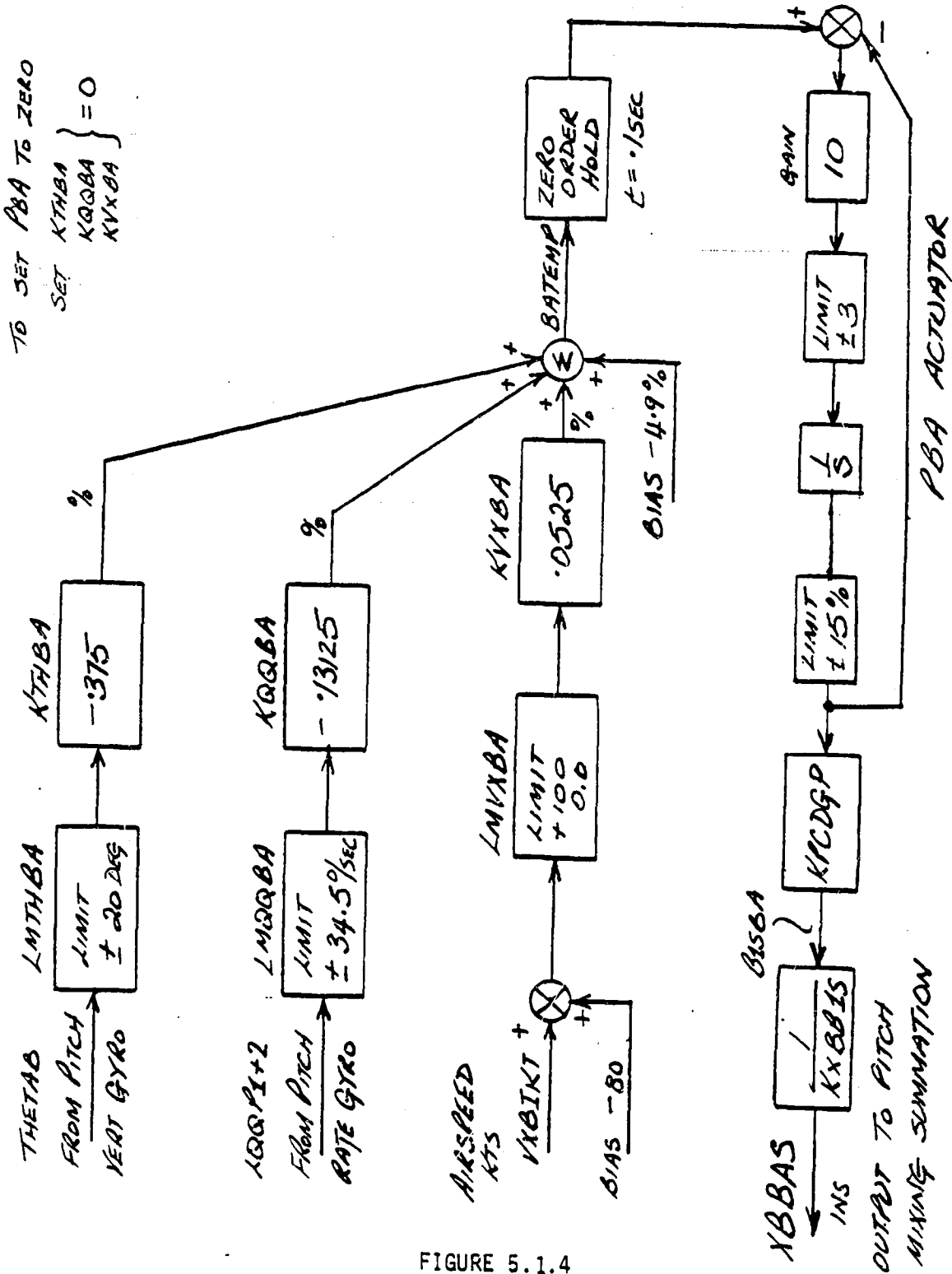
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BLACK HAWK YAW SAS CHANNELS



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BLACK HAWK PITCH BIAS ACTUATOR

FIGURE 5.1.4

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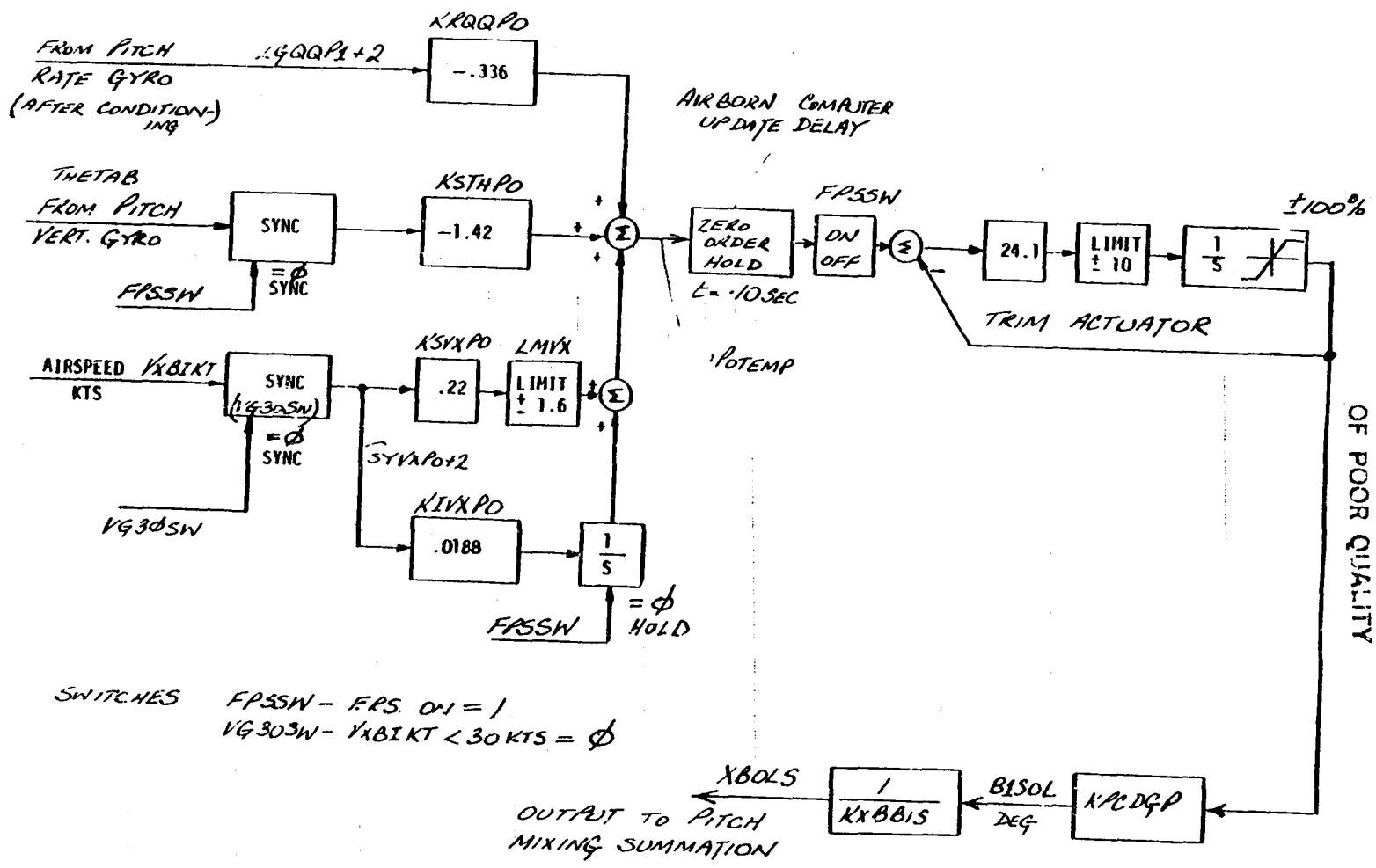
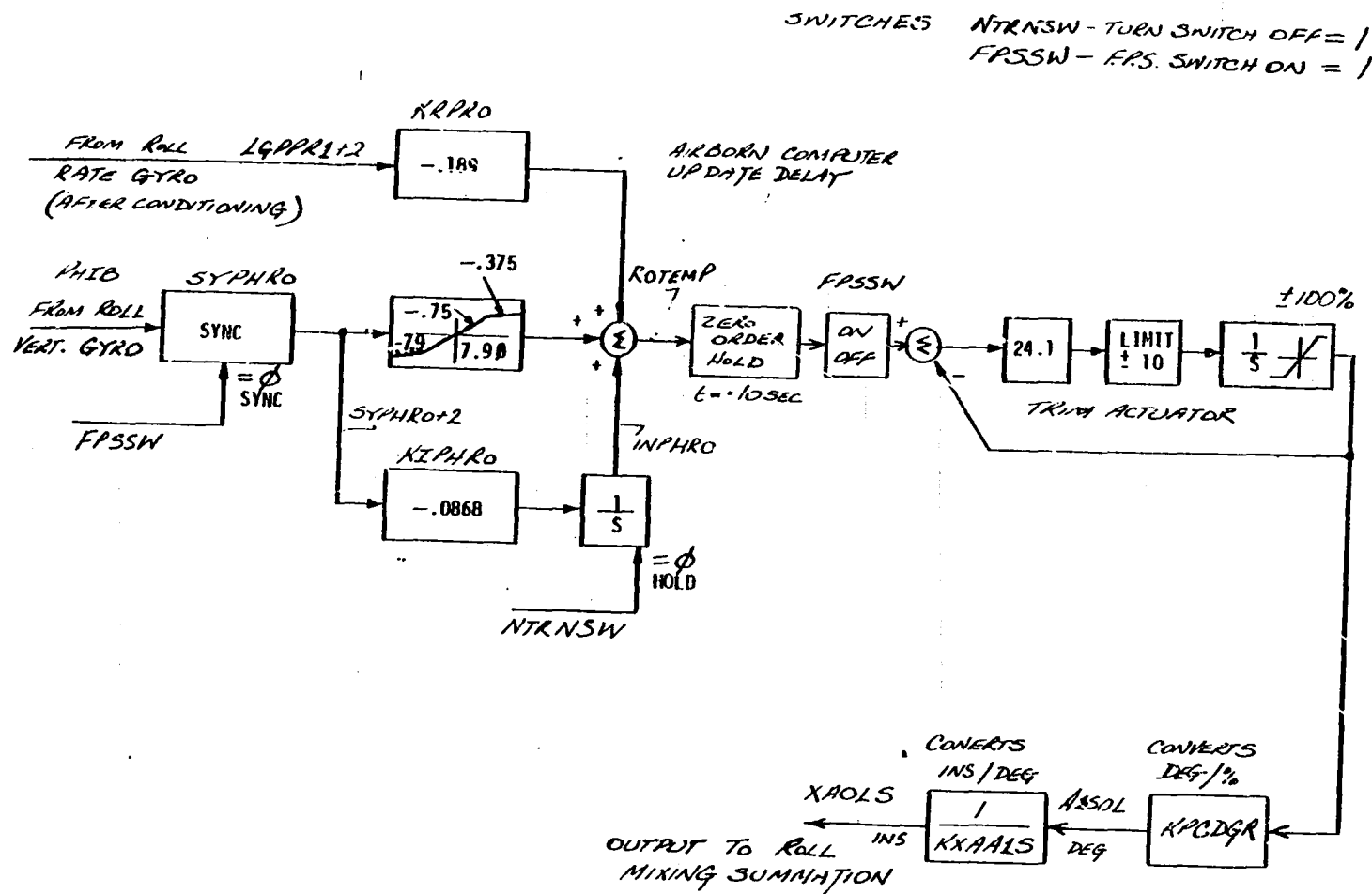


FIGURE 5.1.5(a)

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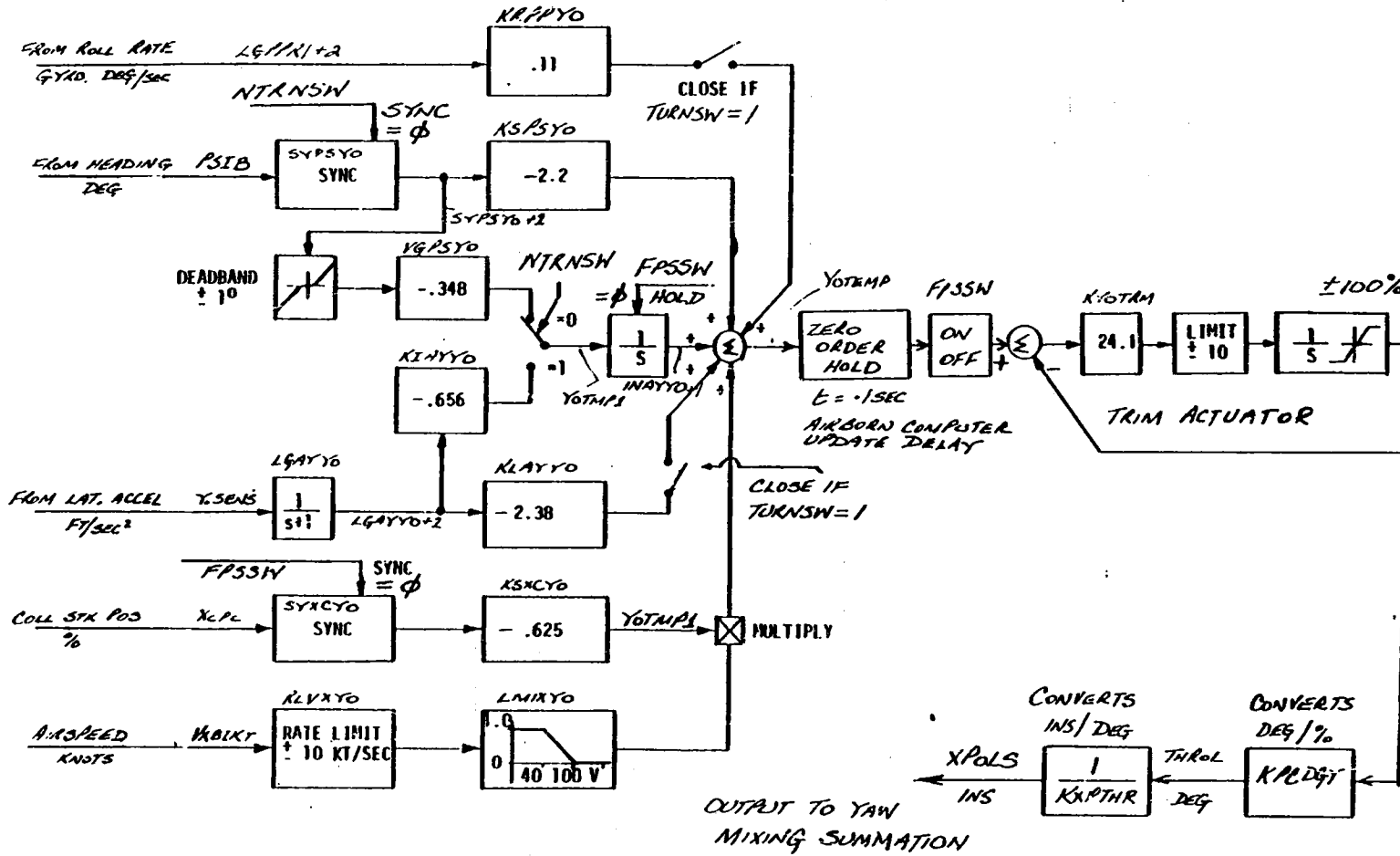
BLACK HAWK PITCH FPS CHANNEL

FIGURE 5.1.5(b)



BLACK HAWK ROLL FPS CHANNEL

FIGURE 5.1.5(c)



BLACK HAWK YAW FPS CHANNEL



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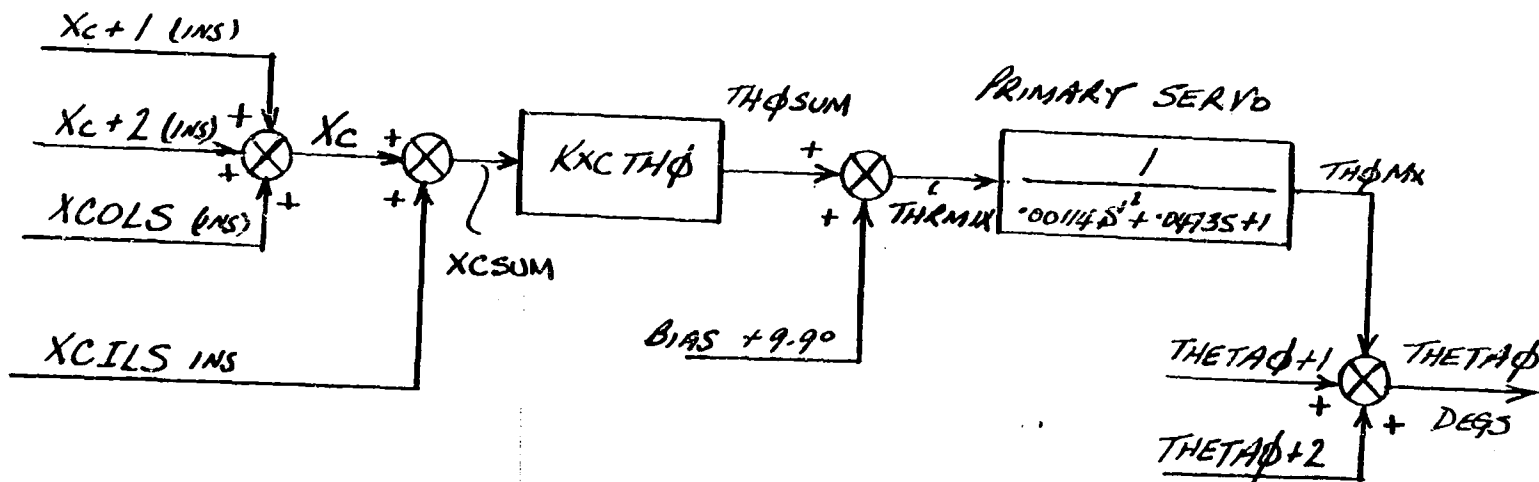


FIGURE 5.1.6(a)

BLACK HAWK COLLECTIVE CONTROL

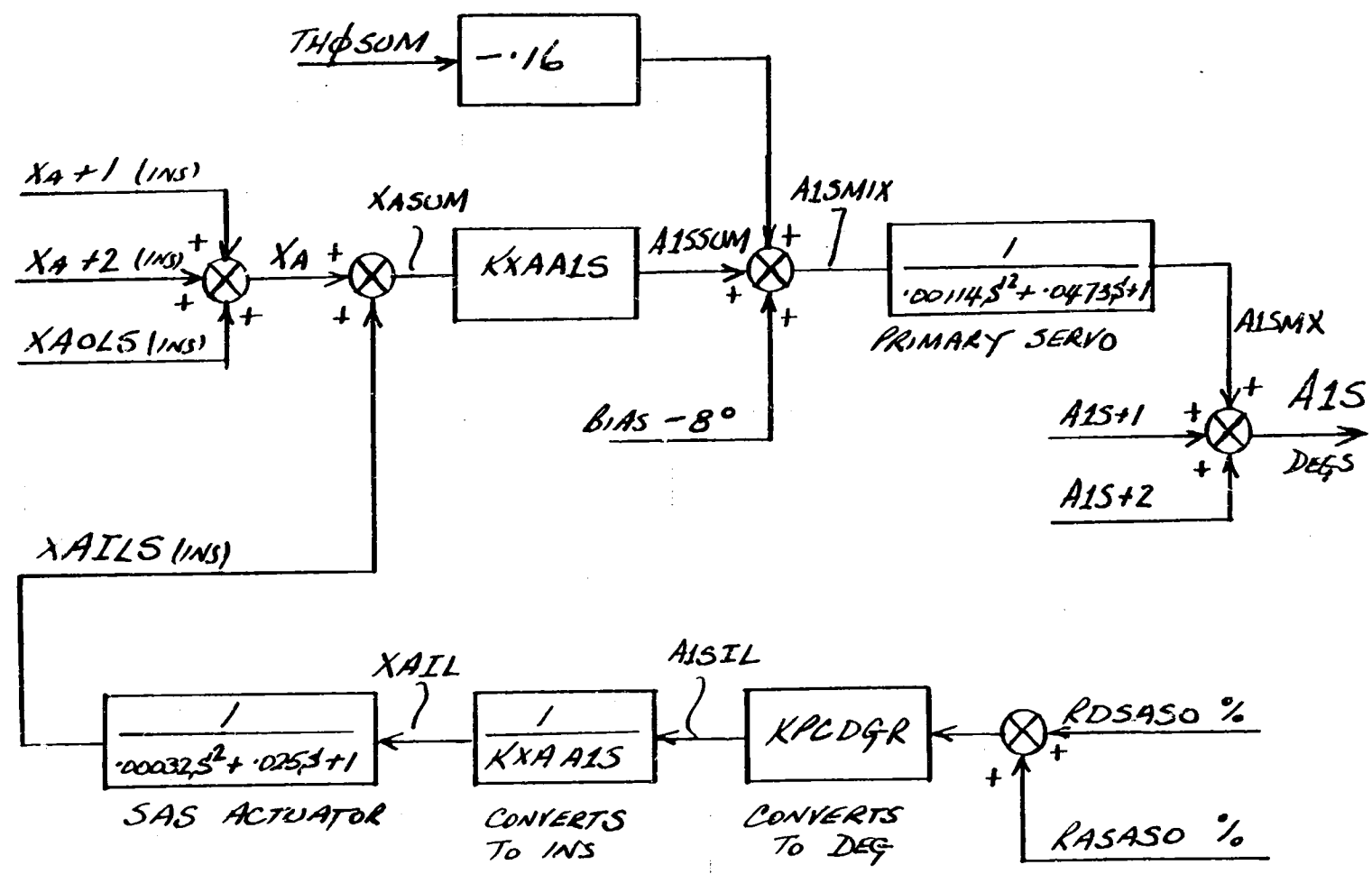
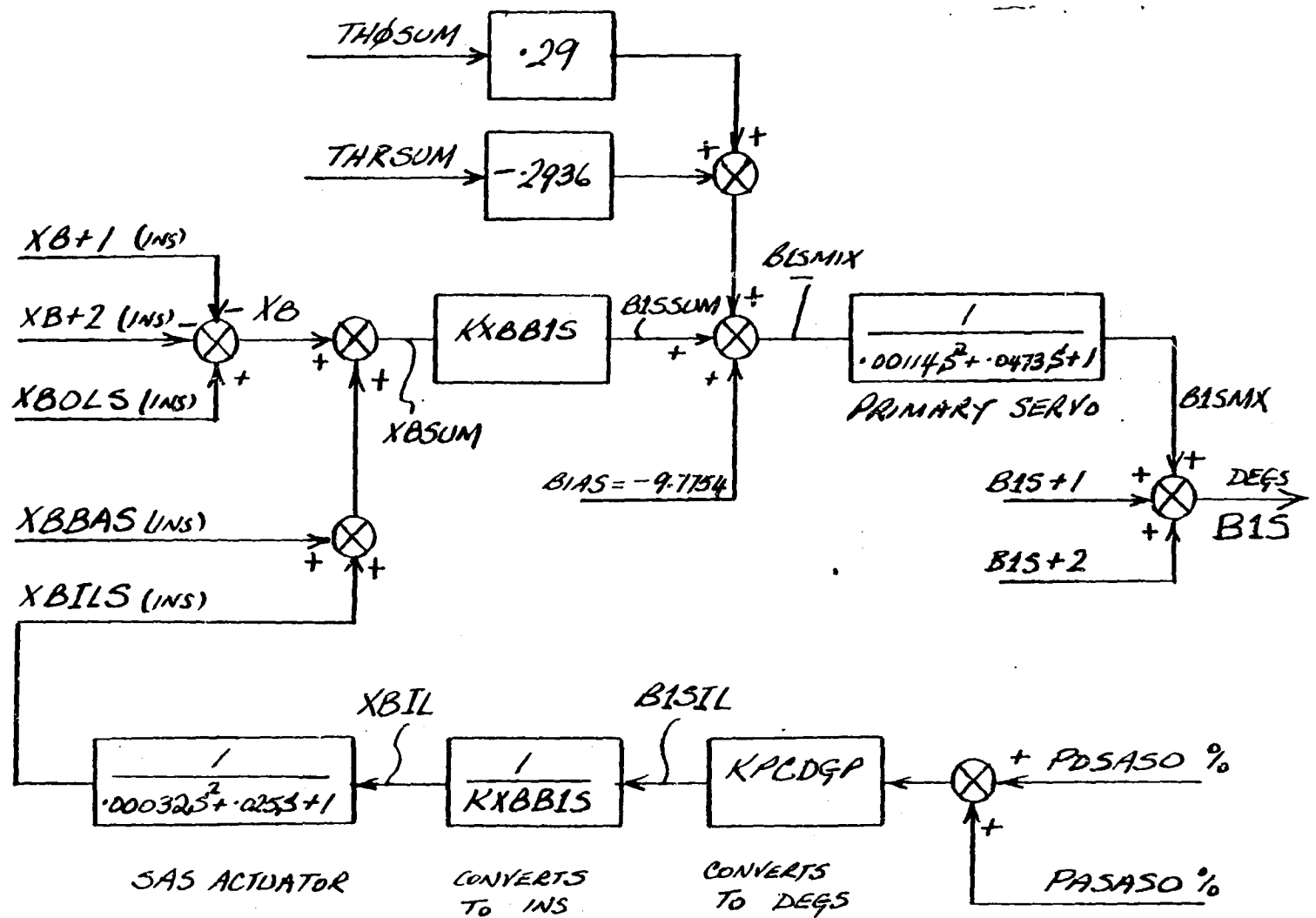


FIGURE 5.1.6(b)

BLACK HAWK LATERAL CYCLIC CONTROL

FIGURE 5.1.6(c)



BLACK HAWK LONGITUDINAL CYCLIC CONTROL

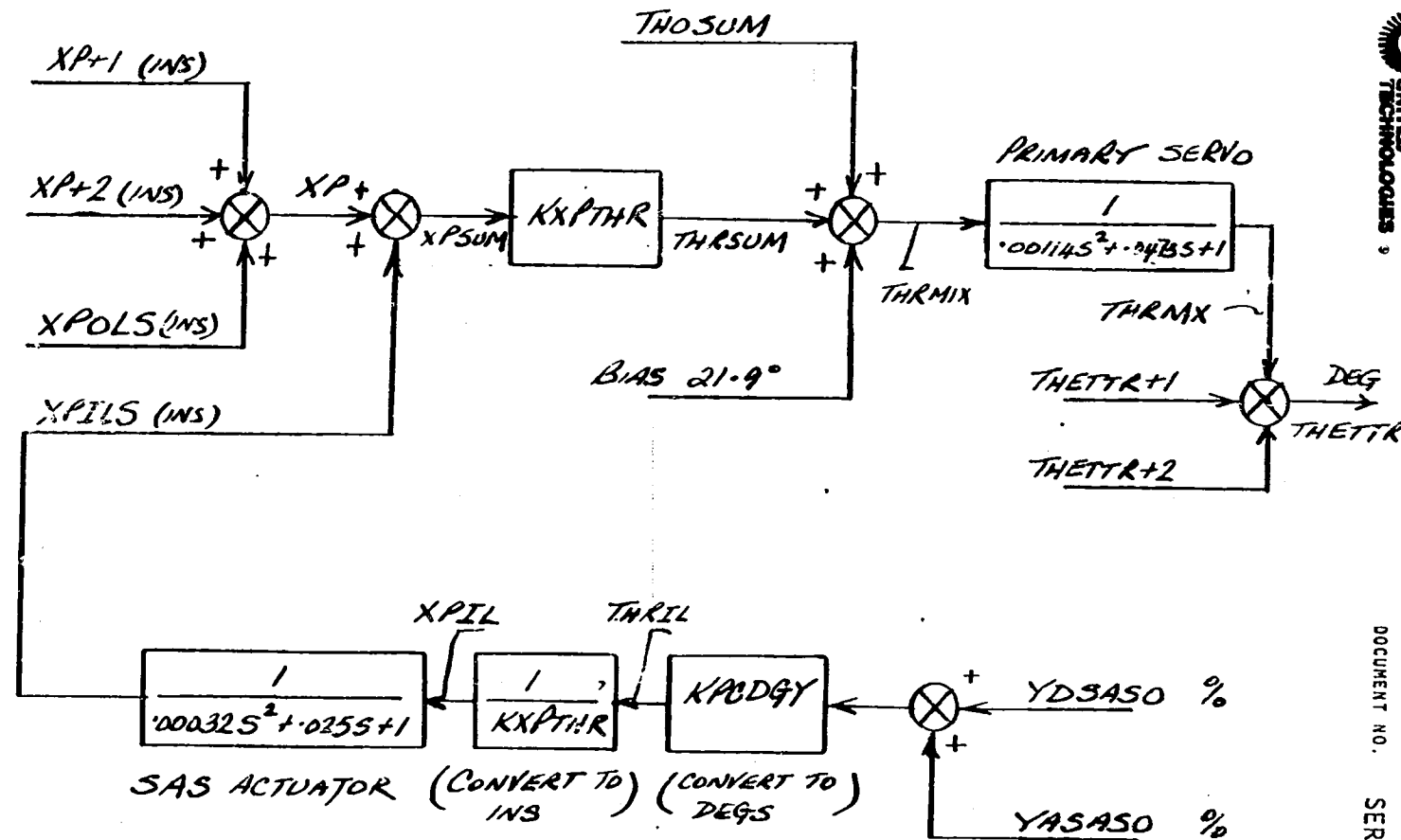


FIGURE 5.1.6(D)

BLACK HAWK TAIL ROTOR COLLECTIVE CONTROL

BLACK HAWK STABILATOR CONTROL SYSTEM

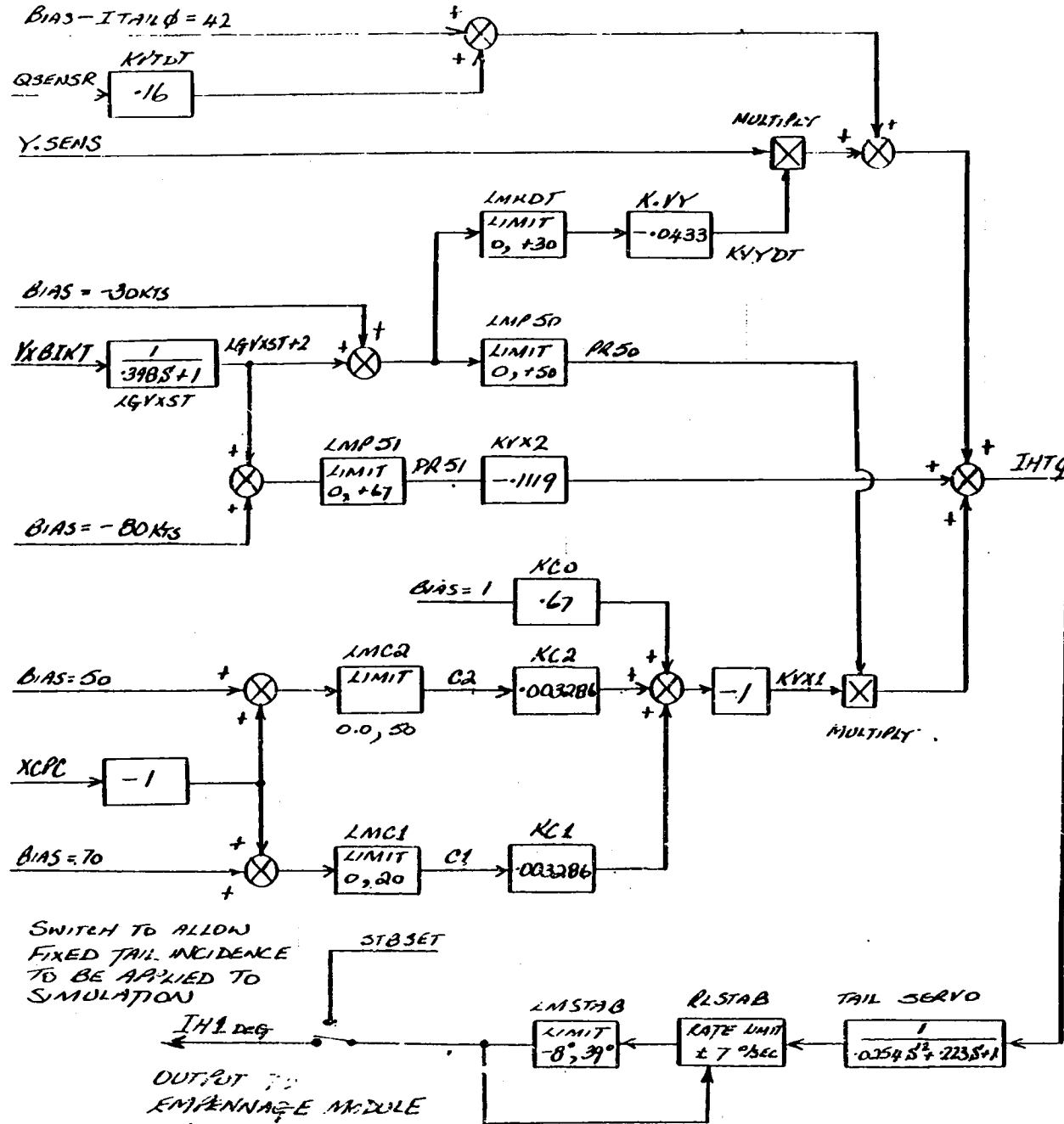


FIGURE 5.1.7
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5.5.3 FLIGHT CONTROLS MODULE INPUT/OUTPUT DATA TRANSFER

INPUT TRANSFER	
PARAMETER	ORIGIN MODULE
Q DEG	MOTION
P DEG	
R DEG	
THETA B	
PHI B	
PSI B	
VXBIKT	
AYPSI	
XA+1	TRIMMER
XB+1	
XC+1	
XP+1	
XA+2	COCKPIT
XB+2	
XC+2	
XP+2	

OUTPUT TRANSFER	
PARAMETER	DESTINATION MODULE
ALS	MAIN ROTOR
BIS	
THETA ϕ	
IHT	EMPENNAGE
IYT	
THETTR	TAIL ROTOR

5.5.4. BLACK HAWK CONTROL SYSTEM GENERAL INPUT

$KPCDGP = -0.283$	PITCH DEG / %
$KPCDGR = 0.16$	ROLL DEG / %
$KPCDGY = -0.298$	YAW DEG / %
$KPCDGC = 0.16$	COLL DEG / %
$KXBBIS = -2.83$	LONG DEG / INCH
$KXAAIS = 1.60$	LAT DEG / INCH
$KXPTR = -5.539$	DIR DEG / INCH
$KXCTH\phi = 1.60$	COLL DEG / INCH

ROTOR HEAD BLADE PITCH TRAVEL LIMITS

FOR GENERAL ANALYTICAL INVESTIGATION HEAD TRAVEL LIMITS ARE INHIBITING. HOWEVER, FOR PILOTED SIMULATION EVALUATION, THE FOLLOWING LIMITS SHOULD BE INTRODUCED TO PROPERLY DEFINE THE RIGGING.

UPPER LIMIT, DEG	LOWER LIMIT, DEG
$AISUL = 8.0$	$AISLL = -8.0$
$BISUL = 16.3$	$BISLL = -12.5$
$TH\phi UL = 25.9$	$TH\phi LL = 9.9$
$THRUL = 36.5$	$THRLL = 4.5$

SEE VOLUME II FOR FURTHER DETAILS

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5.6 ENGINE/FUEL CONTROL MODULE

5.6.1 Module Description

This engine/fuel control model is basically a linearized representation with coefficients which vary as a function of engine operating condition. This model adequately provides for closing the rotor shaft speed loop throughout the normal operating envelope of the helicopter. However, maneuvers which result in significant rotor speed excursions may result in discrepancies in the simulation. All the usual restrictions and assumptions of linear simulation are applicable and should be observed. In an analysis mode deviations from trim are not large, but for pilot-in-the-loop operation some means of continuously synchronizing the steady state engine torque must be developed. This engine module should not be used for engine performance evaluation.

The elements of the model are shown on the simplified block diagram of Figure 6.1.1. They comprise the control interface with Gen. Hel., fuel control, gas turbine, power turbine, and rotor shaft speed degree-of-freedom interface with the Gen. Hel. rotor. This model which derives total S.H.P. at the rotor shaft by appropriately factoring the output from one engine is applicable for evaluations in the governed range. A transmission clutch is represented allowing disconnect of the rotor drive at zero torque required by the rotor. This allows autorotations and recoveries to be executed.

Initialization of the engine/fuel control module is accomplished by using the steady state engine performance required to trim the helicopter simulation in free flight.

A basic background to the complete T.700 engine/fuel control system is given in references 6.6.1 and 6.6.2. A detailed block diagram of the simulation is given in Figure 6.1.2.

The basic engine control system operation is through the interaction of the Electrical (ECU, Figure 6.1.3), and Hydromechanical (HMU) control units. In general, the HMU provides for gas generator speed control and rapid response to power demand. The ECU trims the HMU to satisfy the requirements of the load so as to maintain rotor speed. The Load Demand Spindle (LDS) is a function of collective pitch setting and provides compensation to reduce rotor transient droop. Any steady state errors resulting from inconsistent collective positioning are trimmed by the ECU. In the simulation, this is trimmed by the difference between actual gas generator speed (from steady performance data and that resulting from the collective setting). These characteristics are implemented in the simulation.

In general, isochronous governing of the rotor speed is maintained by developing an error relative to the reference speed and commanding more or less power to stabilize at the required speed. Basically, this process involves the speed error demanding a change in gas generator speed (N_{GG}) via the shaping of the ECU electrical network, Figure 6.1.3. This signal is summed with the LDS input in the HMU, and compared with the actual gas generator speed. The subsequent error, commands changes in fuel flow leading to a higher or lower gas turbine speed and changes in the gas flow. This in turn provides increased or decreased power at the driveshaft from the power turbine. In the simulation, torque output from one engine (Q_{pt}) is derived from three sources. From direct changes in fuel flow, from changes in gas generator speed and as a result of changes in power turbine speed. These increments are summed to obtain a total change in engine torque from trim and subsequently factored by the number of operating engines and engine/rotor gearing ratio, to obtain engine torque output to the rotor shaft.

The interface with the main program is presented on Figures 6.1.1 and 6.1.2. The rotor can be visualized in simple terms as a damper responding to changes in rotor speed. However, the significant effect of the rotor relates to the changes in torque loading as a result of control inputs and changing states. Rotor shaft accelerations result from torque differences in output from the power turbine and torque required by the rotor. The simulation is initialized at trim such that the rotor is at the input speed and $\Delta Q=0$. A clutch is modelled, Figure 6.1.4, which will disengage the rotor from the engine at a zero rotor torque level. When Gen Hel is being executed in combination with the engine, total engine torque must replace rotor torque as a reaction in the airframe equations.

