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Date 24 May 1989

EVALUATION AND ANALYSIS OF
THE F-16B FLIGHT CONTROL SYSTEM

A Thesis

Presented for the

Master of Science

Degree

The University of Tennessee, Knoxville

David Craig Dykhoff

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ABSTRACT

The F-16 was the first tactical aircraft to incorporate a "fly-by-wire" flight control system. This concept provided unprecedented flexibility and capability to control system designers; unfortunately, it also brought many unforeseen problems.

In 1987, the author enjoyed the privilege of flying the F-16 for the purposes of gathering quantitative data and forming qualitative opinion. The conclusions presented in this thesis are based on that data and analysis of the flight control system.

Several unique characteristics were uncovered during the evaluation, as were several serious problems. The general conclusion is that the designers of the F-16 oversimplified and probably misunderstood the intricacies of the interface between the pilot and airplane.

Fly-by-wire flight controls will undoubtedly be the standard for tactical aircraft of the future; however, our understanding of the complex man/machine synergism is critical to their successful development.

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LIST OF ABBREVIATIONS

AGL	Above Ground Level
AOA	Angle of Attack (a.k.a. alpha)
ARI	Aileron-Rudder Interconnect
BDU-33	designation for 25 lb practice bomb
CCIP	Continuously Computed Impact Point
CG	Center of Gravity
deg	degrees
ECA	Electronic Component Assembly
EPU	Emergency Power Unit
FLCC	Flight Control Computer
FLCS	Flight Control System
FLCP	Flight Control Panel
fpm	feet per minute
ft	feet
g	acceleration of 1 gravity
g LOC	g induced Loss of Consciousness
HQR	Handling Qualities Rating (see appendix D)
HUD	Heads Up Display
hz	hertz (a.k.a. cycles per second)
IMN	Indicated Mach Number
in	inches
KCAS	Knots Calibrated Airspeed
KIAS	Knots Indicated Airspeed
KT	Knots

lb	pounds
LEF	Leading Edge Flap
LOC	Loss of Consciousness
MAC	Mean Aerodynamic Chord
mils	milliradians
MPO	Manual Pitch Override
msec	milliseconds
nm	nautical miles
N_z	Normal Load Factor
PIO	Pilot Induced Oscillation
PMG	Permanent Magnet Generator
psi	pounds per square inch
rad	radians
REO	Radar Electro-Optical Display
sec	seconds
TED	Trailing Edge Down
TEU	Trailing Edge Up

CHAPTER I

INTRODUCTION

1.1 Background

An aircraft's flight control system is the mechanism by which aircraft attitude, and resultantly flight path, is governed. The moments necessary to establish and maintain aircraft attitude are generated by aerodynamic reactions on control surfaces. These surfaces generally deflect in some manner to allow adjustment of the above mentioned moments. Pilot inputs to the control system are made through cockpit controls, which generally consist of a stick (or yoke) for longitudinal and lateral control, and pedals for directional control.

Conventional control systems produce control surface deflections proportional to the deflection of the cockpit controls. All early systems and some current ones utilize mechanical linkages to effect this process.[1] The first step in the progression of control system sophistication was the introduction of "boosted" systems, which employ

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[1] Loftin, A. Quest for Performance: The Evolution of Modern Aircraft. Washington: NASA Scientific and Technical Information Branch, 1985.

hydraulic power to increase the forces available for control surface deflection.[2]

Stability and control augmentation systems were the next development step. These were the first to violate the concept of proportionality between cockpit control deflection and control surface deflection. With the design intent of improved flying qualities, these systems normally make control system inputs (in parallel with the pilot) based on aircraft body rates, aircraft accelerations or pilot inputs.[3]

The latest development is the advent of electronic flight control systems. In these "fly-by-wire" systems, cockpit control deflection becomes essentially a parameter that conveys pilot intention to the flight control computer. Based on a myriad of inputs from aircraft sensors, which may include aircraft body parameters (rates and accelerations), air data (dynamic pressure, airspeed, Mach number), aircraft status (weight, loading, configuration and center of gravity), and mission phase (landing, cruise, combat), these control systems determine

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[2] Richardson, R. "Aircraft Flight Control Systems". USNTPS Notes, 1987.

[3] Mondey, D. The Complete Illustrated Encyclopedia of the World's Aircraft. Secaucus: Chartwell Books, Inc., 1978.

the control surface deflections required to satisfy pilot demands.[4]

1.2 Identification of Subject

The F-16 was the first tactical aircraft to employ a fly-by-wire control system. No mechanical connection between cockpit controls and flight control surfaces was incorporated. The cockpit controls themselves were also unique, in that they deflected very little when driven and relied on force sensors to detect pilot inputs.[5]

Free of the limitations and compromises inherent in conventional control systems, the F-16 provided designers with the opportunity to create the ultimate flight control system. Aircraft overstress and departure from controlled flight could be eliminated, and optimum airplane response could be programmed for every control input.[6] Such was the perception that prevailed until the aircraft got off the drawing board and into the air.

The first F-16 flight almost terminated in a crash immediately following takeoff due to an uncontrollable

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[4] Huenecke, K. Modern Combat Aircraft Design. Anapolis: Naval Institute Press, 1987.

[5] Jones, S. U.S. Fighters. Fallbrook: Aero Publishers, 1975.

[6] Whitford, R. Design for Air Combat. London: Jane's Publishing Co., 1987.

lateral pilot induced oscillation[7]. Further flight testing revealed that the man/machine interface that occurs during many mission phases was much more complicated than originally assumed.[8] Several problems were remedied prior to service delivery, but several remain uncorrected.

This thesis does not attempt to chronicle the development of the F-16 flight control system; rather, it presents an objective evaluation from a pilot perspective and suggests ways to remedy the problems discovered. The design of the F-16 was extremely radical for its time, and it represents a developmental milestone; as such, it is an ideal candidate for such a study.

This thesis assumes a basic understanding of aircraft stability and control, as well as familiarity with basic flight test techniques and nomenclature.

1.3 Work Chronology

The testing executed for this thesis was performed in late 1987 as part of a Phase IIA Developmental Testing Evaluation which constituted the author's final project at

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[7] Parrag, M.; Knotts, L. Augmented Aircraft Handling Qualities. Buffalo: CALSPAN Corp., 1987.

[8] Smith, E. "Evaluating the Flying Qualities of Today's Fighter Aircraft," Design Criteria for the Future of Flight Controls (Proceedings of the Flight Dynamics Laboratory Flight Control Symposium 2-5 March 1982). AFWAL-TR-82-3064.

the United States Naval Test Pilot School. Such an evaluation is concerned only with the identification of deficiencies.

This thesis was composed during the fall and spring semesters of 1989. Data from the above mentioned evaluation was utilized, upon which the author has attempted to expand by further defining the deficiencies, determining their mission impact, analyzing their causes and suggesting remedies for them.

CHAPTER II

DESCRIPTION OF FLIGHT CONTROL SYSTEM [9][9] [10]

The F-16 incorporates a four channel, computer controlled, analog fly-by-wire flight control system (FLCS) that hydraulically positions control surfaces. Electrical signals generated by a stick, rudder pedals and a manual trim panel are input to a flight control computer (FLCC) which also receives signals from the air data system, flight control rate gyros and accelerometers. The FLCC commands the appropriate deflection of the primary control surfaces, which include all movable horizontal tails, flaperons and a rudder. Roll coordination is provided by an aileron-rudder interconnect, and limiters are incorporated for all three axes. Three axis trim commanded by the pilot is also accomplished through the flight control computer. Secondary flight controls include automatic leading edge flaps and a speedbrake. Flight control surfaces are shown in appendix A, figure 1.

2.1 Cockpit Controls and Displays

Pitch and roll commands are initiated through the side-mounted control stick shown in appendix A, figure 2. The stick is a two axis, force sensing, minimum displacement

[9] Numbers in brackets refer to similarly numbered references in the Bibliography.

unit mounted on the right side of the cockpit with a slight clockwise rotation. A wristrest assembly is mounted on the right sidewall aft of the stick. Directional commands are made through the force-sensing rudder pedals, which also generate brake and nose wheel steering signals. Rudder pedal feel is provided by mechanical springs. Deflection displacements and forces are presented in table I.

Table I
MAXIMUM COCKPIT CONTROL DISPLACEMENTS

Control	Deflection Direction	Force for Maximum Command (1)		Maximum Deflection Displacement (2)
		Gear Up	Gear Down	
Control Stick	Aft	25 lb		1/4 in
	Forward	17 lb		1/8 in
	Left	17 lb	12 lb	1/4 in
	Right			
Rudder Pedals	Left	110 lb	110 lb	5/8 in
	Right			

(1) Forces determined from transfer functions based on trim for 1g, wings level flight.
(2) Displacements measured during ground testing.

Trim for all three axes is commanded by dials on a manual trim panel mounted on the left console. Longitudinal and lateral trim may also be commanded through a thumbswitch mounted on the control stick. The manual trim panel, which also incorporates trim indicators and a trim/autopilot disconnect switch, is shown in appendix A, figure 3.

The Flight Control Panel (FLCP), located on the left console, houses various switches and indicators related to flight control functions. Two of the switches, LE FLAPS and ALT FLAPS, provide control of the leading edge flaps and flaperons, respectively. The autopilot panel contains the switches used to establish pitch and roll attitude hold. The FLCP and autopilot panels are shown in appendix A, figure 4. Additional FLCS switches are listed with their locations and functions in table II.

Table II
FLCS SWITCHES

Switch Designation	Location	Function
Paddle Switch	Front Side of Control Stick	When paddle switch depressed in cockpit designated by stick selector switch, flight control inputs from other cockpit locked out
Stick Select	Left Console	
Manual Pitch Override	Left Console	Allows pilot to assume manual control of horizontal tails
Stores Config.	Left Auxiliary Console	Changes FLCS limits to accommodate different loadings
Nose Wheel Steering	Right Side of Control Stick	Engages/Disengages Nose Wheel Steering
Speedbrake Switch	Left Side of Throttle	Extends/Retracts Speed Brakes

Several FLCS indicator lights are incorporated to warn the pilot of FLCS failures. A dual malfunction in one of the electrical control axes, a servoactuator, or the AOA portion of the air data system is annunciated by a red DUAL FC FAIL warning light located on the right upper edge of the glareshield. A FLT CONT SYS caution light, located on the caution light panel on the right auxiliary console, illuminates when a failure occurs in the FLCS. A LEF FLAP light, also located on the caution panel, illuminates when a failure has occurred in the leading edge flap command electronics or when the leading edge flaps have been commanded to lock.

2.2 Primary Flight Controls

The primary flight control system employs quadruplex flight control computer channels with quadruplex signal inputs, except for gun compensation and autopilot inputs which are non-redundant. Electrical isolation is incorporated with the intent of preventing a single failure from affecting more than one branch. Voting performed on the four command output signals is intended to identify a faulty channel and permit continued flight with the good channels. There are no mechanical linkages between the cockpit and the control surfaces. Integrated servoactuators, which are powered by both hydraulic systems, deflect the control surfaces to their commanded positions.

2.2.1 Longitudinal Flight Path Control

Longitudinal control is accomplished through symmetrical deflection of the all-moveable horizontal tails, which deflect a maximum of 21 degrees trailing edge down and 21 degrees trailing edge up.[10] Longitudinal control laws are implemented by the flight control

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[10] Dennison, K.; Phase IIA Developmental Testing of the F-16B Airplane. Patuxent River: NATC, 1988.

computer, which determines a "pitch command error signal" from the following inputs:

(1) longitudinal stick force,

(2) angle of attack (AOA),

(3) dynamic pressure,

(4) pitch rate,

(5) roll inputs commanding the horizontal tails,

(6) pitch trim,

(7) normal acceleration,

(8) roll rate (for AOA limiting), and

(9) autopilot inputs.

The "pitch command error signal" is gain adjusted then fed through a proportional plus integral feedback loop, which maintains trim flight and provides airplane stability. Maximum command signals are restricted by a load factor limiter for forward stick commands and an AOA/load factor limiter (which varies the available AOA with load factor) for aft stick commands. A pitch command limiter is incorporated with the intent of preventing pitch control saturation during full deflection commands. A detailed block diagram of the flight control system control laws is presented as appendix A, figure 5. Control law gains are dictated by airplane configuration as specified in table III.