ENGINE/FUEL CONTROL/DRIVE REPRESENTATION

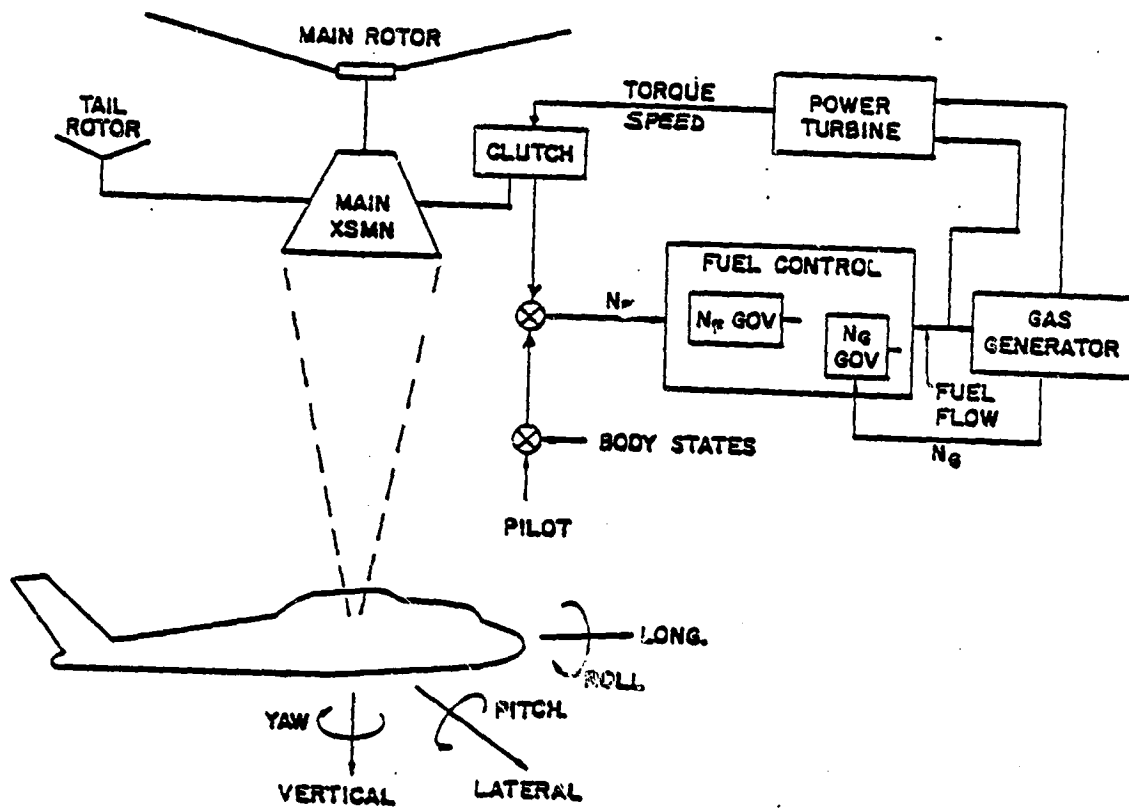
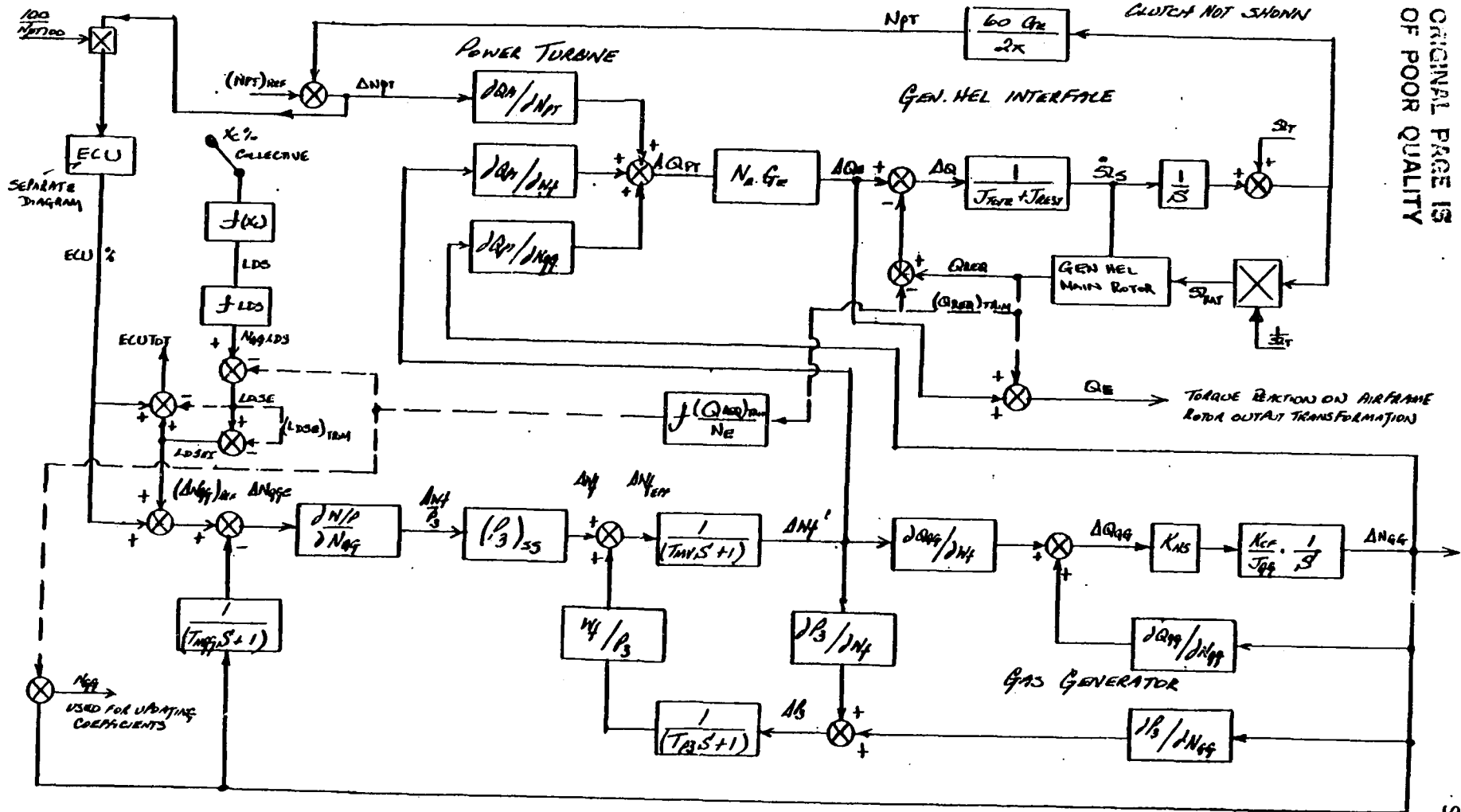


FIGURE 6.1.1

ENGINE/FUEL CONTROL MODULE



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OF POOR QUALITY

FIGURE 6.1.2

5.6.5

SER 70452

FUEL CONTROL
ELECTRIC CONTROL UNIT

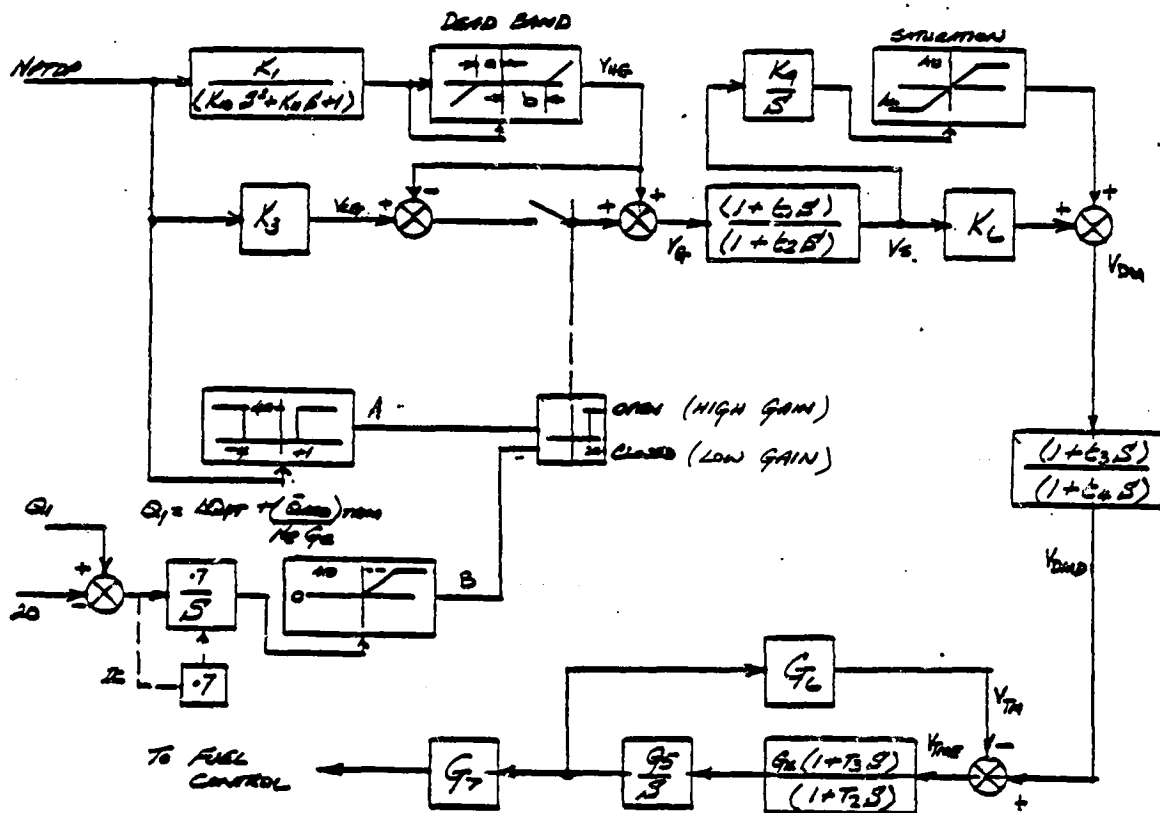


FIGURE 6.1.3

ROTOR DEGREE OF FREEDOM

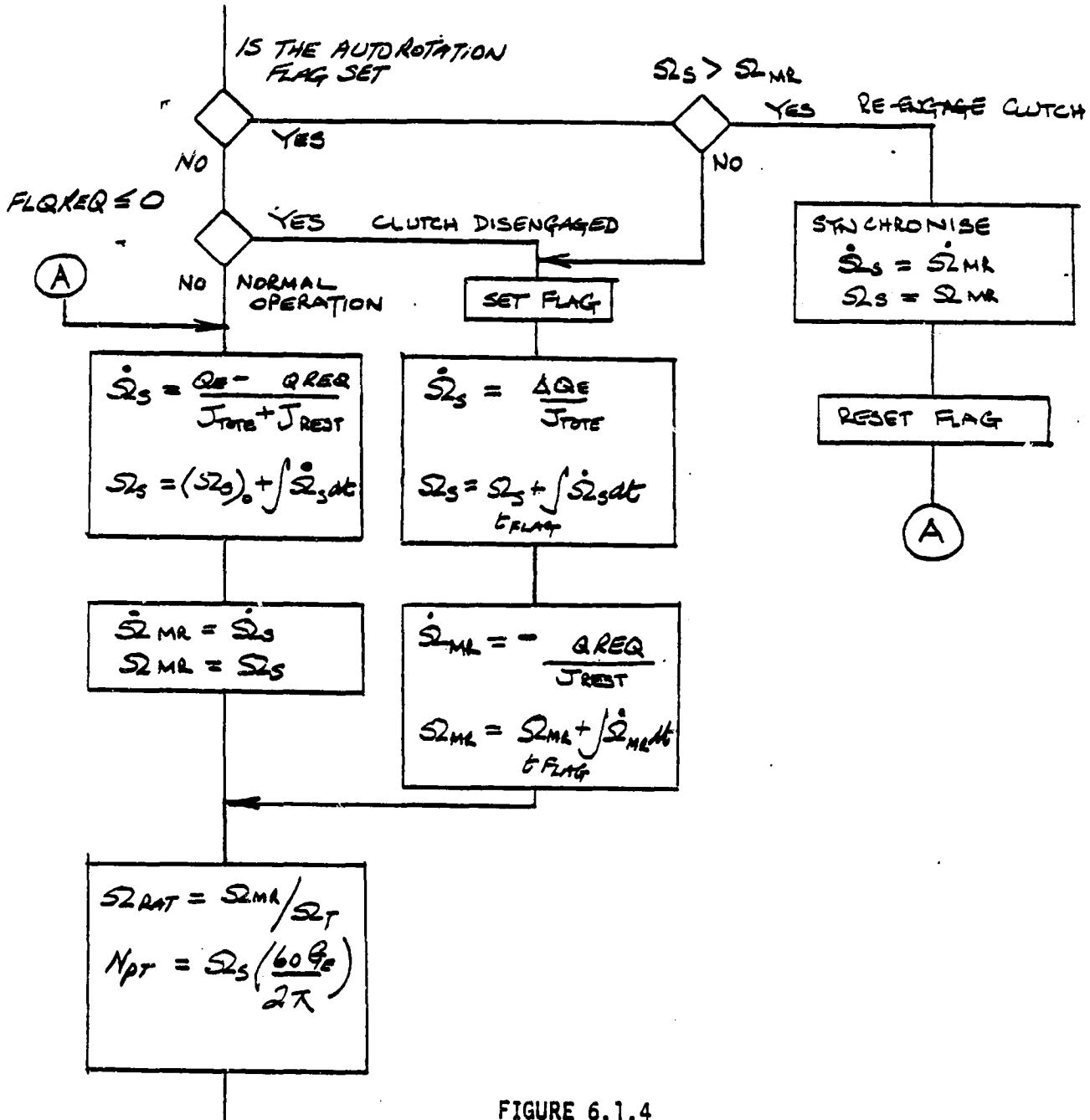


FIGURE 6.1.4

5.6.2 ENGINE/FUEL CONTROL MODULE EQUATIONS

ENGINE/FUEL CONTROL SELECTION LOGIC

A SOFTWARE SWITCH SHOULD BE PROVIDED TO
SELECT THE ENGINE/FUEL CONTROL MODULE

IF THE ENGINE FUEL CONTROL SWITCH IS NOT SET
DO NOT EXECUTE THIS MODULE, RETURN TO MAIN PROGRAM

$$\begin{aligned} \text{SET } S_{2MR} &= S_{2T} \\ Q_E &= Q_{MR} \end{aligned}$$

IF THE ENGINE/FUEL CONTROL SWITCH IS SET EXECUTE
THE FOLLOWING EQUATIONS.

THE FOLLOWING EQUATIONS ARE EXECUTED IN THE TRIMMING MODE

TOTAL TORQUE REQUIRED TO TRIM

$$(\bar{Q}_{REQ})_{TRIM} = Q_{TRIM} (1 + K_Q) \quad , \quad QRQ_{TRIM}$$

WHERE

$$Q_{TRIM} = \frac{(K_{FRQ} - 1)}{K_{FRQ}} Q_{TRIM}^{(b-1)} + \frac{1}{K_{FRQ}} Q_{HMR_E}$$

GAS GENERATOR SPEED FOR TRIM

$$(N_{GG})_{TRIM} = f\left(\frac{(\bar{Q}_{REQ})_{TRIM}}{N_E}\right) \quad , \quad GGSS$$

FROM TABLE 6.5.1, FIGURE 6.5.1

LOAD DEMAND SPINDLE (LDS) OUTPUT

$$N_{GG-LDS} = f(LDSCAM, X_C) \quad , \quad N_{GG-LDS}$$

FROM FIGURES 6.5.2, 6.5.3 TABLES 6.5.2, 6.5.3

WHERE $X_C = (X_{C+1}) + (X_{C+2})$

ELECTRONIC CONTROL UNIT (ECU) BIAS AT TRIM

$$(LDSE)_{TRIM} = (N_{GG-LDS})_{TRIM} - (N_{GG})_{TRIM} \quad , \quad LDSTRM$$

FUEL FLOW AT TRIM

$$(TOTFUL)_{TRIM} = \sum f(N_{Gg})_{TRIM} \text{ per engine, SUM 2}$$

FROM TABLE 6.5.4, FIGURE 6.5.4

REFERENCE POWER TURBINE SPEED

$$(N_{PT})_{REF} = \Omega_{250} * \left(\frac{60 G_E}{2\pi} \right) \text{ RPM}, N_{PTREF}$$

WHERE $\Omega_{250} = \Omega_T$ UNLESS HELICOPTER IS TRIMMED
IN AUTOROTATION WITH SPLIT NEEDLES

$$(N_{PTP})_{REF} = (N_{PT})_{REF} \left(\frac{100}{N_{PT100}} \right) \%, N_{PTREF}$$

THE FOLLOWING EQUATIONS ARE EXECUTED IN THE
COMPUTE MODE - INITIALISE / UPDATE ALL TIME VARYING COEFFICIENTS

$$FLQREQ = \frac{Q_{HMR}(1+K_Q)}{(.3S+1)}$$

$$QREQ = Q_{HMR}(1+K_Q)$$

LOAD DEMAND SPINDLE OUTPUT (LDS)

$$N_{GG LDS} = f(LDSCAM, X_C) \quad , \quad LDS$$

FROM TABLES 6.5.2, 3 FIGURES 6.5.2, 3

LDS INCREMENTAL INPUT FROM TRIM.

$$LDSEI = N_{GG LDS} - (N_{GG})_{TRIM} - (LDSE)_{TRIM}$$

ELECTRICAL CONTROL UNIT

POWER TURBINE SPEED ERROR

$$N_{PTDP} = (N_{PTDP})_{REF} - N_{PT} \frac{100}{(N_{PT100})}$$

SPEED ERROR SHAPING

$$I_{DB1}(S) = N_{PTDP} \frac{K_1}{(K_{10}S^2 + K_{11}S + 1)} \quad , \quad I_{DB1}$$

IF $I_{DB1} \leq -a$

$I_{DB1} \geq b$

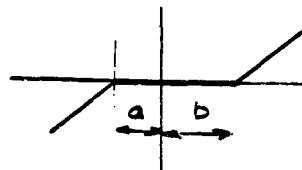
$-a < I_{DB1} < b$

$V_{HG} = I_{DB1} + a$

$V_{HG} = I_{DB1} - b$

$V_{HG} = 0$

IE DEAD BAND



$$V_{LQ} = N_{PTDA} \cdot K_3 \quad , V_{LQ}$$

ERROR SIGNAL TRANSFER LOGIC

IF $A \geq 20$ OR $B \geq 20$ (SWITCH OPEN)

$$V_G = V_{HG} \quad , V_G V_G$$

OTHERWISE (SWITCH CLOSED)

$$V_G = V_{LQ} + V_Q$$

WHERE

IF $-4 \leq N_{PTDA} \leq 1$, $A = 0$

IF $N_{PTD} < -4$ OR $N_{PTDA} > 1$, $A = 40$

$$Q_{INT} = \frac{.7}{S} \left\{ \Delta Q_{PT} + \frac{(\bar{Q}_{REQ})_{TRIM}}{N_E \cdot G_E} - 20 \right\} \quad , Q_{INT}$$

IF $0 \leq Q_{INT} \leq 40$, $B = Q_{INT}$, BBBBB

IF $Q_{INT} < 0$, $B = 0$

IF $Q_{INT} > 40$, $B = 40$

NOTE THAT Q_{INT} MUST BE INITIALISED AT TRIM VALUE OF Q_{INT} I.E. $\Delta Q_{PT}(t-1) = 0$

$$V_S(s) = \left(\frac{1 + t_1 s}{1 + t_2 s} \right) V_G \quad , \quad V_{SKS}$$

$$V_{SS}(s) = \frac{K_9 \cdot V_S}{s} \quad , \quad V_{SS}$$

IF $V_{SS} > A_U$ SET $V_{SS} = A_U$

IF $V_{SS} < A_L$ SET $V_{SS} = A_L$

$$V_{DM} = V_{SS} + K_6 V_S \quad , \quad V_{DM}$$

$$V_{DMD}(s) = \frac{(1 + t_3 s)}{(1 + t_4 s)} \cdot V_{DM} \quad , \quad V_{DMD}$$

$$M(s) = (V_{DMD} - V_{TM}) G_4 \frac{(1 + T_3 s)}{(1 + T_2 s)} \quad , \quad MAMA$$

WHERE $V_{TM} = G_6 \text{INS}_{(t-1)}$

$$\text{INS}(s) = M \frac{G_5}{s} \quad , \quad \text{INS}$$

$$\text{ECU}(s) = G_7 \text{INS} \quad , \quad \text{ECU}$$

TOTAL ECU OUTPUT

$$ECU_{TOT} = ECU(S) + LDSEI - (LDSE)_{TRIM}, \quad ECU_{OUT}$$

GAS GENERATOR REFERENCE SPEED CHANGE REQUIRED.

$$\Delta N_{GG REF} = ECU(S) + LDSEI, \quad GGRPMR$$

GAS GENERATOR SPEED ERROR

$$\Delta N_{GGE} = (\Delta N_{GG})_{REF} - \frac{\Delta N_{GG}(z^{-1})}{(T_{NGG} S + 1)}, \quad GGRPME$$

INCREMENTAL FUEL FLOW DEMANDED

$$\Delta W_f = \left[\frac{P}{3} \right]_{SS} \cdot \left(\frac{\partial W_f / P}{\partial N_{GG}} \right) \Delta N_{GGE}, \quad FUEL$$

$\left(\frac{P}{3} \right)_{SS}$ FROM TABLE 6.55, FIGURE 6.5.5

GAS GENERATOR SECTION

CHANGE IN COMPRESSOR DISCHARGE PRESSURE

$$\Delta P_3 = \left[\frac{\partial P_3}{\partial W_f} \right] \cdot \Delta W_f'_{(t-1)} + \left[\frac{\partial P_3}{\partial N_{GG}} \right] \cdot \Delta N_{GG}_{(t-1)}$$

$\frac{\partial P_3}{\partial W_f}, \frac{\partial P_3}{\partial N_{GG}}$ FROM TABLE 6.5.6, FIGURES 6.5.6 6.5.7

EFFECTIVE INCREMENTAL FUEL FLOW DEMANDED

$$\Delta W_{fEFF}(S) = \Delta W_f + \left\{ \left[\frac{\Delta W_f}{\Delta P_3} \right] \Delta P_3 \right\} \frac{1}{(T_{P3}S + 1)}, \text{ INCWF}$$

$\frac{\Delta W_f}{\Delta P_3}$ FROM TABLE 6.5.6, FIGURE 6.5.8

ACTUAL METERED FUEL FLOW

$$\Delta W_f'(S) = \frac{\Delta W_{fEFF}}{(T_{MV}S + 1)}, \text{ ACTWF}$$

TOTAL FUEL FLOW (OUTPUT ONLY)

$$TOTFUL = (TOTFUL)_{TRM} + \Delta W_f', \text{ TOTFUL}$$

GAS GENERATOR ACCELERATING TORQUE

$$\Delta Q_{GG} = \left[\frac{\partial Q_{GG}}{\partial W_f} \right] \cdot \Delta W_f' + \left[\frac{\partial Q_{GG}}{\partial N_{GG}} \right] \cdot \Delta N_{GG}_{(t-1)}, \text{ GGD}$$

$\frac{\partial Q_{GG}}{\partial W_f}, \frac{\partial Q_{GG}}{\partial N_{GG}}$ FROM TABLE 6.5.6, FIGURES 6.5.9, 6.5.10

INCREMENTAL CHANGE IN GAS GENERATOR SPEED FROM TRIM

$$\Delta N_{GG} = K_{HS} \cdot \left(\frac{K_{CF}}{J_{GG}} \right) \int \Delta Q_{GG} dt, \text{ GGRAMD}$$

ACTUAL GAS GENERATOR SPEED (FOR UPDATING TIME VARYING COEFFICIENTS)

$$N_{GG} = (N_{GG})_{TRIM} + \Delta N_{GG}, \text{ GGRPM}$$

POWER TURBINE SECTION

$$\Delta N_{PT} = N_{PTOP} * \left(\frac{N_{PT100}}{100} \right)$$

POWER TURBINE TORQUE PER ENGINE FROM TRIM

$$\Delta Q_{PT} = \left[\frac{\partial Q_{PT}}{\partial W_f} \right] \cdot \Delta W_f' + \left[\frac{\partial Q_{PT}}{\partial N_{PT}} \right] \cdot \Delta N_{PT} + \left[\frac{-\partial Q_{PT}}{\partial N_{GG}} \right] \cdot \Delta N_{GG}, \text{ PTQ}$$

$\frac{\partial Q_{PT}}{\partial W_f}$, $\frac{\partial Q_{PT}}{\partial N_{PT}}$, $\frac{\partial Q_{PT}}{\partial N_{GG}}$ FROM TABLE 6.5.6, FIGURES 6.5.11
6.5.12
6.5.13

TOTAL INCREMENTAL TORQUE OUTPUT

(AT ROTOR SPEED)

$$\Delta Q_E = N_E \cdot G_E \cdot \Delta Q_{PT}, \text{ PE}$$

TOTAL TORQUE OUTPUT

$$Q_E = \Delta Q_E + (\bar{Q}_{REQ})_{TRIM}, \text{ QHEG}$$

THIS IS THE TORQUE REACTION ON THE AIRFRAME

ROTOR SPEED DEGREE OF FREEDOM

NORMAL OPERATION

$$\dot{\Omega}_s = \frac{Q_E - Q_{REQ}}{(J_{TOT} + J_{REST})}$$

$$\Omega_s = \Omega_{s0} + \int \dot{\Omega}_s dt$$

WHERE $\Omega_{s0} = \Omega_T$

THEN $\dot{\Omega}_{MR} = \dot{\Omega}_s$

$$\Omega_{MR} = \Omega_s$$

$$\Omega_{RAT} = \Omega_{MR} / \Omega_T$$

$$N_{AT} = \Omega_s \cdot \left(\frac{60 Q_E}{2\pi} \right)$$

$$N_{PT} = N_{PT} \left(\frac{100}{N_{PT100}} \right)$$

CLUTCH DISENGAGEMENT

IF ROTOR CLUTCH WAS ENGAGED ON THE PREVIOUS PROGRAM PASS, AND

IF $FLQ_{REQ} \leq 0$

DISENGAGE THE CLUTCH AND SET FLAG

THEN

$$\dot{S}_S = \frac{\Delta Q_E}{J_{TOTE}}$$

$$S_S = S_{S \text{ FLAG}} + \int S_S dt$$

$$\dot{S}_{MR} = - \frac{Q_{REQ}}{J_{REST}}$$

$$S_{MR} = S_{MR \text{ FLAG}} + \int S_{MR} dt$$

$S_{S \text{ FLAG}}$, $S_{MR \text{ FLAG}}$ ARE THE VALUES OF S_S AND S_{MR} WHEN THE DISENGAGE FLAG IS SET.

CLUTCH ENGAGEMENT

IF THE ROTOR CLUTCH WAS DISENGAGED ON THE PREVIOUS PROGRAM PASS, AND

IF $S_S \geq S_{MR}$, RE-ENGAGED THE CLUTCH
RESET FLAG

SYNCHRONISE $\dot{S}_{S_e} = \dot{S}_{MR_e}$

$$S_{S_e} = S_{MR_e}$$

CONTINUE AS FOR NORMAL OPERATION

5.6.3 ENGINE MODULE INPUT/OUTPUT DATA TRANSFER

INPUT TRANSFER	
PARAMETER	ORIGIN MODULE
QHMR	MAIN ROTOR
XC+1	TRIMMER
XC+2	COCKPIT

OUTPUT TRANSFER	
PARAMETER	DESTINATION MODULE
OMR.MR	MAIN ROTOR
OMR.MR	
QH BEG	

5.6.4

NOTATION FOR THE ENGINE/FUEL CONTROL MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
$(\bar{Q}_{REQ})_{TRIM}$	QRQTRM	1b FT	Total torque required at trim.
Q_{TRIM}	QTRIM	1b FT	Filtered torque required by the main rotor.
K_Q	KTRQ	-	Factor to account for tail rotor torque plus losses
Q_{MR}	QHMR	1b FT	Actual rotor torque required.
N_{GG}	GGRPM	% RPM	Gas generator speed
$(N_{GG})_{TRIM}$	GGSS	% RPM	Gas generator speed at trim.
N_E	NETOT	-	Number of engines.
N_{GGLDS}	NGGLDS	% RPM	Load demand spindle output.
L_{DSCAM}	LDS	DEG	Load demand spindle rotation.
X_C	XCPC	% COLL	Collective stick position.
$(L_{DSE})_{TRIM}$	LDSTRM	% RPM	Synchronized LDS output at trim.
$(TOTFUL)_{TRIM}$	SUM2	LB/HR	Total fuel flow.
$(N_{PT})_{REF}$	NPTREF	RPM	Power turbine reference speed, RPM. (Set by sync with rotor speed at trim).
N_{PT100}	NPT100	RPM	100% power turbine speed
L_{DSEI}			LDS incremental input from trim.
N_{PTD}	NPTDP	% RPM	Power turbine speed error.
N_{PT}	NPT	% RPM	Actual power turbine speed.
Q_{TRIM}	QTRIM	1b FT	Filtered Rotor Torque
$FLQREQ$	FLQREQ	1b FT	Filtered Rotor Torque used in the clutch model.

NOTATION FOR THE ENGINE/FUEL CONTROL MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
I _{DB1}	IDB1		Electronic Control Unit Shaping Network Variables
V _{HG}	VHG		
V _{LG}	VLG		
V _G	VGVG		
Q _{INT}	QINT		
B	BBBBB		
V _S	VSVS		
V _{SS}	VSS		
V _{DM}	VDM		
V _{DMD}	VDMD		
M	MAMA		
V _{TM}	VTM		
I _{NS}	INS		
E _{CU}	ECU	% RPM	
G _E	GER	-	Power turbine to rotor gearing.
ECU _{TOT}	ECUTOT	% RPM	Total ECU output
K ₁₋₋₋₋₋₁₁	K ₁₋₋₋₋₋₁₁	-	ECU gain constants
t ₁₋₋₋₋₋₄		SECS	ECU time constants
T _{2, T3}		SECS	ECU time constants
a, b,		-	ECU dead band definition
A, B		-	ECU logic variables
G ₄₋₋₋₇		-	ECU gain constants
Δ ^N GG REF	GGRPMR	% RPM	Demanded change in G.G. reference speed.

5.6,4 (Cont'd.)

NOTATION FOR THE ENGINE/FUEL CONTROL MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
L_{DSEI}		% RPM	Increment in LDS output from trim.
ΔN_{GGE}	GGRPME	% RPM	G.G. speed error.
T_{NGG}	TNGG	SEC.	SPD servo time constant
ΔW_f	FUEL	LB/HR	Incremental fuel flow demanded
$\Delta W_f'$	ACT WF	LB/HR	Actual metered increment in fuel flow.
ΔP_3	P3D		Change in compressor discharge pressure.
T_{P3}	TP3	SEC.	Lag in fuel feedback loop.
ΔW_{Eff}	INCF	LB/HR	Compensated demand in fuel flow
T_{MV}	TMV	SEC.	Fuel metering valve time constant
ΔQ_{GG}	GGQD	LB/FT	Gas generator accelerating torque
ΔN_{GG}	GGRPMD	% RPM	Incremental change in gas generator speed.
K_{CF}		-	Conversion factor.
J_{GG}		SLUGS FT ²	Inertia of G.G. rotor.
ΔQ_{PT}	PTQ	LB FT	P.T. incremental torque output per engine.
ΔQ_E	PTE	LB FT	P.T. incremental torque output (at rotor speed)
Q_E	QHEG	LB FT	P.T. total torque output
Ω_s	OSHAFF.	RADS/SEC ²	Rotor shaft acceleration
Ω_s	OSHAFT	RADS/SEC	Actual shaft speed.
Ω_{SO}	OMSFO	RADS/SEC	Initial shaft speed.
ΔNPT	NPTD		Incremental P.T. Speed From Ref.

NOTATION FOR THE ENGINE/FUEL CONTROL MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
Ω_0	OMMRO	RADS/SEC	Trim rotor speed.
$\dot{\Omega}_{MR}$	OMRAT.	RADS/SEC ²	Rotor hub acceleration
Ω_{MR}	OMGRR.	RADS/SEC	Rotor hub speed.
Ω_{RAT}		-	Rotor speed relative to trim.
J_{TOTE}	JTOTE	SLUGS FT ²	Inertia upstream of clutch
J_{REST}	JREST	SLUGS FT ²	Inertia downstream of clutch less blades
KHS	KHS		Gas generator heat sink factor

5.6.5 BLACKHAWK / T700 ENGINE MODULE INPUT DATA

INPUT CONSTANTS

K_R	=	.1	T_{NEG}	=	.02 SEC
K_{FRQ}	=	15	$\frac{\partial N_{EG}}{\partial N_{GT}}$	=	.25
N_E	=	2	T_{P3}	=	.01
G_E	=	76.0216	$\frac{K_{GF}}{J_{GG}}$	=	.479 % RPM / FT LB
S_{LT}	FROM ROTOR MODULE				
N_{PT100}	=	20,000 RPM	K_{HS}	=	.65
K_1	=	3.0 V/%	J_{TOTE}	=	690.6 SLUGS FT ²
K_2	=	.6 V/%	J_{REST}	=	943.9 SLUGS FT ²
K_3	=	.059 V/FT LB	N_{GG100}	=	44700 RPM
K_4	=	.2 V/V	T_{MV}	=	.004 SEC
K_5	=	.08 V/V-SEC.			
K_{10}	=	.020408			
K_{11}	=	.1			
a	=	0.4 V			
b	=	0.8 V			
t_1	=	.55 SEC			
t_2	=	.107 SEC			
t_3	=	.08 SEC			
t_4	=	1.58 SEC			
A_U	=	0			
A_L	=	-3.5			
G_4	=	564 ma/V			
G_5	=	.0159 ins/ma			
G_6	=	56 V/ins			
G_7	=	336 %/in			
T_2	=	.2 SEC			
T_3	=	.04 SEC			

INPUT FUNCTIONS

TABLE 6.5.1 STEADY STATE ENGINE PERFORMANCE

$(\bar{Q}_{RED})_{TRIM} / NE$ 16 FT	$(N_{99})_{TRIM}$ % RPM
0	70
6000	84.6
12000	89.9
18000	94.0
24000	97.2
30000	100.1
36000	102.6

TABLE 6.5.2 COLLECTIVE/STATIC DROOP COMPENSATION RIGGING

Xc %	LDS DEGS
0	0
10	8.5
20	17.0
30	25.5
40	34.0
50	42.5
60	51.0
70	59.5
80	68.0
90	76.5
100	85.0

TABLE 6.5.3 LOAD DEMAND SPINDLE CAM OUTPUT

LDS DEGS	N _{GG} LDS %
0	78.6
20	85.0
40	91.3
60	97.8
80	100.0
100	100.0

$N_{GG} = 100\%$ @ $LDS = 67^\circ$

TABLE 6.5.4 STEADY FUEL FLOW REQUIRED

(N _{GG}) _{SS} %	(TOT FUL) _{SS} lb/hr
65	120
70	140
75	166
80	210
85	280
90	390
95	575
100	790

TABLE 6.5.5 DISCHARGE PRESSURE FOR STEADY CONDITIONS

N_{gg} %	$(P_3)_{SS}$ lb/in ²
70	69
75	78
80	94
85	118
90	149
95	184
100	226

TABLE 6.5.6 TIME VARYING COEFFICIENTS

N_{gg} %	$\frac{\partial P_3}{\partial W_f}$	$\frac{\partial P_3}{\partial N_{gg}}$	$\frac{\Delta W_f}{\Delta P_3}$	$\frac{\partial Q_{gg}}{\partial W_f}$	$\frac{\partial Q_{gg}}{\partial N_{gg}}$
UNITS	$\frac{lb/in^2}{lb/hr}$	$\frac{lb/in^2}{\% RPM}$	$\frac{lb/hr}{lb/in^2}$	$\frac{Ft lb}{lb/hr}$	$\frac{Ft lb}{\%}$
70	.051	5.10	2.60	.29	-1.10
75	.071	4.30	2.50	.285	-2.1
80	.078	3.97	2.40	.283	-3.1
85	.072	4.20	2.32	.273	-4.3
90	.089	4.73	2.49	.261	-5.65
95	.080	4.97	2.75	.200	-7.20
100	.075	1.71	3.20	.173	-4.62

TABLE 6.5.6 (CONTINUED)

N_{99} %	$\frac{\delta QPT}{\delta N_{99}}$	$\frac{\delta QA}{\delta N_{99}}$	$\frac{\delta QPT}{\delta N_{99}}$
UNITS	$\frac{FT LB}{lb/hr}$	$\frac{FT LB}{RPM}$	$\frac{FT LB}{\% RPM}$
70	.24	.0045	0
75	.325	.0065	4.8
80	.358	.0086	7.0
85	.363	.0094	8.3
90	.387	.0107	9.9
95	.380	.0132	11.8
100	.331	.0151	5.5

BLACK HAWK STEADY STATE ENGINE PERFORMANCE

MAP NAME: MAPA1
 MAP TYPE: UVA
 INPUT VARIABLE(S): QRQTRM/ENGINE
 OUTPUT VARIABLE: GGSS
 PRIMARY MAP:

0.00 LOWER LIMIT
 36000.00 UPPER LIMIT
 5000.00 DELTA

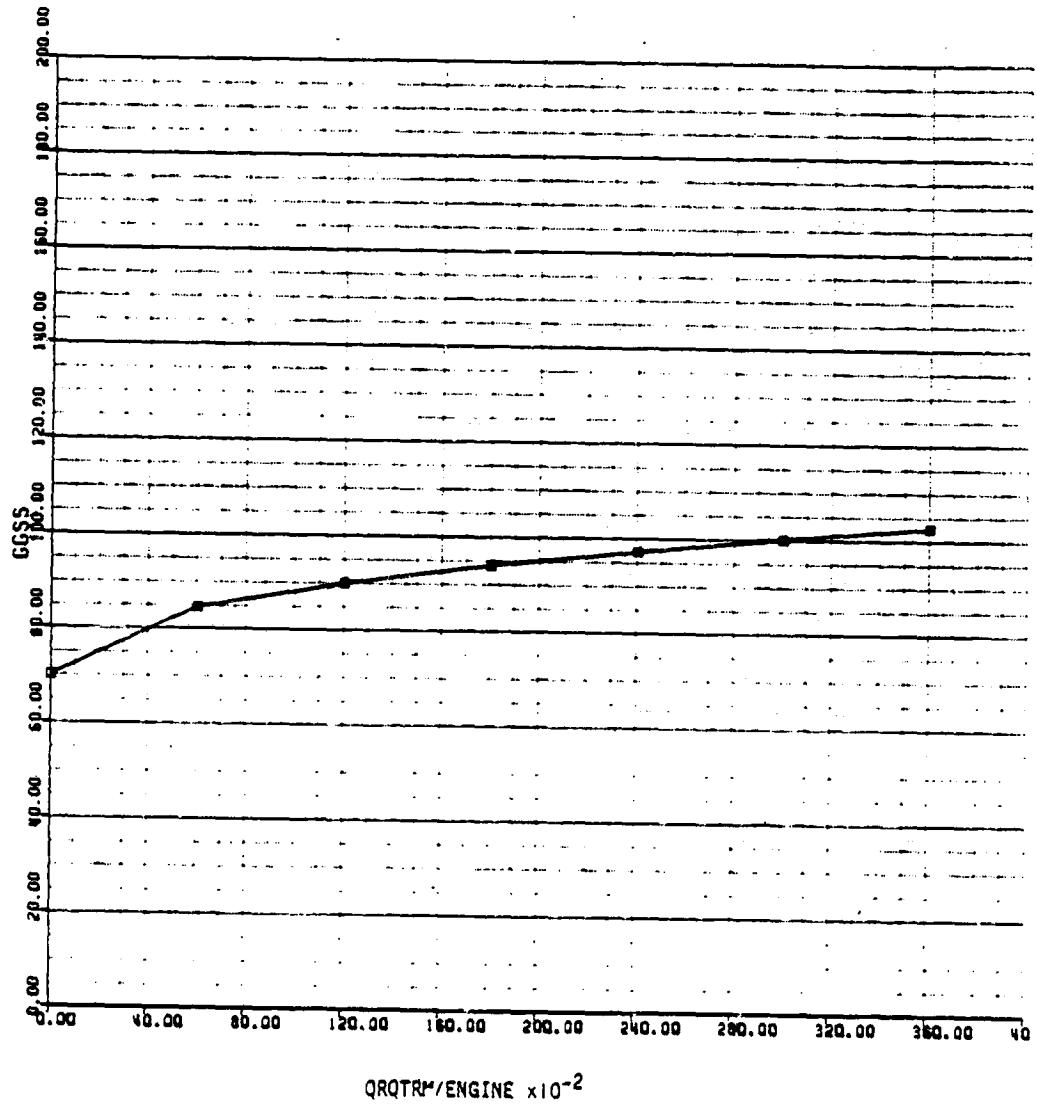


FIGURE 6.5.1

BLACK HAWK COLLECTIVE STATIC DROOP COMPENSATOR RIGGING

MAP NAME: MAPC1
 MAP TYPE: UVA
 INPUT VARIABLE(S): RO
 OUTPUT VARIABLE: LOS
 PRIMARY MAP: " "

0.00	LOWER LIMIT
100.00	UPPER LIMIT
10.00	SCALE

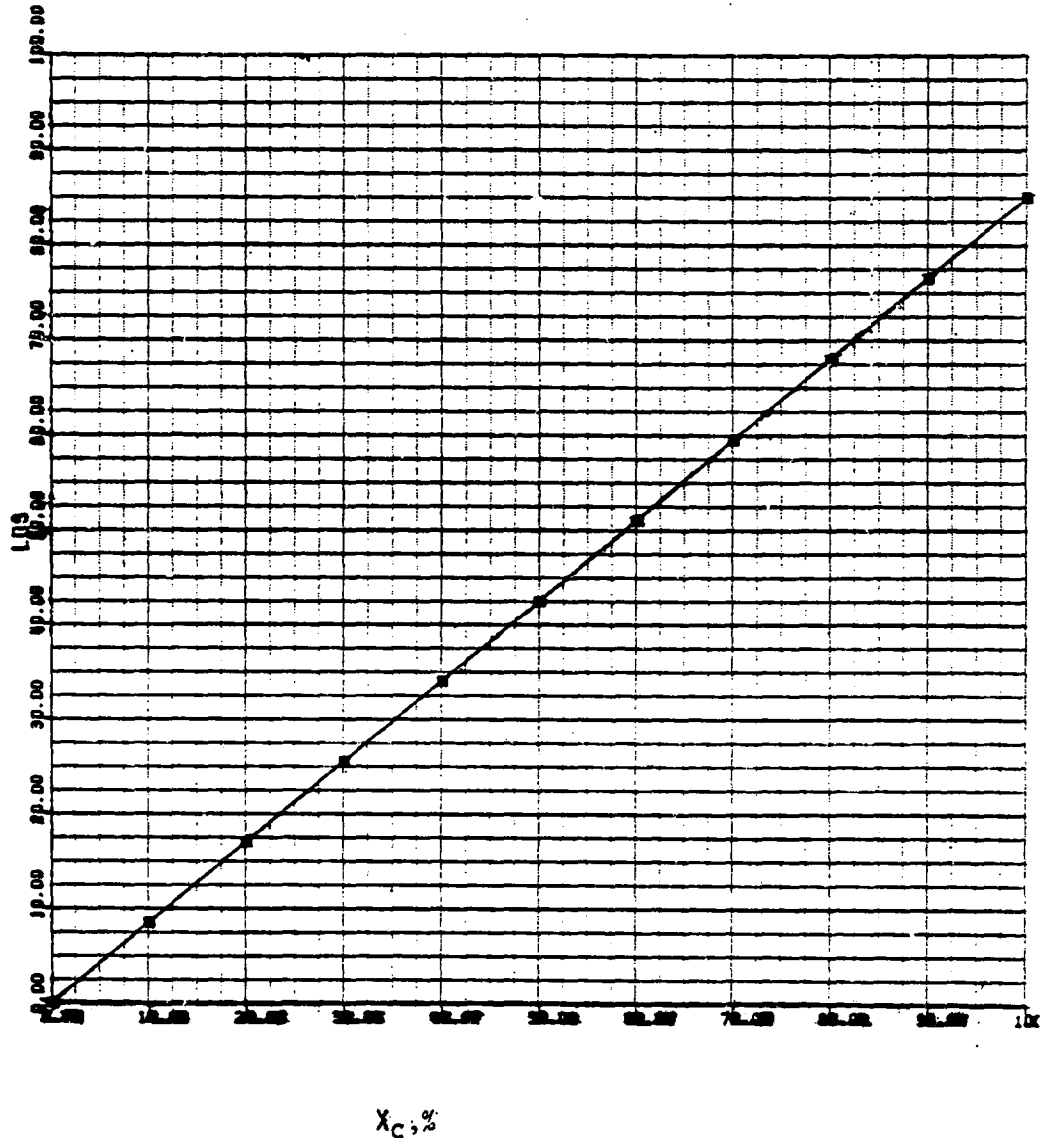


FIGURE 6.5.2

BLACK HAWK - LOAD DEMAND SPINDLE CAM OUTPUT

MAP NAME: MAPC2
 MAP TYPE: UVA
 INPUT VARIABLE(S): LDS
 OUTPUT VARIABLE: NCGLOS
 PRIMARY MAP:
 0.00 LOWER LIMIT
 100.00 UPPER LIMIT
 20.00 DELTA

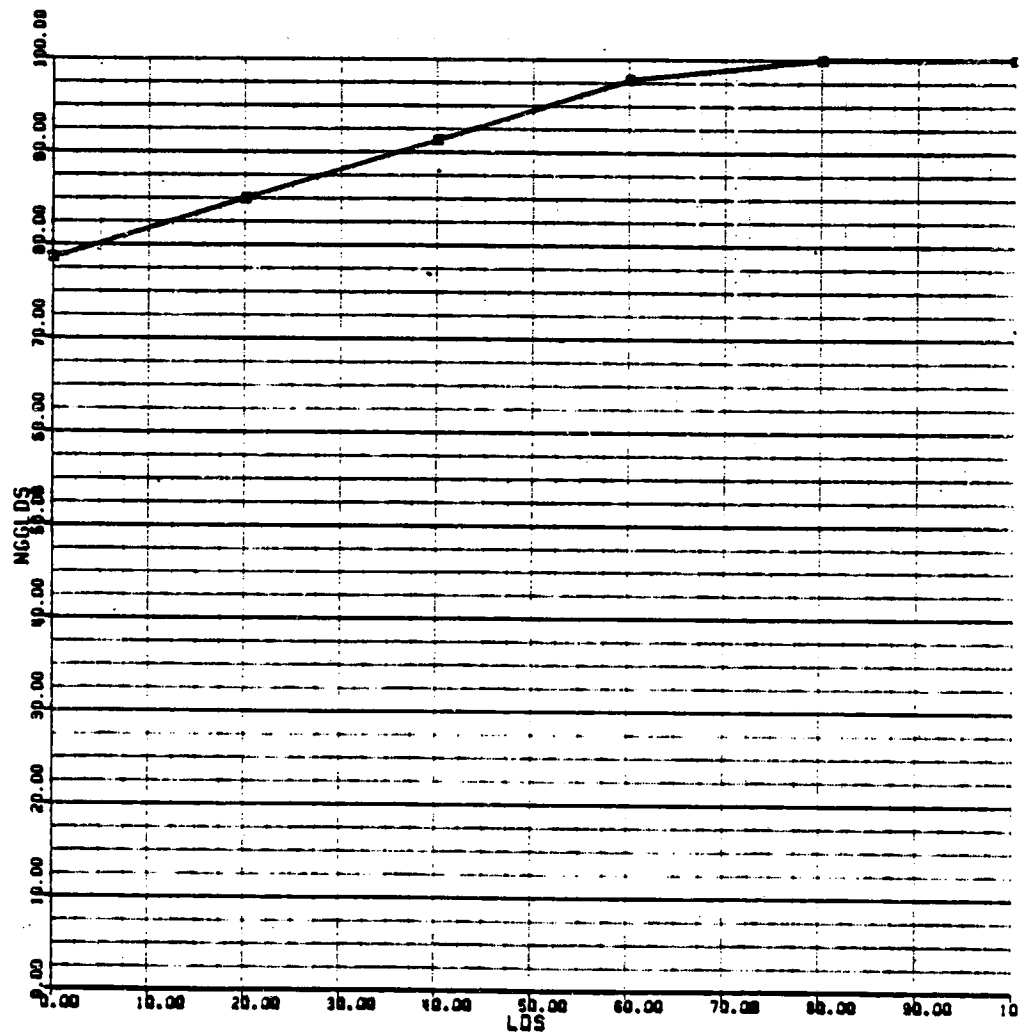


FIGURE 6.5.3

BLACK HAWK-ENGINE STEADY STATE FUEL FLOW REQUIRED

MAP NAME: MAPA2
 MAP TYPE: UVR
 INPUT VARIABLE(S): GGSS
 OUTPUT VARIABLE: SUM2
 PRIMARY MAP:
 70.00
 100.00
 5.00
 LOWER LIMIT
 UPPER LIMIT
 CONST.

(TOTFUL)_{SS} ~ (NGG)_{SS}%

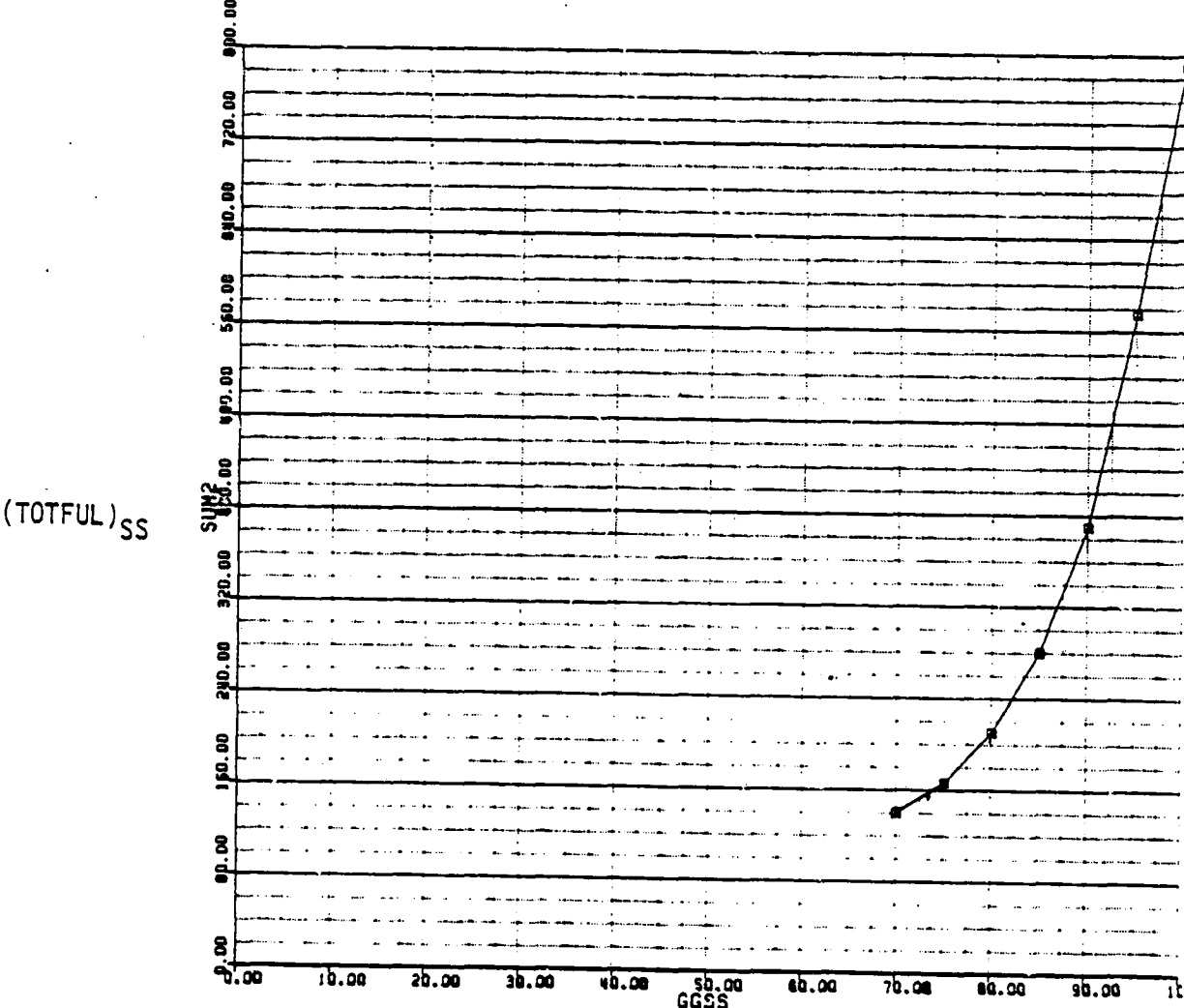


FIGURE 6.5.4

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BLACK HAWK - ENGINE DISCHARGE PRESSURE FOR STEADY CONDITIONS

MAP NAME: MAPB2
 MAP TYPE: UVR
 INPUT VARIABLE(S): GGPM
 OUTPUT VARIABLE: KFUN+2
 PRIMARY MAP:
 70.00
 100.00
 5.00

CURVE LIMIT
 UPPER LIMIT
 LOWER LIMIT

(P₃)_{SS}

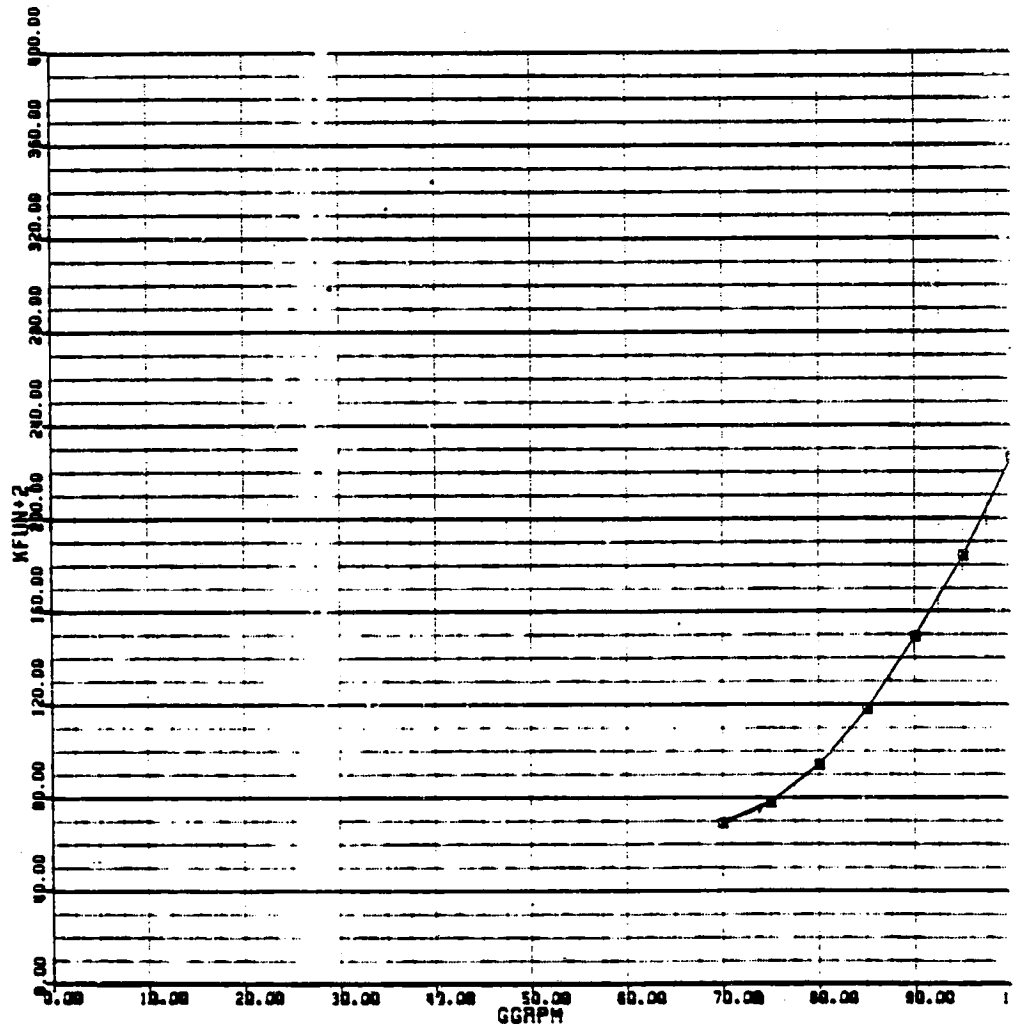


FIGURE 6.5.5

BLACK HAWK - ENGINE TIME VARYING COEFFICIENT

MAP NAME: MAP84
 MAP TYPE: UVR
 INPUT VARIABLE(S): GGPM
 OUTPUT VARIABLE: KFUN→4

PRIMARY MAP: 70.00
 100.00
 5.00

LOWER LIMIT
 UPPER LIMIT
 CENTER

$$\frac{\partial P_3}{\partial W_F} \sim NGG\%$$

$$\frac{\partial P_3}{\partial W_F}$$

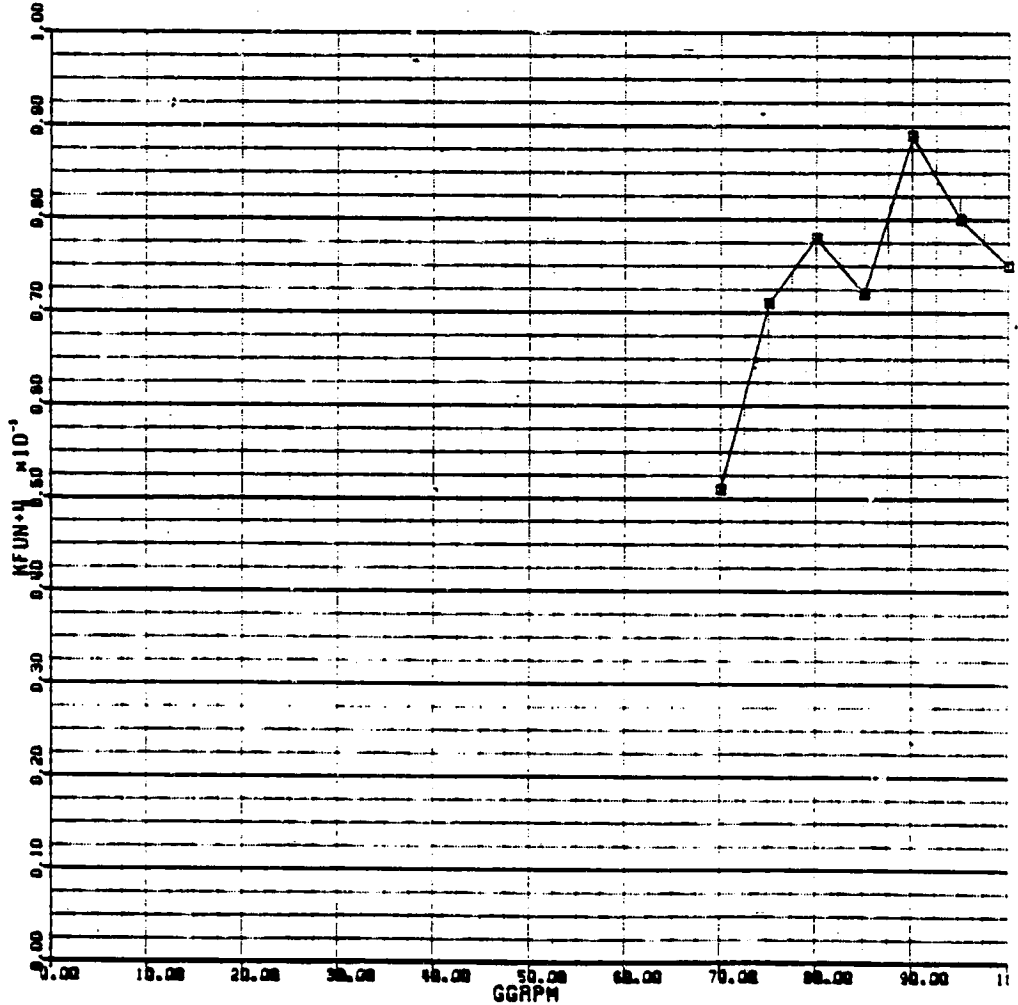


FIGURE 6.5.6

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BLACK HAWK - ENGINE TIME VARYING COEFFICIENT

MAP NAME: MAPB7
 MAP TYPE: UVR
 INPUT VARIABLE(S): GGPM
 OUTPUT VARIABLE: KFUN+7
 PRIMARY MAP:

70.00
 100.00
 5.00

$\frac{\partial P_3}{\partial N_{GG}} \sim N_{GG}\%$

$\frac{\partial P_3}{\partial N_{GG}}$

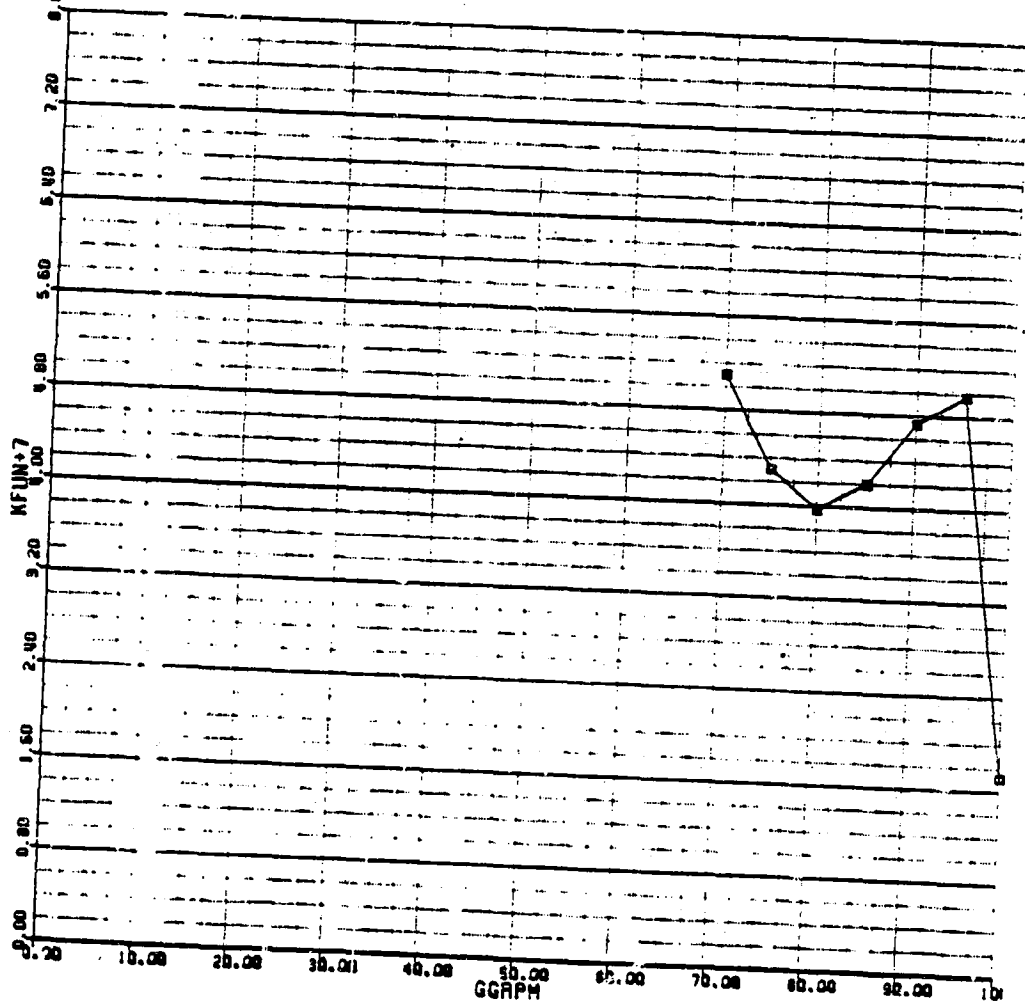


FIGURE 6.5.7

BLACK HAWK - ENGINE TIME VARYING COEFFICIENT

$$\frac{\Delta W_F}{\Delta P_3} \sim NGG\%$$

MAP NAME: MAP03
 MAP TYPE: UVA
 INPUT VARIABLE(S): GCRPM
 OUTPUT VARIABLE: KFUN+3
 PRIMARY MAP: 70.00 LOWER LIMIT
 100.00 UPPER LIMIT
 5.00 DELTA

$$\frac{\Delta W_F}{\Delta P_3}$$

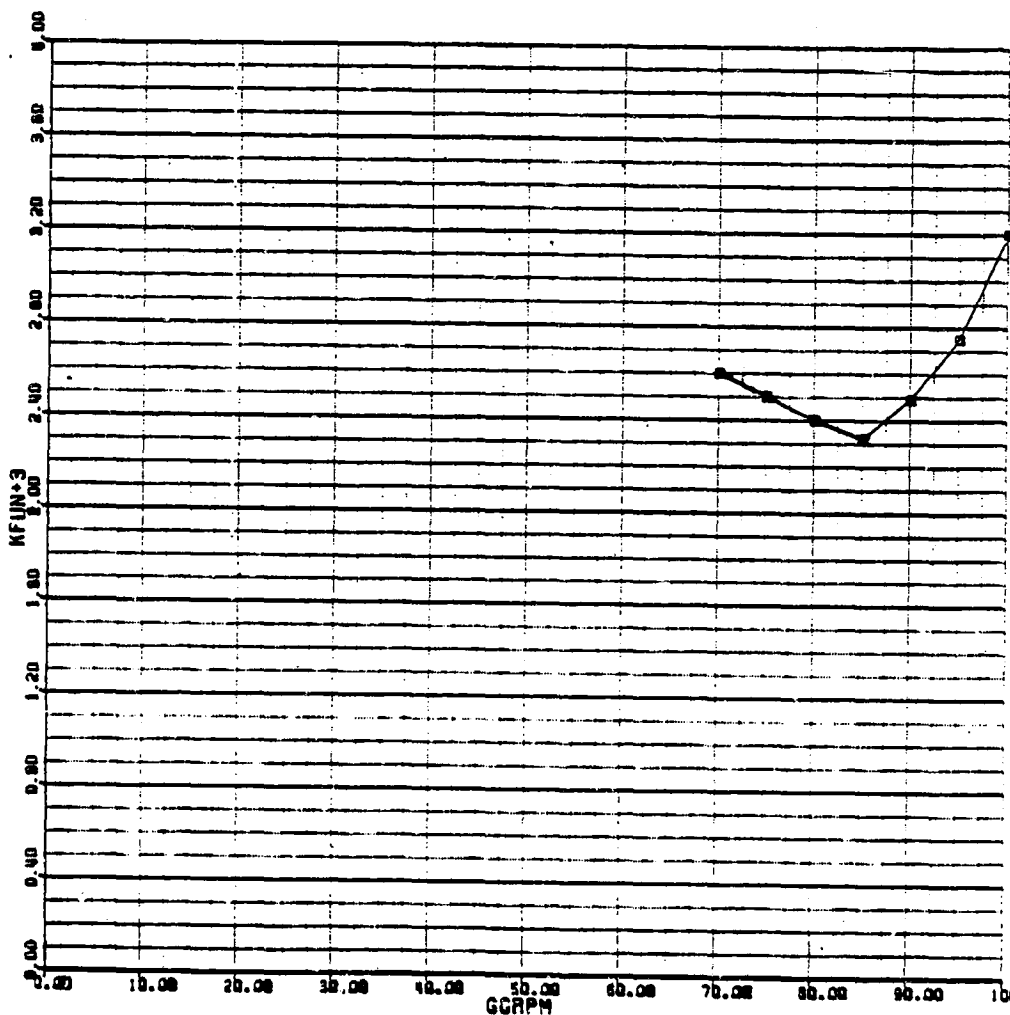


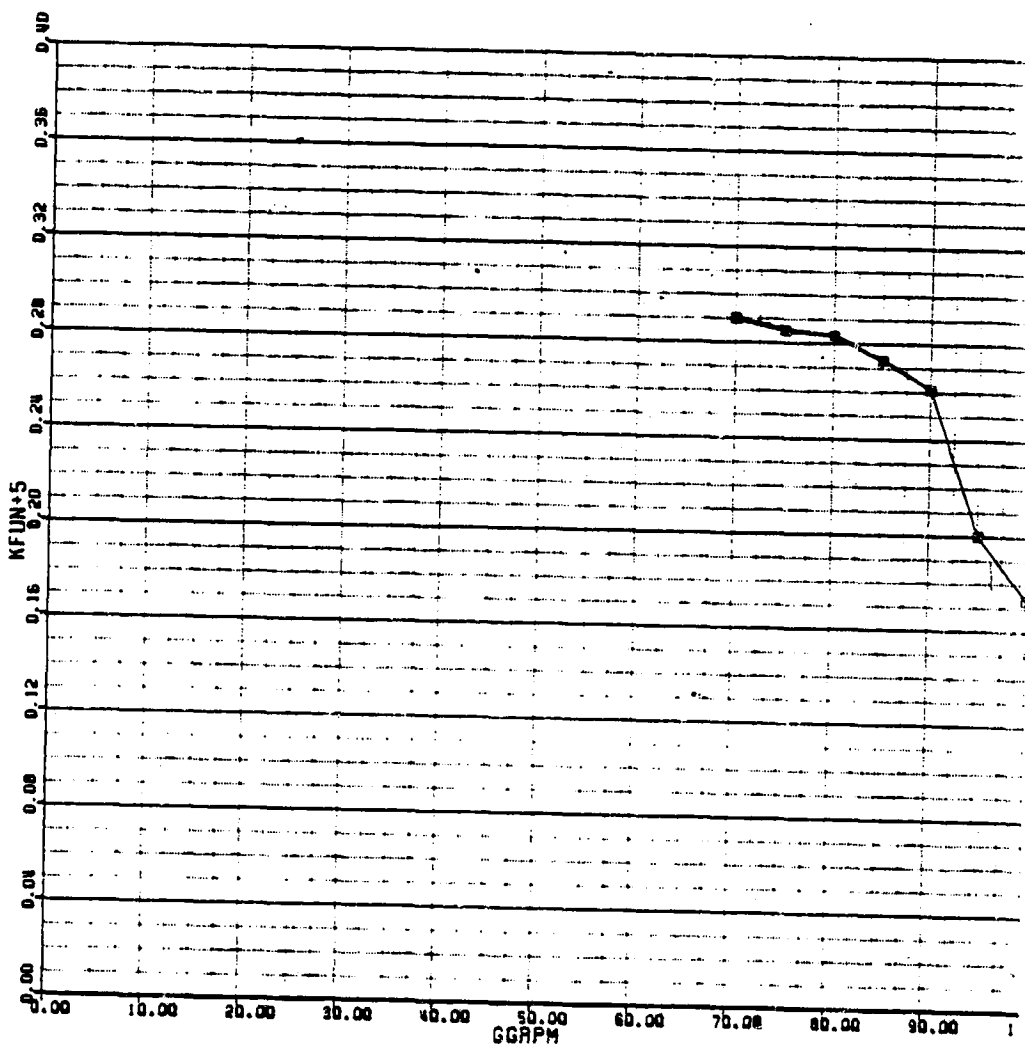
FIGURE 6.5.3

BLACK HAWK - ENGINE TIME VARYING COEFFICIENT

MAP NAME: MAP85
 MAP TYPE: UVR
 INPUT VARIABLE(S): GGRPM
 OUTPUT VARIABLE: KFUN+5
 PRIMARY MAP:

70.00
 100.00
 5.00

$\frac{\partial Q_{GG}}{\partial W_F}$ ~ NGG%



$\frac{\partial Q_{GG}}{\partial W_F}$

FIGURE 6.5.9

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OF POOR QUALITY

BLACK HAWK - ENGINE TIME VARYING COEFFICIENT

MAP NAME: MAP86
 MAP TYPE: UVA
 INPUT VARIABLE(S): GGPRM
 OUTPUT VARIABLE: KFUN+6
 PRIMARY MAP:

70.00 LOWER LIMIT
 100.00 UPPER LIMIT
 5.00 DELTA

$\frac{\partial Q_{GG}}{\partial N_{GG}} \sim N_{GG}\%$

$\frac{\partial Q_{GG}}{\partial N_{GG}}$

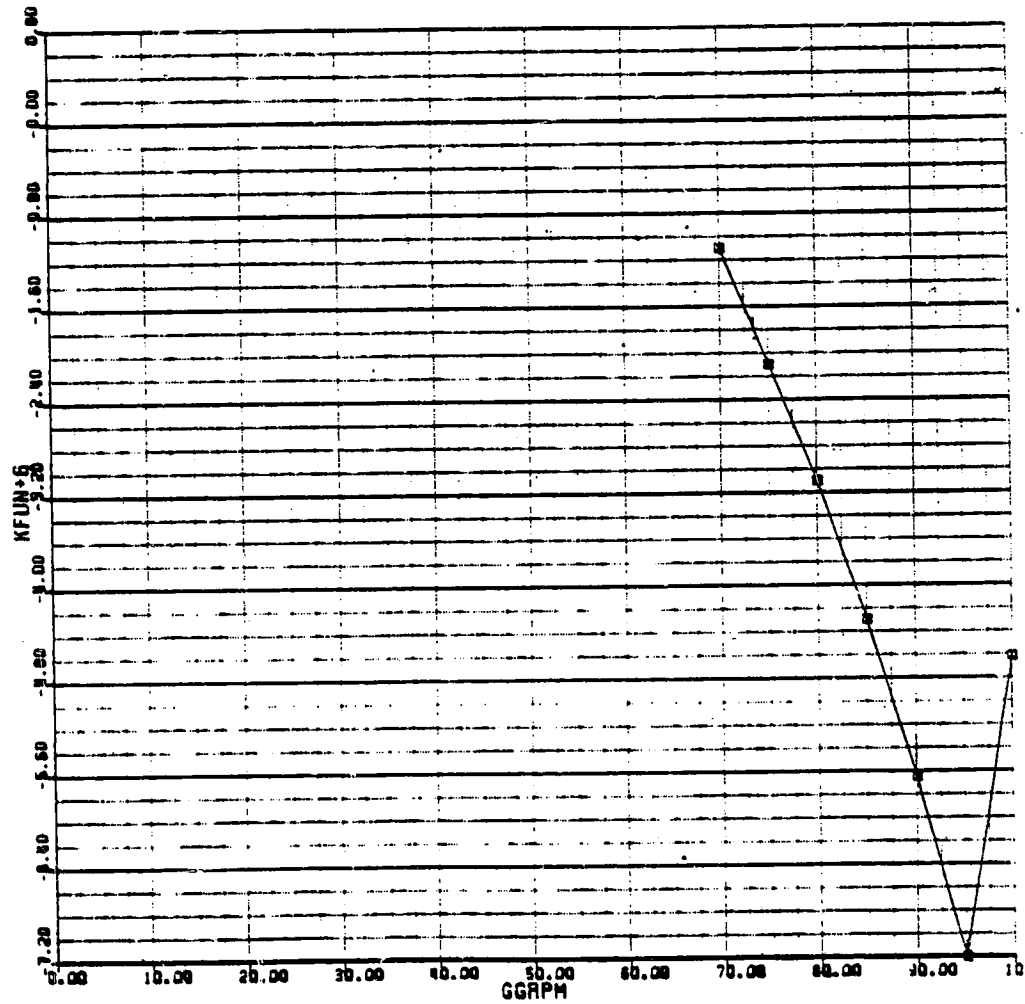


FIGURE 6.5.10

ORIGINAL PAGE 13
OF POOR QUALITY

BLACK HAWK - ENGINE TIME VARYING COEFFICIENT

MAP NAME: MAP88
 MAP TYPE: UVR
 INPUT VARIABLE(S): GGPM
 OUTPUT VARIABLE: KFUN+8
 PRIMARY MAP:

70.00 ← LOWER LIMIT
 100.00 ← UPPER LIMIT
 5.00 ← DELTA

$\frac{\partial Q_{PT}}{\partial W_F} \sim N_{GG\%}$

$\frac{\partial Q_{PT}}{\partial W_F}$

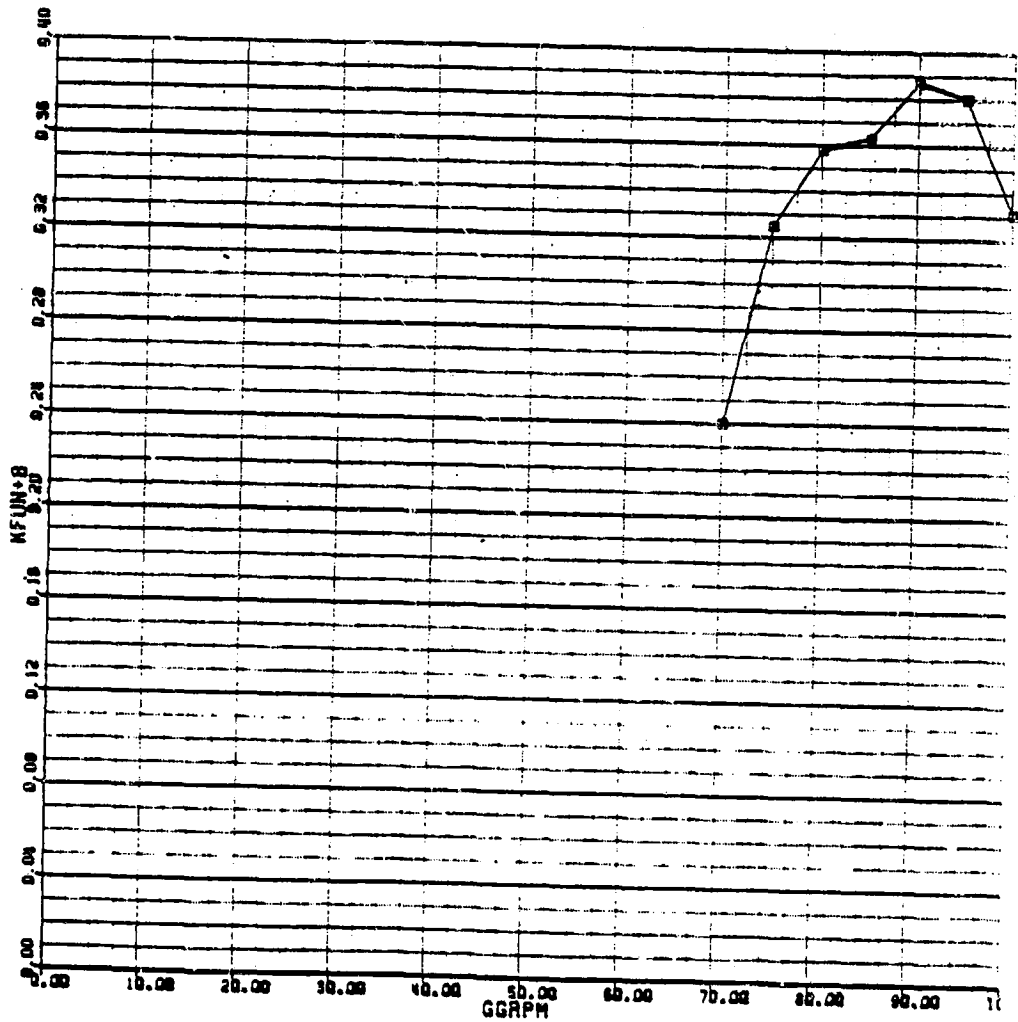


FIGURE 6.5.11

BLACK HAWK - ENGINE TIME VARYING COEFFICIENT

MAP NAME: MAPB10
 MAP TYPE: UVA
 INPUT VARIABLE(S): GGAPM
 OUTPUT VARIABLE: KFUN+10

PRIMARY MAP: 70.00 LOWER LIMIT
 100.00 UPPER LIMIT
 5.00 (0.0, 1.0)

$\frac{\partial Q_{PT}}{\partial N_{PT}} \sim Ngg\%$

$\frac{\partial Q_{PT}}{\partial N_{PT}}$

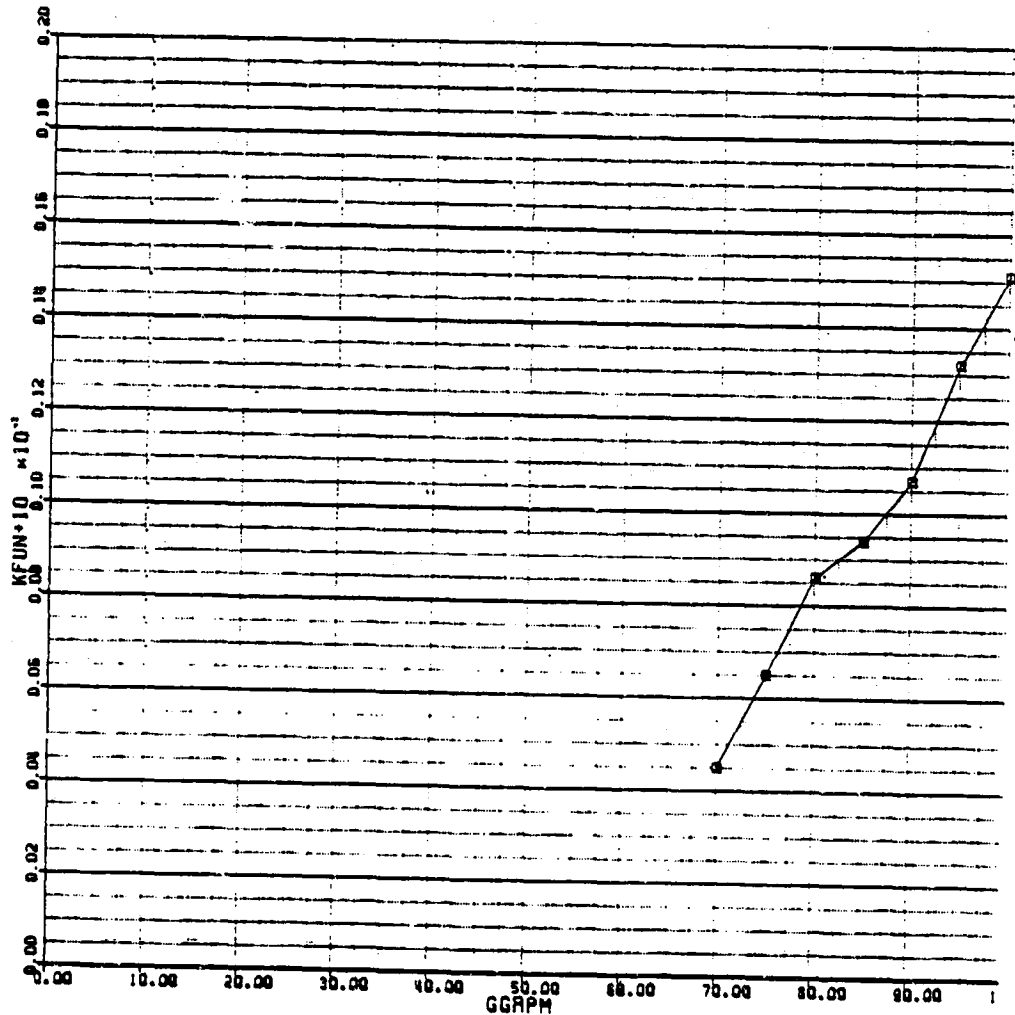


FIGURE 6.5.12

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BLACK HAWK - ENGINE TIME VARYING COEFFICIENT

MAP NAME: MAP89
 MAP TYPE: UVA
 INPUT VARIABLE(S): GGAPM
 OUTPUT VARIABLE: KFUN+9
 PRIMARY MAP:

$\frac{\partial Q_{PT}}{\partial N_{GG}} \sim N_{GG}\%$ 70.00 LOWER LIMIT
 100.00 UPPER LIMIT
 5.00 DELTA

$\frac{\partial Q_{PT}}{\partial N_{GG}}$

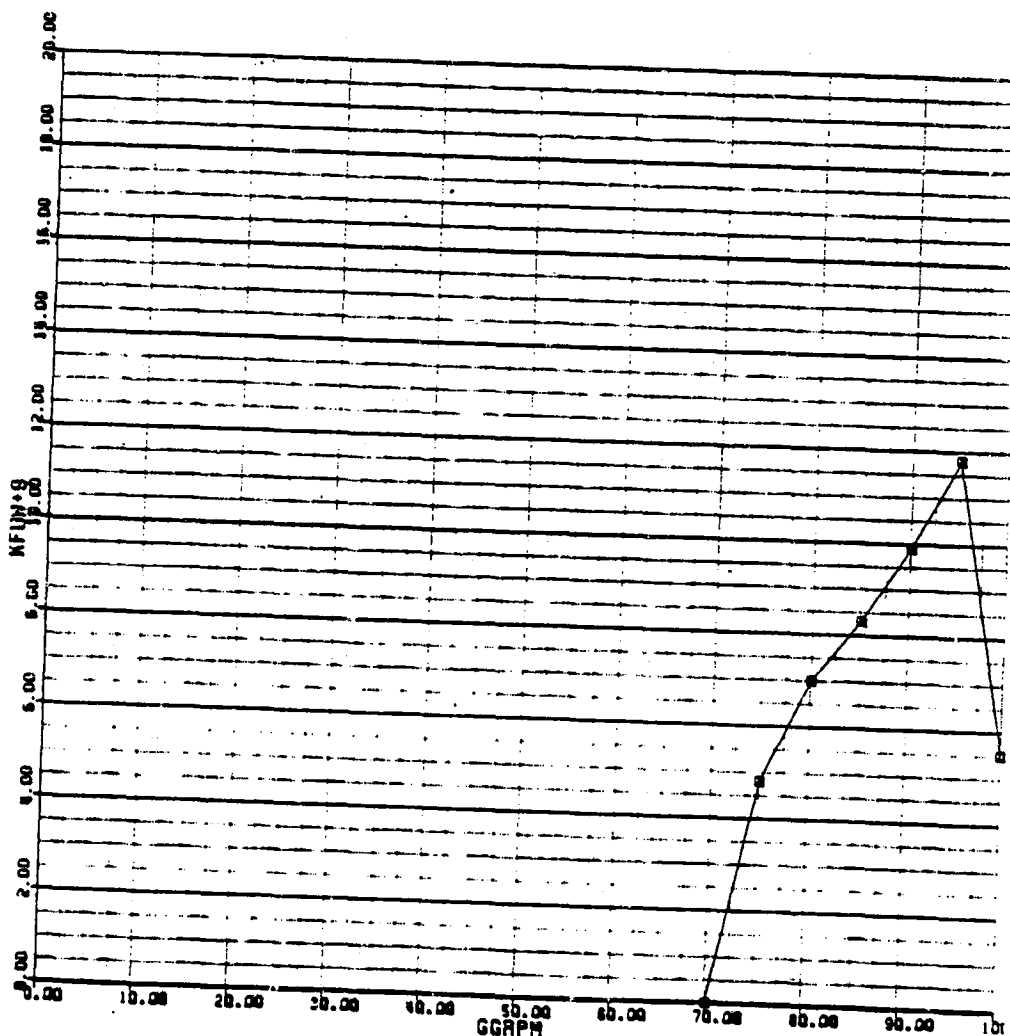


FIGURE 6.5.13

5.6.6 References

1. T700 Fuel and Control System, J. J. Curran,
AHS National Forum Paper, May 1973
2. General Electric, T700 Operations Manual



5.7 LANDING INTERFACE MODULE

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5.7 LANDING INTERFACE MODULE5.7.1 Module Description

This generalized representation consists of a landing gear force reaction model complete with all necessary space/body geometry calculations to track a free helicopter landing onto a level ground plane. The landing gear is represented by separate non-linear tire and strut dynamic characteristics as shown in Figure 7.1.1. Tire in-ground-plane loads are developed as a non-linear function of the tire deflection and normal load. These forces are adjusted depending on the friction criteria which determines tire skid characteristics at the deck surface. Finally, strut loads are summed with other external forces and moments at the helicopter CG.

Axes Systems. Two axes systems are used in this landing interface module as shown in Figure 7.1.2. All landing gear forces and moments are formulated in axes parallel to the primary body axes system passing through the CG. The space axes system of which the ground plane is set at WFLD is used to determine the landing gear proximity to the ground. Inplane friction forces are checked in the space axes system.

Landing gear geometry. In the equations defining the geometry of the landing gear, Figure 7.1.3: It is assumed that the strut moves along the line parallel to the helicopter Z axis. No account is taken of drag linkage constraints which cause the axle to move in an arc in the X-Z plane. This geometry together with the Gen Hel calculated position of the helicopter C.G. position in space, is used to establish the location of the tire, axle reference and gear reference points for each gear in space axes. These coordinates are used later to determine the proximity of tire contact and subsequent tire and strut deflections.

Determination of tire contact. The determination of tire contact, for an arbitrary orientation of the helicopter relies on establishing the length of a normal line from the ground plane to the axle reference position. When the distance along the gear line becomes less than the tire radius, contact has occurred. Subsequently, this difference is defined as radial tire deflection. In practice tire contact can occur at any point on the width of the tire. In the model the contact point is assumed to be at the center of the tire tread, irrespective of the distortion resulting from radial or axial loading.

Determination of tire inplane deflections and loads. In order to establish the degree of inplane deflection and corresponding loading on the tire, it is necessary to track the intersection of the landing gear line relative to the ground plane. Essentially two equations of the gear line, (each projected into a two dimensional plane) are solved for the intersection coordinates with the ground plane. When contact for an individual tire is established, the point of intersection is retained and on subsequent iterations through the program, tire deflection is determined by comparing the new and old gear line intersections. The coordinates of the initial contact are retained until the tire leaves the ground or is modified by the tire slipping. The latter aspect is discussed later. Following the transformation of the deflections into helicopter body axes, the three components of deflection at each tire are used to enter the "tire characteristics data file," to obtain the three components of tire force.

Determination of tire contact conditions. Following the determination of tire forces from the helicopter/ground plane relative motion, a test of the ability of the inplane friction forces to resist the applied forces, without slipping, must be established. The tire forces obtained in helicopter axes must be transferred to the ground plane for the friction check. Classical friction considerations provide for a coefficient of static friction and a coefficient for sliding friction. In the former case (when brakes are set), the maximum amount of inplane load which can be resisted without slipping, is proportional to the coefficient of friction and the normal loading. When this level is exceeded, motion will result. Then the force resisting the motion will depend on a reduced (sliding) coefficient of friction and the normal force. In practice, there is a smooth transition between the two conditions. However, the model assumes a discrete change. A further assumption is that the test for frictional loads in the model assumes that X and Y inplane components can be tested separately. In practice, the resultant force determines slip conditions. This latter assumption was made to simplify the model and facilitate the introduction of brakes. When the brakes are activated, it is assumed that the wheels are locked. For brakes off, a very low coefficient of friction is introduced into the tire X direction. The wheel degree of freedom is not currently represented and therefore spin up (say on landing) inertia loads are not calculated. If slip is not occurring, calculated tire forces are passed unchanged. If slippage is occurring, the inplane forces are set to the value for sliding friction.