Table III
FLCS GAINS

Designation	Configuration / Failure State
Cruise Gains	Landing Gear Handle in UP Position and Alternate Flap Handle in NORM Position and Air Refuel Door in CLOSED Position
Slow Flight Gains	Landing Gear Handle in DOWN Position or Alternate Flap Handle in EXTEND Position or Air Refuel Door in OPEN Position
Standby Gains	Dual Failure of Total Pressure Inputs or Dual Failure of Static Pressure Inputs or Dual Channel FLCS Failure

2.2.1.1 Cruise gains control laws. Below 15 degrees AOA, the longitudinal control laws are designed to yield a normal load factor corresponding to longitudinal stick force, which is sometimes referred to as a "g command system". Since stick force commands load factor and not angle of attack, speed stability is intended to be neutral. The designed pitch command gradient, which is intended to be invariant with flight condition, is shallower for stick forces less than 6.0 lb as presented in table IV.

Table IV
DESIGN PITCH COMMAND GRADIENTS

Direction	Longitudinal Stick Force (lb)	Local Stick Force per Normal Load Factor (lb/g)
Aft	0 to 1.75	0 (Breakout)
	1.75 to 6.0	9.6
	6.0 to 32	2.2
Forward	0 to 1.75	0 (Breakout)
	1.75 to 6.0	9.6
	6.0 to 32	2.8

Maximum negative load factor is restricted by the limiter and varies between -4.0 and -1.0 depending on dynamic pressure as depicted in appendix A, figure 5. Positive AOA/load factor limiter boundaries change as a function of the stores configuration switch and are shown in appendix A, in figure 6.

Above 15 degrees AOA, the FLCS changes the pitch control law to command a blend of normal load factor and angle of attack. The design result is positive static stability with the intent to provide high AOA/low airspeed warning. Maneuver gradients are scheduled as a function of the change in normal load factor per unit change in AOA (Nz/alpha).

2.2.1.2 Slow flight gains control laws. Below 10 degrees AOA, the longitudinal control laws are designed to yield a pitch rate corresponding to longitudinal stick force, which is sometimes referred to as a "pitch rate command system". Maneuver gradients are shallower for stick forces less than 6.0 lb in either direction. Positive pitch rate commands are limited to a maximum of 16 deg/sec and are not affected by the stores configuration switch. Negative pitch commands are limited to a maximum of -8 deg/sec. Above 10 degrees AOA, the FLCS changes to a blended pitch rate and AOA command system.

2.2.1.3 Longitudinal trim. The longitudinal trim system allows the pilot to adjust the commanded load factor for trimmed flight (i.e. no longitudinal stick commands). A stack of four potentiometers, located in the manual trim panel, produces a quadruplex "g command signal" that is summed with stick inputs upstream of the flight control computer. The pilot adjusts the potentiometers with a manual trim wheel on the panel, or with a thumbswitch on the control stick which commands an electrical gear head motor. The motor is connected to the potentiometers through a slip clutch. A trim disconnect switch disables the thumbswitch circuit but does not affect manual trim wheel adjustment. The slip clutch is intended to allow manual adjustment of the trim wheel in the event of an electrical motor jam. An indicator needle on the manual control panel displays the commanded trim load factor. Design specifics are presented in table V.

Table V
LONGITUDINAL TRIM SYSTEM DESIGN PARAMETERS

Trim Range	Indication Sensitivity	Maximum Trim Rate (Ground)
+/- 2.4 g	0.675 g/dot	2.7 deg/sec horiz. tail

2.2.2 Lateral Flight Path Control

2.2.2.1 Roll rates. Lateral control is accomplished through asymmetric deflection of both the all-moveable horizontal tails and the trailing edge flaperons. The ratio of degrees horizontal tail deflection to degrees trailing edge flaperon deflection changes with mach number according to the schedule shown appendix A, figure 5, F10. Maximum deflections for these surfaces is presented in table VI.

Table VI
ROLL CONTROL SURFACE MAXIMUM DEFLECTIONS

Control Surface	Maximum Deflection	
	Trailing Edge Down	Trailing Edge Up
Differential Trailing Edge Flaperon	20 deg	23 deg
Differential Horizontal Tail	25 deg	25 deg

In cruise configuration, the neutral flaperon position is streamlined. Roll commands deflect the flaperons equal amounts in opposite directions until the downward deflecting flaperon reaches 20 degrees TED, after which the downward deflecting flaperon stops and the upward deflecting flaperon continues to its 23 degree TEU limit. Use of the flaperons as high lift devices (described in section 2.3.2) shifts their neutral position downward since asymmetric (roll) and symmetric (high lift) deflection commands are summed. As a result, downward deflection available for roll control is reduced. With the landing gear handle DOWN, the flaperons assume a 20 degree TED neutral position and only the upward deflecting flaperon moves in response to roll commands.

Lateral control laws are implemented by the flight control computer, which determines a roll rate error signal from the following inputs:

(1) lateral stick force,

(2) roll rate,

(3) dynamic pressure,

(4) angle of attack,

(5) pitch inputs commanding the horizontal tails,

(6) roll trim,

(7) autopilot inputs.

The lateral control laws are designed to effect a roll rate corresponding only to lateral stick force and invariant with flight condition. The designed roll command force gradient is shallower for stick forces less than 6 lb as presented in table VII.

Table VII
DESIGN ROLL COMMAND FORCE GRADIENTS

Direction	Lateral Stick Force (lb)	Stick Force per Commanded Roll Rate (lb/deg-sec)
Either	0 to 1	0 (Breakout)
	1 to 5	0.2
	5 to 9	0.067
	9 to 17.6	0.035

When cruise gains are in effect, a roll rate limiter can permit a maximum rate of 324 degrees per second. This maximum commanded roll rate is decreased if (1) equivalent airspeed is less than 170 KT (2) AOA is greater than 15 degrees or (3) stabilator deflection is greater than 5 degrees. The lowest limit value (most restricted maximum roll rate) is designed to be 80 degrees per second. Gradients and limiter functions are similar with slow flight gains in effect (defined in table III) except that the maximum limiter value is designed to be 167 degrees per second and does not change with flight condition. A detailed block diagram of FLCS control laws is presented as appendix A, figure 5.

2.2.2.2 Roll mode response. A non-linear lag filters lateral stick force commands. A longer lag is applied to stick inputs away from neutral than is applied to inputs when the stick is returning to neutral. The intended result is a longer roll mode time constant when the stick is being displaced than when it is being released. Filter response characteristics predict roll mode time constants in the vicinity of 400 and 200 msec for roll initiation and termination, respectively. For roll rate commands of less than 20 deg/sec, the designed time constant for roll in is the same as for roll out (200 msec).

2.2.2.3 Lateral trim system. The lateral trim system functions are identical to those of the longitudinal trim system, except that the pilot is selecting a trim condition roll rate vice a trim condition load factor. Design specifics are presented in table VIII.

Table VIII
LATERAL TRIM SYSTEM DESIGN PARAMETERS

Trim Range	Indication Sensitivity	Maximum Trim Rate (Ground)
+/- 66.7 deg/sec	18.76 deg/sec-dot	0.54 deg/sec flaperon

2.2.3 Directional Flight Path Control

Directional control is provided by the deflection of a single rudder with deflection limits of 30 degrees left and right. Directional control laws are implemented by the FLCS, which sums rudder pedal force inputs with a blend of lateral acceleration (for command augmentation) and washed out yaw rate (for stability augmentation) to determine a rudder command signal. Directional trim and aileron/rudder interconnect signals are added to this command signal to determine rudder deflection. Rudder pedal force gradient is linear (3.2 degrees/lb) and is presented in appendix A, figure 5. Rudder pedal inputs are gain adjusted at high angles of attack, with attenuation beginning at 14 degrees AOA and authority decreasing linearly to zero at 26 degrees AOA. With the stores configuration switch in the CAT III position, the AOA values for full and zero authority change to 3 and 15 degrees, respectively. Above 29 degrees AOA the FLCS uses a lagged yaw rate signal to command rudder deflection with the intention of preventing a departure/spin.

2.2.3.1 Directional trim. The directional trim system allows the pilot to adjust the controls-free position of the rudder. Implementation is similar to the pitch and roll axes, except the cockpit control on the manual trim panel is a dial instead of a thumbwheel and no electrical trim provisions are provided on the control stick. Design specifics are presented in table IX.

Table IX
DIRECTIONAL TRIM SYSTEM DESIGN PARAMETERS

Trim Range	Indication Sensitivity	Maximum Trim Rate (Ground)
+/- 12 deg	2.0 deg/dot	120 deg/sec rudder

2.2.3.2 Aileron-to-rudder interconnect (ARI). The ARI is designed to provide turn coordination and roll-yaw decoupling over a wide range of flight conditions. A "turn coordination required" signal is generated using indicated Mach number and shaped angle of attack. This signal is then used to gain-adjust the aileron command signal, and the result is summed in the yaw command channel. The ARI signal is removed from the yaw command computation when angle of attack exceeds 29 degrees or wheel speed is greater than 60 kt.

2.2.3.3 Gun compensation. The gun compensation function is intended to automatically correct for the reaction forces caused by gun firing. Since the gun is mounted off-center, these forces include a yawing moment, a sideforce and a rearward force. A non-redundant signal from the gun motor is lagged, gain adjusted and summed with roll and yaw axis commands with the intent of allowing precise tracking during gun firing operations. Additionally, disruption of the tracking solution would normally occur when the yaw command channel makes a rudder input in response to the lateral acceleration it senses during gun firing; a sideforce canceller is incorporated with the intent of eliminating this yaw command and preventing the tracking disturbance.

2.2.4 Standby Gains

The FLCS utilizes standby gains for flight control computations in the event of a dual air data failure or dual channel FLCS failure. Control responses are tailored for a default flight condition of approximately 600 KCAS with the landing gear handle UP and 230 KCAS with the landing gear handle DOWN. The leading edge flaps are set at zero degrees unless the landing gear handle is DOWN or the alternate flap switch is set to EXTEND, in which case they deflect to 15 degrees TED. Trailing edge flap operation is not affected. Operation on standby gains should illuminate the STBY GAINS, LE FLAPS and FLT CONT SYS lights and possibly the ADC light depending on the failure. The STBY GAINS light cannot be reset in flight, and the FLCS will continue to operate on standby gains even if the failure clears.

2.3 Secondary Flight Controls

2.3.1 Leading Edge Flaps (LEF)

Full span leading edge flaps are intended to improve performance, rudder effectiveness and buffet characteristics at high angles of attack in addition to providing increased lift for takeoff and landing. The electronic component assembly (ECA) schedules LEF extension as a function of angle of attack, Mach number and pressure altitude. The ECA computes triplex LEF command signals, which are supplied to dual redundant LEF command servos and electrical drive motors. The motors turn a torque tube and mechanical actuators in each wing to extended the LEFs. Deflection limits are presented in table X.

Table X
LEADING EDGE FLAP DEFLECTION LIMITS

Pressure Altitude (ft)	Deflection Limits (deg LED) (1)	
	UPPER	LOWER
Below 10,000 (2)	0	25
10,000 to 15,000	0 to -2 (3)	25
Above 15,000	-2	25
Above 20,000 and > Mach 1.0	FIXED AT - 2	

(1) Deflection limits measured in degrees of leading edge down, using the streamlined position as a 0 degree reference.
(2) With weight on wheels, or wheel speed greater than 60 kt and throttle at idle, LEF deflection fixed at -2 degrees.
(3) Upper deflection limit changes linearly from 0 to -2 degrees as altitude increases from 10 to 15,000 ft.
(4) With a dual air data failure, the system reverts to STANDBY mode and fixes the LEF deflection at 15 degrees for slow flight configuration and 0 degrees for cruise configuration (defined in table III).

LEF extension schedule is presented as appendix A, figure 7. Asymmetry sensing/braking mechanisms, mounted at the outboard end of each drive torque tube, are intended to prevent asymmetric LEF extension. Pulses from cam switches in the sensing/braking mechanisms are compared in the ECA, which is designed to disable the LEF command servo (locking the LEF) if it detects a miscompare. The LEF is locked in its present position in a similar manner when the pilot positions the LEF switch to the LOCK position. In either case, the ECA illuminates the LEF FLAPS caution light.