Reinitialization of the original tire contact point as a result of slippage. Under conditions of no-slip the tire inplane deflection is developed from consecutive calculations of the gear line intersection with the ground plane. During slip conditions, the contact point for the tire moves and the initialization of the gear-line intersection must be revised to reflect the tire movement and establish a new value for the contact point to be used on the next pass through the program.

Acceleration of the landing gear strut. Under steady conditions, the loads transferred to the airframe by the strut will be equal to the tire reactions. However, under transient conditions, the acceleration of the unsprung mass can modify the loading. Under light loading condition (where a significant portion of helicopter weight is reacted by the rotor), the strut operates in a preload range. Under these conditions, where tire reaction load is less than the strut preload setting, tire loads are transferred to the airframe with zero strut deflection. Once the preload setting is exceeded, the strut (unsprung mass) is accelerated depending on its own dynamic characteristics, the tire applied loads and the unsprung mass. Note that logic precludes the equation flow reverting back to the preload mode until the natural transient provides a zero strut deflection condition. The strut is assumed to have velocity squared damping and an isothermal air spring. A software switch is provided which allows bypassing the strut calculation. Finally, the strut loads are transferred to the helicopter CG for summation with other external forces.

LANDING GEAR
TIRE AND STRUT REPRESENTATION

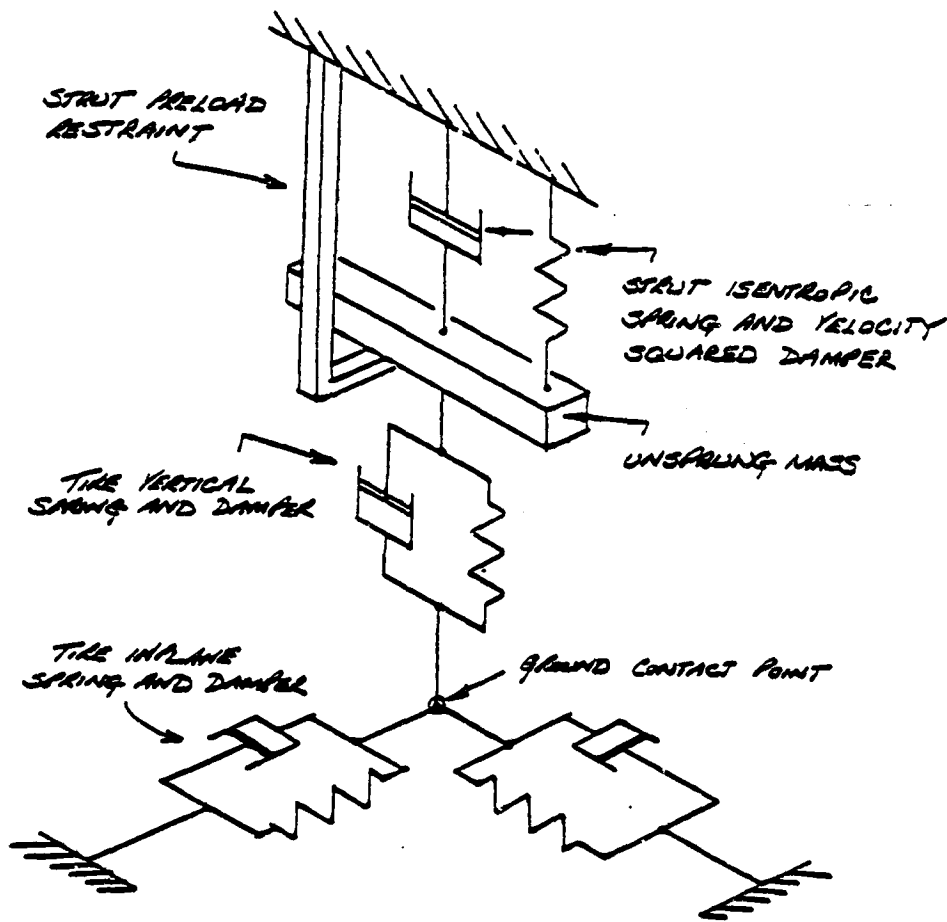
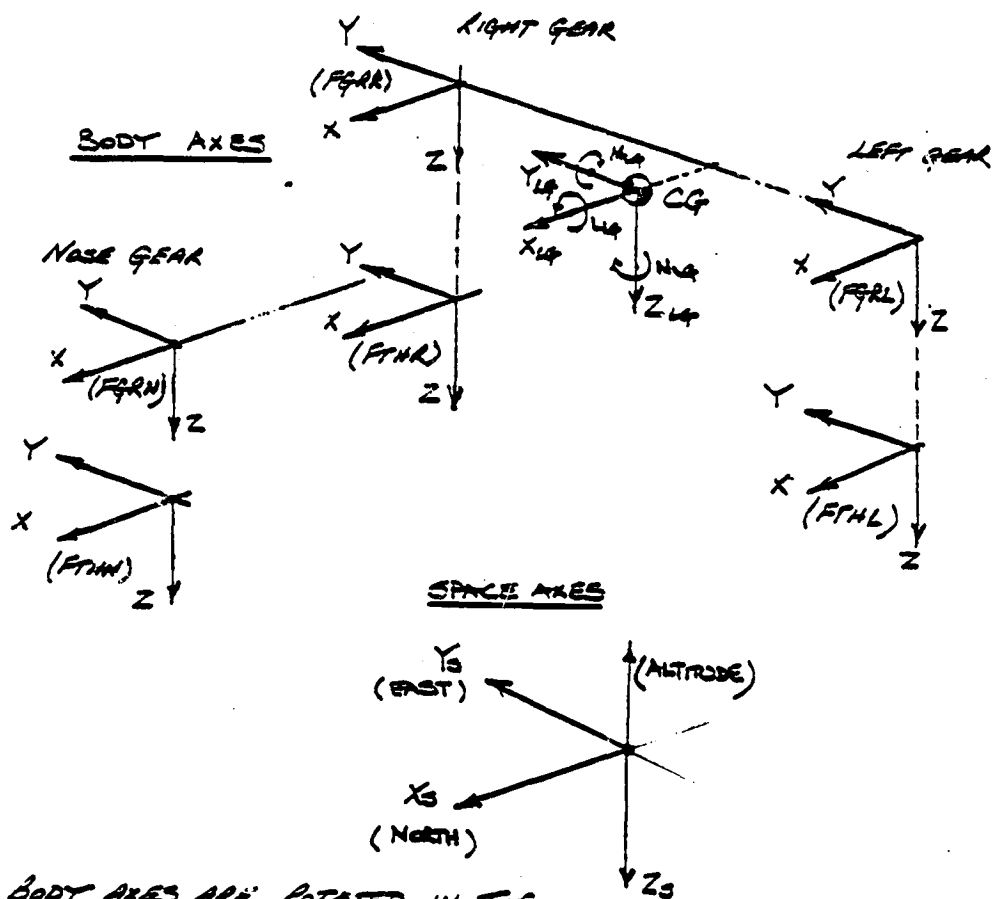


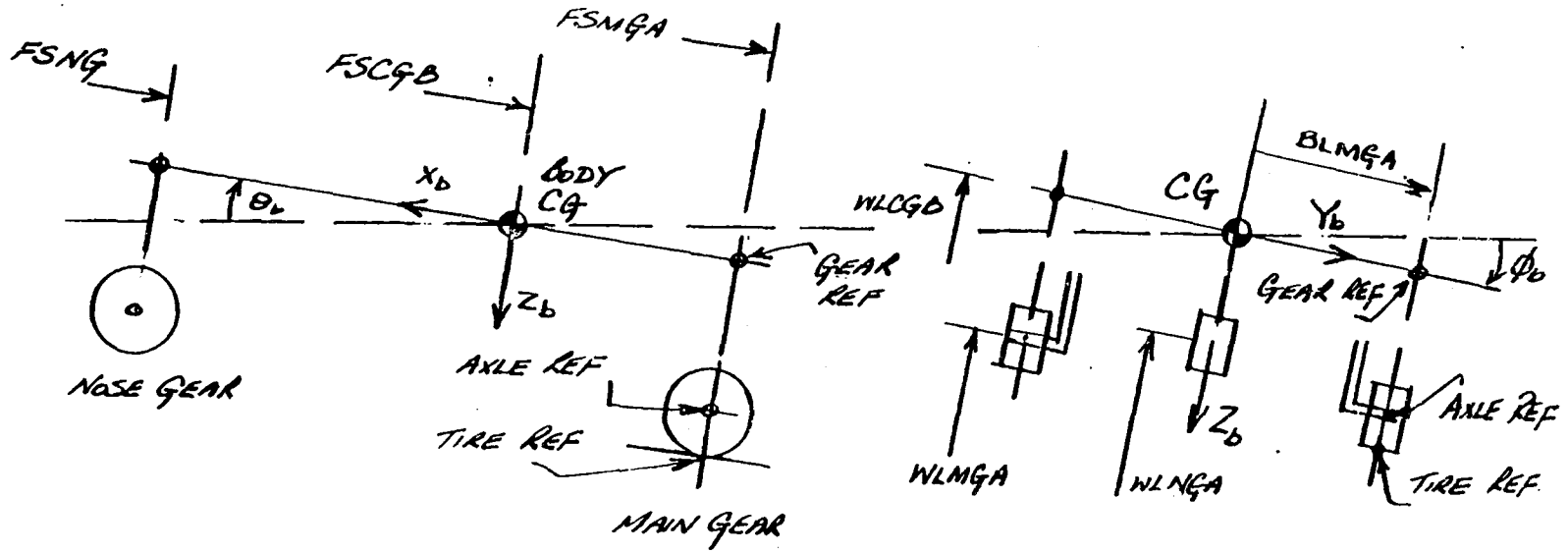
FIGURE 7.1.1

LANDING MODULE AXES SYSTEM



BODY AXES ARE ROTATED IN THE
SEQUENCE: ψ_0 ABOUT Z_0 , θ_0 ABOUT Y_0 , ϕ_0 ABOUT X_0
TO REACH THE SPACE AXES SYSTEM

FIGURE 7.1.2



LANDING GEAR GEOMETRY
Figure 7.1.3

LANDING INTERFACE EQUATION FLOW

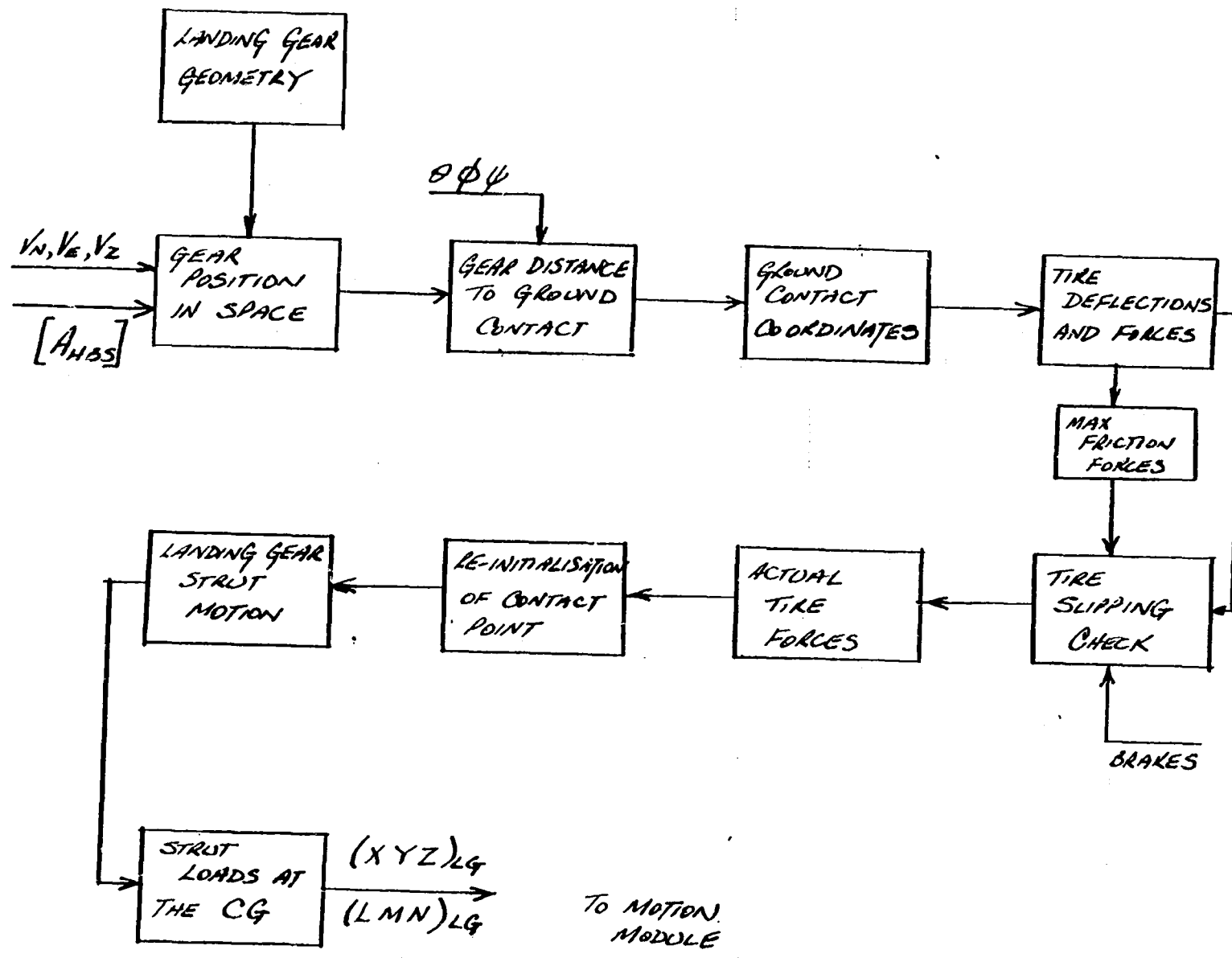


FIGURE 7.1.4

TO MOTION
MODULE

5.7.2 LANDING INTERFACE MODULE EQUATIONS

HELICOPTER CENTER OF GRAVITY POSITION IN SPACE AXES

V_N, V_E, V_Z ARE CALCULATED IN MAIN HELICOPTER PROGRAM

$$X_{SH} = X_{SH0} + \int V_N dt \quad , SH$$

$$Y_{SH} = Y_{SH0} + \int V_E dt \quad , SH + 1$$

$$Z_{SH} = Z_{SH0} - \int V_Z dt \quad , SH - 2$$

V_Z +VE UP IN MAIN PROGRAM

LANDING GEAR GEOMETRY

$FSCGB, WLCGB$ ARE CALCULATED IN THE MAIN HELICOPTER PROGRAM

NOSE GEAR :

$$X_{NGA} = \frac{(FSCGB - FSNGA)}{12} \quad , NGA$$

$$Y_{NGA} = 0 \quad , NGA + 1$$

$$Z_{NGA} = \frac{(WLCGB - WLNGA0)}{12} \quad , NGA + 2$$

RIGHT MAIN GEAR :

$$X_{RMGA} = \frac{(FSCGB - FSMGA)}{12} \quad , RMGA$$

$$Y_{RMGA} = \frac{B_{RMGA}}{12} \quad , RMGA + 1$$

$$Z_{RMGA} = \frac{(WLCGB - WLMGA0)}{12} \quad , RMGA + 2$$

PROJ. NO.	NAME	DATE	SIMPOSKY AIRCRAFT	REV.	TEMP.	FORM
DRAWING	TITLE		MATERIAL			
DRAWING			SCALE			

LEFT MAIN GEAR:

$$X_{LMGA} = X_{RMGA} \quad , LMGA$$

$$Y_{LMGA} = Y_{RMGA} \quad , LMGA + 1$$

$$Z_{LMGA} = Z_{RMGA} \quad , LMGA + 2$$

THESE COORDINATES ARE FOR THE FULLY EXTENDED STRUTS AT THE AXLE LOCATION

FREE EXTENSION POSITIONS OF GEAR IN SPACE AXES:

TIRE REFERENCE POSITIONS

NOSE GEAR:

$$\begin{bmatrix} X \\ Y \\ Z \end{bmatrix}_{NTSO} = \begin{bmatrix} A_{HBS} \\ X_{NGA} \\ Y_{NGA} \\ Z_{NGA} + R_{TA}/12 \end{bmatrix} + \begin{bmatrix} X \\ Y \\ Z \end{bmatrix}_{SH} \quad \begin{matrix} , NTSO \\ , NTSO + 1 \\ , NTSO + 2 \end{matrix}$$

RIGHT GEAR:

$$\begin{bmatrix} X \\ Y \\ Z \end{bmatrix}_{RMTSO} = \begin{bmatrix} A_{HBS} \\ X_{RMGA} \\ Y_{RMGA} \\ Z_{RMGA} + R_{TM}/12 \end{bmatrix} + \begin{bmatrix} X \\ Y \\ Z \end{bmatrix}_{SH} \quad \begin{matrix} , RMTSO \\ , RMTSO + 1 \\ , RMTSO + 2 \end{matrix}$$

LEFT GEAR:

$$\begin{bmatrix} X \\ Y \\ Z \end{bmatrix}_{LMTSO} = \begin{bmatrix} A_{HBS} \\ X_{LMGA} \\ Y_{LMGA} \\ Z_{LMGA} + R_{TM}/12 \end{bmatrix} + \begin{bmatrix} X \\ Y \\ Z \end{bmatrix}_{SH} \quad \begin{matrix} , LMTSO \\ , LMTSO + 1 \\ , LMTSO + 2 \end{matrix}$$

Prepared	NAME	DATE	TITLE	Page	TEMP	PERM.
	JL		SIKORSKY AIRCRAFT	3		
Checked						Model
Drawing						Report No.
<u>AXLE REFERENCE POSITION IN SPACE</u>				NOTE: SIMILARITY TO THE REF. POSITIONS		
<u>NOSE GEAR</u>						
X	=	A _{NAS}	X _{NAS}	+	X	NAS
Y	=	A _{NAS}	Y _{NAS}	+	Y	NAS+1
Z	=	A _{NAS}	Z _{NAS}	- S _{NAS}	Z	NAS+2
<u>RIGHT GEAR</u>						
X	=	A _{NAS}	X _{ENG}	+	X	RAS
Y	=	A _{NAS}	Y _{ENG}	+	Y	RAS+1
Z	=	A _{NAS}	Z _{ENG}	- S _{ENG}	Z	RAS+2
<u>LEFT GEAR</u>						
X	=	A _{NAS}	X _{ENG}	+	X	LAS
Y	=	A _{NAS}	Y _{ENG}	+	Y	LAS+1
Z	=	A _{NAS}	Z _{ENG}	- S _{ENG}	Z	LAS+2

Prepared	NAME	DATE	TITLE	REV.	TEMP	PCAM
Checked				SIKOPSKY AIRCRAFT		
Drawing				Model		
				Recon No.		

GEAR REFERENCE POSITION IN SPACE

NOTE: SIMILARITY
TO THE REF. EQ^T

NOSE GEAR

$$\begin{array}{|c|} \hline X \\ \hline Y \\ \hline Z \\ \hline \end{array} = A_{HBS} \begin{array}{|c|} \hline X \\ \hline Y \\ \hline Z \\ \hline \end{array} + \begin{array}{|c|} \hline X \\ \hline Y \\ \hline Z \\ \hline \end{array} \quad M7$$

$\begin{array}{|c|} \hline X \\ \hline Y \\ \hline Z \\ \hline \end{array}$
 $\begin{array}{|c|} \hline X \\ \hline Y \\ \hline Z \\ \hline \end{array}$
 $\begin{array}{|c|} \hline X \\ \hline Y \\ \hline Z \\ \hline \end{array}$

RIGHT GEAR

$$\begin{array}{|c|} \hline X \\ \hline Y \\ \hline Z \\ \hline \end{array} = A_{HBS} \begin{array}{|c|} \hline X \\ \hline Y \\ \hline Z \\ \hline \end{array} + \begin{array}{|c|} \hline X \\ \hline Y \\ \hline Z \\ \hline \end{array} \quad M8$$

$\begin{array}{|c|} \hline X \\ \hline Y \\ \hline Z \\ \hline \end{array}$
 $\begin{array}{|c|} \hline X \\ \hline Y \\ \hline Z \\ \hline \end{array}$
 $\begin{array}{|c|} \hline X \\ \hline Y \\ \hline Z \\ \hline \end{array}$

LEFT GEAR

$$\begin{array}{|c|} \hline X \\ \hline Y \\ \hline Z \\ \hline \end{array} = A_{HBS} \begin{array}{|c|} \hline X \\ \hline Y \\ \hline Z \\ \hline \end{array} + \begin{array}{|c|} \hline X \\ \hline Y \\ \hline Z \\ \hline \end{array} \quad M9$$

$\begin{array}{|c|} \hline X \\ \hline Y \\ \hline Z \\ \hline \end{array}$
 $\begin{array}{|c|} \hline X \\ \hline Y \\ \hline Z \\ \hline \end{array}$
 $\begin{array}{|c|} \hline X \\ \hline Y \\ \hline Z \\ \hline \end{array}$

Prepared	NAME	DATE	TITLE	Page	FORM	FORM
			SIKORSKY AIRCRAFT			
Checked						
Drawn						
<u>DISTANCE OF NORMAL FROM GROUND PLANE TO AXLE</u>						
<u>REFERENCE POINT</u>						
$D_{NNA} = Z_{NAS} + WLFD$, D_{NNA}
$D_{NRA} = Z_{RAS} + WLFD$, D_{NRA}
$D_{NLA} = Z_{LAS} + WLFD$, D_{NLA}
<u>DIRECTION COSINES OF GEAR LINE</u>						
THE GEAR LINE OF DEFLECTION IS ASSUMED TO BE PARALLEL TO AIRCRAFT BODY AXES. THUS THE DIRECTION COSINES ARE EQUIVALENT TO THE 3RD COLUMN OF A_{HNS} FOR A UNITY VECTOR						
$\cos \alpha_{NG} = \sin \theta_0 \cos \phi_0 \cos \psi_0 + \sin \phi_0 \sin \psi_0$, $\cos \alpha_{NG}$
$\cos \beta_{NG} = \sin \theta_0 \cos \phi_0 \sin \psi_0 - \sin \phi_0 \cos \psi_0$, $\cos \beta_{NG}$
$\cos \gamma_{NG} = \cos \theta_0 \cos \phi_0$, $\cos \gamma_{NG}$
$\cos \alpha_{NR} = \cos \alpha_{NR} = \cos \alpha_{NG}$						
$\cos \beta_{NR} = \cos \beta_{NR} = \cos \beta_{NG}$						
$\cos \gamma_{NR} = \cos \gamma_{NR} = \cos \gamma_{NG}$						

Prepared	NAME	DATE	SIKORSKY AIRCRAFT	FIG. NO.	REV.
Checked					
Drawing					

CHECK FOR TIRE CONTACT WITH GROUND PLANE

IF

$$\delta_{TNG} = \left(\frac{R_{TN}}{12} + \frac{DNNA}{\cos \gamma_{NG}} \right) \geq 0, \quad DL_{TNG}$$

$$\delta_{TRG} = \left(\frac{R_{TR}}{12} + \frac{DNRA}{\cos \gamma_{RG}} \right) \geq 0, \quad DL_{TRMG}$$

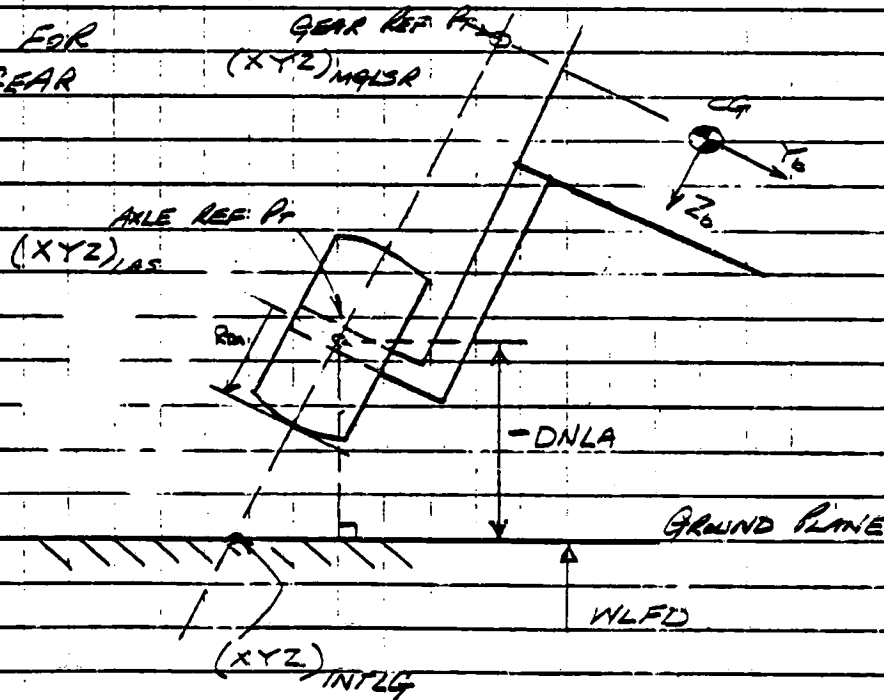
$$\delta_{TLG} = \left(\frac{R_{TL}}{12} + \frac{DNLA}{\cos \gamma_{LG}} \right) \geq 0, \quad DL_{TLMG}$$

SET CONTACT FLAG FOR CORRESPONDING GEAR

IF $\delta_{TNG}, \delta_{TRG}, \delta_{TLG} < 0$

RETURN TO MAIN HELICOPTER PROGRAM

DRAWN FOR
LEFT GEAR



Prepared	NAME	DATE	SIKORSKY AIRCRAFT	FIG.	VIEW	PERM.
Checked	TITLE			VOLUME		
Drawing				RECORD NO.		

INTERSECTION OF GEAR LINE WITH GROUND PLANE

$$(X_{INT})_{NG, RG, LG} = (a \cdot Z_{INT} + b)_{NG, RG, LG} \quad \begin{matrix} INTNG \\ INTRMG \\ INTLMG \end{matrix}$$

$$(a)_{NG} = \left(\frac{X_{NGSR} - X_{NAS}}{Z_{NGSR} - Z_{NAS}} \right) \quad b_{NG} = (X_{NAS} - a_{NG} Z_{NAS})$$

$$(a)_{RG} = \left(\frac{X_{RGRS} - X_{RAS}}{Z_{RGRS} - Z_{RAS}} \right) \quad b_{RG} = (X_{RAS} - a_{RG} Z_{RAS})$$

$$(a)_{LG} = \left(\frac{X_{LGLS} - X_{LAS}}{Z_{LGLS} - Z_{LAS}} \right) \quad b_{LG} = (X_{LAS} - a_{LG} Z_{LAS})$$

$$(Y_{INT})_{NG, RG, LG} = (c \cdot Z_{INT} + d)_{NG, RG, LG} \quad \begin{matrix} INTNG + 1 \\ INTRMG + 1 \\ INTLMG + 1 \end{matrix}$$

$$(c)_{NG} = \left(\frac{Y_{NGSR} - Y_{NAS}}{Z_{NGSR} - Z_{NAS}} \right) \quad d_{NG} = (Y_{NAS} - c_{NG} Z_{NAS})$$

$$(c)_{RG} = \left(\frac{Y_{RGRS} - Y_{RAS}}{Z_{RGRS} - Z_{RAS}} \right) \quad d_{RG} = (Y_{RAS} - c_{RG} Z_{RAS})$$

$$(c)_{LG} = \left(\frac{Y_{LGLS} - Y_{LAS}}{Z_{LGLS} - Z_{LAS}} \right) \quad d_{LG} = (Y_{LAS} - c_{LG} Z_{LAS})$$

$$(Z_{INT})_{NG, RG, LG} = -WLFD \quad \text{FOR FIXED, LEVEL GROUND PLANE}$$

PROJECT	NAME	DATE	SIKORSKY AIRCRAFT	FIG.	TEMP	FORM
CHECKED			TITLE			
DRAWING				PART NO.		

INITIALISATION OF GROUND CONTACT POINT

IF 'CONTACT FLAG' HAS BEEN SET ON THE CURRENT PASS THROUGH THE PROGRAM THEN SET

$$[(XYZ)_{ID}]_{NG, RG, LG} = [(XYZ)_{INT}]_{NG, RG, LG} \begin{matrix} , IDNG, +1, +2 \\ , IDRG, +1, +2 \\ , IDLG, +1, +2 \end{matrix}$$

TO INITIALISE POINT OF CONTACT ON THE GROUND PLANE

IF 'CONTACT FLAG' IS NOT SET

$$[(XYZ)_{ID}]_{NG, RG, LG} = 0$$

TIRE DEFLECTIONS IN GROUND PLANE AXES (SPACE AXES)

$$(\Delta X)_{NG, RG, LG} = (X_{INT})_{NG, RG, LG} - (X_{ID})_{NG, RG, LG} \begin{matrix} , DELNG \\ , DELRMG \\ , DELLMG \end{matrix}$$

$$(\Delta Y)_{NG, RG, LG} = (Y_{INT})_{NG, RG, LG} - (Y_{ID})_{NG, RG, LG} \begin{matrix} , DELNG+1 \\ , DELRMG+1 \\ , DELLMG+1 \end{matrix}$$

$$(\Delta Z)_{NG, RG, LG} = (Z_{INT})_{NG, RG, LG} - (Z_{ID})_{NG, RG, LG} \begin{matrix} , DELNG+2 \\ , DELRMG+2 \\ , DELLMG+2 \end{matrix}$$

NOTE: 1. $(\Delta Z) = 0$ CODE AS A CHECK

2. THE CONTACT FLAG MAY BE SET SEVERAL TIMES DURING THE COURSE OF A LANDING AS THE GEAR BOUNCES IN AND OUT OF CONTACT.

PROJECT	NAME	DATE	SUBJECT: AIRCRAFT	COND	TEMP	PERM
DESIGNED	TITLE			NOTE		
DRAWING				REVISION		

TIRE DEFLECTIONS IN HELICOPTER BODY AXES

$$\begin{bmatrix} \Delta X \\ \Delta Y \\ \Delta Z \end{bmatrix} = A_{HSB} \begin{bmatrix} \Delta X \\ \Delta Y \\ \Delta Z \end{bmatrix} + \begin{bmatrix} 0 \\ 0 \\ \delta_T \end{bmatrix}$$

, DEL HNG
 , DEL HRG
 , DEL HLG

HNG NG
 HRG RG
 HLG LG

TIRE FORCES IN HELICOPTER BODY AXES

$$X_{FTHN} = -N_{TN} [K_{TXN} \Delta X_{HNG} + G_{DN} K'_{TXN} \Delta X_{HNG}] \quad , \quad F_{THN}$$

$$Y_{FTHN} = -N_{TN} [K_{TYN} \Delta Y_{HNG} + G_{DN} K'_{TYN} \Delta Y_{HNG}] \quad , \quad F_{THN+1}$$

$$Z_{FTHN} = -N_{TN} [K_{TZN} \Delta Z_{HNG} + G_{DN} K'_{TZN} \Delta Z_{HNG}] \quad , \quad F_{THN+2}$$

$$(X)_{FTHR} = -N_{TN} \left[\frac{K_{TXR}}{L} \Delta X_{HRG} + G_{DN} \frac{K'_{TXR}}{L} \Delta X_{HRG} \right] \quad , \quad F_{THR}$$

FTHL

$$(Y)_{FTHR} = -N_{TN} \left[\frac{K_{TYR}}{L} \Delta Y_{HRG} + G_{DN} \frac{K'_{TYR}}{L} \Delta Y_{HRG} \right] \quad , \quad F_{THR+1}$$

FTHL+1

$$(Z)_{FTHR} = -N_{TN} \left[\frac{K_{TZR}}{L} \Delta Z_{HRG} + G_{DN} \frac{K'_{TZR}}{L} \Delta Z_{HRG} \right] \quad , \quad F_{THR+2}$$

FTHL+2

WHERE $\Delta X = (\Delta X_t - \Delta X_{(t-1)}) \frac{1}{\Delta T}$

$\Delta Y = (\Delta Y_t - \Delta Y_{(t-1)}) \frac{1}{\Delta T}$

$\Delta Z = (\Delta Z_t - \Delta Z_{(t-1)}) \frac{1}{\Delta T}$

THIS TIRE DAMPING
COMPONENT IS FILTERED
THROUGH A FIRST ORDER
LAG. $T = .01$

NOTE VERTICAL TIRE LOADS ARE PROVIDED AS LOAD ~ DEFLECTION NON LINEAR MAPS. IN ORDER TO PRESERVE CALCULATION LOOP INTEGRITY AN ARTIFICIAL K'_{TZR} IS DERIVED (=MAP OUTPUT / ΔZ). THIS STIFFNESS SHOULD NOT BE CONFUSED WITH K'_{TZR} WHICH IS A LOCAL SLOPE TIRE INPLANE STIFFNESS MAYBE EQUATIONS OR MAPS.

Desig'd	NAME	DATE	SIKORSKY AIRCRAFT	Page	TEMP	PERM
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TIRE FORCES IN THE GROUND PLANE (ALIGNED ALONG HELICOPTER BODY AXES)

TEMPORARILY SET $\psi_0 = 0$

$$\begin{bmatrix} X \\ Y \\ Z \end{bmatrix} = \begin{bmatrix} A_{HBS} \\ \psi_0 = 0 \end{bmatrix} \begin{bmatrix} X \\ Y \\ Z \end{bmatrix}$$

F_{TDN}	F_{THN}	F_{TDN} , +1, +2
F_{TDR}	F_{THR}	F_{TDR} , +1, +2
F_{TDL}	F_{THL}	F_{TDL} , +1, +2

LOGIC FOR MAIN GEAR BRAKES

IF BRAKES ARE SET $K_{HBS} = B_{BK}$

IF BRAKES ARE OFF $K_{HBS} = N_{BK}$

MAXIMUM REACTION THAT CAN BE SUSTAINED IN THE X DIRECTION

$$\begin{pmatrix} X \end{pmatrix} \begin{matrix} F_{TDNM} \\ F_{TDRM} \\ F_{TDLM} \end{matrix} = K_{HBS} \cdot M (171) \begin{matrix} F_{TDN} \\ F_{TDR} \\ F_{TDL} \end{matrix} \begin{matrix} F_{TDNM} \\ F_{TDRM} \\ F_{TDLM} \end{matrix}$$

NOTE $K_{HBS} = N_{BK}$ ALWAYS FOR NOSE WHEEL

Prepared	JR	NAME	DATE	SIKORSKY AIRCRAFT	Page	TEMP	PERM
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CHECK FOR LONGITUDINAL TIRE SLIP

• IF $(|X|)_{F_{DN}, F_{DR}, F_{DL}} > (X)_{F_{DNM}, F_{DRM}, F_{DRM}}$

THEN TIRE IS SLIPPING, SET SLIP FLAG AND

SET $(X)_{F_{DN}, F_{DR}, F_{DL}} = \begin{bmatrix} X_F \\ X_F \end{bmatrix} \mu_{SLIP} (X)_{F_{DNM}, F_{DRM}, F_{DRM}}$
TDR
TDL

• IF $(|X|)_{F_{DN}, F_{DR}, F_{DL}} < (X)_{F_{DNM}, F_{DRM}, F_{DRM}}$

THEN TIRE IS NOT SLIPPING

SET $(X)_{F_{DN}, F_{DR}, F_{DL}} = (X)_{F_{DNM}, F_{DRM}, F_{DRM}}$

MAXIMUM REACTION THAT CAN BE SUSTAINED IN THE
LATERAL DIRECTION

$(Y)_{F_{DNM}, F_{DRM}, F_{DRM}} = \mu (|Z|)_{F_{DN}, F_{DR}, F_{DL}}$

CHECK FOR LATERAL TIRE SLIP

• IF $(|Y|)_{F_{DN}, F_{DR}, F_{DL}} > (Y)_{F_{DNM}, F_{DRM}, F_{DRM}}$

THEN TIRE IS SLIPPING, SET SLIP FLAG AND

SET $(Y)_{F_{DN}, F_{DR}, F_{DL}} = \begin{bmatrix} Y_E \\ Y_F \end{bmatrix} \mu_{SLIP} (Y)_{F_{DNM}, F_{DRM}, F_{DRM}}$
TDR
TDL

• IF $(|Y|)_{F_{DN}, F_{DR}, F_{DL}} < (Y)_{F_{DNM}, F_{DRM}, F_{DRM}}$

THEN TIRE IS NOT SLIPPING

SET $(Y)_{F_{DN}, F_{DR}, F_{DL}} = (Y)_{F_{DNM}, F_{DRM}, F_{DRM}}$

Project	NAME	DATE	SIKORSKY AIRCRAFT	PAGE	TEMP	ROOM
Checked	TITLE			FILE		
Drawn	Description			Reference		

CORRECTED TIRE LOADS IN HELICOPTER BODY AXES

TEMPORARILY SET $\psi_0 = 0$

$$\begin{bmatrix} X \\ Y \\ Z \end{bmatrix} = \begin{bmatrix} A_{HSB} \\ \psi_0 = 0 \end{bmatrix} \begin{bmatrix} X \\ Y \\ Z \end{bmatrix}$$

$\begin{matrix} F_{HN} \\ F_{HR} \\ F_{HL} \end{matrix}$
 $\begin{matrix} F_{DN} \\ F_{DR} \\ F_{DL} \end{matrix}$
 $\begin{matrix} F_{UN}, +1, +2 \\ F_{UR}, +1, +2 \\ F_{UL}, +1, +2 \end{matrix}$

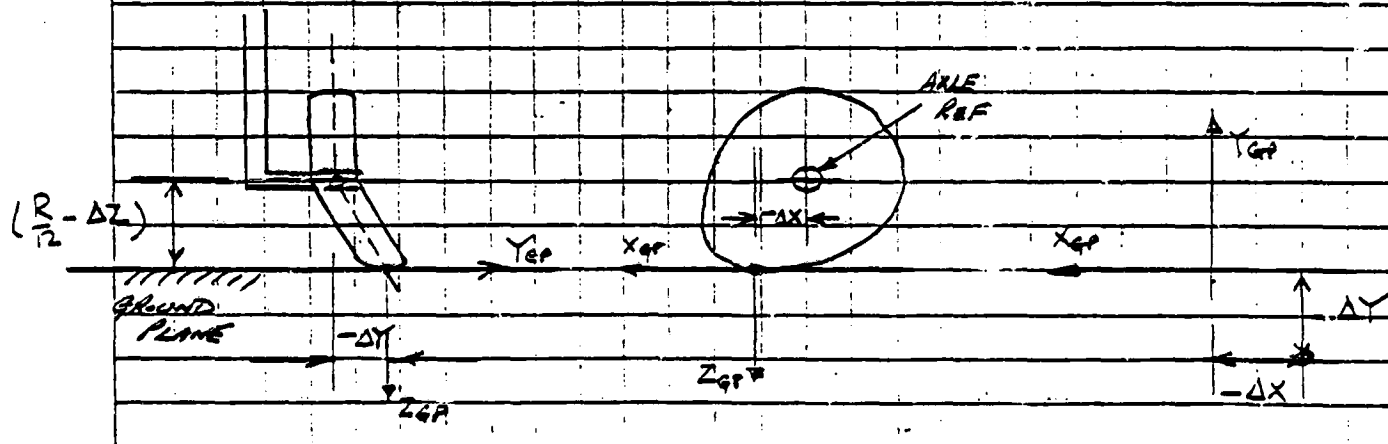
TIRE LOADS IN HELICOPTER AXES AT THE AXLE REFERENCE POINT

$$\begin{bmatrix} X \\ Y \\ Z \end{bmatrix} = \begin{bmatrix} X \\ Y \\ Z \end{bmatrix}$$

$\begin{matrix} F_{HN} \\ F_{HR} \\ F_{HL} \end{matrix}$
 $\begin{matrix} F_{DN} \\ F_{DR} \\ F_{DL} \end{matrix}$

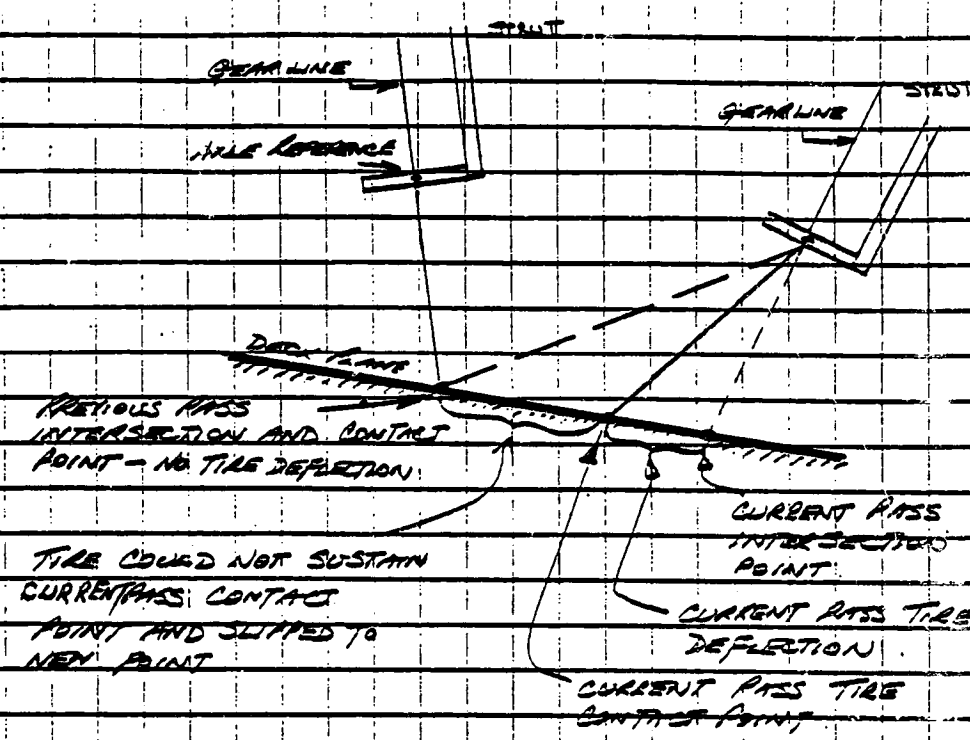
$$\begin{bmatrix} L \\ M \\ N \end{bmatrix} = \begin{bmatrix} Z F_{HN} (-\Delta Y) \frac{HNG}{HLG} - Y F_{HN} \left(\frac{R_{TN,M} - \Delta Z}{T} \frac{HNG}{HLG} \right) \\ Z F_{HR} (-\Delta X) \frac{HNG}{HLG} + X F_{HR} \left(\frac{R_{TN,M} - \Delta Z}{T} \frac{HNG}{HLG} \right) \\ Y F_{HL} (-\Delta X) \frac{HNG}{HLG} - X F_{HL} (-\Delta Y) \frac{HNG}{HLG} \end{bmatrix}$$

$\begin{matrix} F_{HN} \\ F_{HR} \\ F_{HL} \end{matrix}$
 $\begin{matrix} F_{DN} \\ F_{DR} \\ F_{DL} \end{matrix}$



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Drawing						Report No.

REINITIALISATION OF GEAR LINE / DECK INITIAL INTERSECTION POINT FOR NEXT PROGRAM PASS



IF TIRE IS NOT SLIPPING, JUMP OVER THE FOLLOWING EQUATIONS

$$\text{SET } (\Delta X, \Delta Y)_{SLIP} = 0$$

IF TIRE IS SLIPPING IN THE X DIRECTION THE TIRE DEFLECTION IS GIVEN BY

$$(\Delta X_{SLIP})_{N,R,L} = - \frac{(X)_{FTHN, FTHR, FTHL}}{K_{TNN, TNR, TNL}} \begin{matrix} DELSDN \\ DELSDR \\ DELSDL \end{matrix}$$

IF TIRE IS SLIPPING IN THE Y DIRECTION THE TIRE DEFLECTION IS GIVEN BY

$$(\Delta Y_{SLIP})_{N,R,L} = - \frac{(Y)_{FTHN, FTHR, FTHL}}{K_{TNN, TNR, TNL}} \begin{matrix} DELSDN + 1 \\ DELSDR + 1 \\ DELSDL + 1 \end{matrix}$$

Prepared	NAME	DATE	SIKORSKY AIRCRAFT	FIG.	TEMP	REV.
Checked	TITLE			MODEL		
Drawing				FIGURE NO.		

AMOUNT OF WHEEL SLIP IN THE GROUND PLANE

$$\begin{array}{|c|} \hline \Delta X \\ \hline \Delta Y \\ \hline \Delta Z \\ \hline \end{array} = A_{HAS} \begin{array}{|c|} \hline \Delta X \\ \hline \Delta Y \\ \hline \Delta Z \\ \hline \end{array}$$

$\begin{matrix} \text{SLPSN} \\ \text{SLPSR} \\ \text{SLPSL} \end{matrix}$

 $\begin{matrix} \text{SLPSN} \\ \text{SLPSR} \\ \text{SLPSL} \end{matrix}$

ΔX	ΔY	ΔZ
ΔX	ΔY	ΔZ
ΔX	ΔY	ΔZ

ΔX : DELSSN, DELSSR, DELSSL
 ΔY : +1, +1, +1
 ΔZ : +2, +2, +2

INITIAL INTERSECTION POINT AS MODIFIED BY TIRE SLIP
FOR NEXT PROGRAM PASS

$$(X_{ID})_{\substack{NGO \\ RGO \\ LGO}} = (X_{INT})_{\substack{NG \\ RG \\ LG}} - (\Delta X_{SLPS})_{\substack{N \\ R \\ L}}$$

ΔX_{SLPS} : IDNG
 ΔX_{SLPS} : IDRG
 ΔX_{SLPS} : IDLG

$$(Y_{ID})_{\substack{NGO \\ RGO \\ LGO}} = (Y_{INT})_{\substack{NG \\ RG \\ LG}} - (\Delta Y_{SLPS})_{\substack{N \\ R \\ L}}$$

ΔY_{SLPS} : IDNG+1
 ΔY_{SLPS} : IDRG+1
 ΔY_{SLPS} : IDLG+1

ACCELERATION OF THE UNSPRUNG WHEEL MASS

NOSE GEAR

- IF $Z_{FTAN} > -F_{PREN}$
 SET $\ddot{\delta}_{NG} = \dot{\delta}_{NG} = \delta_{NG} = 0$, DL_{NG} , DL_{NG} , DL_{NG}
 $F_{STAN} = -Z_{FTAN}$, F_{STRN}

- IF $Z_{FTAN} \leq -F_{PREN}$ THIS LOGIC SHOULD BE RETAINED IF $\delta_{NG} > 0$

$$\ddot{\delta}_{NG} = -\frac{1}{M_{NG}} \{ Z_{FTAN} + F_{STRN} \}$$

$$\dot{\delta}_{NG} = \int \ddot{\delta}_{NG} dt$$

$$\delta_{NG} = \int \dot{\delta}_{NG} dt$$

FOR NEXT PROGRAM PASS

$$F_{STRN} = \frac{\dot{\delta}_{NG}}{|\dot{\delta}_{NG}|} \{ C_{NG} (\dot{\delta}_{NG})^2 + F_{DNNGO} \} + F_{STFN}$$

WHERE

$$F_{STFN} = \left\{ \frac{C_{PVN}}{\left(\frac{C_{PVN}}{F_{PREN}} \right) - \delta_{NG}} \right\}$$

, F_{STRN}
 , F_{STFN}

DENOMINATOR LIMITED TO 7.01

RIGHT MAIN GEAR (LEFT GEAR SIMILAR)

- IF $Z_{FTAR} > -F_{PREM}$

$$\text{SET } \ddot{\delta}_{RG} = \dot{\delta}_{RG} = \delta_{RG} = 0 \quad , DL..RG, DL.RG, DLRG$$

$$F_{STRR} = -Z_{FTAR} \quad , F_{STRR}$$

- IF $Z_{FTAR} \leq -F_{PREM}$ THIS LOGIC SHOULD BE RETAINED IF $\delta_{RG} > 0$

$$\ddot{\delta}_{RG} = -\frac{1}{M_{MG}} \{ Z_{FTAR} + F_{STRR} \}$$

$$\dot{\delta}_{RG} = \int \ddot{\delta}_{RG} dt$$

$$\delta_{RG} = \int \dot{\delta}_{RG} dt$$

FOR NEXT PROGRAM PASS

$$F_{STRR} = \frac{\dot{\delta}_{RG}}{|\dot{\delta}_{RG}|} \{ C_{RG} (\dot{\delta}_{RG})^2 + F_{DRG0} \} + F_{STRR}$$

WHERE

$$F_{STRR} = \left\{ \frac{C_{PVM}}{\left(\frac{C_{PVM}}{F_{PREM}} \right) - \delta_{RG}} \right\}$$

DENOMINATOR LIMITED TO $> .01$



GEAR LOADS AT THE GEAR REFERENCE POINT

- IF STRUT LOADS ARE TO BE BYPASSED (STLDSW ≠ 0)

$$\begin{pmatrix} Z \\ FGRN \\ FGRR \\ FGRL \end{pmatrix} = \begin{pmatrix} Z \\ FTAN \\ FTAR \\ FTAL \end{pmatrix} \begin{matrix} , FG RN + 2 \\ , FG RR + 2 \\ , FG RL + 2 \end{matrix}$$

- IF STRUT LOADS ARE TO BE USED (STLDSW = 0)

$$\begin{pmatrix} Z \\ FGRN \\ FGRR \\ FGRL \end{pmatrix} = - (F) \begin{pmatrix} STRN \\ STRR \\ STRL \end{pmatrix}$$

VERTICAL GEAR LOAD FILTERS

$$\begin{pmatrix} Z \\ FGRN \\ FGRR \\ FGRL \end{pmatrix} = \begin{pmatrix} Z \\ FGRN \\ FGRR \\ FGRL \end{pmatrix} * \frac{1}{(1 + T_{FGRS})}$$

MOMENTS AT REFERENCE POINT

$$\begin{pmatrix} L \\ FGRN \\ FGRR \\ FGRL \end{pmatrix} = -Y \begin{pmatrix} FTAN \\ FTAR \\ FTAL \end{pmatrix} \begin{pmatrix} Z NGA - \delta NG \\ RMGA \\ LMGA \end{pmatrix} - \begin{pmatrix} \delta NG \\ RG \\ LG \end{pmatrix} + L \begin{pmatrix} FTAN \\ FTAR \\ FTAL \end{pmatrix} \begin{matrix} , FG RN + 3 \\ , FG RR + 3 \\ , FG RL + 3 \end{matrix}$$

$$\begin{pmatrix} M \\ FGRN \\ FGRR \\ FGRL \end{pmatrix} = X \begin{pmatrix} FTAN \\ FTAR \\ FTAL \end{pmatrix} \begin{pmatrix} Z NGA - \delta NG \\ RMGA \\ LMGA \end{pmatrix} - \begin{pmatrix} \delta NG \\ RG \\ LG \end{pmatrix} + M \begin{pmatrix} FTAN \\ FTAR \\ FTAL \end{pmatrix} \begin{matrix} , FG RN + 4 \\ , FG RR + 4 \\ , FG RL + 4 \end{matrix}$$

$$\begin{pmatrix} N \\ FGRN \\ FGRR \\ FGRL \end{pmatrix} = N \begin{pmatrix} FTAN \\ FTAR \\ FTAL \end{pmatrix} \begin{matrix} , FG RN + 5 \\ , FG RR + 5 \\ , FG RL + 5 \end{matrix}$$

GEAR LOADS AT THE HELICOPTER CG

$$X_{LG} = X_{FTAN} + X_{FTAR} + X_{FTAL} \quad , X_{LG}$$

$$Y_{LG} = Y_{FTAN} + Y_{FTAR} + Y_{FTAL} \quad , Y_{LG}$$

$$Z_{LG} = Z_{FGRN} + Z_{FGRR} + Z_{FGRL} \quad , Z_{LG}$$

$$L_{LG} = Y_{RMGA} (Z_{FGRR} - Z_{FGRL}) + L_{FGRN} + L_{FGRR} + L_{FGRL} \quad , L_{LG}$$

$$M_{LG} = -Z_{FGRN} X_{NGA} - (Z_{FGRR} + Z_{FGRL}) X_{RMGA} + M_{FGRN} + M_{FGRR} + M_{FGRL} \quad , M_{LG}$$

$$N_{LG} = Y_{FTAN} X_{NGA} + (Y_{FTAR} + Y_{FTAL}) X_{RMGA} + (X_{FTAL} - X_{FTAR}) Y_{RMGA} + N_{FGRN} + N_{FGRR} + N_{FGRL} \quad , N_{LG}$$

5.7.3 LANDING INTERFACE MODULE INPUT/OUTPUT DATA TRANSFER

INPUT TRANSFER	
PARAMETER	ORIGIN MODULE
FSCGB WLCGB	MAIN ROTOR
VN VE VZ	MOTION
THETAB PHIB PSIB	
[AHBS]	

OUTPUT TRANSFER	
PARAMETER	DESTINATION MODULE
XLG YLG ZLG LLG MLG NLG	MOTION

5.7.4

NOTATION FOR GEN. HEL. LANDING SIMULATION

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
X_{SH}	SH	FT	Helicopter center of gravity position in space axes.
Y_{SH}	SH+1	FT	
Z_{SH}	SH+2	FT	
X_{SH0}	SH0	FT	Initial position of helicopter in space axes.
Y_{SH0}	SH0+1	FT	
Z_{SH0}	SH0+2	FT	
V_N	V_N	FT/SEC	Helicopter space axes velocities.
V_E	V_E	FT/SEC	
V_Z	V_Z	FT/SEC	
FSCGB	FSCGB	INS.	Helicopter c.g. less rotor.
WLCGB	WLCGB	INS.	
FSNGA	FSNGA	INS.	Fuse. stat. of nose gear strut.
FSMGA	FSMGA	INS.	Fuse. stat. of main gear strut.
BLMGA	BLMGA	INS.	Butt. stat. of main gear strut

NOTATION FOR GEN. HEL. LANDING SIMULATION

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
WLNGAO	WLNGAO	INS	WL. position of free extension of the axle.
WLMGAO	WLRMGA	INS	
X _{NGA}	NGA	FT	Position of freely extended axles in body axes.
X _{MGA}	RMGA, LMGA,	FT	
Y _{NGA}	NGA+1	FT	
Y _{RMGA}	RMGA+1	FT	
Y _{LMGA}	LMGA+1	FT	
Z _{NGA}	NGA+2	FT	
Z _{MGRA}	RMGA+2	FT	
Z _{MGLA}	LMGA+2	FT	
R _{TN}	RTNF	INS	Nominal nose tire radius.
R _{TM}	RTMF	INS	Nominal main tire radius.
[A _{HBS}]	Defined in the Motion Module		Helicopter body to space axes transformation matrix.
X _{NTSO}	NTSO	FT	Nose tire reference position in space axes under free extension.
Y _{NTSO}	NTSO+1	FT	
Z _{NTSO}	NTSO+2	FT	
X _{RMTSO}	RMTSO	FT	Right tire reference position in space axes under free extension.
Y _{RMTSO}	RMTSO+1	FT	
Z _{RMTSO}	RMTSO+2	FT	

NOTATION FOR GEN. HEL. LANDING SIMULATION

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
X _{LMTSO}	LMTSO	FT	Left tire reference position in space axes under free extension.
Y _{LMTSO}	LMTSO+1	FT	
Z _{LMTSO}	LMTSO+2	FT	
δ _{NG}	DLNG	FT	Nose strut deflection Right Left
δ _{RG}	DLRG	FT	
δ _{LG}	DLLG	FT	
X _{NAS}	NAS	FT	Nose axle reference position in space axes.
Y _{NAS}	NAS+1	FT	
Z _{NAS}	NAS+2	FT	
X _{RAS}	RAS	FT	Right axle reference position in space axes.
Y _{RAS}	RAS+1	FT	
Z _{RAS}	RAS+2	FT	
X _{LAS}	LAS	FT	Left axle reference position in space axes.
Y _{LAS}	LAS+1	FT	
Z _{LAS}	LAS+2	FT	
X _{NGSR}	NGSR	FT	Nose gear reference position in space axes
Y _{NGSR}	NGSR+1	FT	
Z _{NGSR}	NGSR+2	FT	

NOTATION FOR GEN. HEL. LANDING SIMULATION

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
X MGRSR	RMGSR	FT	Right gear reference position in space axes
Y MGRSR	RMGSR+1	FT	
Z MGRSR	RMGSR+2	FT	
X MGLSR	LMGSR	FT	Left gear reference position in space axes
Y MGLSR	LMGSR+1	FT	
Z MGLSR	LMGSR+2	FT	
WLFD	WLFD	FT	Height of ground plane.
d_{NNA}	DNNA	FT	Normal distance from the ground plane to axle reference point for nose, right and left gear.
d_{NRA}	DNRA	FT	
d_{NLA}	DNLA	FT	
$\cos \alpha_{NG}$	CSANG		Direction cosines of gear line.
$\cos \beta_{NG}$	CSBNG		
$\cos \gamma_{NG}$	CSGNG		
δ_{TNG}	DLTNG	FT	Nose tire radial deflection.
δ_{TRG}	DLTRG	FT	Right tire radial deflection.
δ_{TLG}	DLTLG	FT	Left tire radial deflection.
(a) NG, RG, LG			Coefficients of gear line equation projections in 2D for the nose, right and left landing gear.
(b) NG, RG, LG			
(c) NG, RG, LG			
(d) NG, RG, LG			

NOTATION FOR GEN. HEL. LANDING SIMULATION

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
$(X_{ID})_{NGO}$	IDNG	FT	Initial deck contact positions before distortion of tires.
$(Y_{ID})_{NGO}$	IDNG+1	FT	
$(Z_{ID})_{NGO}$	IDNG+2	FT	
$(X_{ID})_{RGO}$	IDRG	FT	
$(Y_{ID})_{RGO}$	IDRG+1	FT	
$(Z_{ID})_{RGO}$	IDRG+2	FT	
$(X_{ID})_{LGO}$	IDLG	FT	
$(Y_{ID})_{LGO}$	IDLG+1	FT	
$(Z_{ID})_{LGO}$	IDLG+2	FT	
$(X_{INT})_{NG}$	INTNG	FT	Intersection of gear line with deck plane in space axes for the nose gear.
$(Y_{INT})_{NG}$	INTNG+1	FT	
$(Z_{INT})_{NG}$	INTNG+2	FT	
$(X_{INT})_{RG}$	INTRMG	FT	Intersection of gear line with deck plane in space axes for the right gear.
$(Y_{INT})_{RG}$	INTRMG+1	FT	
$(Z_{INT})_{RG}$	INTRMG+2	FT	
$(X_{INT})_{LG}$	INTLMG	FT	Intersection of gear line with deck plane in space axes for for left gear.
$(Y_{INT})_{LG}$	INTLMG+1	FT	
$(Z_{INT})_{LG}$	INTLMG+2	FT	
$(\Delta X)_{NG}$	DELNG	FT	Tire deflections in the deck plane for the nose gear.
$(\Delta Y)_{NG}$	DELNG+1	FT	
$(\Delta Z)_{NG}$	DELNG+2	FT	
$(\Delta X)_{RG}$	DELRNG	FT	Tire deflections in the deck plane for the right gear.
$(\Delta Y)_{RG}$	DELRNG+1	FT	
$(\Delta Z)_{RG}$	DELRNG+2	FT	

NOTATION FOR GEN. HEL. LANDING SIMULATION

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
$(\Delta X)_{LG}$	DELLMG	FT	Tire deflections in the deck plane for the left gear.
$(\Delta Y)_{LG}$	DELLMG+1	FT	
$(\Delta Z)_{LG}$	DELLMG+2	FT	
$(\Delta X)_{HNG}$	DELHNG	FT	Tire deflections in helicopter body axes for nose gear.
$(\Delta Y)_{HNG}$	DELHNG+1	FT	
$(\Delta Z)_{HNG}$	DELHNG+2	FT	Tire deflections in helicopter body axes for right gear.
$(\Delta X)_{HRG}$	DELHRG	FT	
$(\Delta Y)_{HRG}$	DELHRG+1	FT	
$(\Delta Z)_{HRG}$	DELHRG+2	FT	Tire deflection in helicopter body axes for left gear.
$(\Delta X)_{HLG}$	DELHLG	FT	
$(\Delta Y)_{HLG}$	DELHLG+1	FT	
$(\Delta Z)_{HLG}$	DELHLG+2	FT	
$[A_{HSB}]$	$(= [A_{HBS}]^T)$		Helicopter space to body axes transformation matrix.
$(KTX)_{N, R, L}$	KTXN,R,L	lb/FT	Tire 3 component stiffness coefficients - Note that these can be replaced by loading maps.
$(KTY)_{N, R, L}$	KTYN,R,L	lb/FT	
$(KTZ)_{N, R, L}$	KYZN,R,L	lb/FT	
$(X)_{FTHN}$	FTHN	LB	Tire forces in helicopter body axes for the nose gear.
$(Y)_{FTHN}$	FTHN+1	LB	
$(Z)_{FTHN}$	FTHN+2	LB	

NOTATION FOR GEN. HEL. LANDING SIMULATION

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
(X) _{FTHR}	FTHR	LB	Tire forces in helicopter body axes for the right gear.
(Y) _{FTHR}	FTHR+1	LB	
(Z) _{FTHR}	FTHR+2	LB	
(X) _{FTHL}	FTHL	LB	Tire forces in helicopter body axes for the left gear.
(Y) _{FTHL}	FTHL+1	LB	
(Z) _{FTHL}	FTHL+2	LB	
d_M	DMDM	INCHES	Main gear tire diameter
d_N	DNDN	INCHES	Nose gear tire diameter
P_M	PMRPMR	lb/in ²	Main gear tire pressure
P_{RM}	RAT PRM	lb/in ²	Main gear rated tire pressure
N_{TN}	NTN	-	No. of tail gear wheels
N_{TM}	NTM	-	No. of main gear wheels
P_N	PNPN	lb/in ²	Nose gear tire pressure
P_{RN}	RAT PRN	in/in ²	Nose gear rated tire pressure
(X) _{FTDN}	FTDN	LB	Tire forces in the space axes but aligned along helicopter X axis for the nose gear.
(Y) _{FTDN}	FTDN+1	LB	
(Z) _{FTDN}	FTDN+2	LB	
(X) _{FTDR}	FTDR	LB	Tire forces in the space axes but aligned along helicopter X axis for the right gear.
(Y) _{FTDR}	FTDR+1	LB	
(Z) _{FTDR}	FTDR+2	LB	
(X) _{FTDL}	FTDL	LB	Tire forces in the space axes but aligned along helicopter X axis for the left gear.
(Y) _{FTDL}	FTDL+1	LB	
(Z) _{FTDL}	FTDL+2	LB	

NOTATION FOR GEN. HEL. LANDING SIMULATION

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
$K_{\mu B}$	KMUXX		Factor for brakes
BRK	KMUXBK		Value of $K_{\mu B}$ brakes on.
NBRK	KMUXO		Value of $K_{\mu B}$ brakes off.
μ	KMU		Longitudinal tire coef. of friction.
X_{FTDNM}	FTDNM	LB	Max. friction load that can be sustained in the X direction.
X_{FTDRM}	FTDRM	LB	
X_{FTDLM}	FTDLM	LB	
K_{SLIP}	KSLIP		Factor for sliding friction.
Y_{FTDNM}	FTDNM+1	LB	Max. friction load that can be sustained in the Y direction.
Y_{FTDRM}	FTDRM+1	LB	
Y_{FTDLM}	FTDLM+1	LB	
$((\Delta X)_{SLP})_N$	DELSPLN	FT	Amount of tire deflection sustained during slip conditions in the X direction.
$((\Delta X)_{SLP})_R$	DELSPLR	FT	
$((\Delta X)_{SLP})_L$	DELSPLL	FT	
$(\Delta Y_{SLP})_N$	DELSPLN+1	FT	Amount of tire deflection sustained during slip conditions in the Y direction.
$(\Delta Y_{SLP})_R$	DELSPLR+1	FT	
$(\Delta Y_{SLP})_L$	DELSPLL+1	FT	

NOTATION FOR GEN. HEL. LANDING SIMULATION

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
(X) FTAN	FTAN	LB	Tire forces and moments at the axle reference position, nose gear.
(Y) FTAN	FTAN+1	LB	
(Z) FTAN	FTAN+2	LB	
(L) FTAN	FTAN+3	FT	
(M) FTAN	FTAN+4	FT	
(N) FTAN	FTAN+5	FT	
(X) FTAR	FTAR	LB	Tire forces and moments at the axle reference position, right gear.
(Y) FTAR	FTAR+1	LB	
(Z) FTAR	FTAR+2	LB	
(L) FTAR	FTAR+3	FT	
(M) FTAR	FTAR+4	FT	
(N) FTAR	FTAR+5	FT	
(X) FTAL	FTAL	LB	Tire forces and moments at the axle reference position, left gear.
(Y) FTAL	FTAL+1	LB	
(Z) FTAL	FTAL+2	LB	
(L) FTAL	FTAL+3	FT	
(M) FTAL	FTAL+4	FT	
(N) FTAL	FTAL+5	FT	
(Δ X) SLPSN	DELSSN	FT	Amount of slippage of gear intersection point is space axes, nose gear.
(Δ Y) SLPSN	DELSSN+1	FT	
(Δ Z) SLPSN	DELSSN+2	FT	

NOTATION FOR GEN. HEL. LANDING SIMULATION

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
$(\Delta X)_{SLPSR}$	DELSSR	FT	Amount of slippage of gear intersection point is space axes, right gear.
$(\Delta Y)_{SLPSR}$	DELSSR+1	FT	
$(\Delta Z)_{SLPSR}$	DELSSR+2	FT	
$(\Delta X)_{SLPSL}$	DELSSL	FT	Amount of slippage of gear intersection point is space axes, left gear.
$(\Delta Y)_{SLPSL}$	DELSSL+1	FT	
$(\Delta Z)_{SLPSL}$	DELSSL+2	FT	
$(\ddot{\delta})_{NG}$	DL..NG	FT	Wheel axle motion along strut line
$(\ddot{\delta})_{RG}$	DL..RG	FT	
$(\ddot{\delta})_{LG}$	DL..LG	FT	
$(\dot{\delta})_{NG}$	DL.NG	FT	
$(\dot{\delta})_{RG}$	DL.RG	FT	
$(\dot{\delta})_{LG}$	DL.LG	FT	
$(\delta)_{NG}$	DLNG	FT	
$(\delta)_{RG}$	DLRG	FT	
$(\delta)_{LG}$	DLLG	FT	
F_{PREN}	FPREN	LB	Nose gear strut preload.
F_{PREM}	FPREM	LB	Main gear strut preload.
F_{STRN}	FSTRN	LB	Strut force transferred to airframe for the nose, right and left gear.
F_{STRR}	FSTRR	LB	
F_{STRL}	FSTRL	LB	
M_{NG}	MASNG	SLUGS	Unsprung mass of nose gear.
M_{MG}	MASMG	SLUGS	Unsprung mass of main gear.

NOTATION FOR GEN. HEL. LANDING SIMULATION

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
C_{NG}	CNG	LB/(FT/SEC) ²	Velocity squared damping for nose and main gear struts
C_{MG}	CMG	LB/(FT/SEC) ²	
RTRTT	RTRTT	FT/SEC	Velocity break points on strut damping
LFRTT	LFRTT	FT/SEC	
CRGH	CRGH	LB/(FT/SEC) ²	Corresponding damping coef. of velocity squared damping for the main gear.
CLGH	CLGH	LB/(FT/SEC) ²	
CRGL	CRGL	LB/(FT/SEC) ²	
CLGL	CLGL	LB/(FT/SEC) ²	
KOLEOM	KOLEOM	LB-FT	Isothermal air spring coef. main gear.
	KSP+7	FT	Free extension of main gear.
	KSP+10	FT	
KOLEON	KOLEON	LB-FT	Isothermal air spring coef. nose gear
	KSP+6	FT	Free extension of nose gear.
(X) FGRN	FGRN	LB	Gear forces and moment at the gear reference positions - nose gear.
(Y) FGRN	FGRN+1	LB	
(Z) FGRN	FGRN+2	LB	
(L) FGRN	FGRN+3	FT	
(M) FGRN	FGRN+4	FT	
(N) FGRN	FGRN+5	FT	
(X) FGRR	FGRR	LB	Gear forces and moment at the gear reference positions - right gear.
(Y) FGRR	FGRR+1	LB	
(Z) FGRR	FGRR+2	LB	
(L) FGRR	FGRR+3	FT	
(M) FGRR	FGRR+4	FT	
(N) FGRR	FGRR+5	FT	

NOTATION FOR GEN. HEL. LANDING SIMULATION

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
(X) FGRL	FGRL	LB	Gear forces and moment at the gear reference positions - left gear.
(Y) FGRL	FGRL+1	LB	
(Z) FGRL	FGRL+2	LB	
(L) FGRL	FGRL+3	FT	
(M) FGRL	FGRL+4	FT	
(N) FGRL	FGRL+5	FT	
X _{LG}	XLG	LB	Total gear loads at the helicopter CG in body axes.
Y _{LG}	YLG	LB	
Z _{LG}	ZLG	LB	
L _{LG}	LLG	FT	
M _{LG}	MLG	FT	
N _{LG}	NLG	FT	

5.7.5 BLACKHAWK LANDING MODULE INPUT DATA

MAIN GEAR

TIRE INPUT CONSTANTS:

- $R_{TM} = 12.65''$
 - $P_M = 140 \text{ PSI}$
 - $P_{RM} = 110 \text{ PSI}$
 - $N_{TM} = 1$
 - $F_{SMGA} = 297.43$
 - $W_{LMGA} = 184.8$
 - $B_{LMGA} = 53.2$
 - $M_{MG} = 2.6708 \text{ SLUGS}$
 - $G_{DR} = .003 (= G_{DL})$
- STRUT INPUT CONSTANTS:

- $C_{PYM} = 3465 \text{ FT/LBS}$
- $F_{PREM} = 3000 \text{ LB}$
- $\gamma = 1.34$

IF $|\dot{S}|_{RG, LG} \leq 1.4167 \text{ ft/sec}$

$C_{RG, LG} = 1382.4 \text{ lb} \frac{\text{sec}^2}{\text{ft}^2}$

$\frac{F_{DRGO}}{D_{LG0}} = 0$

IF $|\dot{S}|_{RG, LG} > 1.4167 \text{ ft/sec}$

$C_{RG, LG} = 216.0 \text{ lb} \frac{\text{sec}^2}{\text{ft}^2}$

$\frac{F_{DRGO}}{D_{LG0}} = 2340.98$

TABLE 7.5.1
SOLUTION FOR CUBE ROOT IN TIRE INPLANE STIFFNESS EQUATIONS

$$\text{LET } P = \left(\frac{12 \cdot \Delta Z_{HRG}, HRG, HLG}{DTN, TM} \right)$$

RANGE $0 \leq P \leq .02$	
P	$P^{1/3}$
0	0
.002	.12599
.004	.15874
.006	.18171
.008	.20000
.010	.21544
.012	.22894
.014	.24101
.016	.25198
.018	.26207
.020	.27144

RANGE $.02 < P \leq .3$	
P	$P^{1/3}$
.020	.27144
.04	.34200
.06	.39149
.08	.43089
.10	.46416
.12	.49324
.14	.51925
.16	.54288
.18	.56462
.20	.58480
.22	.60368
.24	.62145
.26	.63825
.28	.65421
.30	.66943

(MAIN GEAR CONT)

TIRE LONGITUDINAL STIFFNESS :

$$K_{TXR} = .53 D_{TM} (P_M + 4 P_{RM}) \left(\frac{12 \Delta Z_{HRG}}{D_{TM}} \right)^{1/3} \cdot 12 \quad 16/ft$$

$$K_{TXL} = .53 D_{TM} (P_M + 4 P_{RM}) \left(\frac{12 \Delta Z_{HLG}}{D_{TM}} \right)^{1/3} \cdot 12 \quad 16/ft$$

TABLE 7.5.2 TIRE LATERAL STIFFNESS $0 \leq \Delta Z_{HRG, HLG} \leq .11667$.

	P_M PSI		
	90	120	150
$12 * \Delta Z_{HRG} $ $12 * \Delta Z_{HLG} $	K_{TYR} K_{TYL}	16/INCH	
0	0	0	0
.1	950	1030	1090
.2	1260	1340	1430
.3	1430	1560	1700
.4	1560	1740	1920
.5	1630	1840	2080
.6	1680	1940	2210
.7	1710	2000	2310
.8	1730	2060	240
.9	1740	2080	2530
1.0	1740	2100	2560
1.1	1735	2155	2530
1.2	1730	2210	2500
1.3	1720	2155	2460
1.4	1710	2120	2420

NOTE: OUTPUT OF THIS MAP TO BE MULTIPLIED BY 12 TO OBTAIN 16/ft

(MAIN GEAR CONT)

LATERAL STIFFNESS (CONT)

$$.11667 \leq \Delta Z_{HRG,HLG} \leq .25$$

	90	120	150
1.4	1710	2120	2420
3.0	1360	1800	2140

NOTE: OUTPUT OF THIS
MAP TO BE MULTIPLIED
BY 12 TO OBTAIN lb/ft

TABLE 7.5.3 TIRE VERTICAL LOAD DEFLECTION DATA

	PM PSI		
	90	120	150
12* ΔZ _{HRG} 12* ΔZ _{HLG}	Z _{FTHR} , Z _{FTHL}		
0	0	0	0
1	3100	3800	4600
2	6900	8800	10600
3	11500	14200	17000
4	16300	19900	23900
5	21000	26300	31600
6	32100	42000	51900
7	48150	63000	77850
8	72220	94490	116760

$$\text{CALCULATE } K_{TZR, TZL} = \frac{Z_{FTHR, FTHL}}{\Delta Z_{HRG, HLG}}$$

FOR ENTRY INTO GENERAL EQUATIONS

(MAIN GEAR CONT)

TABLE 7.5.4. VERTICAL TIRE STIFFNESS (USED IN DAMPING CALCULATIONS)

	PM PSI		
	90	120	150
- ZFTHR; lb	KTZR, lb/inch		
- ZFTHL; lb	KTZL		
0	2970	3600	4170
4000	3710	4630	5390
8000	4320	5240	6030
12000	4600	5510	6300
16000	4780	5710	6500

NOTE OUTPUT OF THIS
MAP TO BE MULTIPLIED
BY 12 TO OBTAIN lb/in.

ORIGINAL PAGE IS
OF POOR QUALITY

TAIL GEAR

TIRE INPUT CONSTANTS :

$R_{TN} = 7.5''$ $D_{TN} = 15''$
 $P_N = 90 \text{ PSI}$
 $P_{RN} = 110 \text{ PSI}$
 $N_{TN} = 1$
 $FSNGA = 644.62$
 $WLNGA = 180.4$
 $BLNGA = 0$
 $MNG = 1.0 \text{ SLUGS}$
 $GDN = .003$

STRUT INPUT CONSTANTS :

$C_{PVN} = 2208.0 \text{ FT } b$
 $F_{AREN} = 1251.0$
 $\gamma = 1.34$

$C_{NG} = 66.72$
 $F_{DNG0} = 0.0$

(TAIL GEAR CONT)

TIRE LONGITUDINAL STIFFNESS

$$K_{TXN} = .53 D_{TN} (P_N + 4 P_{RN}) \left(\frac{12 \Delta Z_{HNG}}{D_{TN}} \right)^{1/3} \cdot 12 \text{ lb/ft}$$

TIRE LATERAL STIFFNESS

$$K_{TYN} = 19.7 (Z_{FTHN})^{1/2} \cdot 12 \text{ lb/ft}$$

TABLE 7.5.5 VERTICAL LOAD ~ DEFLECTION DATA

	P _N PSI		
	90	120	150
12 * ΔZ _{HNG}	Z _{FTHN} , lb		
0	0	0	0
.5	950	1150	1380
1.0	2100	2500	2900
1.5	3350	4000	4690
2.0	4700	5600	6500
2.5	6300	7450	8600
3.0	10000	11800	13700
3.5	18300	21900	25800
4.0	26600	32000	38000
4.5	34900	42100	50100
5.0	43200	52200	62300

(TAIL GEAR CONT)

VERTICAL STIFFNESS FOR USE IN THE TIRE DAMPING EQUATIONS
TABLE 7.5.6

Z_{FTAN}	K_{TZN}
0	2040
4000	2630
8000	4050
12000	6000
16000	6000

NOTE OUTPUT OF THIS MAP TO
BE MULTIPLIED BY 12 TO
OBTAIN lb/ft.

GENERAL LANDING MODULE INPUT DATA

- WLFD = 0
- μ = 0.6
- BRK = 1.0
- NBRK = .01
- KSLIP = .85

SOLUTION FOR CUBE ROOT IN THE INCLINE
STIFFNESS EQUATIONS

$$P = \frac{W \sin \theta}{D \sin \theta}$$

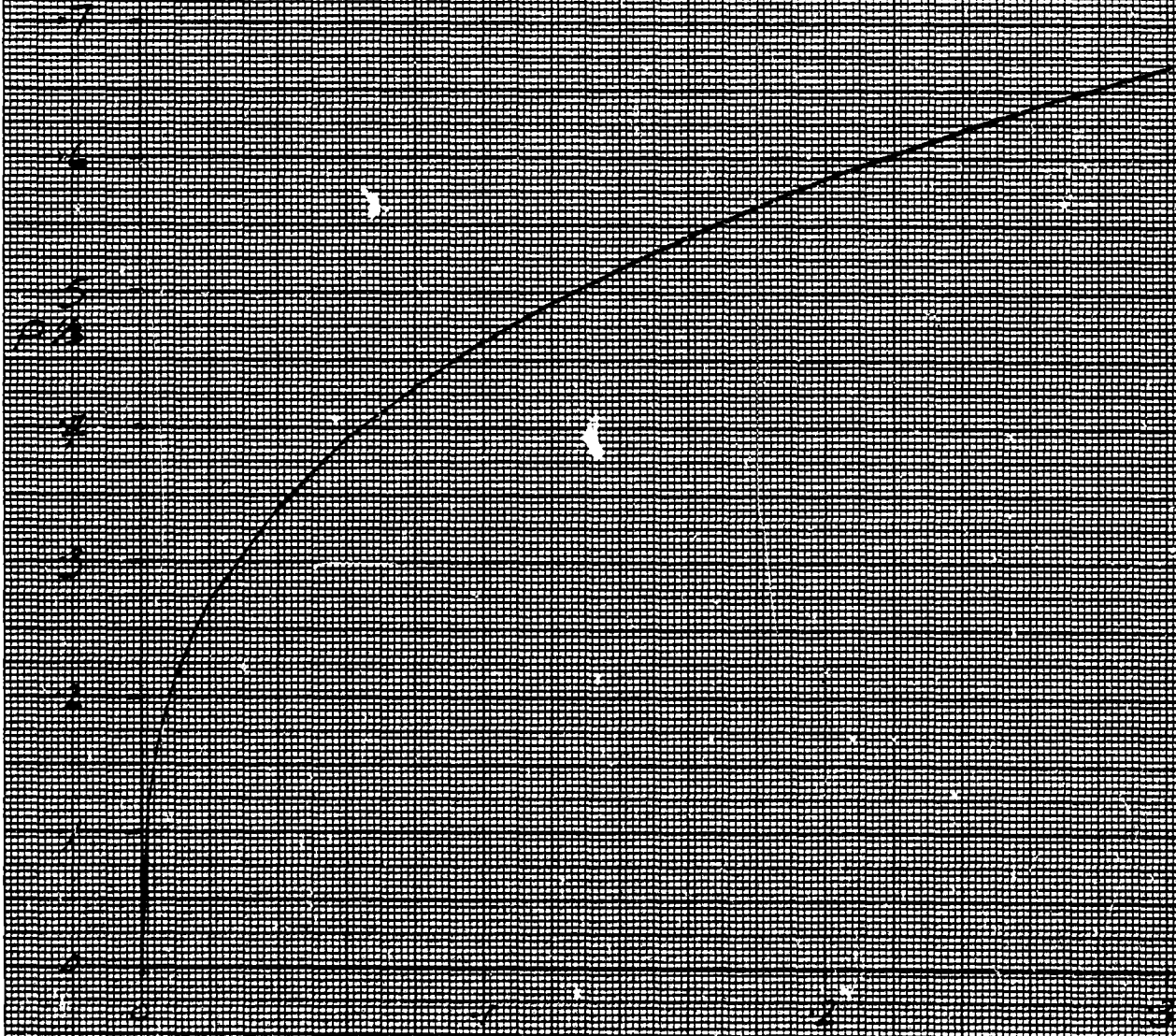
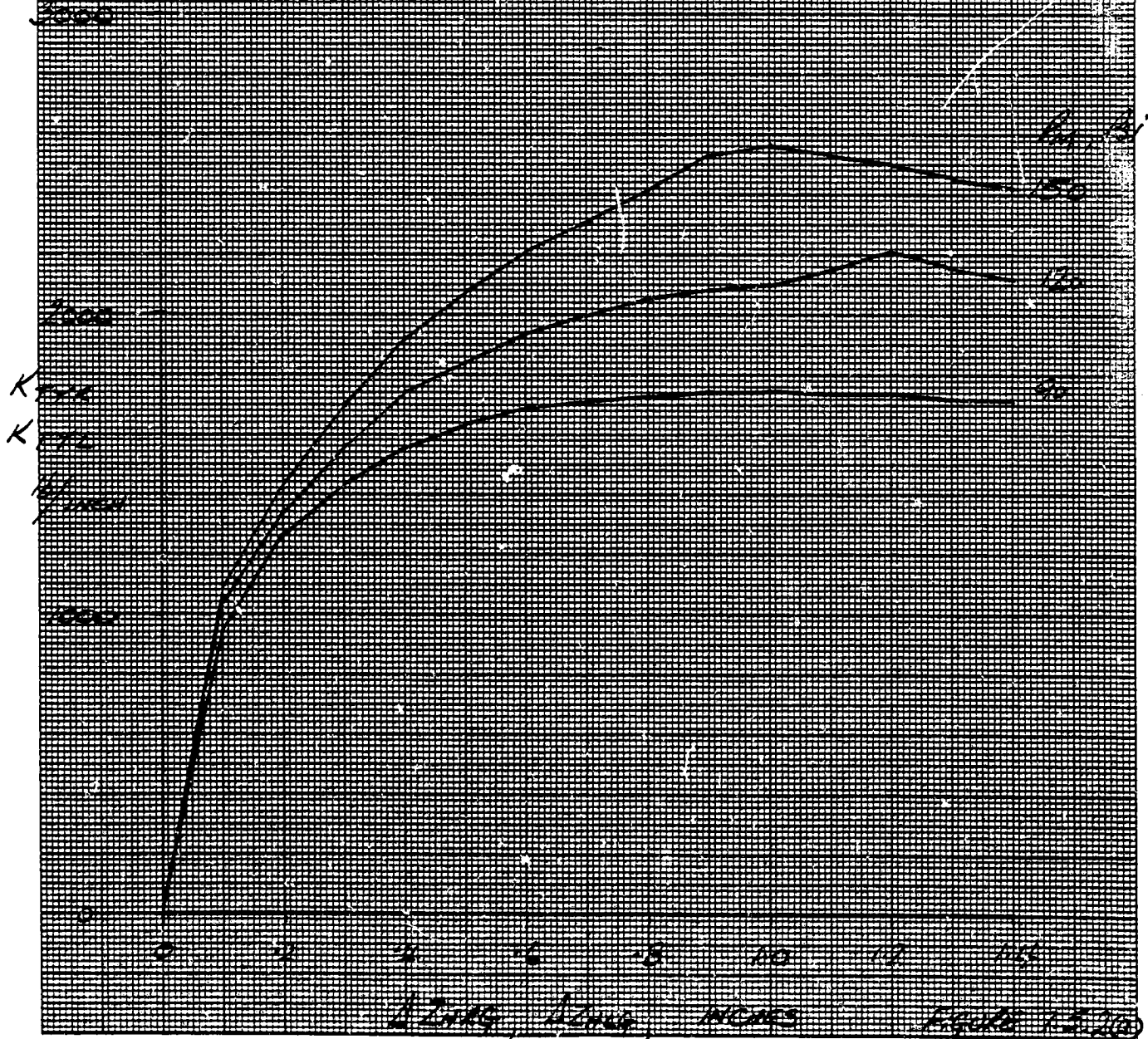


FIGURE 7.5.1

SOURCE 10 X 10 TO THE CENTIMETER AS 8014-01

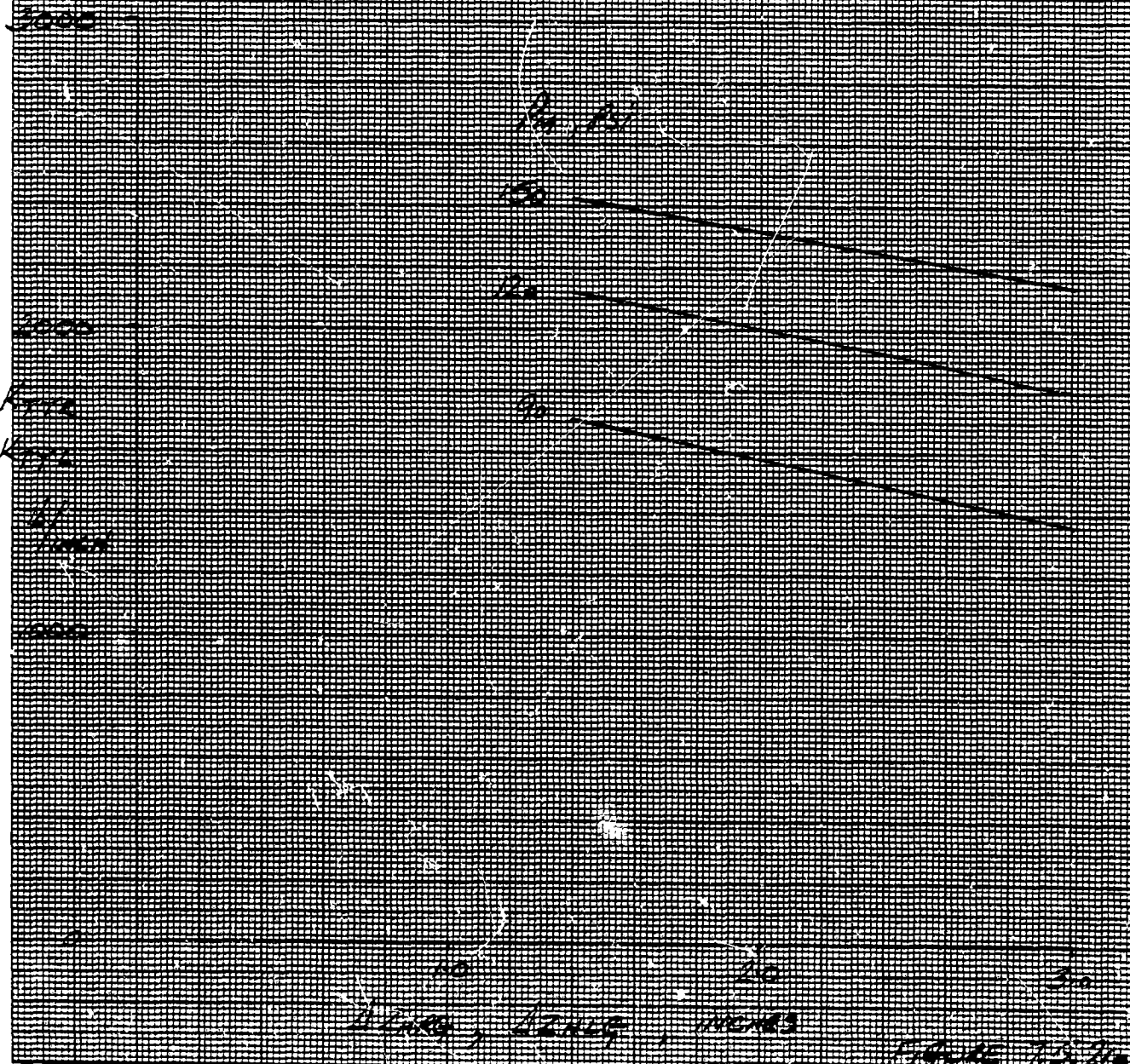
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*BLACK HAWK - MAIN LANDING GEAR
TIRE LATERAL STIFFNESS*