2.3.2 Trailing Edge Flaperons

Three-fourths span trailing edge flaperons are mounted on the inboard trailing edge of the wing and function both as ailerons and high lift devices. The trailing edge flaps are commanded to extend by placing the landing gear handle to DOWN, the alternate flap switch in EXTEND, or the air refuel door switch in OPEN. Any of the above conditions cause the flaperons extend to their downward limit of 20 deg TED if airspeed is below 240 KCAS; above 240 KCAS commanded deflection is reduced linearly with dynamic pressure up to 370 KCAS where no deflection occurs. Flaperons are programmed to 2 deg TEU in the transonic region with the intent of reducing transonic and supersonic drag.

2.3.3 Speed Brakes

Speed brake surfaces consist of two pairs of clamshell surfaces located adjacent to the engine nozzle and inboard of the stabilator surfaces. Each of the four surfaces is positioned by a double acting hydraulic actuator powered by hydraulic system A. Extension limits are presented in table XI.

Table XI
SPEED BRAKE EXTENSION LIMITS

Configuration	Extension Limit (deg)
Landing Gear Handle UP	60
Landing Gear Handle DOWN, Weight Off Nose Gear	43 (1)
Landing Gear Handle DOWN, Weight On Nose Gear	60

(1) 43 degree limit in this configuration can be overridden by holding speed brake switch in the open (aft) position, in which case 60 degree extension is available.

A total of six closure seals are actuated by the motion of the speed brake. Speed brake extension and retraction are commanded by a three position switch on each throttle. The open (aft) position is spring loaded to return to the off (center) position; the closed (forward) position is detented. Either cockpit has manual command of the speedbrakes if the other cockpit switch is in the off (center) position. If one of the switches is in the closed (forward) position, the other cockpit switch must be held in the open (aft) position or the speed brakes will retract.

2.4 Post-Departure Flight Control Operation

2.4.1 Yaw Rate Limiter

When angle of attack exceeds 29 degrees, a yaw rate limiter overrides lateral stick, rudder pedal and ARI inputs. The limiter applies rudder opposite and flaperon into the sensed yaw rate until AOA drops below 29 degrees with the intent of improving spin resistance. No similar provisions are provided for yaw rate limiting during inverted departures.

2.4.2 Manual Pitch Override (MPO)

Placing the MPO switch to OVRD allows the pilot to override the negative g limiter. If angle of attack is above 29 degrees, the MPO override also overrides the positive g/AOA limiter and enables rudder pedal inputs. The MPO system is designed to provide the pilot with the control authority required to recover from upright and inverted deep stalls.

2.5 Related Aircraft Systems

2.5.1 Autopilot System

The autopilot system is designed to provide attitude hold or altitude hold in the pitch axis and attitude hold or heading select in the roll axis. Any combination of functions may be selected using the two position ROLL and PITCH switches on the miscellaneous panel. The master

AUTOPILOT switch is electrically held to the ON position and goes to the OFF position if any of the following occur:

(1) TRIM/AP DISC switch to DISC;

(2) AIR REFUEL switch to OPEN;

(3) ALT FLAPS switch to EXTEND;

(4) Landing gear handle to DOWN;

(5) STBY GAINS light illuminated.

Autopilot operation is interrupted (but the AUTOPILOT switch remains ON) while the paddle switch on the stick is held depressed.

2.5.2 Nosewheel Steering System

Nosewheel steering is electrically controlled and hydraulically actuated using system B pressure. Rudder pedal force commands nosewheel deflection, which reaches a maximum of 32 degrees in each direction. If nosewheel steering is engaged with force on the rudder pedals, the nosewheel will be rapidly driven to the commanded deflection. Nosewheel steering is disengaged when the nose landing gear strut is extended, and is not available following alternate landing gear extension.

2.5.3 Hydraulic Systems

Two independent hydraulic pumps supply pressure to two 3,000 psi hydraulic systems (designated A and B). Both systems operate simultaneously to supply hydraulic power to the primary flight controls (including the trailing edge flaperons) and leading edge flaps. The speed brake is powered by system A, and the nosewheel steering is powered by system B. Each system incorporates a FLCS accumulator to provide additional hydraulic power during rapid surface deflection and to furnish temporary hydraulic power while the emergency power unit (section 2.5.5) comes up to speed following a dual hydraulic failure. A diagram of the hydraulic power system is presented as appendix A, figure 8.

2.5.4 Electrical System

The primary FLCS electrical power supply system includes a dedicated FLCS permanent magnet generator (PMG), two dual channel converter/regulators, and four inverters. The engine driven FLCS PMG is the primary power source during normal operations. It has four outputs, one for each branch of the FLCS, and is designed to generate sufficient power to operate the FLCS at 40 percent RPM. Converter/regulators and inverters condition the FLCS PMG output to provide usable power to the four FLCS channels. Backup power sources include the main generator, the EPU generator (section 2.5.5), the EPU PMG, the airplane battery and the FLCS batteries.

2.5.5 Emergency Power Unit

The EPU is a self-contained system that is designed to simultaneously provide emergency hydraulic pressure to system A and emergency electrical power to the FLCS. The EPU is started using hydrazine and is sustained by engine bleed air with automatic hydrazine augmentation if bleed air is insufficient. The EPU is designed to activate automatically when both hydraulic system pressures fall below 1000 psi or the main generator disconnects from the bus system. Manual actuation of the EPU is designed to be available regardless of failure conditions.

CHAPTER III

FLIGHT TEST RESULTS

3.1 Scope of Tests

Flying qualities were evaluated during four flights totalling 4.6 hours duration under daylight visual conditions. Flight tests were conducted within the limits prescribed by the F-16 A/B Flight Manual [7], the F-16 Combined Test Force and the Air Force Flight Test Center and were flown in the Edwards Complex I and II areas of Restricted Area R2508. Mechanical characteristics were evaluated during a one hour ground evaluation.

Airplane loadings, which included two pilots and full internal fuel (5700 lb), utilized two external store configurations which were designated A and B as specified in table XII.

Table XII
F-16B TEST LOADINGS

Loading Desig.	Station	Suspension/ Store (1)	Drag Count (Units)	Takeoff G.W. (lb)	Loading Category (2)
A	-	None	7	21,900 (3)	I
B	3	1 x SUU-20/ 6 x BDU-33	55	23,750	III
	7	1 x SUU-20/ 6 x BDU-33			
	1	1 x ATM-9P			
	9	1 x ATM-9P			

(1) Both configurations included wingtip AIM-9 launcher rails (stations 1 and 9).
(2) Flight control system gains and limiter values were changed by a Stores Configuration switch, the position of which is dictated by the Loading Category (I or III).
(3) Takeoff weight shown for loading designation A is an average for the three flights using that loading.

Airplane test configurations are presented as appendix B, table I and detailed test conditions are presented as appendix B, table II. Airplane gross weight varied between 23,550 and 17,500 lb, and center of gravity (CG) position varied between 32.8 and 37.2% mean aerodynamic chord (MAC), which is relatively aft within the contractor recommended limits of 23.4 to 38.5% MAC. Representative weight and balance forms for each of the two loadings are presented as appendix C. Test envelope boundaries are presented as table XIII.

Table XIII
F-16B TEST ENVELOPE

Config/ Loading Category	Pressure Altitude (ft)	Airspeed (KIAS)	Load Factor (Nz)	Angle of Attack (deg)	Side Slip (deg)
Gear Up CAT I	1000 AGL to 41,200	102 to 710	-0.4 to +9.0	-1 to 26	10
Gear Up CAT III	550 AGL to 21,400	205 to 510	+0.5 to +5.2	1 to 15	3
Gear Dn	Field Elev to 10,300	95 to 270	+0.3 to +2.1	-1 to 21	10

3.2 Method of Test

Flying qualities and stall characteristics were evaluated in accordance with the FIXED WING STABILITY AND CONTROL TEST MANUAL [36]. Manufacturer recommended procedures [7] for takeoff, climb, cruise, descent and landing were used in establishing flight conditions. Handling Qualities Ratings (HQRs) were assigned in accordance with the Cooper-Harper Scale (appendix D) published in reference [25].

Flight parameter data were recorded from production cockpit instrumentation. Side slip angles were estimated using side slip release heading comparison. Control stick forces were measured in the front cockpit using a hand held force gauge (0-30 lb). Due to the shape of the control stick, push and pull forces had to be measured at different vertical positions on the control stick, and were therefore normalized as illustrated in appendix A, figure 10. Control stick displacements were measured with a ruler during ground testing, as the extremely small displacements made inflight measurement infeasible. Rudder pedal forces and displacements were estimated. Data was recorded on pilot kneeboard cards, portable voice tape recorders and an installed production airborne video tape recorder capable of recording information presented on the Head Up Display (HUD) or the Radar Electro-Optical Display (REO). All data was reduced by hand.

3.3 Control System Mechanical Characteristics

Mechanical characteristics of the control system were quantitatively evaluated during ground testing and verified inflight. Plots of control displacement vs. control force for all three axes are presented as appendix E, figure 1. Mechanical characteristics are summarized in table XIV.

Table XIV
MECHANICAL CHARACTERISTICS

Control Axis (Direction)	Breakout+Friction (lb)			Maximum Deflection	
	Measured	Specification		Displacement (in)	Force Req. (lb)
		Max	Min		
Longitud. (PULL)	1 3/4	3	1/2	1/4	20
Longitud. (PUSH)	1 3/4	3	1/2	1/8	19
Lateral (LEFT)	1	2	1/2	7/32	11
Lateral (RIGHT)	1	2	1/2	1/4	14
Direction. (EITHER)	10	7	1	5/8	100

(1) Specification column refers to MIL-F-8785C, Flying Qualities of Piloted Airplanes, 5 November 1980. While specification compliance was not the intent of this thesis, the above limits were considered enlightening for purposes of comparison and are therefore presented.

All cockpit controls were force sensing, minimum displacement units as discussed in section 2.1. Control centering for all three axes was positive and absolute, position freeplay was negligible and no oscillations were observed after a sharp input to any of the controls. Rudder breakout was qualitatively assessed as slightly high but not objectionable. Apparent impact of the minimum displacement type controls on high gain mission tasks is discussed in paragraphs addressing those tasks.

3.4 Longitudinal Flying Qualities

3.4.1 Dynamic Longitudinal Stability

3.4.1.1 Configuration CRUISE long period. Long period (phugoid) characteristics in configuration CR were qualitatively evaluated during maximum range cruise and loiter flight phases. Quantitative evaluation was also performed by observing the controls-free airplane response following a 15 KIAS deceleration from trimmed flight speeds of 250 and 120 KIAS at 15,000 ft. The airplane exhibited no long period oscillation when trimmed for airspeeds that corresponded to angles of attack (AOA) less than 15 degrees. When trimmed for 120 KIAS/19 degrees AOA, the airplane exhibited a slightly divergent long period oscillation with a natural frequency of 0.18 rad/sec and a predicted time to double amplitude of approximately 15 minutes. A graphical depiction is presented as appendix E, figure 2. No pilot suppression of the long period was required during mission tasking. Lack of a long period oscillation at AOAs below 15 degrees was apparently the result of the FLCS maintaining the trim load factor vice the trim angle of attack (as would a conventional control system) following control release. The long period oscillation that occurs at high AOA will not have substantial mission impact, as the flight conditions under which it could be excited will occur only during slow speed air combat, the dynamics of which will preclude a sustained phugoid oscillation. Configuration CR long period characteristics are therefore satisfactory.

3.4.1.2 Configuration POWER APPROACH long period. Long period (phugoid) characteristics in configuration PA were evaluated qualitatively during simulated instrument approach and quantitatively by observing the controls-free airplane response following a 15 KIAS deceleration from a trimmed flight speed of 135 KIAS/11 deg AOA at 7,000 ft. Control release following deceleration resulted in a smooth nose down pitch rotation, acceleration to airspeeds in excess of trim, and stabilization at a more nose down pitch attitude, faster airspeed and lower AOA than trim. No pilot suppression of long period oscillation was required during normal mission tasking. AOA feedback in the FLCS was apparently not sufficiently strong to maintain AOA as the aircraft accelerated, and as the AOA dropped below 10 degrees the FLCS apparently reverted to a pitch rate command system with no tendency to return the airplane to trim airspeed or AOA. Long period oscillation will not present a problem to the pilot executing an instrument approach. Long period characteristics in configuration PA are therefore satisfactory.

3.4.1.3 Short period. Short period response was evaluated qualitatively during air-to-air and air-to-ground tracking, simulated instrument approach and landing flare. Quantitative evaluation was performed at the flight conditions presented in table XV.

Table XV
LONGITUDINAL SHORT PERIOD TEST POINTS

Configuration	Pressure Altitude	Indicated Airspeed
Cruise	41,000 ft	0.85 IMN
	20,000 ft	245 KIAS
	15,000 ft	405 KIAS
	3,000 ft	420 KIAS
Power	15,000 ft	510 KIAS
Combat	15,000 ft	710 KIAS
Power Approach	8,000 ft	136 KIAS

Response to pitch doublets was essentially deadbeat at all flight conditions tested. Initial pitch response was rapid when the pilot aggressively repositioned the nose during air-to-ground tracking. Pitch attitude adjustments during landing flare could be expeditiously accomplished with no requirement to over-drive the longitudinal control. Longitudinal short period characteristics are satisfactory.

3.4.2 Non-Maneuvering Longitudinal Stability

3.4.2.1 Configuration CRUISE static stability.

Longitudinal static stability, as indicated by the variation of longitudinal stick force with airspeed in unaccelerated flight, was evaluated in configuration CR by stabilizing at various airspeeds at 15,000 ft pressure altitude. Qualitative evaluation was performed during acceleration runs and simulated mission tasking. No variation in steady state stick force was observed during level acceleration from 170 to 700 KIAS or during air-to-ground weapons delivery dive acceleration from 200 to 450 KIAS. Stabilized point data was taken during a deceleration from 280 to 100 KIAS and is presented as appendix E, figure 3. All tests indicated neutral static non-maneuvering stability for angles of attack below 15 degrees, presumably a result of the intended "g command" pitch control laws effected by the FLCS. Positive non-maneuvering static stability was indicated by the requirement for increasing aft stick force to stabilize at airspeeds slower than 153 KIAS (15 degrees AOA). Stick force gradient for these slower speeds was steep, resulting in 16 lb aft stick force required to maintain 100 KIAS/23-25 deg AOA. The neutral static stability through most of the envelope reduced pilot workload during dynamic tactical situations, but the absence of speed cues dictated frequent monitoring of the airspeed indicator when close airspeed control was required during cruise flight. The strike fighter pilot will not be required to retrim the airplane while tracking a target during dive acceleration, and will be warned of a high AOA situation during an air-to-air engagement by the requirement for increasing aft stick force. Longitudinal non-maneuvering static stability is therefore satisfactory.

3.4.2.2 Configuration POWER APPROACH static stability. Longitudinal static stability was evaluated in configuration PA by stabilizing at airspeeds between 110 and 172 KIAS at 9,000 ft after trimming for a normal approach speed of 135 KIAS/11 degrees AOA. Data are presented as appendix E, figure 3. Neutral static stability was indicated for speeds above 135 KIAS (10 - 11 degrees AOA), presumably a result of the intended "pitch rate command" control laws effected by the FLCS. Positive non-maneuvering static stability was indicated by the requirement for increasing aft stick force to stabilize at airspeeds slower than 135 KIAS (11 degrees AOA). Stick force gradient was moderately steep for the slower airspeeds, resulting in 14 lb aft stick force required to maintain 110 KIAS/18 deg AOA. The increasing aft stick force required to maintain altitude will alert the pilot to an unintentional deceleration that occurs during instrument approach; as such, longitudinal non-maneuvering static stability in configuration PA is satisfactory.

3.4.3 Maneuvering Characteristics

3.4.3.1 Maneuvering stability. Static maneuvering stability, as indicated by the requirement for increasing aft stick force to increase load factor, was evaluated in configurations P and PA during steady turns and symmetrical pushovers. Test conditions and data are presented as appendix E, figures 4 and 5, and are summarized in table XVI.

Table XVI
MANEUVERING LONGITUDINAL CONTROL FORCES

Config.	Airspeed/Altitude (KIAS/ft)	Stick Force per Load Factor (local gradient, lb/g)
P	400 / 15,000	PULL: 2.8 to 3.9 PUSH: 8.7 to 9.9
P	250 / 15,000	PULL: 3.5 to 4.0
PA	135 / 8,000	PULL: 15.3 to 17.8 PUSH: 11

Positive maneuvering stability was indicated at all flight conditions tested. Quantitative testing revealed a small stick force gradient change as load factor increased beyond two but the transition was not noticed by the pilot. Two test points performed at 250 KIAS confirmed that stick force gradient in configuration CR was essentially invariant with airspeed as intended by FLCS design. Stick force gradients required to command load factors less than one (trim load factor) were considerably higher than those for required for load factors greater than one, possibly due to the force measurement and normalization technique utilized (described in METHOD OF TEST, section 3.2). The strike fighter pilot will encounter no unpredictable stick force per g gradients during mission tasking; as such, the control forces in maneuvering flight in configurations P and PA are satisfactory.

3.4.3.2 Dynamic control forces. Dynamic maneuvering stability was evaluated qualitatively during simulated air combat maneuvering and quantitatively during sudden pullups in configuration P at 15,000 ft/400 KIAS. Test conditions and data are presented as appendix E, figure 4. Load factor was applied at an onset rate similar to that observed during stick pumping at the longitudinal short period frequency. Peak longitudinal stick forces required in the sudden pullups were 55 to 85% higher than those corresponding to the same load factor in a steady turn. Step input of large stick forces seemed to result in a step increase of load factor followed by a small ramp increase, which would be consistent with the intended proportional plus integral pitch control law. Simulated break turns did not result in unexpectedly high load factors. The fighter pilot will find the airplane pitch response to sudden stick inputs predictable. Dynamic control forces are therefore satisfactory.

3.4.3.3 Load factor onset rate. Load factor onset rate was qualitatively evaluated for g induced loss of consciousness potential in configuration P at 15,000 ft, 0.85 and 0.93 IMN. Following a full aft stick input at either airspeed, load factor increased to approximately seven in less than one second (estimated) followed by a continued increase to nine in less than two seconds. The evaluation pilot experienced "tunnel vision" during the test despite straining maneuvers performed prior to and during the turns. During simulated air combat maneuvering, there was no requirement to increase load factor gradually when initiating a hard turn since the FLCS g limiter prevented airframe overstress. The extremely high load factor onset rate available was within the range documented as presenting a high probability of g induced loss of consciousness for periods in excess of twenty seconds.[11] No warning regarding g induced loss of consciousness is included in the F-16B Flight Manual [7]. No loss of consciousness warning or terrain avoidance systems were incorporated in the airplane. When he encounters a hostile airplane in a low altitude, high speed scenario the fighter pilot will command a full aft stick turn, lose consciousness and fail to regain situational awareness in time to prevent the airplane from impacting the ground. Analysis and recommendations regarding this problem are discussed in Chapter IV.

3.4.4 Longitudinal Trimmability

Longitudinal trimmability was evaluated qualitatively during all mission tasks in configurations CR, D, P and PA. No long term drift of longitudinal trim was noted. Trim authority was sufficient to quickly reduce level flight

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[11] U.S. Naval Flight Surgeon's Manual, 2nd Edition, 1978, U.S. Government Printing Office, Chapter 2.

control forces to zero during all test events. Longitudinal trim was overly sensitive at all airspeeds, however, as the smallest trim increment that could be commanded often resulted in an overshoot. Since the longitudinal trim was designed as a "g command" system, trim inputs were not required with speed change, and were primarily used as a fine adjustment of hands-off normal acceleration. As a result, only very small longitudinal trim changes were required, making the trim rate seem inappropriately high. Workload will be increased during all cruise mission phases when the longitudinal trim sensitivity causes the pilot to make several trim inputs to accurately accomplish a small trim adjustment. The excessive sensitivity of the longitudinal trim system is a nuisance which could be corrected by slowing the trim rate.

3.4.5 Longitudinal Control During Takeoff and Landing

3.4.5.1 Bounce tendencies following touchdown.

Longitudinal characteristics subsequent to touchdown were evaluated in configuration L during eight touch-and-go and four full stop landings on Runway 22. Winds ranged from calm to 11 knots and were never more than 10 degrees off runway heading. The airplane displayed a severe tendency to bounce following touchdown, even at rates of descent less than 100 fpm. Landing gear characteristics appeared to be at least partially responsible, as light extension damping and short compression travel of the struts made the landing gear seem "springy". Additionally, substantial excess lift was available at touchdown due to a high landing speed (45 KIAS above stall) which was the result of a 13 degree angle of attack limit[12] imposed to prevent runway contact of the lower empennage. Eliminating the bounce required that rate of descent be meticulously minimized prior to touchdown (discussed in section 3.4.5.2), followed by slight forward stick pressure upon runway contact. The replacement student transitioning to the F-16 will require extra dual flights in the two seat model while he is instructed in the non-standard landing technique required to avoid bouncing after touchdown, making training more expensive and time consuming.

3.4.5.2 Flight path control during landing flare. Flight path control was evaluated in configuration L during eight touch-and-go and four full stop landings on Runway 22. Winds ranged from calm to 11 knots and were never more than 10 degrees off runway heading. Maintaining a sink rate sufficiently small to prevent a bounce (section 3.4.5.1) was extremely difficult, requiring rapid (3 per second) moderate (5 lb) push and pull longitudinal stick inputs (HQR-6, appendix D). A PIO of +/- 2 degrees pitch attitude often occurred as the pilot attempted to establish

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[12] Flight Manual, USAF Series F-16A/B, T.O. 1F-16A-1. Fort Worth: General Dynamics, 1988.

the extremely shallow flight path vector required. Deviation from the desired touchdown point averaged 500 to 1000 ft for initial landings, decreasing to 300 ft for the last two landings. Stick force appeared to primarily command a pitch rate, which provided positive control of pitch attitude but made precise flight path control difficult. Analysis of the PIO tendencies of this airplane are presented in Chapter IV. The student pilot will require extra sorties in the replacement training unit to acquire acceptable landing proficiency and touchdown predictability.

3.4.5.3 Pitch attitude control during takeoff and aerodynamic braking. Pitch attitude control was evaluated during four takeoffs, two each in configurations TO1 and TO2, and four roll outs using aerodynamic braking in configuration L. Maximum wind was 12 kt with crosswind component never exceeding 2 kt. During three of the takeoff tests, a longitudinal pilot induced oscillation (PIO) of +/- 1 deg occurred for two cycles when the pilot attempted to set the takeoff attitude. A similar PIO occurred when the pilot attempted to adjust the aerodynamic braking attitude during roll out. PIO tendencies of this airplane are analyzed in Chapter IV. The concentration required to suppress the PIO will increase pilot workload during the already intensive takeoff and landing phases.

3.4.6 Longitudinal Control During Air-to-Air Refueling

Longitudinal control was qualitatively evaluated during air-to-air refueling in configuration IFR at 23,000 ft/270 KIAS and formation flight in configuration cruise at 23,000 ft, 230 to 280 KIAS. A longitudinal PIO often occurred during close formation flight, particularly when small stepdown corrections were initiated. The oscillation had a frequency of 1 1/2 hz, a peak magnitude of +/- 2 deg, damped to zero in an average of three cycles, and was accompanied by peak stick forces estimated at 5 lb (push and pull). The PIO could be suppressed if the pilot worked hard at making small, smooth inputs. Maintaining stepdown +/- 2 ft during air-to-air refueling was fairly difficult, requiring frequent (2/sec) small (3 lb) smooth stick inputs (HQR-4, appendix D). On a long range attack mission, the attack pilot will be prematurely fatigued by the concentration required to perform his enroute refueling. PIO tendencies of this airplane are analyzed in Chapter IV.

3.5 Lateral-Directional Flying Qualities

3.5.1 Dynamic Lateral-Directional Stability

3.5.1.1 Dutch roll mode.

3.5.1.1 Configurations CRUISE, POWER and COMBAT Dutch roll. Dutch roll characteristics were evaluated by observing airplane response to rudder doublets at the test points presented in table XVII.