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SQUARE 10 X 10 TO THE CENTIMETER AS 8014-01

Black Hawk - Main Landing Gear
THE LATERAL STIFFNESS



SQUARE 10 X 10 TO THE CENTIMETER AS 8010-01

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FIGURE 7.3.21

BUMPER HAWK - MAIN LANDING GEAR

TIRE VERTICAL ROAD DEFLECTION

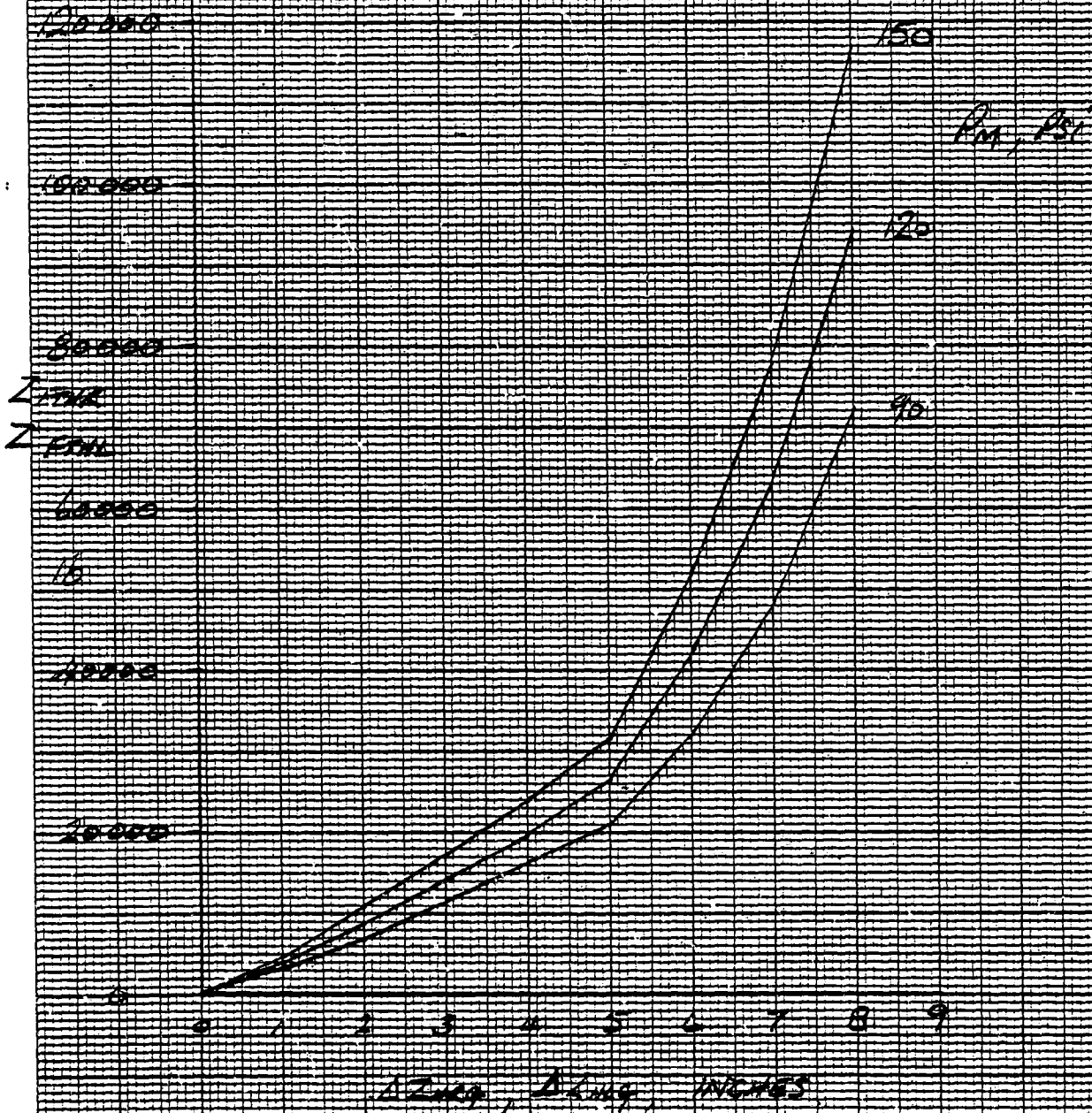


FIGURE 15.3

BLACK HAWK - MAIN LANDING GEAR
TIRE VERTICAL STIFFNESS
(used only for tire bending)

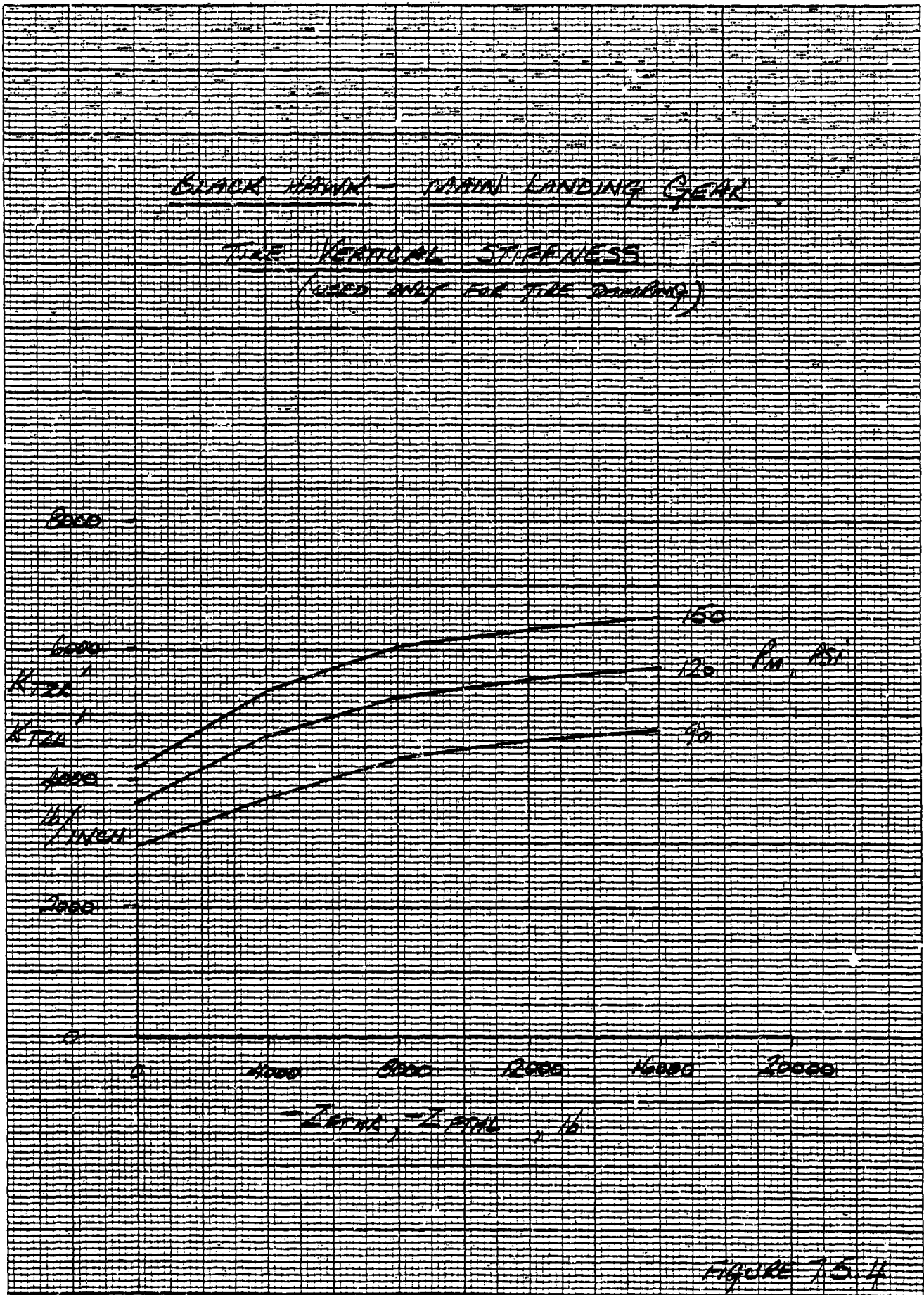


FIGURE 75-4

SERIAL 10 X 10 THE HALF INCH AS 89-3-81

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GRAPH PAPER

Black Hawk - Tail Landing Gear
Tire Vertical Load Deflection Data

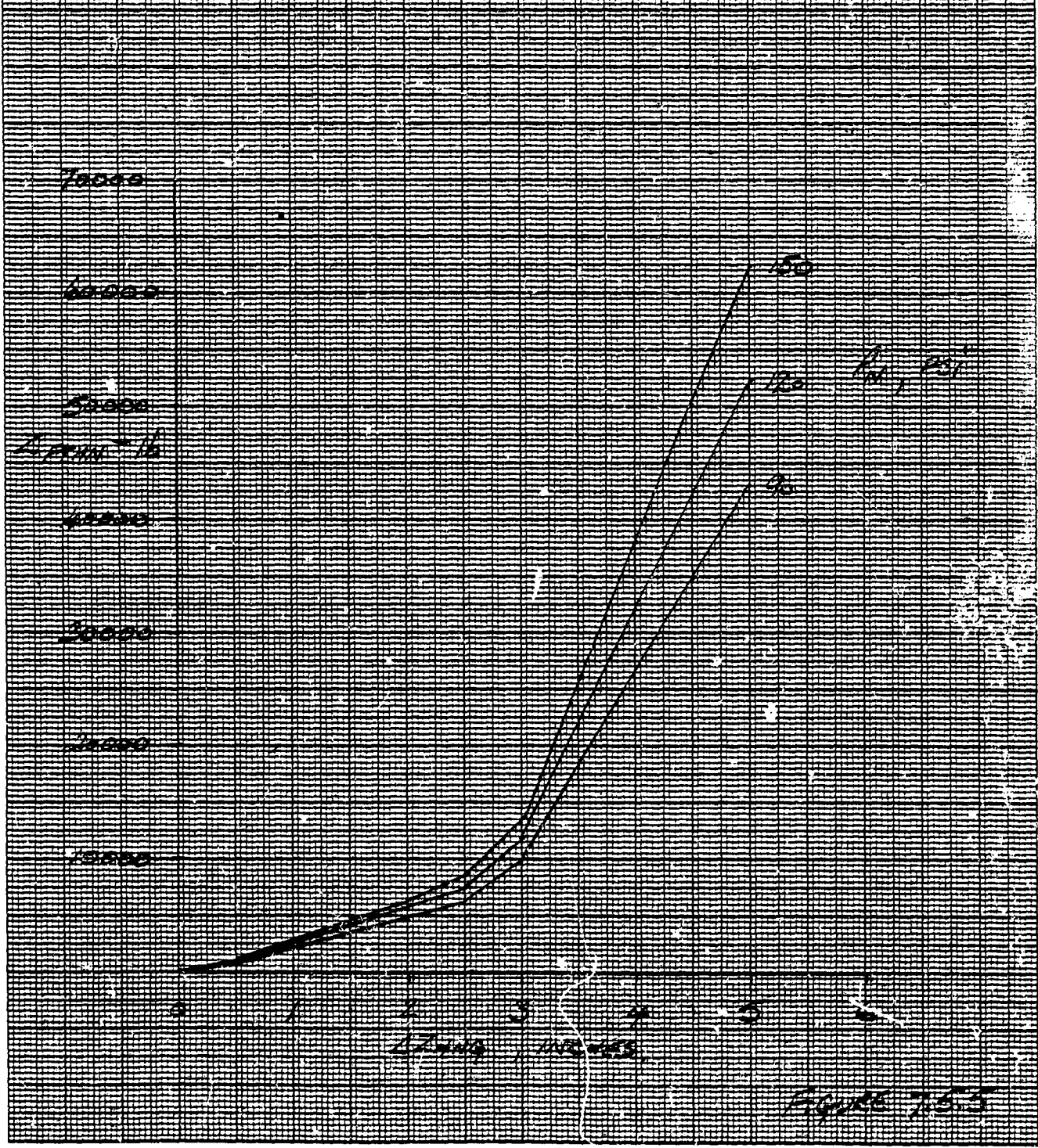


FIGURE 7-53

[Handwritten signature]

SOURCE 10 1 10 30 THE CONTINIZER AS 0014-01

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Pittsford, N.Y.

BLACK MARK - TAIL LANDING GEAR

TIRE VERTICAL STIFFNESS

(USED ONLY IN DAMPING CALCULATION)

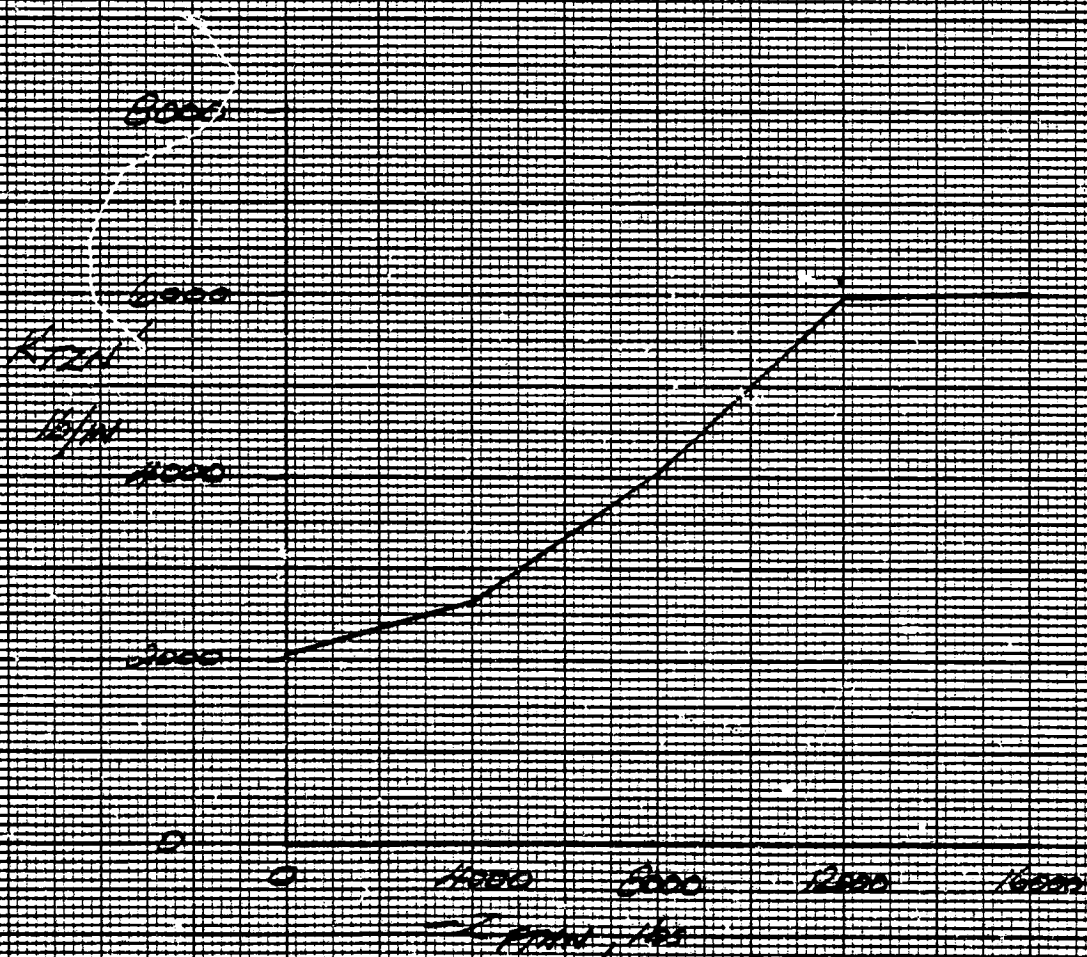


FIGURE 1.5.6

GRAPHIC 10 X 10 THE MILLS AS 8013-41

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GRAPH PAPER

5.8	GROUND EFFECT MODULE	
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5.8.1 Module Description

This module simulates the effect of ground proximity on a helicopter by modifying the main rotor downwash equation.

The modification factor was derived from Black Hawk hover-power flight test results (Reference 5.8.6-1).

The modification factor is reduced as flight speed is increased at constant height by a wake angle correction

It is likely that recent and on-going research on this subject (Reference 5.8.6-2) will lead eventually to an empirical model describing the local rotor inflow velocity due to the hyperbolically shaped ground vortex and its image that has been shown to exist at low speeds. Unfortunately, sufficient data for correlation of such a model has not yet been published.

5.8.2 GROUND EFFECT MODULE EQUATIONS

$$DWSHMR = \frac{KCT \cdot CTA}{2 \mu_{TOT}} \left\{ \frac{1}{1 + \frac{TDND \cdot S}{\mu_{TOT}}} \right\} \cdot K_{GE}$$

$$K_{GE} = \left[1 + .115 \left(\frac{R}{Z} \right)^2 \left(\frac{\Delta}{\mu_{TOT}} \right) \right]^{-2/3}$$

$$Z = \frac{(WLMR - WLQND)}{12} - \int V_Z dt.$$

The part in square brackets [] is the ground proximity effect (usually switched to 1 when $Z > 5 \times R$), the rest of the equation is as quoted in the main rotor downwash section.

5.8.3 GROUND EFFECTS MODULE INPUT/OUTPUT DATA TRANSFER

INPUT TRANSFER	
PARAMETER	ORIGIN MODULE
DWSHMR	MAIN ROTOR MOTION
LAMMR	
UTOTMR	
VZ	

OUTPUT TRANSFER	
PARAMETER	DESTINATION MODULE
DWSHMR	MAIN ROTOR

5.8.4 NOTATION FOR THE GROUND EFFECT MODULE

SYMBOL USED IN EQUATIONS	PROGR/M MNEMONIC	UNITS	DESCRIPTION
C_T	CTHAMR	-	Main rotor thrust coefficient
DW_0	DWSHMR		Main rotor uniform self induced velocity
KCT	KCTMR	-	Thrust gain for uniform downwash
R	RMR	FT	Main rotor radius
S		-	Laplace operator
T_{DWO}	TDWOMR	SEC	Time constant for uniform downwash calculation
Z		FT	Height of rotor hub above ground
λ	LAMBMR		Main rotor inflow ($=\mu_{Z_s} - D_{W_0}$)
μ_{TOT}	UTOTMR	-	Total main rotor inflow ($=[\mu_{X_s}^2 + \mu_{Y_s}^2 + \lambda^2]^{1/2}$)
μ_{X_s}	MUXSMR	-	Velocity in X direction, shaft axes
μ_{Y_s}	MUYSMR	-	Velocity in Y direction, shaft axes
μ_{Z_s}	MUZSMR	-	Velocity in Z direction, shaft axes
WLMR	WLMR	IN	Waterline Station of Rotor Hub
WLGND	WLGND	IN	Waterline Station of the Ground (Nominal)
VZ	VZ	FT/SEC	Vertical Velocity

5.8.5 Input Data

No separate input data is required. The parameters .115 and $2/3$ in equation 5.8.2-1, were empirically derived from data in Reference 5.8.6-1 which is pertinent to Black Hawk aircraft.

5.8.6 References

1. LTC Alan R. Todd to Mr. R. Merritt, Sikorsky Aircraft
Letter "Free hover performance" 22 October 1976
2. Sheridan & Wiesman "Aerodynamics of Helicopter Flight
Near the Ground" AHS Paper 77.33-04 May, 1977

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5.9 Gust Module

5.9.1 Description

This module produces local air velocities at all rotor-blade segment positions and at the fuselage and empennage aerodynamic centers, caused by various types of gusts. The input gusts may be discrete or continuous.

The output velocities are calculated in the local axis-system in which they are to be used. Thus the fuselage and tail terms are in body axes while those at the blade segments are in blade flapped and lagged axes.

The discrete gusts are characterized by having a front line designating the beginning of a disturbance, travelling through the atmosphere at velocity V_{HW} on a compass bearing ψ_N (Figure 5.9.1.1). Behind this front line, the change in air velocity can assume a step or rounded ramp or pulse shape, characterized by a magnitude V_{HGAMP} and rise-time G_{HDIS} (Figure 5.9.1.2). Since the aircraft is defined to be flying initially at V_{KTS} on compass bearing ψ_{BO} , the gust front can sweep across the aircraft (or be penetrated by the aircraft) from any direction and at any speed, by manipulating the relative speeds and compass bearings.

The continuous gusts are represented by the Dryden Model. The gust is assumed to be aligned with the aircraft heading such that only longitudinal and vertical vectors are defined. This is a necessary restriction, which avoids frequency shifts that would invalidate the gust spectrum model. At aircraft trim speeds greater than 50 ft/sec, a stationary gust field is traversed by the aircraft; at trim speeds less than 50 ft/sec, the gust field traverses the aircraft at 50 ft/sec. While the latter process is not a very appealing artifice mathematically, it is necessary in order to avoid unrealistic time lags in the gust definition equations and flying the aircraft in a constant up or down-draught at hover.

A penetration distance G (ft.) is defined which represents the distance that the aircraft reference hub centroid has penetrated into the gust. Initially the gust front is assumed to be close to the edge of the rotor disk in the upwind direction. A rearward approaching gust is not possible without further manipulation of the model. The propagation rate of the gust field across the rotor is defined by V_{FLD} , which then allows the penetration of any blade-segment, to be calculated (GPR_{jk} , see figure 5.9.1.3).

5.9.1 Description (Continued)

For the discrete gusts GPR_{jk} is introduced explicitly into the profile equations and the gust increment can be added directly. However, for the continuous gust case a table of gust values, determined at fixed time (displacement) intervals, (TABINK) are available. These data are obtained by passing independent, Gaussian, unit R.M.S. noise signals, produced by passing uniformly distributed white noise through the inverse Gaussian distribution function (Reference 5.9.6-2), through the Dryden filters. The assumption is made that perturbations in speed and heading are small enough such that the time constants in the gust functions can be considered constant, and that data can be loaded into the tables at fixed displacement intervals. These assumptions are consistent with the frequency/airspeed constraints of the Dryden Model application.

The gust velocities generated by either discrete or continuous functions, are in an axis set defined by the horizontal gust front. Three transformation matrices are required to orient into the blade axes and one for the body axes. These velocities are added to the other local velocities at the various stations.

The vertical gust component contributes to the total inflow of the main rotor. Because a basically uniform inflow is assumed in the Gen Hel rotor simulation, the change in inflow can be simulated by adding a representative velocity over the whole disc. The division of the blade into segments in Gen Hel is such as to give equal areas of swept annuli, making it possible to take a simple mean as the representative velocity of the additional inflow. Following the transformation of the z-direction component from space to (through body) shaft axes, the mean incremental velocity (VGAVMR) can be added to the downwash equation to compensate the total inflow.

The gust velocity at the fuselage is assumed to be the same value experienced at the rotor hub. The empennage velocities are based on the hub velocity with an approximate time delay to account for the penetration of the specific component.

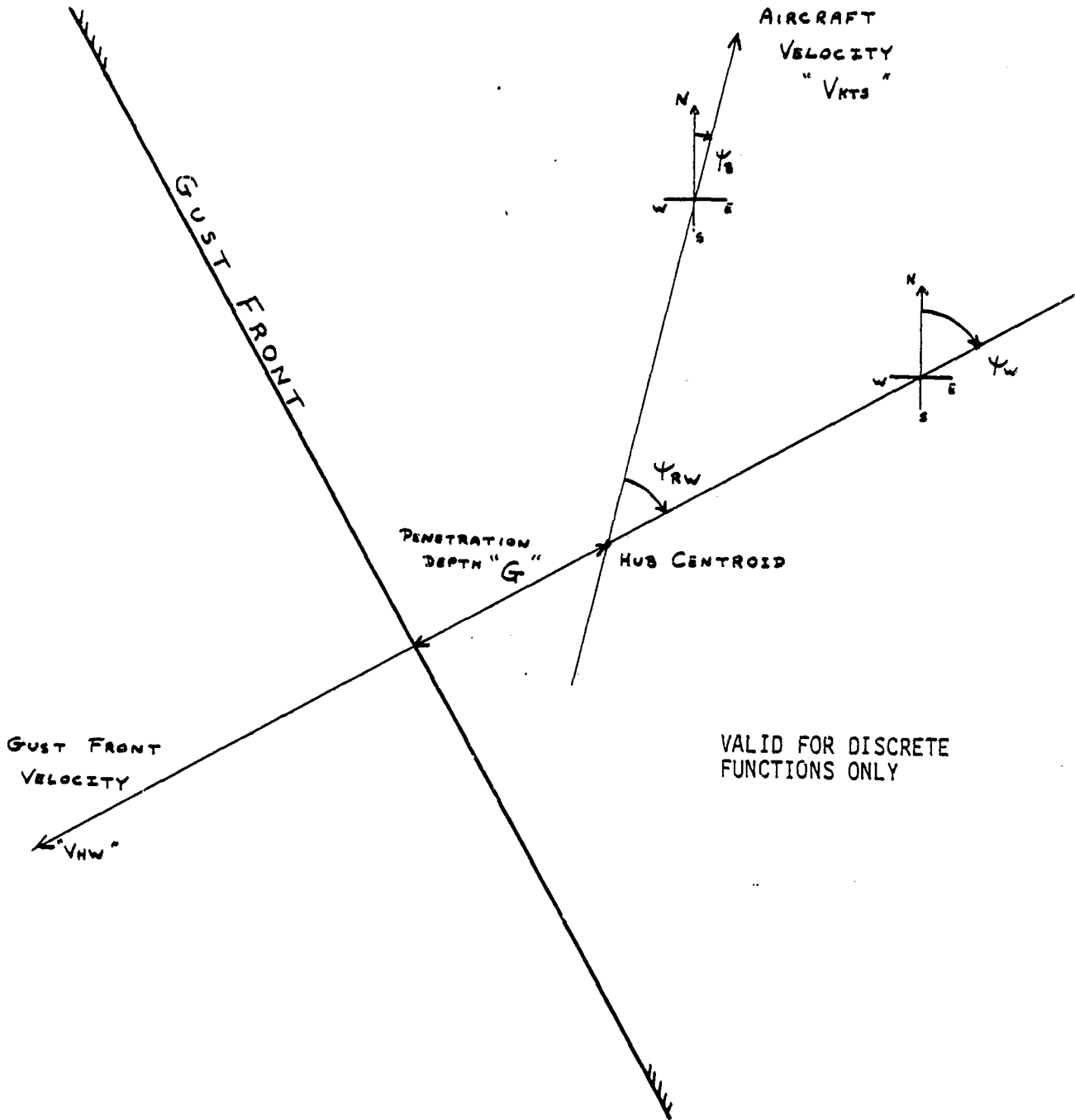
This model of gust penetration may be criticized on the grounds that the point fuselage airload used is not consistent with the detailed penetration of the rotor blades. Fixed wing simulations commonly overcome the distribution problem by introducing correlated non-inertial rotational rates due to gust (Reference 5.9.6-1).

5.9.1 Description (Continued)

It is felt that the separation of the tail used here, accounts for a sufficiently large portion of the non-rotor component gust distribution effects and that further terms (for which there are little if any data) are unnecessary. In the same vein it is sufficiently accurate (and convenient in the implementation) to assume the aerodynamic center of the fuselage to be at the c.g.

Discrete lateral gusts may be handled by orientation of ψ . However, the continuous gust case is invalidated by using any value other than zero for ψ_w , due to apparent frequency shifts in the gust spectrum.

GUST FRONT GEOMETRY



VALID FOR DISCRETE FUNCTIONS ONLY

Figure 5.9.1.1

GUST SELECTION (f₀₋₄)

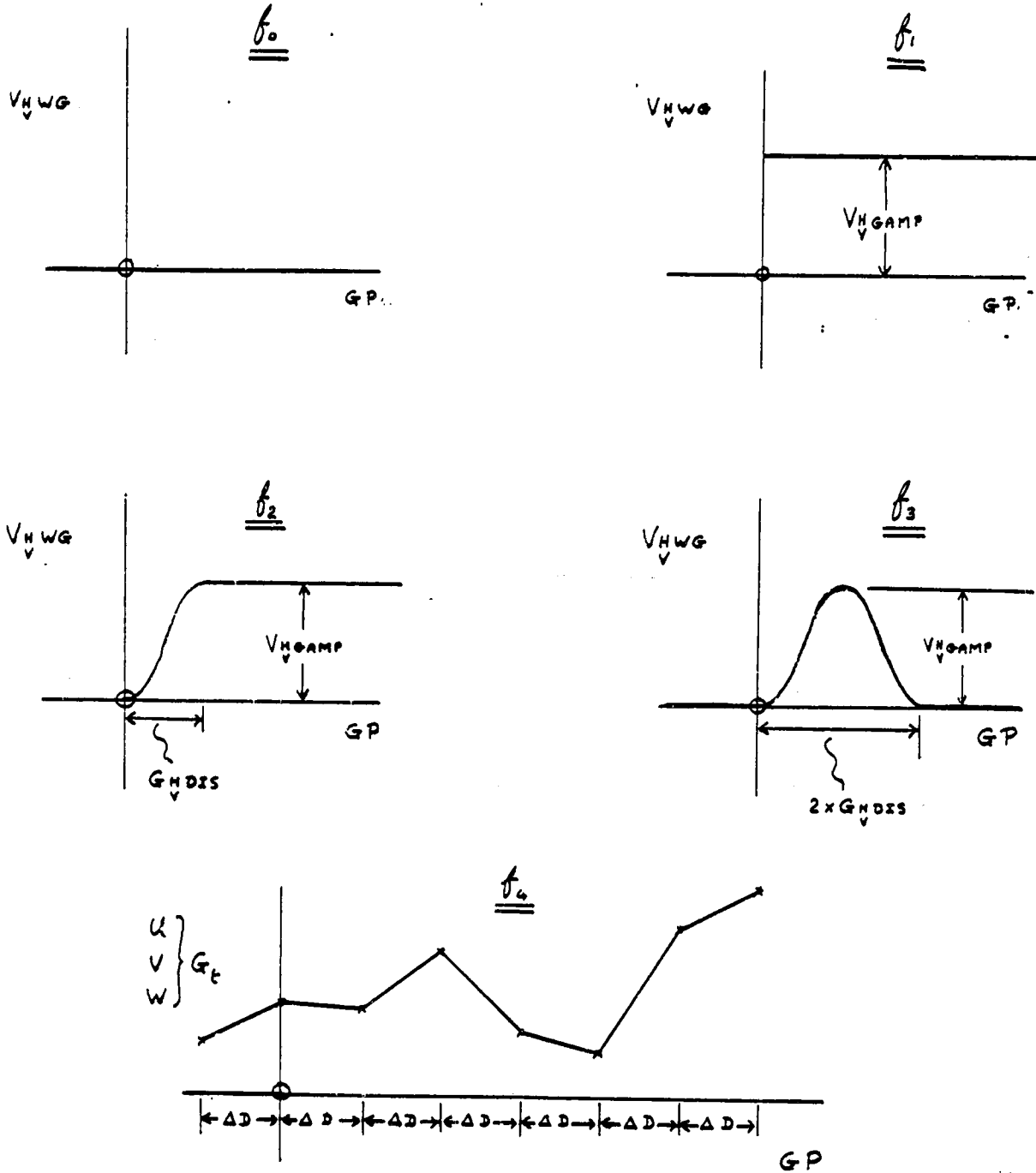


Figure 5.9.1.2

GUST PENETRATION GEOMETRY

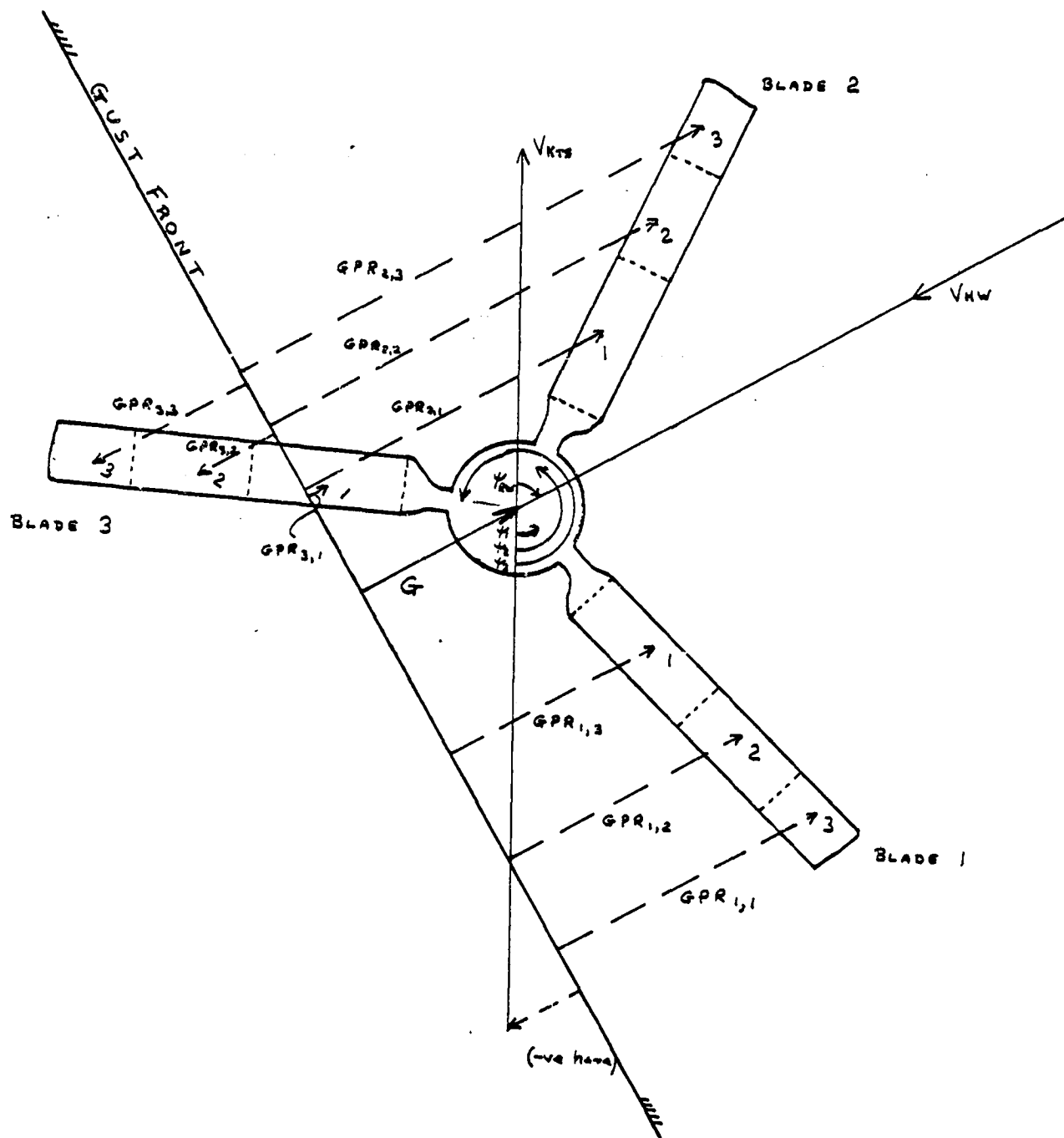


Figure 5.9.1.3

5.9.2 GUST MODULE EQUATIONS

AIRCRAFT HEADING

$$\psi_0 = \psi_{00} + 57.3 \int \dot{\psi}_0 dt$$

(PSIG)

RELATIVE GUST DIRECTION

$$\psi_{RN} = \psi_N - \psi_0$$

(PSIRN)

AIRCRAFT VELOCITY

$$V_{KT} = \left[(V_{XB} + V_{XG})^2 + (V_{YB} + V_{YG})^2 \right]^{1/2} * 1.591715 \text{ (KNOTS)}$$

(CALCULATED IN MOTION MODULE)

GUST PROPOGATION VELOCITY

$$V_{FLD} = V_{KT} (1.689) \cos \psi_{RN} + V_{HW}$$

$V_{HW} =$ INPUT FOR DISCRETE GUST FUNCTIONS ($f_{0,1,2,3}$)

$V_{HW} = 50 - 1.689 V_{KT} \cos \psi_{RN}$, $0 \leq V_{HW}$
FOR THE CONTINUOUS GUST FUNCTIONS (f_4)

GUST TABLE SET UP PARAMETERS

$$ND = V_{FLD} * TIME \dots \text{I.C. MODE ONLY}$$

(TABINR)

$$\Delta FT = \left(\frac{F_{STR} + R_{TR}}{12} \right) - \left(\frac{F_{SMR} + R_{MR}}{12} \right) \dots \text{I.C. MODE ONLY}$$

$$INPNT = \left(\frac{MAXPNT - 1}{2} \right) + \left[\frac{G_0}{TABINK} \right]_{\text{FIX. PT}} \dots \text{I.C. MODE ONLY}$$

WHERE $G_0 = -R_{MR}$

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INITIALISATION OF TABLE PARAMETERS

$$\frac{G_P}{(TABRUN)} = 0$$

$$ICUPD = \left[\frac{G_0 + RMR + \Delta FT}{TABINK} \right]_{FIX. A}$$

CLEAR TABLES TABLEH 0 → MAXPNT-1
 TABLEV 0 → MAXPNT-1

HUB CENTROID PENETRATION DISTANCE

$$\frac{G'}{(GPRM)} = G_0 + \int_0^{\xi} V_{FLD} dt$$

ROTOR BLADE-SEGMENT TABLE LOOK-UP POINTER

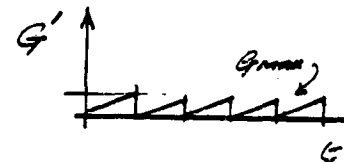
$$\frac{GPR'_I}{(GPRSP)} = G' - RMR(\xi + y_2)_{IS} \cos(\psi_{MR} + \delta + \psi_{RW})_{IB}$$

WHERE $G' = G - G_0'$

$G_0' = 0$ IN IC.

$G_0' = G$ WHEN $G' \geq G_{MAX}$

$G_{MAX} = (TABMAX)(TABINK)$
(GPRMAX)



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(MAP LOOK-UP PARAMETER)

$$\begin{aligned}
 GPR_I &= GPR_I' , & 0 \leq GPR_I' \leq G_{MAX} \\
 &= GPR_I' + G_{MAX} & GPR_I' < 0 \\
 &= GPR_I' - G_{MAX} & GPR_I' > G_{MAX}
 \end{aligned}$$

DISCRETE GUST FUNCTIONS.

DEFINE $GP = GP + \Delta D$ $GP_{IC} = 0$
(TABROW)

HORIZONTAL GUST $V_{HNG} = f_{l_1}(GP)$ SET INTO TABLE TABLEH

VERTICAL GUST $V_{VNG} = f_{l_2}(GP)$ SET INTO TABLE TABLEV

$l_{1,2} = 0, 1, 2, 3, 4$

\int_0 NO GUST VELOCITY
 $V_{VNG} = 0$

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f1 STEP GUST

$$V_V^H N_G = 0, \quad G_P \leq 0$$

$$V_V^H N_G = V_V^H G_{AMP}, \quad G_P > 0$$

f2 SHAPED RAMP

$$V_V^H N_G = 0, \quad G_P \leq 0$$

$$V_V^H N_G = \frac{1}{2} V_V^H G_{AMP} \left[1 + \cos \left\{ \pi \left(\frac{G_P}{G_V^H DIS} - 1 \right) \right\} \right]$$

$$, \quad 0 < G_P < G_V^H DIS$$

$$V_V^H N_G = V_V^H G_{AMP}, \quad G_P \geq G_V^H DIS$$

f3 SHAPED PULSE

$$V_V^H N_G = 0, \quad G_P \leq 0$$

$$V_V^H N_G = \frac{1}{2} V_V^H G_{AMP} \left[1 + \cos \left\{ \pi \left(\frac{G_P}{G_V^H DIS} - 1 \right) \right\} \right]$$

$$, \quad 0 < G_P < 2 * G_V^H DIS$$

$$V_V^H N_G = 0, \quad G_P \geq 2 * G_V^H DIS$$

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CONTINUOUS GUST FUNCTION

f_4 BASED ON THE DRYDEN MODEL.

V_{FLD} REFERS TO THE IC VALUE.

$$V_{HWG} = \sigma_u \frac{\left(\frac{2}{\pi} \frac{V_{FLD}}{L_u}\right)^{1/2}}{\left(S + \frac{V_{FLD}}{L_u}\right)} * \left(\frac{\pi}{\text{TIME}}\right)^{1/2} * \eta_{1t}$$

$$V_{VWG} = \sigma_v \frac{\left(\frac{3}{\pi} \frac{V_{FLD}}{L_v}\right)^{1/2} \left(S + \left(\frac{11}{3}\right)^{1/2} \frac{V_{FLD}}{L_v}\right)}{\left(S + \frac{V_{FLD}}{L_v}\right)^2} * \left(\frac{\pi}{\text{TIME}}\right)^{1/2} * \eta_{2t}$$

THE η_{1t} ARE INDEPENDENT, GAUSSIAN, UNIT R.M.S NOISE SIGNALS, PRODUCED BY PASSING UNIFORMLY DISTRIBUTED WHITE NOISE THROUGH THE INVERSE OF THE GAUSSIAN DISTRIBUTION FUNCTION (REF 5.9.6-2)

THIS APPROACH COULD BE EXPANDED TO INCLUDE A LATERAL GUST COMPONENT (V_{LWG}). HOWEVER, ITS USE BELOW 50KTS MAY BE QUESTIONABLE.