Table XVII
DUTCH ROLL TEST POINTS

Configuration	Pressure Altitude	Indicated Airspeed
Cruise	40,700 ft	0.85 IMN
	20,200 ft	245 KIAS
	15,100 ft	405 KIAS
	13,900 ft	125 KIAS
	2,900 ft	420 KIAS
Power	15,000 ft	505 KIAS
Combat	15,000 ft	710 KIAS

Qualitative evaluation was also performed during air-to-ground weapons delivery. Dutch roll overshoots were observed only at angles of attack greater than 16 degrees. A rudder doublet performed at 125 KIAS/18 deg AOA resulted in a Dutch roll with a damping ratio of 0.4, a natural frequency of 1 3/4 rad/sec, and a roll to yaw ratio of 3 as presented in appendix B, table III. The Dutch roll was deadbeat at all other flight conditions tested, with no overshoots observed following rudder doublets. Relatively large sideslips (greater than 5 degrees) were required to excite the high AOA Dutch roll, and a high roll to yaw ratio made suppression easy using lateral stick. During simulated air-to-ground tracking, rapid repositioning of the azimuth aim point by 30 mils using rudder inputs did not result in Dutch roll overshoots. The strike fighter pilot will encounter Dutch roll oscillation only during slow speed air combat maneuvering and will easily suppress it with a lateral stick input. Dutch roll characteristics in configurations CR, P and CO are therefore satisfactory.

3.5.1.1.2 Configuration POWER APPROACH Dutch roll. Dutch roll characteristics were evaluated quantitatively by observing airplane response to rudder doublets in configuration PA at 9,500 ft/137 KIAS. Qualitative evaluation was also performed in the visual landing pattern. Small sideslip excursions (less than half rudder, or an estimated five degrees of sideslip) did not excite the Dutch roll. Larger sideslip excursions did produce a Dutch roll with a damping ratio of 0.3, a natural frequency of 1 3/4 rad/sec, and a roll to yaw ratio of 3 as presented in appendix B, table III. High roll-to-yaw ratio and a relatively low frequency made pilot suppression of the Dutch roll oscillation easy. Step rudder inputs and releases made during final approach failed to excite the Dutch roll which consequently did not complicate wing attitude or lineup control. The pilot will not be hampered in his efforts to control heading during instrument approach by Dutch roll oscillations. Dutch roll mode characteristics in configuration PA are therefore satisfactory.

3.5.1.2 Spiral mode. Spiral mode characteristics were evaluated at the test points presented in table XVIII.

Table XVIII
SPIRAL STABILITY TEST POINTS

Configuration	Pressure Altitude	Indicated Airspeed
Cruise	41,000 ft	0.85 IMN
	20,000 ft	245 KIAS
	15,000 ft	405 KIAS
	3,000 ft	420 KIAS
Power	15,000 ft	505 KIAS
Combat	15,000 ft	710 KIAS
Power Approach	10,000 ft	138 KIAS

Spiral mode characteristics were evaluated by releasing controls after utilizing rudder inputs to establish the desired bank angle. Roll rates following control release at angles of bank up to 45 degrees were almost imperceptible, indicating essentially neutral spiral stability for all flight conditions tested. The neutral spiral mode is presumably the result of the control laws which were intended to effect a roll rate command system. No uncommanded roll rates will develop when the pilot is distracted during an instrument approach; as such, spiral mode characteristics are satisfactory.

3.5.1.3 Roll mode.

3.5.1.3.1 Configuration CRUISE Roll mode. Roll mode characteristics were quantitatively evaluated by measuring the time for a steady state roll rate to decay to zero following control stick release. Quantitative evaluation was attempted at the test points listed in table XIX.

Table XIX
ROLL MODE TEST POINTS

Configuration	Pressure Altitude	Indicated Airspeed
Cruise	41,200 ft	0.85 IMN
	20,300 ft	252 KIAS
	15,000 ft	405 KIAS
	3,100 ft	420 KIAS
Power	14,200 ft	515 KIAS
Combat	15,000 ft	705 KIAS

Qualitative evaluation was also performed during simulated air combat maneuvering. Roll rates appeared to reach their steady state values rapidly at all flight conditions tested, and roll rate decay following stick release was even more rapid and characterized as almost instantaneous. Roll rate decay times were too short to measure accurately, but appeared to occur in less than 3/4 sec predicting a roll mode time constant of less than 1/4 sec. Some tendency for roll "ratcheting" was apparent at the 15,000 ft/705 KIAS point but it was not of sufficient severity to impact mission effectiveness. The extremely rapid roll rate decay allowed the pilot to precisely capture his desired angle of bank without the need for anticipation or opposite roll control deflection. Roll mode characteristics in configurations CR, P and CO are therefore satisfactory.

<u>3.5.1.3.2 Configuration POWER APPROACH Roll mode.</u> Roll mode characteristics were quantitatively evaluated in configuration PA at 9,000 ft/135 KIAS using the method described in the previous paragraph. Qualitative evaluation was performed in the visual landing pattern. Decay times for various rates of steady state roll were all between 1.2 and 1.5 sec, predicting a roll mode time constant of 0.4 to 0.5 seconds. Effecting a rapid line-up correction on final approach after an intentional offset was not complicated by delays in

establishing or terminating aggressive roll rates; as such, roll mode characteristics in configuration PA are satisfactory.

3.5.2 Static Lateral-Directional Stability

Static lateral-directional stability characteristics were evaluated during steady heading sideslips. Qualitative evaluations were performed only to observe gross non-linearities or trend reversals, while quantitative evaluation included recording complete data at half and full rudder pedal input in both directions. Flight conditions and corresponding evaluation types are specified in table XX.

Table XX
STEADY HEADING SIDE SLIP TEST POINTS

Configuration	Pressure Altitude	Indicated Airspeed	Evaluation Type
CR	41,000 ft	0.85 IMN	Qualitative
	20,000 ft	250 KIAS	QUANTITATIVE
	15,000 ft	405 KIAS	Qualitative
		150 KIAS	Qualitative
		130 KIAS	Qualitative
		110 KIAS	Qualitative
	3,000 ft	423 KIAS	Qualitative
P	15,000 ft	505 KIAS	Qualitative
PA	10,000 ft	137 KIAS	QUANTITATIVE

Complete test conditions and steady heading side slip data are presented as appendix E, figures 6 and 7.

3.5.2.1 Directional stability. In addition to steady heading sideslips, directional stability was qualitatively evaluated during simulated air combat maneuvering and instrument approach. Positive apparent directional stability and essentially linear rudder pedal force gradients were observed in all configurations tested. Simulated air-to-air gunsight tracking in configuration P at airspeeds between 250 and 400 KIAS was not hampered by inadvertent sideslip. During simulated instrument approaches in configuration PA, the airplane immediately returned to heading following small sideslip perturbations introduced by rudder inputs. Static directional stability is satisfactory.

3.5.2.2 Dihedral effect. In addition to steady heading sideslips, dihedral effect was qualitatively and quantitatively evaluated at various flight conditions during rudder only rolls initiated from wings level flight. Test conditions and data for the quantitative rudder only turns are presented in table XXI.

Table XXI
RUDDER-ONLY ROLL RESPONSE

Config.	Pressure Altitude (ft)	Indicated Airspeed (KIAS)	Sideslip Produced (deg)	Time to Roll 0 to 45 deg (sec)
CR	15,000	400	8	4
		250	10	2
PA	9,000	135	9	2

Full rudder inputs in both configurations initially displayed a quick (1/2 sec) small (1 deg) roll opposite the rudder deflection due to sideforces on the rudder, followed by a slow roll in the direction of rudder deflection as sideslip developed indicating positive apparent dihedral effect. Positive dihedral effect was indicated during steady heading sideslips by the requirement for lateral control deflection in the direction of sideslip as presented in appendix E, figures 6 and 7. During rudder-only rolls, the FLCS appeared to be opposing (but not overcoming) the airplane's aerodynamic tendency for roll opposite the sideslip, which was consistent with the intended lateral control law design of a roll rate command system. Dihedral effect characteristics in configurations CR and PA are satisfactory.

<u>3.5.2.3 Sideforce characteristics.</u> Sideforce characteristics were qualitatively evaluated during low level ingress at 500 ft AGL/420 KIAS and final landing approach in addition to quantitative evaluation during steady heading sideslips. Steady heading sideslip data are presented in appendix E, figures 6 and 7 and are summarized in table XXII.

Table XXII
STEADY HEADING SIDESLIP

Configuration	Sideslip Direction	Maximum Sideslip (deg)	Required Angle of Bank (deg)
Cruise	Left	9	25
	Right	10	26
Power Approach	Left	9	11
	Right	10	12

Sideforce characteristics, as indicated by bank angle change variation with sideslip at steady heading, were positive in all configurations tested. Ride quality during low level ingress was not adversely affected by excessive side forces, but minimal turbulence was encountered (winds 10 KT). During final approach, sufficient sideforce was available to easily effect lineup corrections of +/- 10 ft with wings level using rudder inputs. Sideforce characteristics are satisfactory.

3.5.3 Lateral Control Response

3.5.3.1 Lateral control sensitivity. Lateral control sensitivity was evaluated qualitatively during mission tasking and quantitatively during timed rolls in configurations CR and PA at various flight conditions. Complete test conditions and average lateral sensitivity data are presented as appendix E, figure 8. The highest sensitivity observed was 11 deg/sec-lb, which was not excessively high. Lateral sensitivity in configuration PA was satisfactory and did not complicate bank angle control in the landing pattern. After the initial takeoff, lateral response in configuration P was abrupt, probably due to the short roll mode time constant and minimal position feedback from the minimum displacement control stick. Familiarity with the control system, which was quickly acquired, resulted in smoother roll response which was considered responsive but not overly sensitive or abruptly harsh. Lateral control sensitivity was therefore judged to be satisfactory.

3.5.3.2 Roll control effectiveness.

3.5.3.2.1 Configuration CRUISE roll control effectiveness. Roll control effectiveness was evaluated qualitatively during simulated air combat maneuvering and quantitatively during timed maximum input rolls in configuration CR at various flight conditions. No rudder coordination was used. Test conditions and data are presented in appendix E, figure 9 and are summarized in tables XXIII and XXIV.

Table XXIII
CONFIGURATION CR ROLL CONTROL EFFECTIVENESS
LEVEL FLIGHT

Airspeed (KIAS)	Press. Alt. (ft)	Time to Roll 360 deg (sec)
510	15,000	2.3
400	15,000	1.8
255	41,000	2.1
250	15,000	2.2
140	15,000	5.2
125	15,000	6.0
100	15,000	7.0

Table XXIV
CONFIGURATION CR ROLL CONTROL EFFECTIVENESS
ACCELERATED FLIGHT

Airspeed (KIAS)	Press. Alt. (ft)	Load Factor (Nz)	Time to Roll 360 deg (sec)
400	15,000	4.7	1.8
400	15,000	3.0	1.9
400	15,000	1.0	2.0

No roll rate oscillations were observed. Roll control effectiveness was essentially invariant with load factor and airspeed when airspeed was above 200 KIAS. Maximum commanded roll rates reduced sharply as airspeed was reduced and angle of attack was increased, presumably due to high AOA roll rate limiting by the FLCS. Roll rates available below 150 KIAS were slow and resulted in nosedown attitudes of 15 degrees during a 130 KIAS roll and 30 degrees during a 100 KIAS roll as the angle of bank passed 180 degrees. Since his adversary will be able to reverse more rapidly and gain an advantage during slow speed air combat maneuvering, the fighter pilot will have to avoid that portion of the flight envelope, restricting his tactical options. The slow roll rates in configuration CR are analyzed in Chapter IV.

3.5.3.2.2 Configuration POWER APPROACH roll control effectiveness. Roll control effectiveness in configuration PA was evaluated qualitatively in the visual landing pattern with maximum winds of 12 KT and crosswind component never exceeding 2 KT. Quantitative evaluation was performed during full stick input rolls to 30 degrees angle of bank at 9,000 ft/140 KIAS with a gross weight of 18,700 lb and a C.G. of 33.7% MAC. No rudder coordination was used. Time from control initiation to 30 deg angle of bank was between 1.0 and 1.1 seconds for full input rolls in either direction. Roll rates were sufficiently high to allow the expeditious bank angle changes required to make aggressive line-up corrections during final approach. Roll control effectiveness in configuration PA is satisfactory.