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FUSELAGE PENETRATION OF GUST

$$\begin{matrix} L'_{AC} \\ (L'_{PRAC}) \end{matrix} = \frac{(FSCGB - FSMR)}{12}$$

$$\begin{matrix} GPAC' \\ (GPAC) \end{matrix} = G' - L'_{AC} \cos \psi_{RW}$$

TRANSFORMATION MATRICES

ELEMENTS OF THESE MATRICES ARE USED IN OTHER MODULES, THEY ARE REPEATED HERE FOR COMPLETENESS SAKE

SPACE \rightarrow BODY. (VLNG SET = 0, 2ND COLUMN = 0)

$$\begin{matrix} [T_1] \\ (T1MAXT) \end{matrix} = \begin{bmatrix} \cos \theta_b \cos \psi_{RW} & , & 0 & , & -\sin \theta_b \\ \sin \theta_b \sin \phi_b \cos \psi_{RW} + \cos \phi_b \sin \psi_{RW} & , & 0 & , & \cos \theta_b \sin \phi_b \\ \sin \theta_b \cos \phi_b \cos \psi_{RW} - \sin \phi_b \sin \psi_{RW} & , & 0 & , & \cos \theta_b \cos \phi_b \end{bmatrix}$$

BODY \rightarrow SHAFT

$$\begin{matrix} [T_2] \\ (T2MAXT) \end{matrix} = \begin{bmatrix} \cos i_b & , & 0 & , & -\sin i_b \\ \sin i_b \sin i_\phi & , & \cos i_\phi & , & \cos i_b \sin i_\phi \\ \sin i_b \cos i_\phi & , & -\sin i_\phi & , & \cos i_b \cos i_\phi \end{bmatrix}$$

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SHAFT → BLADE

$$\begin{matrix} [T_3] = \\ (T_{3MAXT}) \end{matrix} \begin{bmatrix} -\sin\beta \cos(\psi+\delta) & , & \sin\beta \sin(\psi+\delta) & , & \cos\beta \\ \sin(\psi+\delta) & , & \cos(\psi+\delta) & , & 0 \\ \cos\beta \sin(\psi+\delta) & , & -\cos\beta \cos(\psi+\delta) & , & \sin\beta \end{bmatrix}$$

GUST VELOCITIES AT THE BLADE SEGMENT

(V_{HWGR} , V_{VWGR} FROM TABLES FOR EACH SEGMENT)

$$\begin{bmatrix} U_{PGMRD} \\ U_{TGMRD} \\ U_{RGMRD} \end{bmatrix}_I = [T_3]_I * [T_2] * [T_1] * \begin{bmatrix} V_{HWGR} \\ 0 \\ V_{VWGR} \end{bmatrix}_I$$

$$U_{PGMR} = \frac{U_{PGMRD}}{R_{MR} \Omega_{MR}}$$

$$U_{TGMR} = \frac{U_{TGMRD}}{R_{MR} \Omega_{MR}}$$

$$U_{RGMR} = \frac{U_{RGMRD}}{R_{MR} \Omega_{MR}}$$

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ROTOR DOWNWASH COMPENSATION

$$\lambda_E = \mu_{z3} - DW_{0E} + V_{GAVMR}$$

$$\text{WHERE } V_{GAVMR} = \frac{\sum_{I=0}^{I=I} V_{ZSGUS}}{(b_{MR} + N_{SS})} \cdot \frac{1}{R_{MR} S_{MR}}$$

AND V_{ZSGUS} IS OBTAINED FROM THE THIRD ROW OF

$$\begin{bmatrix} V_{XSGUS} \\ V_{YSGUS} \\ V_{ZSGUS} \end{bmatrix}_I = \begin{bmatrix} T_2 \end{bmatrix} * \begin{bmatrix} T_1 \end{bmatrix} * \begin{bmatrix} V_{HWGR} \\ 0 \\ V_{VWGR} \end{bmatrix}_I$$

GUST VELOCITIES AT THE FUSELAGE

(V_{HWGAC} , V_{VWGAC} FROM TABLES)

$$\begin{bmatrix} V_{XGWF} \\ V_{YGWF} \\ V_{ZGWF} \end{bmatrix} = \begin{bmatrix} T_1 \end{bmatrix} * \begin{bmatrix} V_{HWGAC} \\ 0 \\ V_{VWGAC} \end{bmatrix}$$

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GUST VELOCITIES AT THE EMPENNAGE

$$V_{XG} \begin{matrix} HT \\ VT \\ TR \end{matrix} = V_{XGWF} \text{ DELAYED BY } \frac{FS \begin{matrix} HT \\ VT \\ TR \end{matrix} - FSCGB}{12 * V_{XB}}$$

$$V_{YG} \begin{matrix} HT \\ VT \\ TR \end{matrix} = V_{YGWF} \quad " \quad "$$

$$V_{ZG} \begin{matrix} HT \\ VT \\ TR \end{matrix} = V_{ZGWF} \quad " \quad "$$

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5.9.3 GUST MODULE INPUT/OUTPUT DATA TRANSFER

INPUT TRANSFER	
PARAMETER	ORIGIN MODULE
LGMR	MAIN ROTOR
PSIMR	
OFSTMR	
KMRBX1	
THETAB	MOTION
PHIB	
PSIB	
YAT	
YB	

OUTPUT TRANSFER	
PARAMETER	DESTINATION MODULE
VGAMR	MAIN ROTOR
UPBMR	
UTGMR	
URGMR	
VXGNF	FUSE LAGE
VYGNF	
VZGNF	
VXGHT	HORIZONTAL TAIL
VYGHT	
VZGHT	
VXGVT	VERTICAL TAIL
VYGVT	
VZGVT	
VXGTR	TAIL ROTOR
VYGTR	
VZGTR	

5.9.4

Notation for Gust Module

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
$\dot{\psi}_b$	PSIDOT	RAD/SEC	Rate of Change of A/C Heading
ψ_b	PSIB	DEG	A/C Heading
ψ_{b0}	PSIBO	DEG	Initial Heading
ψ_{RW}	PSIRW	DEG	Relative Wind Heading
ψ_w	PSIWD	DEG	Wind Heading
VKT	VKT	KNOTS	Airspeed
VHW	VHW	FT/SEC	Wind Speed
VFLD	VFLD	FT/SEC	Gust Propagation Rate
ΔD	TABINK	FT	Gust Table Space
TIME	TIME	SEC	Integration Time Interval
ΔFT	DELFT	FT	Table Initial Dead Space
FSMR	FSMR	INS	Fuse. Station for T.R. Hub
FSTR	FSTR	INS	Fuse. Station for T.R. Hub
R_{MR}	RMR	FT	Radius of Main Rotor
INPPNT	INPPNT	-	Table Loading in I.C. Mode
MAXPNT	MAXPNT	-	Maximum Table Points
GO	GOO	FT	(= $-R_{MR}$)
GP	TABRUN	FT	Table Run Distance
ICUPD	ICUPD	FT	Start of Data in Table

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5.9.4

Notation for Gust Module (Continued)

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
TABLEH	TABLEH	-	Table of Horizontal Velocities
TABLEV	TABLEH	-	Table of Vertical Velocities
G'	GGGPRM	FT	Hub Centroid Penetration Distance
ϵ	KSGMR	ND	Normalized Offset
y_2	KMRBK1	ND	Distance from Hinge to Segment Midpoint
ψ_{MR}	PSIMR	DEG	Rotor Azimuth Position
δ	LGMR	RADS	Lagging Angle
GPR'	GPRSP	FT	Penetration of Any Blade Segment
TABMAX	TABMAX	-	Max Number of Table Locations
GMAX	GGGMAX	FT	Max Distance in the Tables
VHWG	VHWG	FT/SEC	Horizontal Gust Velocity
VVWG	VVWG	FT/SEC	Vertical Gust Velocity
VHGAMP VVGAMP	VHGAMP VVGAMP	FT/SEC) FT/SEC)	Discrete Gust Profile Amplitude Functions
GHDIS GVDIS	GHDIS GVDIS	FT) FT)	Discrete Gust Profile Distance Functions
σ_u			Dryden Filter Inputs
σ_v			
L_u			

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5.9.4

Notation for Gust Module (Continued)

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
L_v			
m_u	GAUSSH	-) Horizontal and Vertical Gaussian Random Number
m_v	GAUSSV	-	
L_{AC}	LPRAC	FT	Rotor to CG Offset
FSCGB	FSCGB	INS	Fuselage Station for C.G.
FSMR	FSMR	INS	Fuselage Stations for Main Rotor
GPAC'	GPACP	FT	Gust Penetration of C.G.
θ_b	THETAB	DEG	Airframe Pitch Attitude
ϕ_b	PHIB	DEG	Airframe Roll Attitude
i_θ		DEG	Long. Rotor Shaft Incidence
i_ϕ		DEG	Lat. Rotor Shaft Incidence
T_1	T1MAXT	-	Space→Body Transf. Matrix
T_2	T2MAXT	-	Body→Shaft Transf. Matrix
T_3	T3MAXT	-	Shaft→Blade Transf. Matrix
VHWGR	VHWGR	FT/SEC	Horizontal Gust Vel. from Table
VVWGR	VVWGR	FT/SEC	Vertical Gust Vel. from Table
UPGMRD	UPGMR	FT/SEC) Three Component Gust Velocities at the Blade Segment
UTGMRD	UTGMR	FT/SEC	

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5.9.4

Notation for Gust Module

SYMBOL USED IN EQUATIONS	PROGRAM MNECMNIC	UNITS	DESCRIPTION
URGMR	URGMR	FT/SEC)
UPGMR	-	ND) Three Component Interference: Components at the Rotor. Summed to Memory UPIMRI, UTIMRI, URMRI.
UTGMR	-	ND	
URGMR	-	ND	
λ_T	LAMBMR	ND	Total Normal Rotor Inflow Velocity
u_{zs}	MUZSMR	ND	Vertical Shaft Velocity, Normalized
Dwo	DWSHMR	ND	Uniform Component of Down-Wash
VGAVMR	VGAVMR	ND	Average Gust Velocity at the Rotor Disk
VZSGUS		FT/SEC	Vertical Shaft Component, Av. Gust Velocity
Ω_T	OMGTMR	RADS/SEC	Rotor Speed
b_{MR}	BMR	-	Number of Rotor Blades
N_{SS}	N_{SS}	-	Number of Segments/Blade
VHWGAC	VHWGAC	FT/SEC) Horizontal and Vertical Components of Gust Velocity at the Fuse C.G.
VVWGAC	VVWGAC	FT/SEC	
VXGWF	VXGWF	FT/SEC) Components of Gust Velocity in Body Axes at the C.G.
VYGWF	VYGWF	FT/SEC	
VZGWF	VZGWF	FT/SEC	

5.9.4

Notation for Gust Module (Continued)

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
Vxb	VXB	FT/SEC	Body Axes Longitudinal Velocity
$\begin{bmatrix} V_{XG} \\ V_{YG} \\ V_{ZG} \end{bmatrix}$	$\begin{bmatrix} VXG \\ VYG \\ VZG \end{bmatrix}$	FT/SEC	Components of Gust Velocity at the Horizontal Tail, Vertical Tail and Tail Rotor.
	HT VT TR		
			HT VT TR

5.9.5 Data

The only data of concern here are the continuous gust power spectrum parameters. All other data is defined elsewhere in Gen Hel or is arbitrary (e.g. V_{HGAMP} and G_{HDIS} describing a discrete gust).

The following values are taken from Reference 5.9.6-1:

	Altitude (h) \leq 1750 ft	Altitude (h) $>$ 1750 ft
L_u	1750	$145 h^{1/3}$
L_w	1750	h
σ_u	$\sqrt{L_u/L_w} \sigma_w$	

(For lateral gusts, $\sigma_v = \sigma_u$ and $L_v = L_u$.)

5.9.6 References

1. "Background Information and User Guide for MIL-F-8785B(ASG)," "Military Specification - Flying Qualities of Piloted Airplanes", AFFDL-TR-69-72, C. R. Chalk, et. al., August 1969.
2. "Modeling Turbulence for Flight Simulation at NASA-Ames", CSCR No. 4, Benton L. Parris, January, 1975.

5.10	HELICOPTER MOTION MODULE	
	CONTENTS	5.10-1
5.10.1	Module Description	5.10-2
	FIGURES	
5.10.1.1	Body Axes System	5.10-3
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5.10.3	Module Input/Output Definition	5.10-13
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5.10 Helicopter Motion Module

5.10.1 Module Description

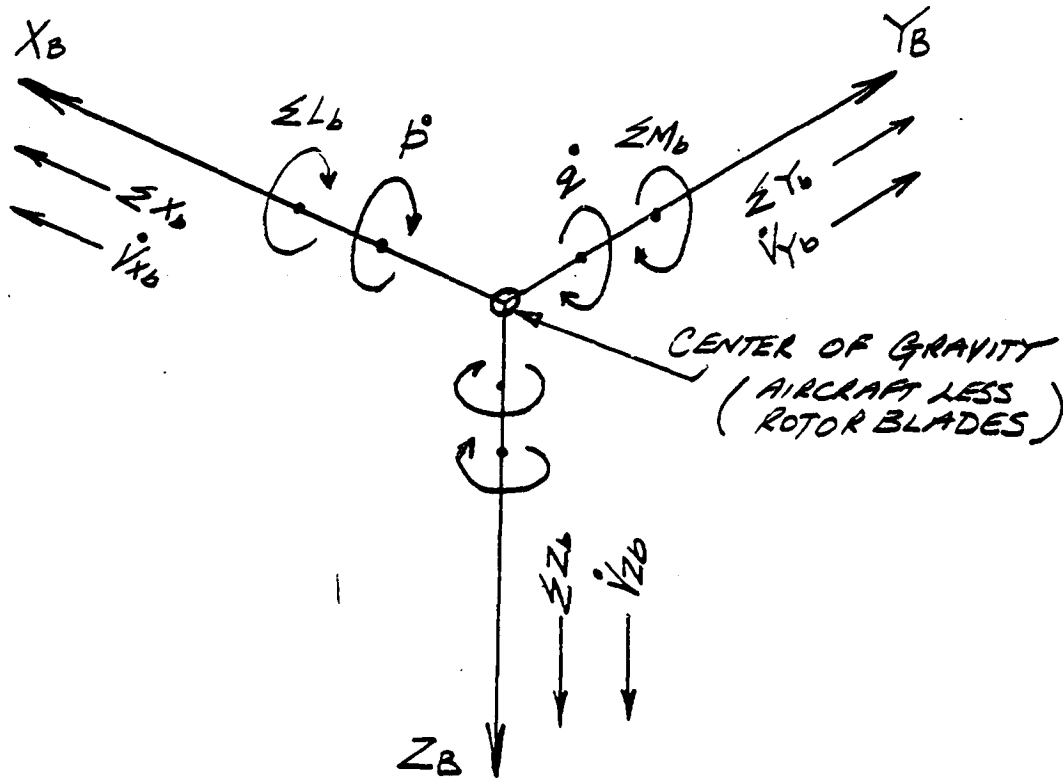
The forces and moments derived from the various simulated components of the aircraft are summed to develop the total external forces and moments at the airframe center of gravity in body axes, Figure 10.1.1. Introduced at this point, for convenience, are the gyroscopic moments resulting from rotating components. These equations are set up to cover any number of items of rotating mass aligned with the three body axes. The total external forces and moments are introduced into the general equations of motion from which the 6 components of acceleration are solved. It will be seen by comparing

with those on page 116 of Reference 10.6.1, that certain small terms have been eliminated. It should be noted that these are the equations of the aircraft less rotor blades, (which have their own degrees of freedom) and the appropriate mass is used. In order to obtain stable solutions under all operational conditions it is recommended that some velocity prediction technique be used in the determination of the body axes velocities, as presented. It should also be noted that since the p and r equation are coupled, they should be solved simultaneously as written. The Euler angular rate equations are given, followed by the body to space velocity axes transformation. Large angles are assumed.

The three component equations of motion for any specified point on the helicopter are presented in general form basically for output. However, lateral acceleration is fed back to the control system. It is recommended that at least two entries to these equations be provided to cover subsequent analyst requirements. The remaining equations in this section are provided for analyst output.

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BODY AXES SYSTEM



X_b, Y_b, Z_b Body Axes System. Origin at the Center of Gravity. X Axis Parallel to Aircraft Center Line.

FIGURE 10.1.1

5.10.2 HELICOPTER MOTION MODULE EQUATIONS

SUMMATION OF EXTERNALLY APPLIED FORCES AND MOMENTS
AT THE CENTER OF GRAVITY

$$SUMXB = X_{MR} + X_{TR} + X_T + X_{WF} + X_{LG} + X_{ADD}$$

$$SUMYB = Y_{MR} + Y_{TR} + Y_T + Y_{NF} + Y_{LG} + Y_{ADD}$$

$$SUMZB = Z_{MR} + Z_{TR} + Z_T + Z_{WF} + Z_{LG} + Z_{ADD}$$

$$SUMLB = L_{MR} + L_{TR} + L_T + L_{NF} + L_{LG} + L_{ADD} + L_{GY}$$

$$SUMMB = M_{MR} + M_{TR} + M_T + M_{NF} + M_{LG} + M_{ADD} + M_{GY}$$

$$SUMNB = N_{MR} + N_{TR} + N_T + N_{WF} + N_{LG} + N_{ADD} + N_{GY}$$

WHERE $H_{XGY} = \sum_{i=1}^N (J_{XGYi} * \omega_{XGYi})$

$$H_{YGY} = \sum_{i=1}^N (J_{YGYi} * \omega_{YGYi})$$

$$H_{ZGY} = \sum_{i=1}^N (J_{ZGYi} * \omega_{ZGYi})$$

$$L_{GY} = H_{YGY}(r) - H_{ZGY}(q)$$

$$M_{GY} = -H_{XGY}(r) + H_{ZGY}(p)$$

$$I_{GY} = H_{XGY}(q) - H_{YGY}(p)$$

(X Y Z L M N)_{ADD} ARE PROVIDED FOR ADDITIONAL
ARBITRARY INPUT

TRANSLATIONAL VELOCITY PREDICTORS

$$\dot{V}_{xb}(t) = V_{xb}(t-1) + \dot{V}_{xb}(t-1) (\Delta t/2)$$

$$\dot{V}_{yb}(t) = V_{yb}(t-1) + \dot{V}_{yb}(t-1) (\Delta t/2)$$

$$\dot{V}_{zb}(t) = V_{zb}(t-1) + \dot{V}_{zb}(t-1) (\Delta t/2)$$

BODY AXES ACCELERATIONS

$$\dot{V}_{XB DOT} = g/W_{BD} (\text{SUMXB} - W_{BD} \sin \theta_b) + r V_{Yb}' - q V_{Zb}'$$

$$\dot{V}_{YB DOT} = g/W_{BD} (\text{SUMYB} + W_{BD} \cos \theta_b \sin \phi_b) + p V_{Zb}' - r V_{Xb}'$$

$$\dot{V}_{ZB DOT} = g/W_{BD} (\text{SUMZB} + W_{BD} \cos \theta_b \cos \phi_b) + q V_{Xb}' - p V_{Yb}'$$

$$P_{DOT} = \frac{I_z}{(I_x I_z - I_{xz}^2)} \left\{ \text{SUMLB} - (I_z - I_y) q r + I_{xz} (p q) \right\}$$

$$+ \frac{I_{xz}}{(I_x I_z - I_{xz}^2)} \left\{ \text{SUMNB} - (I_y - I_x) p q - I_{xz} (r q) \right\}$$

$$Q_{DOT} = \frac{1}{I_y} \left\{ \text{SUMMB} - (I_x - I_z) p r + I_{xz} (r^2 - p^2) \right\}$$

$$R_{DOT} = \frac{I_x}{(I_x I_z - I_{xz}^2)} \left\{ \text{SUMNB} - (I_y - I_x) p q - I_{xz} (r q) \right\}$$

$$+ \frac{I_{xz}}{(I_x I_z - I_{xz}^2)} \left\{ \text{SUMLB} - (I_z - I_y) q r + I_{xz} (p q) \right\}$$

NOTE THESE ANGULAR ACCELERATION EQUATIONS DO NOT INCLUDE
DYNAMIC COMPONENTS REFLECTING LATERAL CG OFFSET

AIRCRAFT VELOCITY AND DISPLACEMENT

BODY AXES VELOCITIES

$$V_{XB} = \int \dot{V}_{XB} dt$$

$$V_{YB} = \int \dot{V}_{YB} dt$$

$$V_{ZB} = \int \dot{V}_{ZB} dt$$

$$P = \int \dot{P} dt + P_0$$

$$Q = \int \dot{Q} dt + Q_0$$

$$R = \int \dot{R} dt + R_0$$

ONLY EXECUTED IN THE
COMPUTE MODE

EULER RATES

$$\dot{\Theta} = 57.3 \{ q \cos \phi_0 - r \sin \phi_0 \}$$

$$\dot{\Psi} = 57.3 \left\{ \frac{r \cos \phi_0 + q \sin \phi_0}{\cos \Theta_0} \right\}$$

ONLY EXECUTED IN
THE COMPUTE MODE

$$\dot{\Phi} = 57.3 p + \dot{\psi}_0 \sin \Theta_0$$

EULER ANGLES

$$\left. \begin{aligned} \text{PHIB} &= \int \dot{\phi}_b dt \\ \text{THETAB} &= \int \dot{\theta}_b dt \\ \text{PSIB} &= \int \dot{\psi}_b dt \end{aligned} \right\} \text{ONLY EXECUTED IN THE COMPUTE MODE}$$

BODY AXES RATES

PDEG = 57.3 (p)

QDEG = 57.3 (q)

RDEG = 57.3 (r)

BODY ATTITUDE FUNCTIONS

SN THEB } DERIVED FROM SINCOS(θ_b) ROUTINE
CS THEB }

SN PHIB } DERIVED FROM SINCOS(ϕ_b) ROUTINE
CS PHIB }

SN PSIB } DERIVED FROM SINCOS(ψ_b) ROUTINE
CS PSIB }

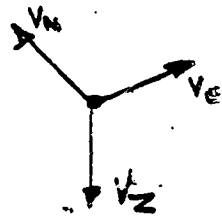
BODY TO SPACE AXES TRANSFORMATION MATRIX

$$[A_{HBS}] = \begin{bmatrix} (\cos \theta_b \cos \psi_b), & (\sin \theta_b \sin \phi_b \cos \psi_b - \cos \phi_b \sin \psi_b), & (\sin \theta_b \cos \phi_b \cos \psi_b + \sin \phi_b \sin \psi_b) \\ (\cos \theta_b \sin \psi_b), & (\sin \theta_b \sin \phi_b \sin \psi_b + \cos \phi_b \cos \psi_b), & (\sin \theta_b \cos \phi_b \sin \psi_b - \sin \phi_b \cos \psi_b) \\ (-\sin \theta_b), & (\cos \theta_b \sin \phi_b), & (\cos \theta_b \cos \phi_b) \end{bmatrix}$$

THIS BODY AXES TO SPACE AXES TRANSFORMATION IS PERFORMED IN THE SEQUENCE $\phi \rightarrow \theta \rightarrow \psi$.

BODY AXES TO SPACE AXES VELOCITY TRANSFORMATION

$$\begin{bmatrix} V_N \\ V_E \\ V_Z \end{bmatrix} = [A_{HBS}] \begin{bmatrix} V_{Xb} \\ V_{Yb} \\ V_{Zb} \end{bmatrix}$$



BODY POSITION IN SPACE

NORTH = $\int (V_N) dt$
 EAST = $\int (V_E) dt$
 ALT = $ALT_0 - \int (V_Z) dt$

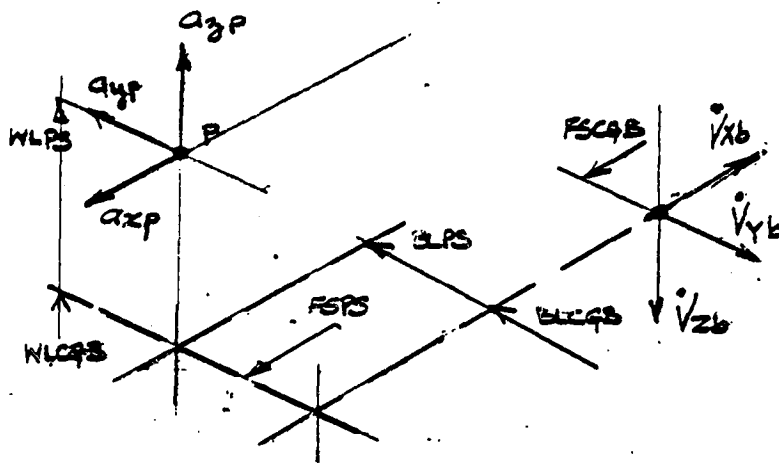
EQUATIONS FOR MOTION AT ANY POINT ON THE AIRFRAME

GEOMETRY

$$l_{ps} = \frac{(FSCGB - FSPS)}{12}$$

$$h_{ps} = \frac{(WLCGB - WLPS)}{12}$$

$$b_{ps} = \frac{(BLCGB - BLPS)}{12}$$



ACCELERATIONS AT A POINT

NOTE: a_{xp} , a_{yp} , a_{zp} ARE ACCELERATIONS THAT A BOB WEIGHT WOULD EXPERIENCE WHICH ARE IN THE OPPOSITE DIRECTION TO \dot{v}_{xb} , \dot{v}_{yb} , \dot{v}_{zb}

$$a_{xps} = + \left(\sum X_B \cdot \frac{g}{W_B} - l_{ps}(q^2 + r^2) + b_{ps}(pq - \dot{r}) + h_{ps}(p\dot{r} + \dot{q}) \right)$$

$$a_{yps} = + \left(\sum Y_B \cdot \frac{g}{W_B} + l_{ps}(pq + \dot{r}) - b_{ps}(p^2 + r^2) + h_{ps}(q\dot{r} - \dot{p}) \right)$$

$$a_{zps} = + \left(\sum Z_B \cdot \frac{g}{W_B} + l_{ps}(p\dot{r} - \dot{q}) + b_{ps}(q\dot{r} + \dot{p}) - h_{ps}(p^2 + q^2) \right)$$

VELOCITIES AT A POINT

$$V_{XPS} = V_{XB} + h_{ps} q - b_{ps} r$$

$$V_{YPS} = V_{YB} + e_{ps} r - h_{ps} p$$

$$V_{ZPS} = V_{ZB} - e_{ps} q + b_{ps} p$$

GRAVITY VECTOR ANGLES

$$\text{LATERAL LATBL} = \tan^{-1} \left(\frac{a_{yp}}{a_{zp}} \right)$$

(ϕ_{gv})

$$\text{LONGITUDINAL LINGBL} = \tan^{-1} \left(\frac{a_{xp}}{a_{zp}} \right)$$

(θ_{gv})

RATE OF CLIMB PARAMETERS

$$V_C = -60 V_Z$$

$$\text{GAMTRU} = -\tan^{-1} \frac{V_Z}{(V_N^2 + V_E^2)^{1/2}}$$

THIS IS THE TRUE ANGLE OF CLIMB

LOAD FACTORS AT THE AIRFRAME CG

$$N_X = \text{SUM } X_B / W_{AD}$$

$$N_Y = \text{SUM } Y_B / W_{AD}$$

$$N_Z = \text{SUM } Z_B / W_{AD}$$

AIR SPEED

$$V_{KT} = [(V_{Xb} + V_{Xg})^2 + (V_{Zb} + V_{Zg})^2]^{1/2} * 0.591715 \quad \text{KNOTS}$$

$$V_{XBIKT} = (V_{Xb} + V_{Xg}) \left\{ \frac{\rho}{0.002578} \right\}^{1/2} * 0.591715 \quad \text{KNOTS INDICATED}$$

FREE STREAM VARIABLES

$$\alpha_{FRE} = \tan^{-1} \frac{(V_{Zb} + V_{Zg})}{(V_{Xb} + V_{Xg})}$$

$$\beta_{FRE} = \tan^{-1} \frac{(V_{Yb} + V_{Yg})}{[(V_{Xb} + V_{Xg})^2 + (V_{Zb} + V_{Zg})^2]^{1/2}}$$

$$Q_{FRE} = \frac{1}{2} \rho [(V_{Xb} + V_{Xg})^2 + (V_{Yb} + V_{Yg})^2 + (V_{Zb} + V_{Zg})^2]$$

5.10.3 MOTION MODULE INPUT/OUTPUT DATA TRANSFER

INPUT TRANSFER	
PARAMETER	ORIGIN MODULE
(...) WB FSCGB HLCGB BLCGB	MAIN ROTOR
(...) WF	FUSELAGE
(...) T	EMPENNAGE
(...) TR	TAIL ROTOR
(...) LG	LANDING
() ADD	ADDITION
β WB	FUSELAGE

OUTPUT TRANSFER		
PARAMETER	DESTINATION MODULE	
VXBDOT VYBDOT VZBDOT P Q R	MAIN ROTOR	
VXB YYB VZB P Q R		
THETAB PHIB PSIB VN VE VZ		FUSELAGE EMPENNAGE TAIL ROTOR
QDEG PDEG RDEG VXBIKT AYPSI THETAB PHIB PSIB		
VZ		LANDING
[A _{WB}]		FLIGHT CONTROL
	GROUND EFFECT	
	LANDING	

5.10.4 NOTATION FOR THE HELICOPTER MOTION MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
SUMXB	SUMXB	LB	Total external force acting at the fuselage c.g. along the X Y and Z axes respectively
SUMYB	SUMYB	LB	
SUMZB	SUMZB	LB	
SUMLB	SUMLB	FT LB	Total external moments acting about the X, Y and Z axes respectively
SUMMB	SUMMB	FT LB	
SUMNB	SUMNB	FT LB	
(...) _{MR}			Main rotor components
(...) _{TR}			Tail rotor components
(...) _{WF}			Fuselage components
(...) _T			Empennage components
(...) _{LG}			Landing Gear components
(...) _{ADD}			Additional Arbitrary Inputs
H _{XGY}	HXGY		Gyroscopic effects, angular Momentum
H _{YGY}	HYGY		
H _{ZGY}	HZGY		
J _{XGY}	JXGY	SLUG FT ²	Rotation inertia about the X, Y and Z axes respectively.
J _{YGY}	JYGY	SLUG FT ²	
J _{ZGY}	JZGY	SLUG FT ²	
W _{XGY}	WXGY	RADS/SEC	Rotational speed of components.
W _{YGY}	WYGY	RADS/SEC	
W _{ZGY}	WZGY	RADS/SEC	
LGY	LGY	FT LB	Gyroscopic effects. Total moments in body axes.
MGY	MGY	FT LB	
NGY	NGY	FT LB	

5.10.4 NOTATION FOR THE HELICOPTER MOTION MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
V_{xb}		FT/SEC	Half pass predicted body axes velocities.
V_{yb}		FT/SEC	
V_{zb}		FT/SEC	
W	WEIGHT	LB	Aircraft gross weight
I_x	IX	SLUGS FT ²	Inertia about body X-axis
I_y	IY	SLUGS FT ²	Inertia about body Y-axis
I_z	IZ	SLUGS FT ²	Inertia about body Z-axis
I_{xz}	IXZ	SLUGS FT ²	Cross coupling inertia
W_{bd}	WTBOD	LB	Weight of the body
F_{SCGB}	FSCGB	INS	Fuselage station of the body c.g.
W_{LCGB}	WLCGB	INS	Waterline station of the body c.g.
B_{LCGB}	BLCGB	INS	Buttline station of the body c.g.
\dot{V}_{xb}	VXBDOT	FT/SEC ²	Accel. along X-axis
\dot{V}_{yb}	VYBDOT	FT/SEC ²	Accel. along Y-axis
\dot{V}_{zb}	VZBDOT	FT/SEC ²	Accel. along Z-axis
\dot{p}	PDOT	RADS/SEC ²	Angular accel. about X-axis
\dot{q}	QDOT	RADS/SEC ²	Angular accel. about Y-axis
\dot{r}	RDOT	RADS/SEC ²	Angular accel. about Z-axis
V_{xb}	VXB	FT/SEC	Vel. along Y-axis
V_{yb}	YXB	FT/SEC	Vel. along Y-axis
V_{zb}	YZB	FT/SEC	Vel. along Z-axis

5.10.4 NOTATION FOR THE HELICOPTER MOTION MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
p	P	RADS/SEC	Angular rate about X-axis
q	Q	RADS/SEC	Angular rate about Y-axis
r	R	RADS/SEC	Angular rate about Z-axis
$\dot{\theta}_b$	THEDOT	DEG/SEC	Pitch rate
$\dot{\phi}_b$	PHIDOT	DEG/SEC	Roll rate
$\dot{\psi}_b$	PSIDOT	DEG/SEC	Heading rate
θ_b	THETAB	DEG	Pitch
ϕ_b	PHIB	DEG	Roll
ψ_b	PSIB	DEG	Yaw
p_D	PDEG	DEG/SEC	Angular rate about X-axis
q_D	QDEG	DEG/SEC	Angular rate about Y-axis
r_D	RDEG	DEG/SEC	Angular rate about Z-axis
	SNTHEB	-	SIN(THETAB)
	CSTHEB	-	COS(THETAB)
	SNPHIB	-	SIN(PHIB)
	CSPHIB	-	COS(PHIB)
	SNPSIB	-	SIN(PSIB)
	CSPSIB	-	COS(PSIB)
[AHBS]	-	-	Body to space axes transformation matrix.
V_N	VN	FT/SEC	Velocity to the north
V_E	VE	FT/SEC	Velocity to the east
V_Z	VZ	FT/SEC	Velocity down

5.10.4 NOTATION FOR THE HELICOPTER MOTION MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
NORTH	NORTH	FT	Position north
EAST	EAST	FT	Position east
ALT	ALT	FT	Altitude
V_C	VC	FT/MIN	Climb velocity
γ_T	GAMTRU	DEG	True climb angle
N_X	NX	G's	Body Axes load factors
N_Y	NY	G's	
N_Z	NZ	G's	
V_{KT}	VKT	KNOTS	Velocity in X-Z plane
α_F	ALFRE	DEG	Free stream air flow variables
β_F	BETFRE	DEG	
q_F	QFRE	LB/FT ²	
FSPS	FSPS _i	INS	Fuselage station of point of interest
WLPS	WLPS _i	INS	Waterline station of point of interest
BLPS	BLPS _i	INS	Buttline station of point of interest
lps		FT	Longitudinal arm
hps		FT	Vertical arm
bps		FT	Lateral arm
A_{xps}	AXPS1	FT/SEC ²	Point #1 accel. along X-axis
A_{yps}	AYPS1	FT/SEC ²	Point #1 accel. along Y-axis
A_{zps}	AZPS1	FT/SEC ²	Point #1 accel. along Z-axis

5.10.4 NOTATION FOR THE HELICOPTER MOTION MODULE

SYMBOL USED IN EQUATIONS	PROGRAM MNEMONIC	UNITS	DESCRIPTION
V _{xps}	VXPS1	FT/SEC	Point #1 vel. along X-axis.
V _{yps}	VYPS1	FT/SEC	Point #1 vel. along Y-axis.
V _{zps}	VZPS1	FT/SEC	Point #1 vel. along Z-axis.
LNGBL	LNGBL1	DEG	Longitudinal ball.
LATBL	LATBL1	DEG	Lateral ball.

5.10.5 BLACK HAWK MOTION MODULE INPUT DATA

1. GYROSCOPIC COMPONENTS

$JXGY_1 = 0.116$	$WXGY_1 = 4188.0$
$JXGY_2 = 0.124$	$WXGY_2 = 2094.0$
$JXGY_3 = 0.082$	$WXGY_3 = 609.0$
$JYGY_1 = 0.715$	$WYGY_1 = 127.1$
$JZGY_1 = 18.0$	$WZGY_1 = -27.0$

2. DOWNWASH CORRECTION TERMS

$$Y_{ADD} = -19.45 Q_{FRE}$$

$$\text{IF } -30^\circ \leq \beta_{NF} \leq 0^\circ, L_{ADD} = -(70 + 1.17 \beta_{NF}^2) Q_{FRE}$$

$$\text{IF } 30 \geq \beta_{NF} \geq 0^\circ, L_{ADD} = -(70 + 5.54 \beta_{NF}^2) Q_{FRE}$$

$$\text{IF } \beta_{NF} > 0 \quad M_{AD} = 51.9 \beta_{NF} \quad \text{LIMIT } M_{AD} < 390.$$

$$M_{ADD} = M_{AD} Q_{FRE}$$

$$\text{IF } \beta_{NF} < 0 \quad M_{AD} = 38.9 \beta_{NF} \quad \text{LIMIT } M_{AD} > -292.$$

$$M_{ADD} = M_{AD} Q_{FRE}$$

FOR $\beta_{NF} \leq -30, \geq 30$ HOLD VALUES FOR $\beta = -30, 30$ RESPECTIVELY

3 INERTIA DATA

$$\begin{aligned} I_x &= 4659.0 \\ I_y &= 38512.0 \\ I_z &= 36796.0 \\ I_{xz} &= 1882.0 \end{aligned}$$

LATERAL ACCELERATION TRANSDUCER

$$\begin{aligned} FSFS &= 307.0 \text{ INS} \\ WLFS &= 263.0 \text{ INS} \\ BLFS &= 0 \text{ INS} \end{aligned}$$

4. GENERAL INPUT

ALTA = 0
RHO = .002378
V_{SOUND} = 1117.0

PICON = 3.1415927
RADCON = 57.29578
RAD SCL = .01745
GRAVITY = 32.174

5.10.6 References

1. Dynamics of Flight, Etkin, B., John Wiley & Sons Inc., 1959.

6.0 DEFINITION OF THE BLACK HAWK COCKPIT

	CONTENTS	6.1
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6.2	Pilot's Primary Controls	6.2
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6.0 DEFINITION OF THE BLACK HAWK COCKPIT

6.1 Overall Cockpit Arrangement

The overall layout of the cockpit environment is shown on the photograph, Figure 1.1. This general view is supported by identification of cockpit components on Figures 1.2a and 1.2b. The cockpit geometric definition, relating to the pilot's seat, controls and instrumentation, is presented on Figures 1.3a, 1.3b and 1.3c as 3-view general arrangement drawings. Diagrams defining the upper and lower consoles are shown on Figures 1.4 and 1.5 respectively. The most significant items are the switching functions on the upper console and the Stabilator/Auto Flight Control Panel on the lower console. The flight instrument panel arrangement is shown on Figure 1.6.

6.2 Pilot's Primary Controls

A diagram of the pilot's cyclic stick grip is presented on Figure 2.1. The grip incorporates the following functions of significance:

Stick trim - provides for lateral and longitudinal stick trim at 1/2"/sec.

Trim release - while depressed the force system is deactivated leaving a limp stick.

A diagram of the pilot's collective stick grip is presented on Figure 2.2. A friction device on the pilot's lever can be turned to adjust the amount of friction and prevent the collective stick from creeping.

6.3 Control Range Characteristics

The control motion, both in terms of control displacement in inches of travel and angular sweep are presented on Figure 3.1. Also provided are the corresponding angular outputs at the rotor head.

6.4 Control Force Feel Characteristics

The control forces, gradients and breakouts for the primary controls are given on Figure 4.1. The values presented were derived from measurements at the 50% control position.

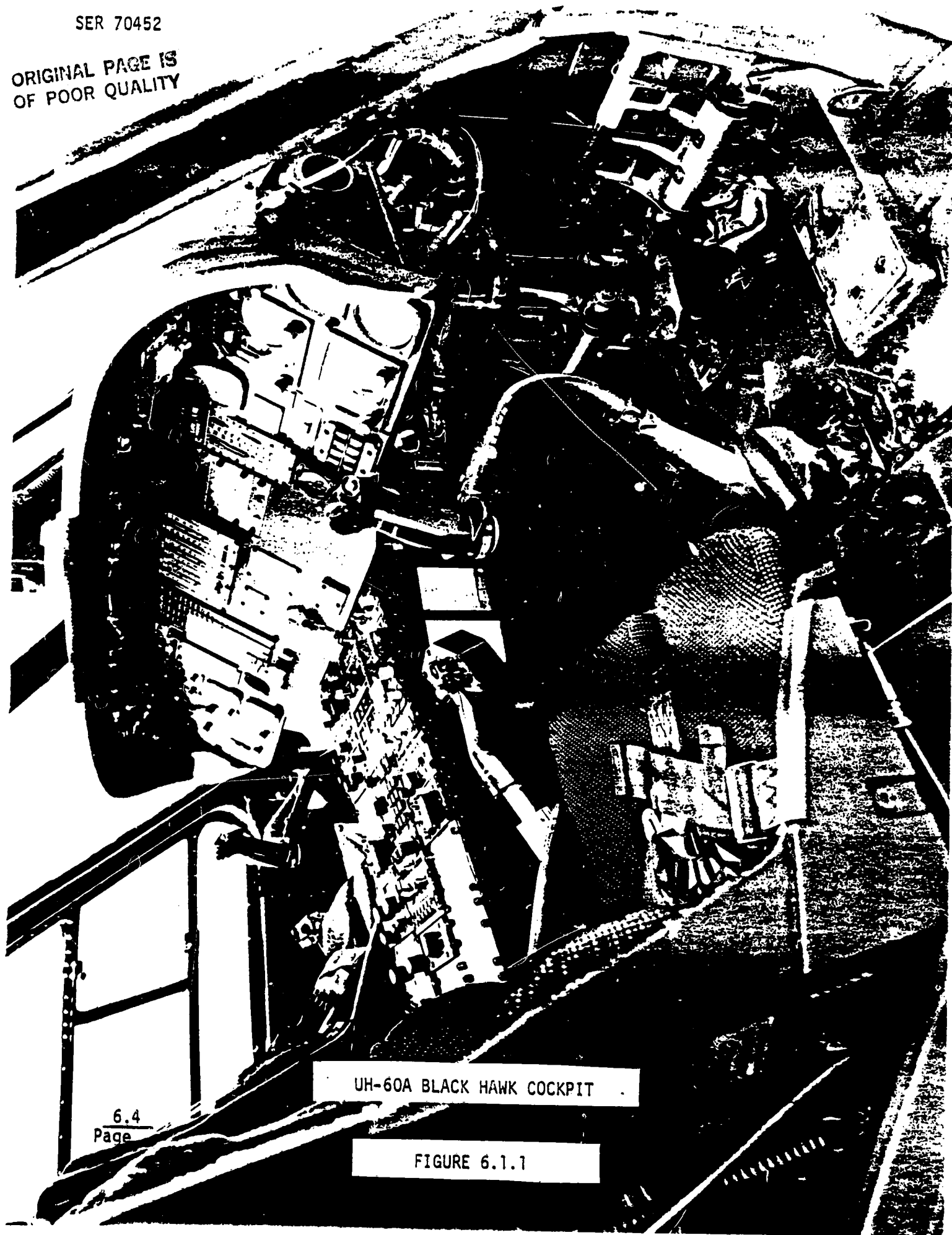
More detailed information concerning all aspects of the cockpit may be obtained from References 6.5.1 through 6.5.4.

6.5 REFERENCES

1. Black Hawk Pilot Manual TM55-1520-237-10
2. Black Hawk MOS 35K Avionics Mechanics Manual
3. Black Hawk Fault Isolation Manual TM55-1520-237-MTF
4. Black Hawk Simulation Flight Training System Device 2038.
NAVTRAEQIPCN 76-C-0086-4001 Volume I ADDENDUM 1.