3.5.3.4 Aileron yaw. Yaw resulting from lateral control input was evaluated in configuration D during air-to-ground weapons delivery, configuration PA during final approach, and configuration CR and PA during half and full deflection aileron-only rolls at various airspeeds. A very small, rapid adverse yaw of one degree maximum was sometimes noticed during aileron-only rolls immediately following lateral control input. There was no requirement for rudder inputs to coordinate aggressive rolls in any configuration. The precise directional control required during air-to-ground weapons delivery was not complicated by aileron yaw when tracking solution was corrected using lateral control inputs. Heading control during final approach was not degraded by aileron yaw after an aggressive correction was made at 1/2 nm following a 400 ft line-up offset. The absence of significant aileron yaw effects was presumably due to the aileron-rudder interconnect, which is designed to provide automatic turn coordination. The absence of undesirable effects such as aileron yaw is discussed in Chapter IV.

3.5.3.3 Roll coupling. Roll coupling tendencies were evaluated qualitatively during simulated air combat maneuvering at 15,000 ft/250 to 350 KIAS. Quantitative testing was performed during full deflection 360 degree rolls in configuration CR, loading designation A, 15,000 ft/400 KIAS at a gross weight of 21,700 and C.G. of 36.4 % MAC. Data are presented in table XXV.

Table XXV
ROLL COUPLING

Load Factor (Nz)			
Start	Minimum	Maximum	Maximum Excursion
1.0	0.7	1.0	-0.3
3.0	2.5	3.1	-0.5
4.7	4.1	4.7	-0.6

Full deflection rolls displayed little roll to pitch coupling, with small load factor decreases exhibited at all test points. Significant load factor changes did not occur during rapid reversals at moderate load factors (2 to 4) during simulated air combat engagements. The fighter pilot will not encounter uncommanded high g flight during tactical rolling maneuvers. Roll to pitch coupling characteristics in configuration CR are satisfactory. The absence of undesirable coupling effects in this airplane is discussed further in Chapter IV.

3.5.4 Directional Control Response

3.5.4.1 Directional control sensitivity. Directional control sensitivity was qualitatively evaluated during air-to-ground weapons delivery in configuration CR and during final approach in configuration PA. Pedal forces were moderate and directional control was not overly sensitive when azimuth tracking solution was adjusted during simulated strafing attacks. Directional control forces were not objectionable when fine adjustments of line-up were made during final approach using rudder inputs; as such, directional control sensitivity is considered satisfactory.

3.5.4.2 Directional control effectiveness.

3.5.4.2.1 Configuration CRUISE directional control effectiveness. Directional control effectiveness in configuration CR was evaluated qualitatively during air-to-ground weapons delivery. Quantitative assessment of side slip available was performed at 15,000 ft during steady heading side slips at various airspeeds. Test conditions and data are presented in appendix E, figure 10 and are summarized in table XXVI.

Table XXVI
SIDE SLIP AVAILABLE
(Full Rudder Pedal Input)

Airspeed (KIAS)	Angle of Attack (deg)	Side Slip Available (deg)
250	4	10
150	15	7
130	17	6
115	20	4
110	23 - 25	0

Side slip available with full rudder pedal input decreased with airspeed after decelerating below approximately 150 KIAS (15 degrees AOA), with no directional reaction observed in response to inputs at 110 KIAS (23-25 deg AOA). The execution of mission tasks was not impeded by the side slip limiting, and directional control effectiveness was sufficient to adjust azimuth aim point +/- 50 mils during simulated strafing runs. The reduction in directional control effectiveness at low airspeeds was presumably the result of the FLCS yaw command laws which were designed to decrease pedal input gains at high angles of attack. Directional control effectiveness in configuration CR is satisfactory.

3.5.4.2.2 Configuration POWER APPROACH directional control effectiveness. Directional control effectiveness was quantitatively evaluated in configuration PA (loading designation A) during steady heading side slips at 10,000 ft/137 KIAS. Qualitative evaluation was performed during final approach. Full rudder pedal input resulted in a sideslip of approximately 10 degrees. Rudder inputs were not normally required during the approach phase, but sufficient directional control was available to generate the side slip required to make small wings level line-up corrections on final. Directional control effectiveness in configuration PA is satisfactory.

3.5.5 Lateral Trimmability

Lateral trimmability was qualitatively evaluated for authority, ease of accomplishment and long term holding during all mission phases in configurations CR, D, P and PA. Maximum external store asymmetry was 900 lb-ft which occurred during air- to-ground weapon delivery (loading designation B with 5 BDU-33s on station 7 and 2 BDU-33s on station 3). Only very small trim changes were required during the mission profile, and trim authority was sufficient to reduce control forces to zero during all test events. No long term drift of the lateral trim system was observed. No lateral trim changes were noted subsequent to separation of BDU-33 practice bombs. Trim rate was satisfactory. Sensitivity was good, as small trim changes could be easily accomplished without overshoot; lateral trimmability is therefore satisfactory.

3.5.6 Directional Trimmability

Directional trimmability was qualitatively evaluated for authority, ease of accomplishment and long term holding during all mission phases in configurations CR, D, P and PA. No changes of directional trim were noted subsequent to separation of BDU-33 practice bombs. Out-of-trim conditions were made apparent by balance ball excursion. Trim authority was sufficient to center the balance ball in all configurations tested. Directional trim changes required during speed or configuration changes were small and easily accomplished. Directional trimmability is satisfactory.

3.5.7 Lateral-Directional Control During Mission Tasks

3.5.7.1 Heading control during roll out. Heading control during angle of bank change was evaluated in configuration PA during simulated instrument approach at 135 KIAS/4,000 ft. During attempts to aggressively capture heading from a 30 degree angle of bank turn, heading slewed two degrees opposite the direction of the turn during rollout, requiring the pilot to overshoot the desired heading slightly prior to initiation of the roll out command. The heading slew was presumably the geometric result of airplane roll about the stability axis, which would be predicted by the design of the FLCS and is intended to reduce kinematic coupling. The heading slew did not significantly complicate heading control while normal corrections were made during instrument approach.

3.5.7.2 Lateral control during formation flight. Lateral control was qualitatively evaluated during formation flight in configuration CR at 23,000 ft, 240 to 280 KIAS with a well defined visual horizon. A lateral pilot induced oscillation (PIO) often occurred when attempting to maintain a close, steady formation position when the lead aircraft rolled in and out of turns. The oscillation had a frequency of approximately 1 1/2 hz, a peak magnitude of +/- 3 deg angle of bank, persisted for between 1 and 3 cycles, and was accompanied by peak lateral stick forces estimated at 4 lb. The PIO could be suppressed if the pilot worked hard at making smooth, small inputs. The PIO appeared to be the result of the control stick (a minimum displacement, force sensing type) which supplied little feedback to the pilot regarding the size of the control inputs being made. During rendezvous of the strike group, the attack pilot will be prematurely fatigued by the concentration level required to maintain formation position. PIO tendencies of this airplane are analyzed in

Chapter IV. Formation flight in configuration PA and/or during instrument meteorological conditions was not attempted during this evaluation but could result in more serious PIO tendencies.

3.5.7.3 Air-to-ground tracking. Lateral-directional pilot induced oscillation tendencies were qualitatively evaluated during air-to-ground weapons delivery in configuration D (loading designation B) at airspeeds between 300 and 450 KIAS, altitudes between 12,000 and 3,500 ft, and weapons delivery modes of Manual and Continuously Computed Impact Point (CCIP). Target area winds were variable at less than 10 kt. Pipper or bomb fall line tracking was easily maintained once established with no tendency toward oscillation. Corrections of azimuth aim point or run-in track tended to be slightly undershooting, with the remainder of the correction easily accomplished using a second correction of smaller magnitude. After an inbound track was established 4 degrees right of the desired run-in line, effecting the correction required to establish the target within 2 mils of the bomb fall line was relatively easy, requiring a 30 deg left angle of bank, a rollout, a 10 degree left angle of bank, and one final rollout all of which were accomplished in 4 seconds (HQR-3, appendix D). Required bank angles were expeditiously established without overshoot. The relatively low concentration required to establish and maintain a weapon delivery track will allow the attack pilot to maintain a visual scan of the area surrounding the target, preserving his threat awareness and survivability. Air-to-ground tracking characteristics are satisfactory.

3.5.7.4 Lateral control during aerodynamic braking. Lateral control during aerodynamic braking was evaluated during four full stop landings in configuration L. Maximum wind was 12 KT with crosswind component never exceeding 2 KT. Detailed test conditions are presented in appendix F, table IX. Small bank angle oscillations occurred during the aerobraking rollout, and intense pilot concentration was required to maintain smooth bank angle control. Lateral control effectiveness was always satisfactory, but establishing the proper control input was difficult, as addressed in the Chapter IV discussion of PIO. Pilot workload during the aerobraking evolution will be increased by the concentration required to maintain his bank attitude. High crosswinds, which were not experienced during this evaluation, could further complicate bank angle control during rollout and make this a more serious deficiency.

3.6 Transient Trim Changes

Transient trim changes were evaluated during the configuration changes listed in table XXVII.

Table XXVII
CONFIGURATION CHANGE TEST POINTS

Configuration	Pressure Altitude (ft)	Airspeed (KIAS)
TO1 to P	2,500	150
TO2 to CO	2,500	150
CR to IFR (1)	27,000	260
CR to DEC (1)	12,000 10,000	190, 280 450
CR to PA (1)	10,000 4,000	260, 175 220
PA to L (1)	3,200	165

(1) The reverse transition was also performed.

A 5 lb aft stick force was required to maintain attitude during the CR to DEC transition at 450 KIAS. No control inputs on any axis were required during any of the other transitions, during which altitude was maintained. The absence of trim transients during configuration changes decreased pilot workload during critical evolutions, and is therefore considered an enhancing characteristic.

3.7 Transonic Characteristics

Transonic characteristics were qualitatively evaluated during a one g acceleration from 0.8 to 1.3 IMN in configuration CO and during a three g deceleration from 1.1 to 0.8 IMN in configuration CR at 15,000 ft. Gross weight averaged 18,900 lb and C.G. averaged 33.9% MAC. Transonic acceleration was brisk. No pitch trim change was noticed as the transonic speed region was transitted in either direction, as load factors remained close to one and three g for the acceleration and deceleration, respectively. No indicated airspeed error transient was noted, and the altimeter displayed an increase in indicated altitude of 600 feet during transonic acceleration and a corresponding decrease during the deceleration. Transonic trim changes will not distract the fighter pilot from tactical decision making as he accelerates while egressing from an engagement. Transonic characteristics are satisfactory.

3.8 Stall Characteristics

3.8.1 Configuration CRUISE Stall

3.8.1.1 Level flight configuration CR stall.

Stall characteristics were evaluated in configuration CR at 15,000 ft with a trim speed of 200 KIAS, a gross weight of 20,100 lb, and a C.G. of 34.8% MAC. Decelerations were performed at one g with pitch attitudes of 10 and 25 degrees nose up and decelerations of one and four KIAS/sec. The airplane did not exhibit a conventional aerodynamic stall, as maximum attainable angle of attack (AOA) was limited by the FLCS. During all decelerations, progressively increasing the AOA above 15 degrees (130 KIAS) resulted in the requirement for increasing aft stick force, a gradual decrease of rudder effectiveness, and increasing airframe buffet that started at 20 degrees AOA (120 KIAS). When 25 to 26 degrees AOA was intercepted (108 KIAS), a slow nose down pitch rotation occurred and no subsequent AOA increase was possible. Flight at the maximum attainable AOA was characterized by moderate airframe buffet, satisfactory lateral control authority, satisfactory nose down longitudinal control authority, zero nose up longitudinal authority, zero directional control authority, and aft stick force estimated at 18 lb. Rate of descent and pitch attitude at maximum AOA were dependent on power setting, and positive rates of climb could be achieved at military power. Angle of attack could be reduced by releasing back stick pressure, resulting in the return of normal control authority as the AOA decreased. No tendency to depart controlled flight was observed. The lack of rudder control authority at maximum AOA seemed abnormal but did not adversely affect aircraft control. During slow speed air combat, the fighter pilot will be made aware of his high angle of attack situation by airframe buffet and the requirement for increasing aft stick force, and will encounter no dangerous flight conditions if he continues to decelerate. Level flight stall characteristics in configuration CR are satisfactory.