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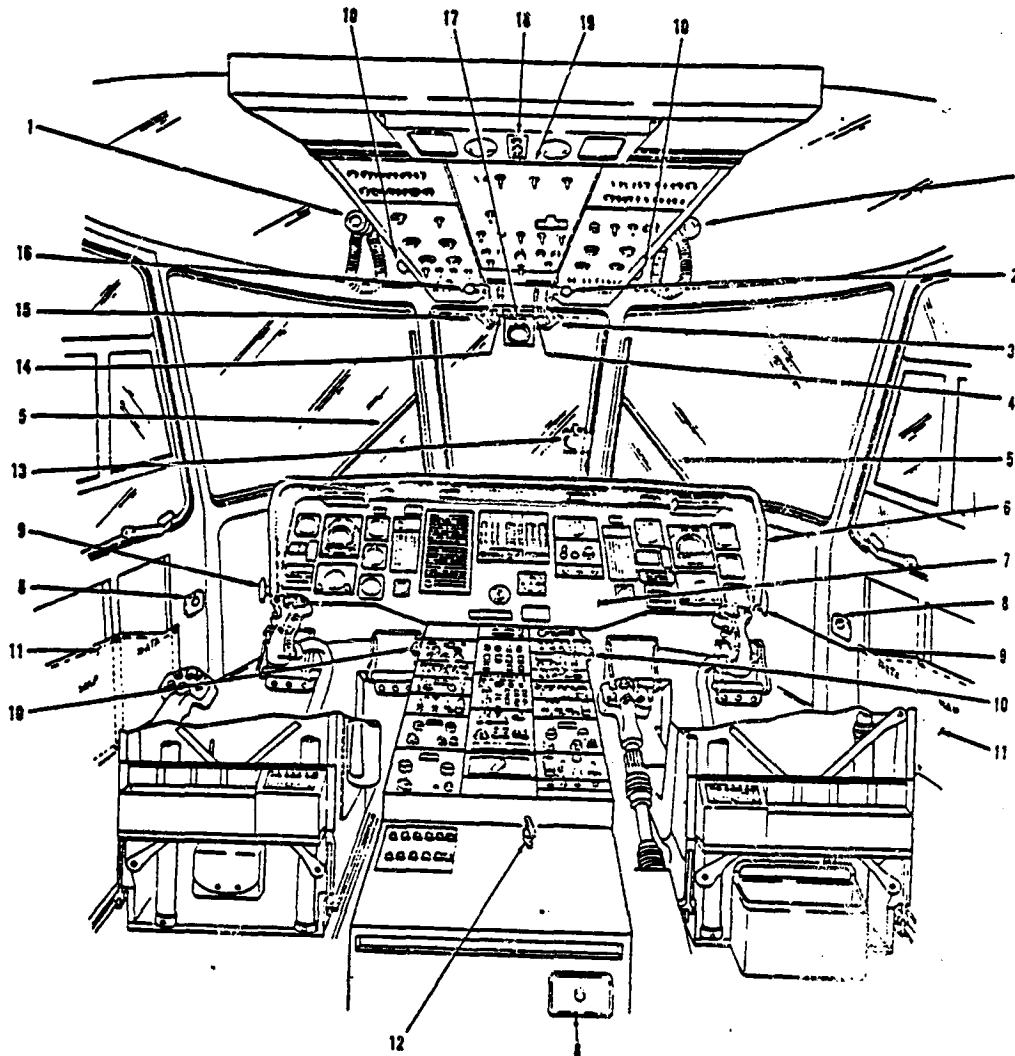


UH-60A BLACK HAWK COCKPIT

6.4
Page

FIGURE 6.1.1

COCKPIT COMPONENT IDENTIFICATION



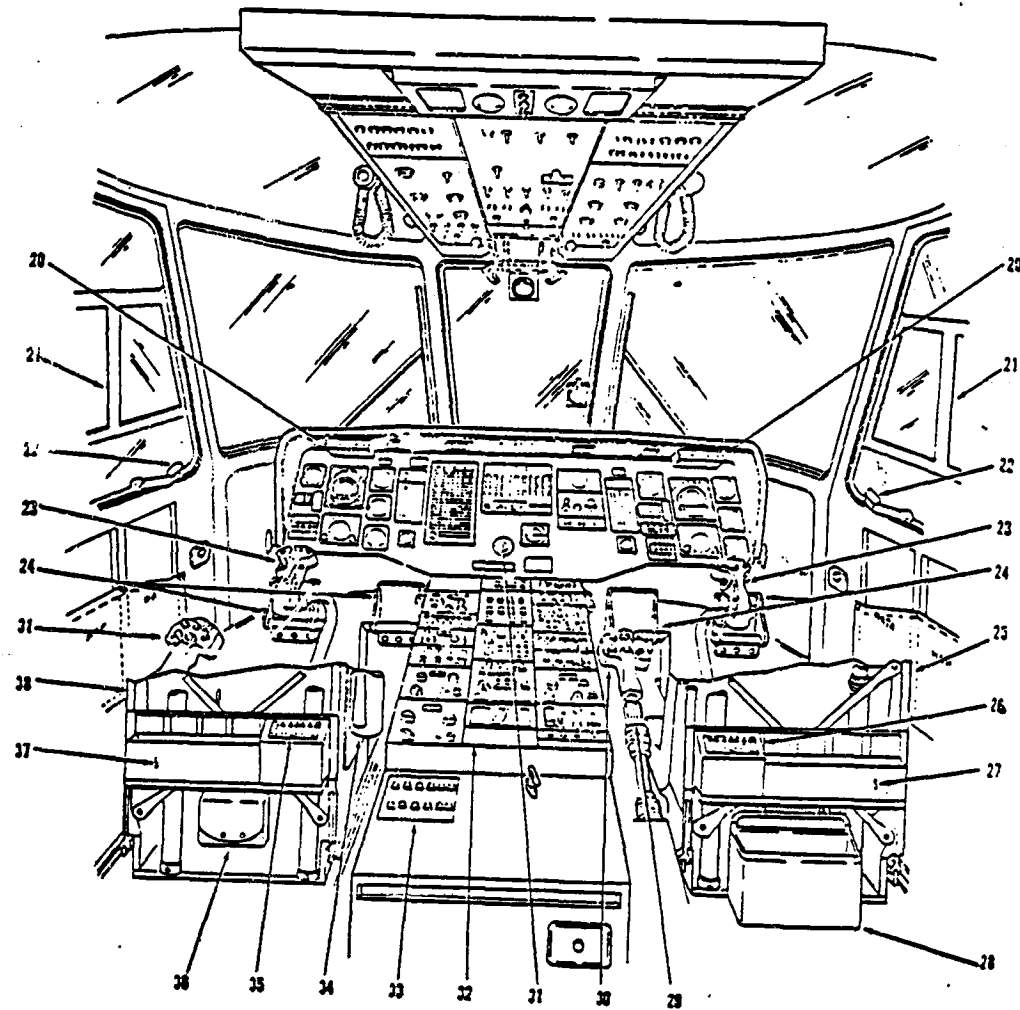
- 1 UTILITY LIGHT
- 2 NO. 2 ENGINE FUEL SELECTOR LEVER
- 3 NO. 2 ENGINE OFF/FINE T-HANDLE
- 4 NO. 2 ENGINE POWER CONTROL LEVER
- 5 WINDSHIELD WIPER
- 6 INSTRUMENT PANEL GLARE SHIELD
- 7 INSTRUMENT PANEL

- 8 ASH TRAY
- 9 PEDAL ADJUST LEVER
- 10 VENT
- 11 MAP/DATA CASE
- 12 PARKING BRAKE LEVER
- 13 STANDBY (MAGNETIC) COMPASS
- 14 NO. 1 ENGINE POWER CONTROL LEVER

- 15 NO. 1 ENGINE OFF/FINE T-HANDLE
- 16 NO. 1 ENGINE FUEL SELECTOR LEVER
- 17 FREE-AIR TEMPERATURE GAGE
- 18 COCKPIT FLOODLIGHT CONTROL
- 19 UPPER CONSOLE

FIGURE 6.1.2(a)

COCKPIT COMPONENT IDENTIFICATION



- | | | |
|-----------------------------------|--|--|
| 20 MASTER WARNING PANEL | 27 CREW CHIEF AMMUNITION/GRENADE STOWAGE COMPARTMENT | 33 BATTERY/BATTERY UTILITY BUS CIRCUIT BREAKER PANEL |
| 21 SLIDING WINDOW | 28 STOWAGE BAG | 34 FIRE EXTINGUISHER |
| 22 COCKPIT DOOR EMERGENCY RELEASE | 29 COLLECTIVE STICK FRICTION CONTROL | 35 GUNNER'S ICS CONTROL PANEL |
| 23 CYCLIC STICK | 30 COLLECTIVE STICK GRIP | 36 FIRST AID KIT |
| 24 DIRECTIONAL CONTROL PEDALS | 31 ENGINE IGNITION KEY LOCK | 37 GUNNER'S AMMUNITION/GRENADE STOWAGE COMPARTMENT |
| 25 PILOT'S SEAT | 32 LOWER CONSOLE | 38 COPILOT'S SEAT |

FIGURE 6.1.2(b)

COCKPIT GEOMETRIC DEFINITION

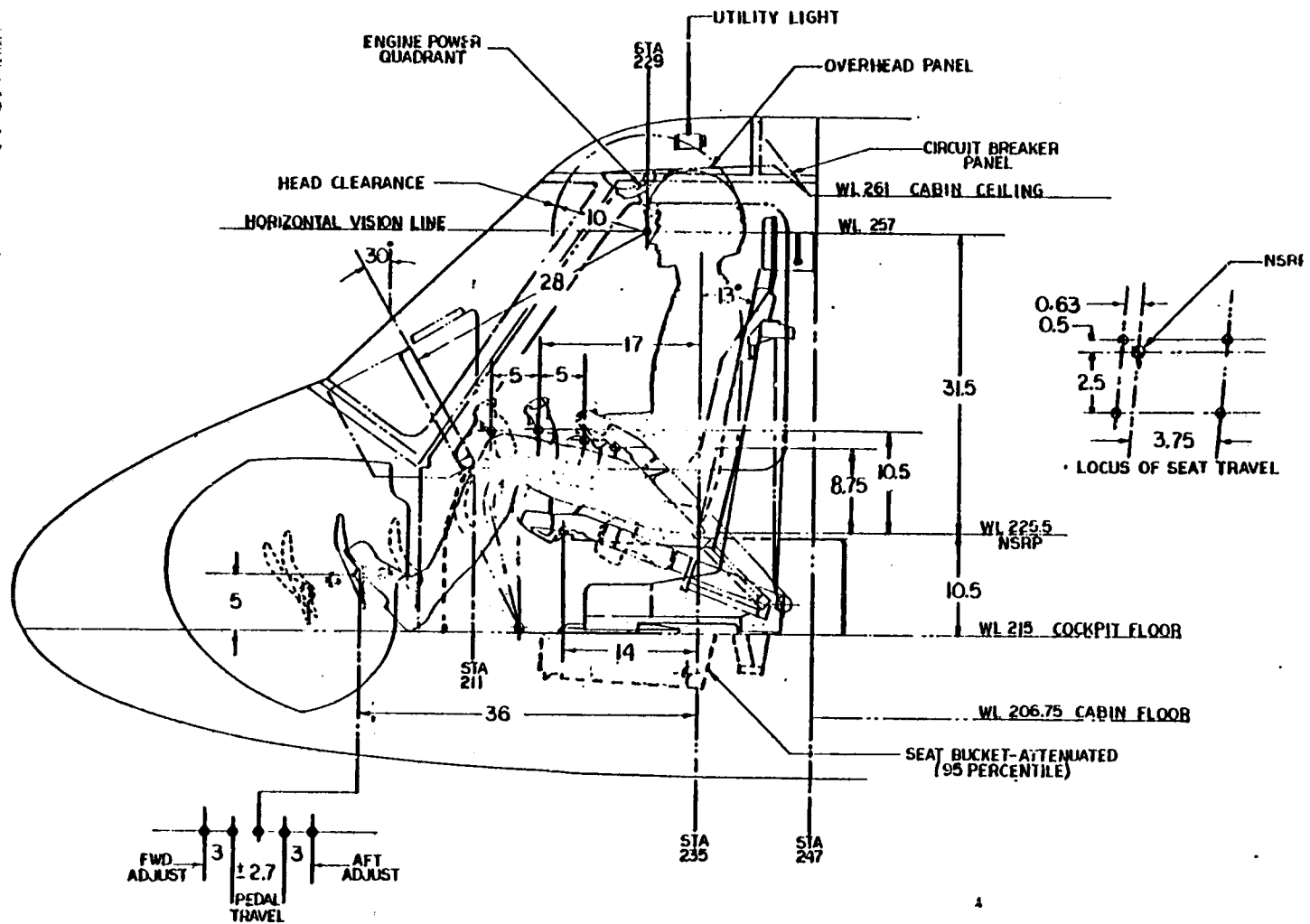


FIGURE 6.1.3(a)

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COCKPIT GEOMETRIC DEFINITION

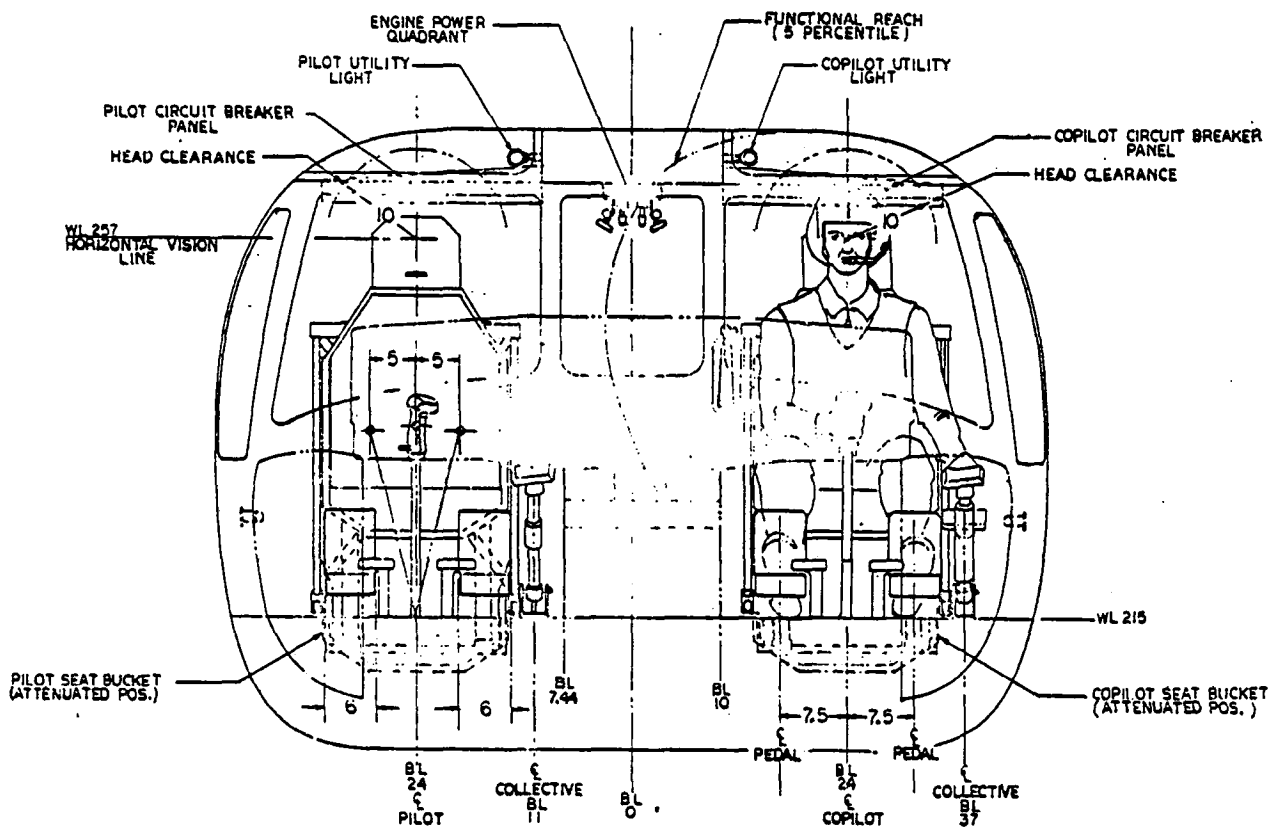


FIGURE 6.1.3(b)

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COCKPIT GEOMETRIC DEFINITION

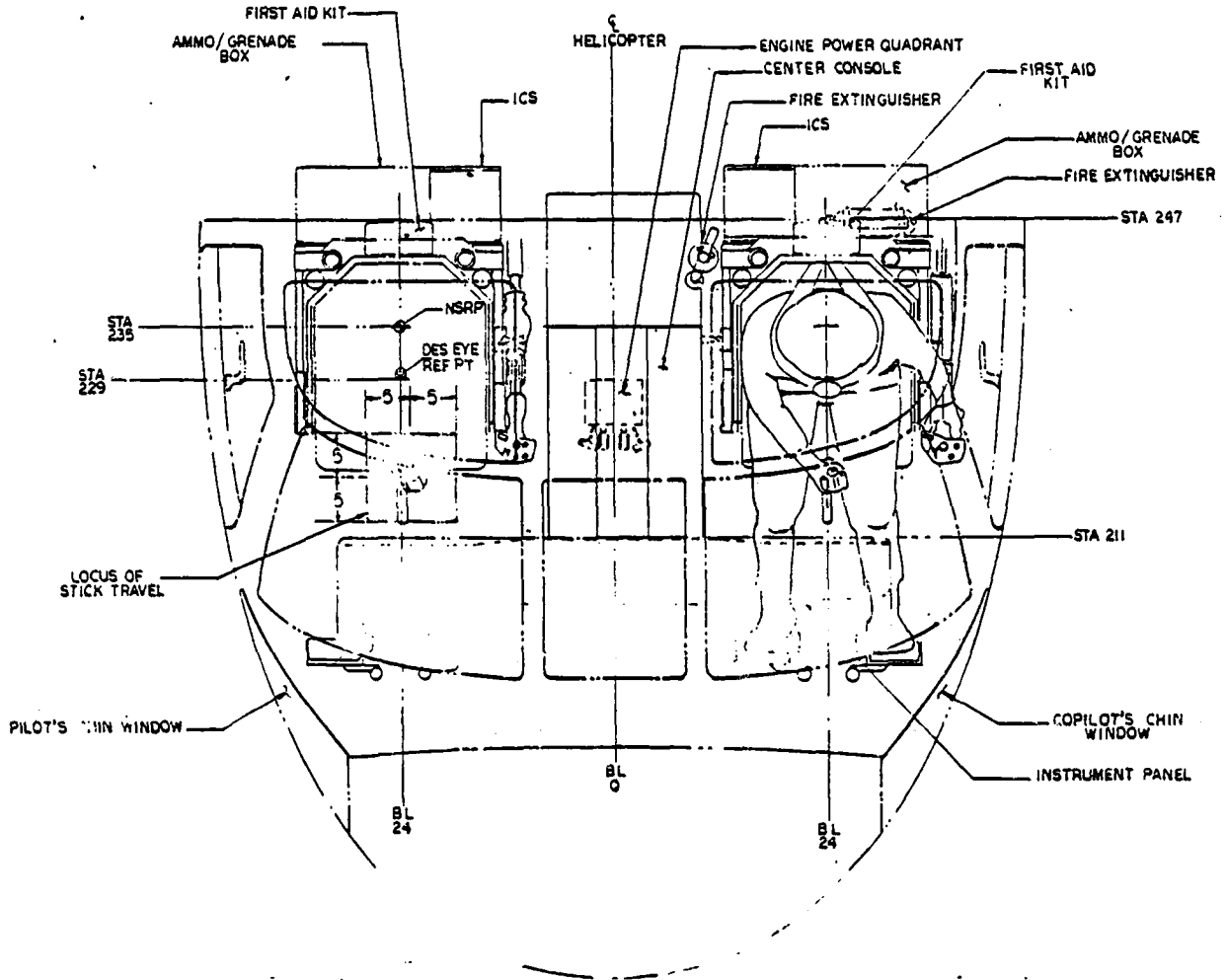
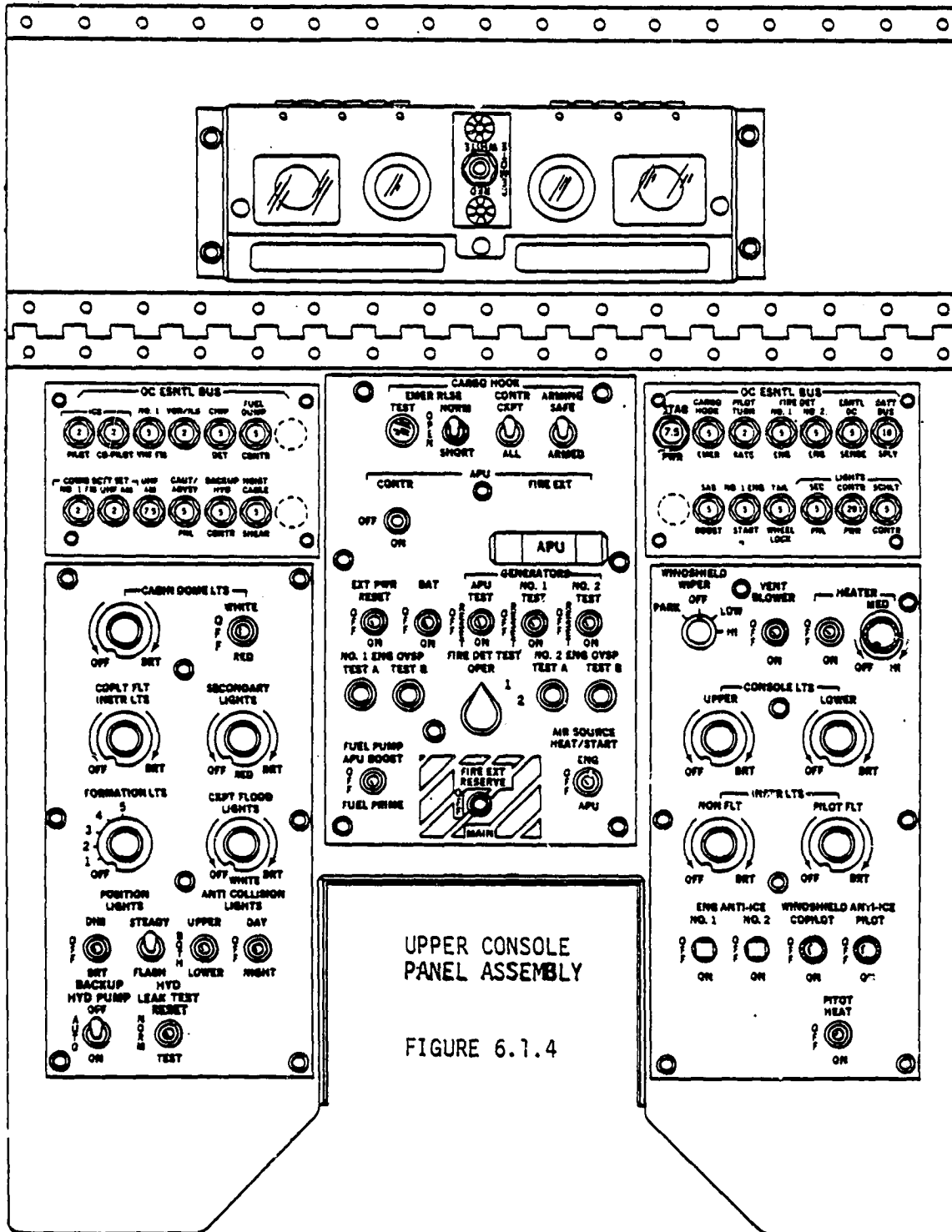
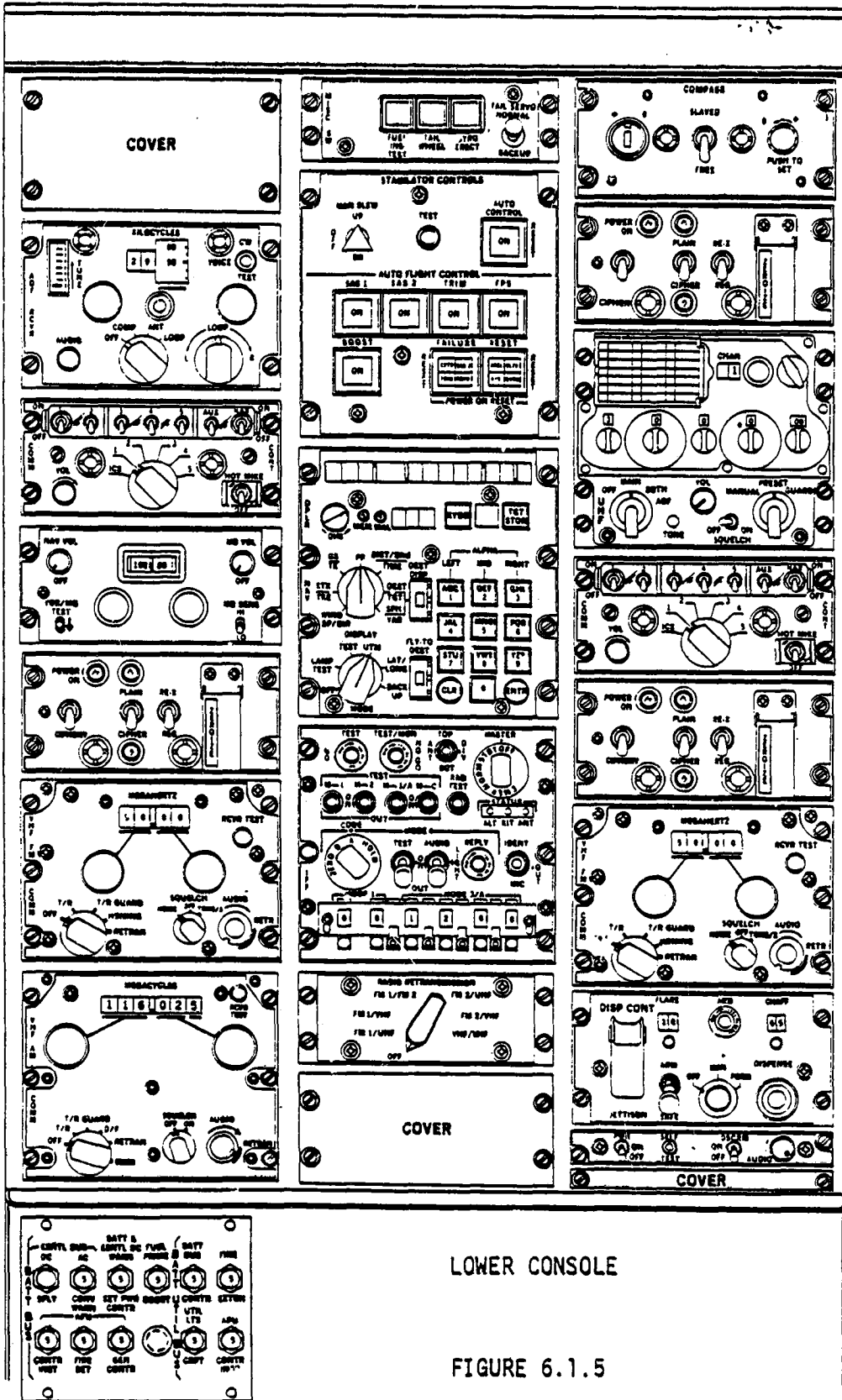


FIGURE 6.1.3(c)





LOWER CONSOLE

FIGURE 6.1.5

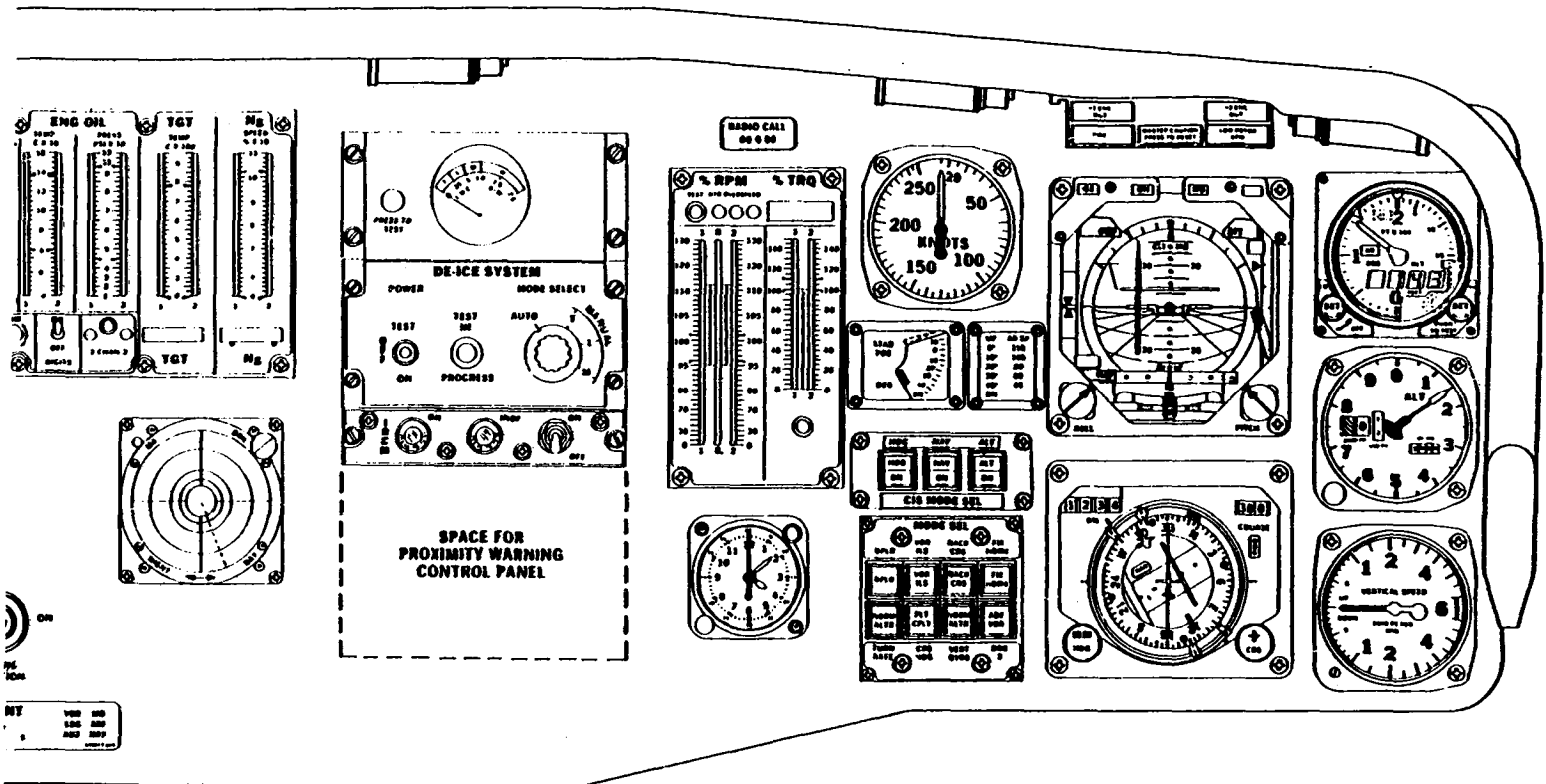
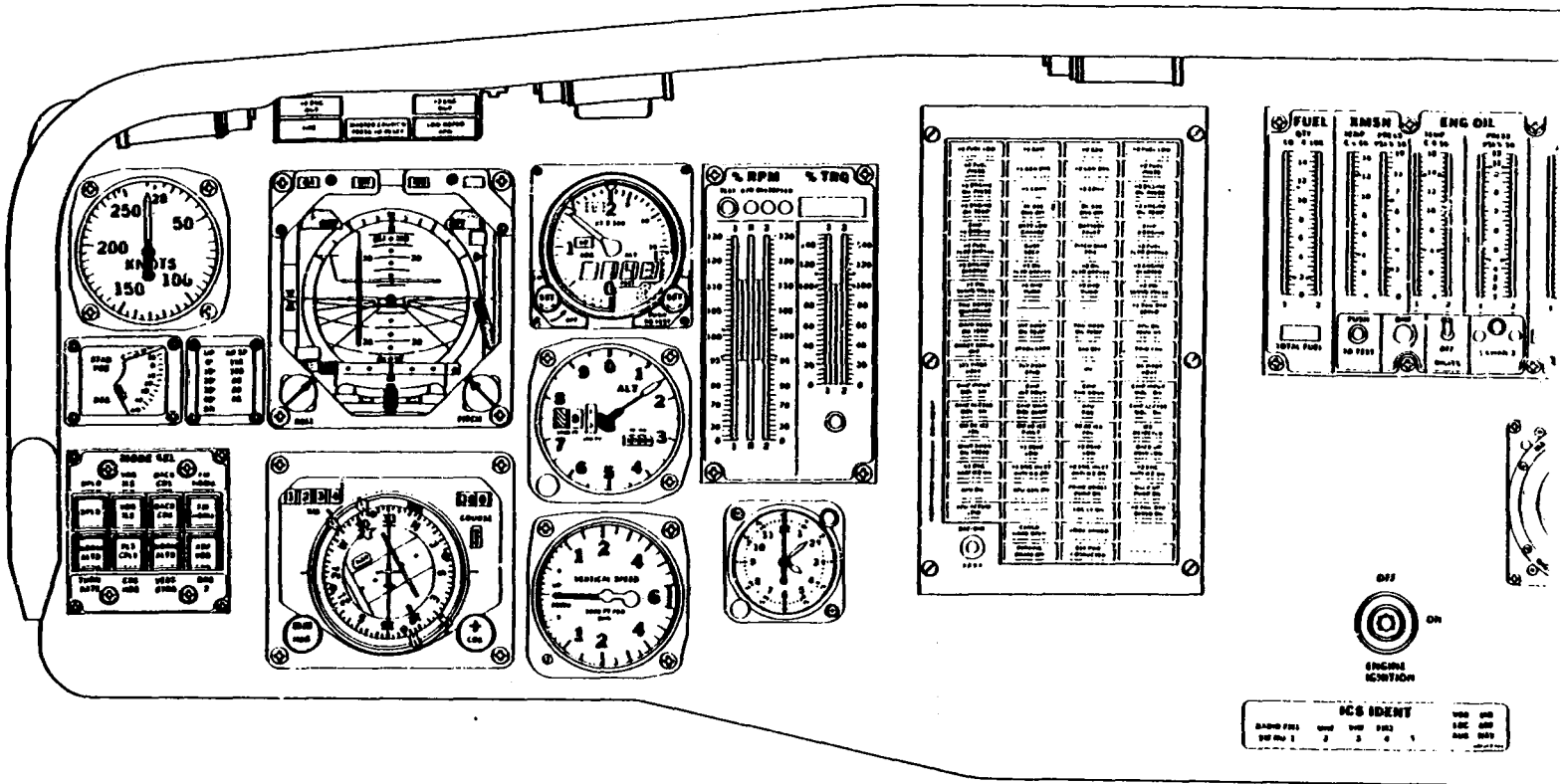
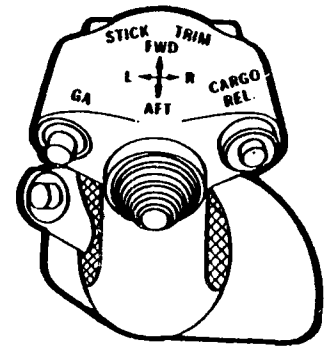


FIGURE 6.1.6



INSTRUMENT PANEL — LEFT HAND SIDE

FIGURE 6.1.6 (Cont'd.)



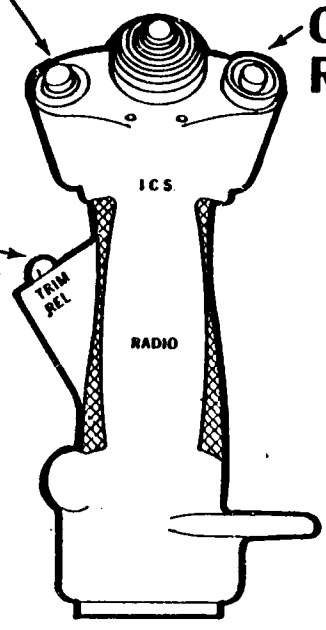
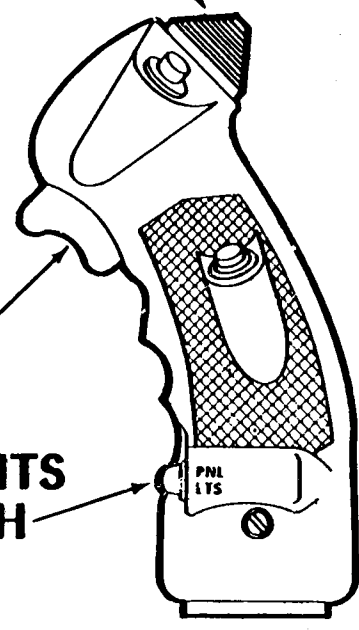
STICK TRIM
GO AROUND
ENABLE SWITCH

CARGO HOOK
RELEASE SWITCH

TRIM
RELEASE
SWITCH

ICS-RADIO
CONTROL

PANEL LIGHTS
KILL SWITCH



CYCLIC STICK GRIP

FIGURE 6.2.1

6.14
PAGE

**EMERGENCY HOOK
RELEASE**

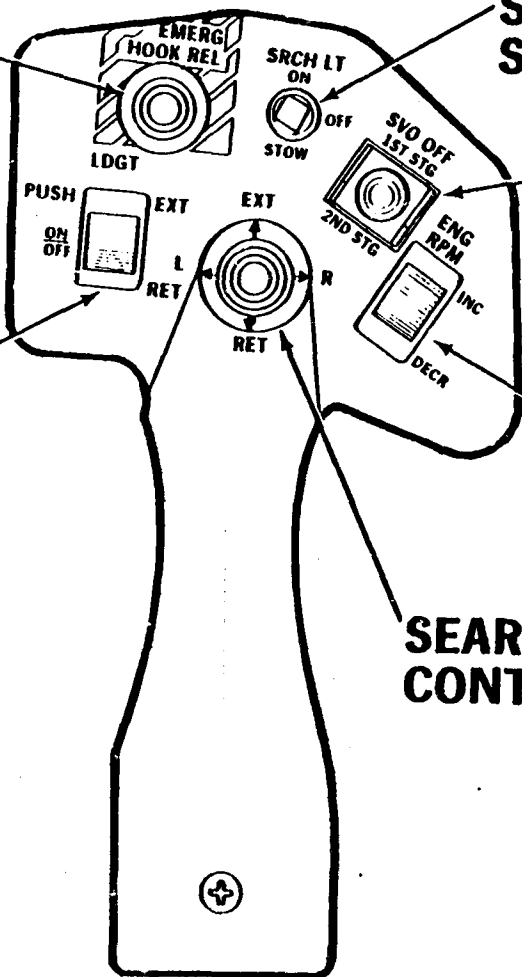
**SEARCHLIGHT
SWITCH**

SERVO SHUT OFF

**LANDING LIGHT
CONTROL**

**ENGINE SPEED
TRIM**

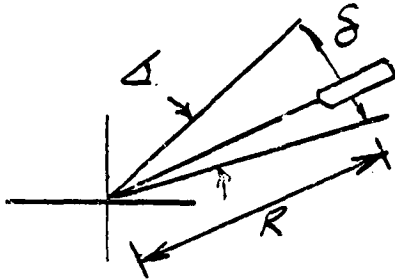
**SEARCHLIGHT
CONTROL**



COLLECTIVE STICK GRIP

FIGURE 6.2.2
6.15
PAGE

BLACK HAWK ANGULAR CONTROL MOTIONS AND DISPLACEMENTS

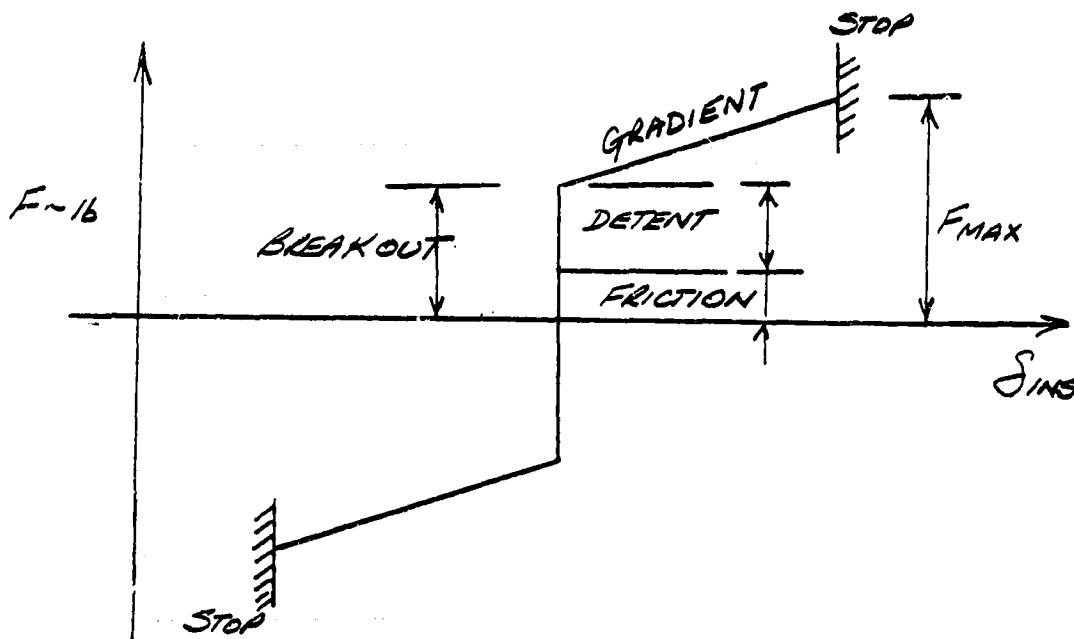


- * High Collective
- ** Low Collective
- *** Mid Collective
- **** With Overtravel
- ***** Considers Control Reduction due to redundant quadrant

CONTROL-POSITION	R(IN.)	δ (IN.)	Δ (DEG)	REFERENCE	OUTPUT @ ROTOR (DEG)
<u>COLLECTIVE</u>	24.0	-	-	Horizontal	θ_{CUFF}
Low	-	0	22.5	Above	9.9
High	-	10.0	46.55	Above	25.9
Δ	-	10.0	24.05	-	16.0
<u>LONGITUDINAL CYCLIC</u>	20.75	-	-	Neutral Cyclic	B_{IS}
Forward *	-	5.0	13.95	Forward	16.5
Pinned**	-	3.89	10.84	Aft	-11.0
Aft ***	-	5.0	13.95	Aft	-12.3
Δ	-	10.0	27.9	-	28.8
<u>LATERAL CYCLIC</u>	24.80	-	-	Vertical	A_{IS}
Left **	-	5.0	11.63	Left	-8.0
Pinned **	-	.96	2.24	Left	-1.54
Right	-	5.0	11.63	Right	8.0
Δ	-	10.0	23.26	-	16.0
<u>PEDAL</u>	15.0	-	-	Vertical	***** θ_{TRCUFF}
Left****	-	2.69	10.33	Left	29.9
Pinned ***	-	0	0	On	15.0
Right ****	-	2.69	10.33	Right	0.1
Δ ****	-	5.38	20.66	-	29.8

FIGURE 6.3.1

BLACK HAWK CONTROL FORCES AND GRADIENTS



DATA DERIVED FROM ATP 27000

CONTROL	STROKE	MAXIMUM TRIM OFF FRICTION	NOMINAL TRIM ON BREAKOUT*	NOMINAL TRIM ON GRADIENT	NOMINAL TRIM ON FMAX
Lateral Cyclic	10.0 in.	.625 lb.	.95 lb.	.54 lb/in	3.65 lb.
Longitudinal Cyclic	10.0 in.	.625 lb.	1.35 lb.	.73 lb/in	5.0 lb.
Pedal (+ Overtravel)	4.92 in. (5.38 in)	4.0 lb.	7.2 lb.	6.5 lb/in	23.2 lb. (24.7 lb)
Collective	10.0 in.	.625 lb	-	-	-

* Breakout (Trim On) = Detent + Friction (Trim Off)
All Values Measured From 50% Control Position

FIGURE 6.4.1