3.8.1.2 Accelerated flight configuration CR stall. Accelerated stall characteristics were evaluated in configuration CR at 15,000 ft with a trim speed of 250 KIAS, a gross weight of 19,900 lb, and a C.G. of 34.7% MAC. Deceleration was performed in a three g turn at a rate of approximately 4 KIAS/sec. Characteristics with increasing angle of attack were similar to those observed during level flight stall with higher airspeeds corresponding to the increased load factor. Buffet levels were moderately higher at all AOAs becoming heavy as limit AOA was approached. A small reduction in lateral authority was noted although authority remained satisfactory. Release of back stick pressure returned the aircraft to buffet free flight. Flight at maximum AOA was controllable and no tendency to depart controlled flight was encountered. When he initiates an aggressive target roll-in, the attack pilot will be made aware of a high AOA situation by airframe buffet and will encounter no dangerous flight conditions if he continues to pull the stick full aft; accelerated stall characteristics in configuration CR are therefore considered satisfactory.

3.6.2 Configuration POWER APPROACH Stall

3.6.2.1 Level flight configuration PA stall. Stall characteristics were evaluated in configuration PA at 7,000 ft with a trim speed of 130 KIAS, a gross weight of 18,400 lb, and a C.G. of 33.3% MAC. Decelerations were performed at one g with decelerations of one and four KIAS/sec. The airplane did not exhibit a conventional aerodynamic stall, as maximum attainable angle of attack (AOA) was limited by the FLCS. During all decelerations, progressively increasing the AOA above 13 degrees (130 KIAS) resulted in the requirement for increasing aft stick force and gradual decrease of rudder effectiveness. Airframe buffet started at 18 units (111 KIAS) and increased slightly with AOA. When 21 degrees AOA was intercepted (95 KIAS), a slow nose down pitch rotation occurred and no subsequent AOA increase was possible. Flight at the maximum attainable AOA was characterized by light to moderate airframe buffet, satisfactory lateral control authority, satisfactory nose down longitudinal control authority, zero nose up longitudinal authority, minimal directional control authority, and aft stick force estimated at 15 lb. Rate of descent and pitch attitude at maximum AOA were dependent on power setting, and a 1,000 fpm rate of climb could be achieved at military power. Angle of attack could be reduced by releasing back stick pressure, resulting in the return of normal control authority as the AOA decreased.

No tendency to depart controlled flight was observed. When he unintentionally decelerates during an instrument approach, the pilot will encounter no dangerous flight conditions and will be able to easily reduce his angle of attack and/or establish a positive rate of climb. Level flight stall characteristics in configuration PA are satisfactory.

3.6.2.2 Accelerated flight configuration PA stall. Accelerated stall characteristics were evaluated in configuration PA at 7,000 ft with a trim speed of 175 KIAS, a gross weight of 18,200 lb, and a C.G. of 33.0% MAC. Deceleration was performed in a two g turn at a rate of approximately 3 KIAS/sec. Characteristics with increasing angle of attack were similar to those observed during level flight stall with higher airspeeds corresponding to the increased load factor. The 21 degree AOA limit was intercepted at 130 KIAS. Buffet levels were moderately higher at all AOAs. Release of back stick pressure returned the aircraft to buffet free flight. Flight at maximum AOA was controllable and no tendency to depart controlled flight was encountered. When he initiates an overly aggressive approach turn, the pilot will encounter no dangerous flight conditions and will be able to easily return the angle of attack to the normal range; resultantly, accelerated stall characteristics in configuration PA are considered satisfactory.

CHAPTER IV

DISCUSSION OF RESULTS

4.1 Advantages of Fly-by-Wire in the F-16

4.1.1 Tailoring of Stability Parameters

In a fly-by-wire flight control system, important stability parameters can be tailored to optimum values. For an example, let us examine airplane longitudinal short period characteristics (section 3.4.1.3). These characteristics (both damping and frequency) have long been considered crucial to mission effectiveness.[13] Appropriate short period characteristics allow the pilot to track a target more precisely, improving the accuracy of both air-to-air and air-to-ground ordnance delivery, as well as enabling the pilot to precisely control his flight path during the final phase of landing.[14] Natural short period characteristics change with parameters such as altitude, weight, mass distribution and, most significantly, airspeed (to which short period frequency can be shown to be

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[13] Berthe, C.; Smith, R. Variable Stability Programs; CALSPAN Corporation: Buffalo, 1986.

[14] Radford, C.; Smith, E. "Approach and Landing Flying Qualities Requirements," Flying Qualities Design Criteria (Proceedings of AFFDL Symposium). AFWAL-TR-80-3067, May 1980.

proportional[15]). With conventional control systems, designers normally endeavor to arrange mass distribution and aerodynamics to effect appropriate natural short period characteristics. Unfortunately these natural characteristics can, at best, be optimized for only one portion of the flight envelope, usually resulting in a less than optimum short period at other flight conditions. The pilot who is flying-by-wire perceives "apparent" short period characteristics rather than the natural characteristics of the airframe he is flying.[16] Since they are functions of the control laws effected by the fly-by-wire system, these "apparent" short period characteristics can be optimized for flight condition and, in the most sophisticated implementations, mission phase. Similar arguments can be made for many other parameters, including those that determine control sensitivity and effectiveness about all three axes.

The F-16's well behaved dynamic longitudinal stability (section 3.4.1), longitudinal maneuvering characteristics (section 3.4.3), roll mode response (3.5.1.3), and lateral control effectiveness (3.5.3.2) can all be directly attributed to properly designed control laws that implement

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[15] Moore, T.S. Airplane Dynamics; USNTPS: Patuxent River MD, 1984.

[16] Parrag, M.; Knotts, L. Augmented Aircraft Handling Qualities. Buffalo: CALSPAN Corp., 1987.

suitable stability parameters about the appropriate control axes throughout the flight envelope.

4.1.2 Elimination of Coupling Effects

Undesirable coupling effects can be minimized by a fly-by-wire flight control system. Since the flight control system can prevent the occurrence of uncommanded sideslip and can actively damp uncommanded oscillation, lateral-directional problems like Dutch roll and aileron yaw (sections 3.5.1.1 and 3.5.4) have been masked from the pilot. With its multitude of input parameters and substantial computational capability, a fly-by-wire system can perform the functions of an aileron-rudder interconnect in a much more elegant and effective manner. The change in load factor that sometimes occurs during rolls, or "roll coupling", has also been eliminated in the F-16 by properly designed control laws (section 3.5.3.5).

4.1.3 Pilot Workload Decrease

Pilot workload can be considerably reduced by a fly-by-wire flight control system. Probably the most significant example would be the elimination of the requirement for longitudinal control input during airspeed changes. Since a conventional control system trims to an angle-of-attack, an airspeed increase requires a nose down command input in order to maintain a constant load factor (assuming positive non-maneuvering longitudinal static stability).[17] With a fly-by-wire system incorporating pitch control laws that trim to a commanded load factor or pitch rate, stick force does not change with airspeed, and the pilot is relieved of the requirement for either a control input or longitudinal retrimming[18] (section 3.4.2.2). When performing bombing, a dive is commenced at a relatively slow airspeed which increases during the dive toward the target. Elimination of the requirement to retrim the airplane during the dive allows the pilot to more effectively concentrate on optimization of his tracking run and avoidance of enemy defenses.

Other examples of workload decreases implemented in the F-16 include completely neutral spiral stability (section 3.5.1.2) and the elimination of transonic pitch trim changes (section 3.7).

4.1.4 Performance Improvement

The incorporation of a fly by wire system can provide design avenues to improved airplane performance. With a conventional control system, airframes must be designed with positive static longitudinal stability to provide

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[17] Moore, T. Static Stability and Control. Patuxent River: USNTPS Notes, 1986.

[18] Parrag, M.; Knotts, L. Augmented Aircraft Handling Qualities. Buffalo: CALSPAN Corp., 1987.

satisfactory handling qualities.[19] Since a fly by wire system shows the pilot an "apparent" stability which can be much different than the natural stability of the airframe, airframes with negative natural stability become a design option. Lower values of trim drag can be attained in airframes with negative longitudinal static stability.[20] This decrease in trim drag extends maximum range and increases the maximum sustainable load factor. Negative static stability also improves the maneuverability of the F-16 by enabling the airframe to generate rapid pitch acceleration and high rates of load factor onset (section 3.4.3.3).

4.2 Problems with the F-16 Fly-by-Wire Implementation

4.2.1 Pilot Induced Oscillation (PIO)

4.2.1.1 Discussion. Analysis of the F-16 flight control system suggests the presence of a time delay of approximately 120 msec in the longitudinal control system and 200 msec in the lateral control system.[21] Since instrumentation was not available, reliable quantification

[19] Moore, T. Airplane Dynamics. Patuxent River: USNTPS Notes, rev. 1984.

[20] Powell, J. Airplane Performance: Fixed Wing Flight Mechanics. Patuxent River: USNTPS Notes, 1985.

[21] Dennison, K. Phase IIA Developmental Testing of the F-16B Airplane. NATC: Patuxent River MD, 1988.

of the delays could not be accomplished during the evaluation. However, response to longitudinal and lateral stick raps supported the existence of those delays and suggested that the above mentioned times were approximately correct. It has been empirically demonstrated that control system delays can seriously degrade flying qualities and cause PIO during high gain tasks and are a source of PIO in augmented flight control systems.[22] The effect of time delays on handling qualities ratings for longitudinal and lateral mission tasking is illustrated in appendix E, figures 11 and 12.

The second factor contributing to the PIO problem was the absence of stick displacement during control inputs. As described in section 3.3, the control stick and rudder pedals were force sensing units that displaced very little during control input. When executing high gain tasks, the pilot, as part of a feedback loop, is constantly making small control inputs, the magnitude and direction of each being based on the response received to previous inputs. With a conventional control stick configuration, immediate feedback regarding the magnitude of each input takes the form of control displacement and control force. Since his

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[22]Bailey, E; Smith, R. "Evaluating the Flying Qualities of Today's Fighter Aircraft," Design Criteria for the Future of Flight Controls (Proceedings of the Flight Dynamics Laboratory Flight Control Symposium 2-5 March 1982). AFWAL-TR-82-3064.

controls provide almost no displacement feedback, the F-
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pilot must rely on force as his only feedback from the controls themselves. Control force feedback is much less positive and precise than displacement feedback;[23] as a result, the magnitude, and sometimes even the existence, of his control input is not positively established in the pilot's mind until he observes airplane response.

The two factors discussed above, control response delay and the lack of control displacement, had the synergistic effect of inadequate control feedback to the pilot which manifested itself in a strong tendency toward pilot induced oscillation (PIO). Since feedback from the controls is minimal, the pilot's only good source of feedback is airplane response, which is delayed by both the control system (discussed above) and the natural response time of the airframe. Such feedback is sufficient when speed and accuracy are not critical, such as when initiating a turn without a specific target angle of bank. However, a pilot attempting a high gain task requires immediate feedback regarding the size of his control input.[24] The delay in airplane response feedback often

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[23]Ball, J. <u>USAF NT-33 Variable Stability Research Aircraft</u>. Buffalo: CALSPAN Corp., 1987.

[24]Parrag, M.; Knotts, L. <u>Augmented Aircraft Handling Qualities</u>. Buffalo: CALSPAN Corp., 1987.

causes the pilot to overdrive the control, since he does not immediately observe the desired response and therefore initiates a second input of greater magnitude. During the rapid iterative inputs required during high gain tasks, airplane response can easily be far out of phase with pilot inputs resulting in the immediate development of pilot induced oscillation (PIO). This insufficient control feedback was the major contributory cause of the problems with longitudinal PIO during inflight refueling (section 3.4.6), takeoff and aerobraking (section 3.4.5.2), and landing flare (section 3.4.5.3); it was also responsible for the lateral PIO experienced during close formation (section 3.5.7.2) and aerobraking (section 3.5.7.4).

4.2.1.2 Recommendations. The PIO problem that occurs during various mission tasks could be eliminated by implementing the following suggestions:

a) Minimize the control system delays in all control axes;

b) Incorporate a control stick that displaces when control inputs are made. The stick natural frequency, damping and force gradient should be typical of those on other modern fighters. Control system inputs could probably still be driven by stick force, and the force sensing character of the current system could be retained; stick displacement would be utilized only to improve feedback to the pilot regarding the magnitude of his control inputs. If further pilot feedback difficulties were encountered with the system suggested above, control inputs based on stick displacement could be incorporated, the required input signals being generated by position transducers on the stick or by modeling stick response in the flight control computer and computing stick position based on stick force.

4.2.2 G Induced Loss of Consciousness (g LOC)

4.2.2.1 Discussion. The high potential for g induced loss of consciousness is a direct result of this airplane's capability for high load factor onset rate (section 3.4.3.3) and high ultimate load factor. Both of these are tactically enhancing characteristics;[25] unfortunately, they push the physiological limits of the human being in the cockpit. We have, in essence, a nine g airplane with a seven g pilot.[26]

4.2.2.2 Recommendations. An easily implemented solution would be the modification of longitudinal control laws to decrease the rate of load factor onset. While it seems wasteful not to take full advantage of the capabilities of this airframe, a small performance penalty and degradation of tactical effectiveness is preferable to

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[25] United States Fighter Weapons School (TOPGUN) Manual, 3rd Edition, May 1986.

[26] U.S. Naval Flight Surgeon's Manual, 2nd Edition, 1978, U.S. Government Printing Office, Chapter 2.

continuing to accept frequent losses of airplane and aircrew attributable to g induced loss of consciousness[27]. Currently accepted g tolerance guidelines[28] predict a substantially decreased probability of g LOC with a load factor onset schedule that allows the initial load factor step to target five gs (instead of the current seven) and then allows the load factor to increase to eight after a total of three seconds (vice the current nine gs after two seconds).

Since the fighter which can generate and maintain the highest load factor may well be the one that wins the fight, limiting airframe performance is not a tactically attractive option; as a result, considerable effort is currently being expended in the area of improving g tolerance. Currently being examined are the areas of aircrew education, physical conditioning and centrifuge training as well as hardware advancements such as reclined seats, improved g garments and positive pressure breathing.[29] The F-16 should certainly be considered a

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[27]Nash, P. "Loss of Consciousness," Weekly Safety Summary No. 24-80 (8-14 June 1980), Naval Safety Center, Norfolk, VA.

[28]Whinnery, J. "$+G_z$ Tolerance Correlation with Clinical Parameters," Aviation Space Environmental Medicine, 50(7):736-741, 1979.

[29]Grissett, J. Physical Fitness Program to Enhance Aircrew G Tolerance (1987 Joint Services G Tolerance Conference). Naval Aerospace Medical Research Laboratory, Pensacola, FL, 1987.

primary candidate for the incorporation any hardware modification that displays the potential for improving g tolerance.

Also being investigated are systems that prevent the catastrophic loss of aircraft and crew following a g LOC episode. Such a system would have the capability to detect an impending or occurring LOC, after which it would issue the flight control commands required to level the wings and commence a pullout to avoid ground impact. The pitch up would be continued until a shallow climb was established, providing sufficient time for the pilot to regain consciousness and reassume control of the airplane. Such systems are currently being investigated, and reliable detection of LOC is proving the most difficult technical problem.[30] The F-16's fly-by-wire control system is the perfect candidate for automatic terrain avoidance systems, since the interface to the control system need only be a simple electrical connection.

4.2.3 Imprecise Flight Path Control During Landing Flare.

4.2.3.1 Discussion. As mentioned in section 3.4.5.1, the control system did not provide sufficiently precise flight path control to prevent a bounce following landing. To fine tune his flight path during the landing flare, a

[30] Lewis, N. "Automatic Aircraft Recovery Following Aircrew Incapacitation" (briefed to the author). NATC Aircrew Systems Laboratory, 1988.

pilot must make small adjustments to lift, which he does by modulating angle-of-attack.[31] A conventional longitudinal control system is well suited to this task, since it provides angle-of-attack command through a second order response.[32] The pilot is generally holding aft stick pressure on the stick during the flare, and is able to modulate his lift (which adjusts his flight path) by making small iterative increases and decreases to that force.

The flight path control task described above is complicated by a fly-by-wire system that commands pitch rate. In such an implementation, stick force controls lift integrally rather than proportionally. When the pilot relaxes pressure following an aft stick input, the aircraft retains the newly commanded pitch attitude and corresponding angle-of-attack (assuming the aircraft is trimmed to zero pitch rate). A conventional system which commands angle-of-attack allows the pilot to command a small, short duration lift increase by making a small increase in aft stick force then quickly decreasing force to the previous level; a pitch rate command system, however, requires a pull, release, push, release sequence

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[31] Radford, C.; Smith, E. "Approach and Landing Flying Qualities Requirements," Flying Qualities Design Criteria (Proceedings of AFFDL Symposium). AFWAL-TR-80-3067, May 1980.

[32] Perkins, C. Airplane Performance Stability and Control. New York: John Wiley & Sons, Inc., 1949.

to accomplish the same task. Resultantly, the iterative lift adjustments required to precisely control flight path are far more difficult to accomplish, and may result in PIO.

As discussed in section 2.2.1.2, the test F-16 had slow flight control laws which commanded a blend of pitch rate and angle-of-attack. Incorporation of the angle-of-attack command portion of the control law may have reduced, but did not alleviate, the problems with flight path control discussed above. Also complicating flight path control during touchdown were the feedback and control system delay characteristics described in section 4.2.1.3.

4.2.3.2 Recommendations. Incorporation of the fixes for the PIO problem (section 4.2.1.2) would probably improve flight path control during landing somewhat. However, new longitudinal control laws for the power approach configuration are required to completely solve this problem.

The F-16 longitudinal control laws were originally implemented to effect a pure pitch rate command system in the power approach configuration, and severe pitch control difficulties were encountered in the landing phase.[33] This evaluator did not fly the airplane with the original

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[33] Parrag, M.; Knotts, L. Augmented Aircraft Handling Qualities. Buffalo: CALSPAN Corp., 1987.

control laws, but would postulate that the difficulties were the result of the phenomena described in the previous section. The new control laws, which command a blend of pitch rate and angle-of-attack, were presumably incorporated to alleviate some of the handling qualities problems encountered during landing. While these new control laws were heralded as an improvement by the manufacturer[34], they fall far short of being a solution. A control law that commands pure angle-of-attack, similar to a conventional control system, should be investigated as a remedy.

4.2.4 High Angle-of-Attack Performance Limits.

While the high angle-of-attack characteristics of this airframe are not a direct result of the flight control system, fly-by-wire is the mechanism that allows the practical utilization of this airframe and its unique stability parameters. A brief discussion of high angle-of-attack performance is therefore presented below.

Angle-of-attack is limited to a maximum of either 25 or 17 degrees, depending on the position of the "stores configuration" switch (section 2.2.1.1). Roll performance is also limited by the flight control system (section 2.2.2.1). These limits represent a tactically significant

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[34] Forgey, D. Flight Control System Functional Description and Troubleshooting Aid. Fort Worth: General Dynamics, 1983.

degradation of slow speed maneuverability (section 3.5.3.2.1). The limits are intended to prevent the airplane from achieving a flight condition that would result in a departure from controlled flight.[35] The potential for departure at relatively benign flight conditions is presumably the result of the same aerodynamics that give the airframe its negative static longitudinal stability and its resulting superb sustained turn capability.

The F-16 compromises slow speed maneuverability for improved sustained turn performance. Such design compromise will probably be required until continuing research and development efforts yield aerodynamic designs capable of stellar performance at both ends of the spectrum. An easy fix for the F-16 high angle-of-attack departure characteristics probably doesn't exist; as such, the high angle of attack restrictions will remain a tactical limitation.

4.2.5 Absence of Speed Cues

4.2.5.1 Discussion. As discussed in section 4.1.3, longitudinal control laws that command load factor can significantly decrease pilot workload by eliminating the requirement to re-trim during speed changes. As discussed

[35] F-16 A/B Flight Manual, USAF T.O. 1F-16A-1, 27 April 1987.

in sections 3.4.3.1 and 3.5.3.2, the F-16 utilizes control laws that provide uniform control effectiveness and sensitivity throughout a wide airspeed range. The combined effect of these two presumably enhancing features is a notable lack of airspeed cues to the pilot. During certain mission phases, this lack of cues can be considered a serious deficiency. For example, with his attention focused on an adversary who is located aft of his wingline, a pilot will not be sufficiently warned that his aircraft is decelerating and he will turn to reengage at an airspeed below optimum, resulting in reduced performance and increased vulnerability.

4.2.5.2 Recommendations. Optimum performance during tactical maneuvering often depends heavily on proper airspeed or angle-of-attack control. While a fly-by-wire system that commands load factor is particularly devoid of pilot airspeed cues, many conventionally controlled airframes provide little tactile feedback of critical flight parameters other than small gages in the cockpit. The F-14, for example, displays no noticeable increase in buffet with angle-of-attack increase above 15 degrees,[36] allowing the pilot to decelerate to a tactically dangerous condition without warning.

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[36] NATOPS Flight Manual, Navy Model F-14A Aircraft, NAVAIR 01-F14AAA-1. Washington: CNO, 1987.

The solution to this problem is a visual or audio warning system to cue the pilot on the status of important flight parameters. Such a system could be an asset to many conventionally controlled aircraft as well. Displays projected on the helmet visor are currently being developed,[37] and cueing for airspeed and angle-of-attack could easily be incorporated into that display presentation. Audio tones also have some merit, but must be carefully integrated from a human factors perspective so as not to overload an already tone-saturated combat cockpit.

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[37]Kaiser Electronics, "Agile Eye for Enhanced Tactical Performance", 1987.

CHAPTER V

GENERAL CONCLUSIONS

5.1 Fly-by-Wire in the Future

Fly-by-wire will undoubtedly be the control system of choice for future tactical aircraft. It offers control system designers the opportunity to:

a. Make an airframe completely user-friendly. Flying qualities can be "programmed" such that the optimal response occurs for every input, maximizing the pilot's combat performance and minimizing his fatigue.

b. Design airframes for maximum aerodynamic efficiency and performance without regard for controllability, which can be effected by the fly-by-wire flight control system.

c. Save weight and space through the elimination of the complex mechanical linkages required by conventional flight control implementations.

5.2 Fly-by-Wire Pitfalls

The fly-by-wire implementation incorporated in the F-16 elegantly solves many traditional flying qualities problems; unfortunately, it introduces several new ones.

The system's most significant afflictions, which include control system delay, insufficient pilot feedback and ineffectual landing longitudinal control laws, become evident only when high gain mission tasks are actually attempted by a real pilot. These problems were not anticipated by the designers and are the consequence of an inadequate appreciation for the complexities the pilot/airplane interface.

5.3 Recommendations for Future Systems

Closer study of the man/machine synergism is critical to the successful development of future electronic flight control systems. The following points are considered crucial to the development of new fly-by-wire systems.

a. The need for simulation, both ground and flight based, cannot be overstated. Parametric modeling of the pilot will undoubtedly improve analytical design capabilities in the future; however, our current understanding of the mechanism by which a pilot processes a multitude of external stimuli is not sufficiently advanced to allow the formation of a pilot model with adequate accuracy. Until such a model is developed, prospective flight control systems must be examined with an actual "pilot in the loop" during early design stages.

b. A data base of fly-by-wire "lessons learned" is crucial to future design efforts. In an overview of several fly-by-wire systems, Smith makes the case that the flying qualities development process has not been working well in the context of experiences related to recent fighter aircraft[38]. In the words of the American philosopher George Santayana, "those who cannot remember the past are condemned to repeat it."

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[38]Smith, R. "Evaluating the Flying Qualities of Today's Fighter Aircraft," Design Criteria for the Future of Flight Controls (Proceedings of the Flight Dynamics Laboratory Flight Control Symposium 2-5 March 1982). AFWAL-TR-82-3064, July 1982.

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APPENDIX A

FIGURES

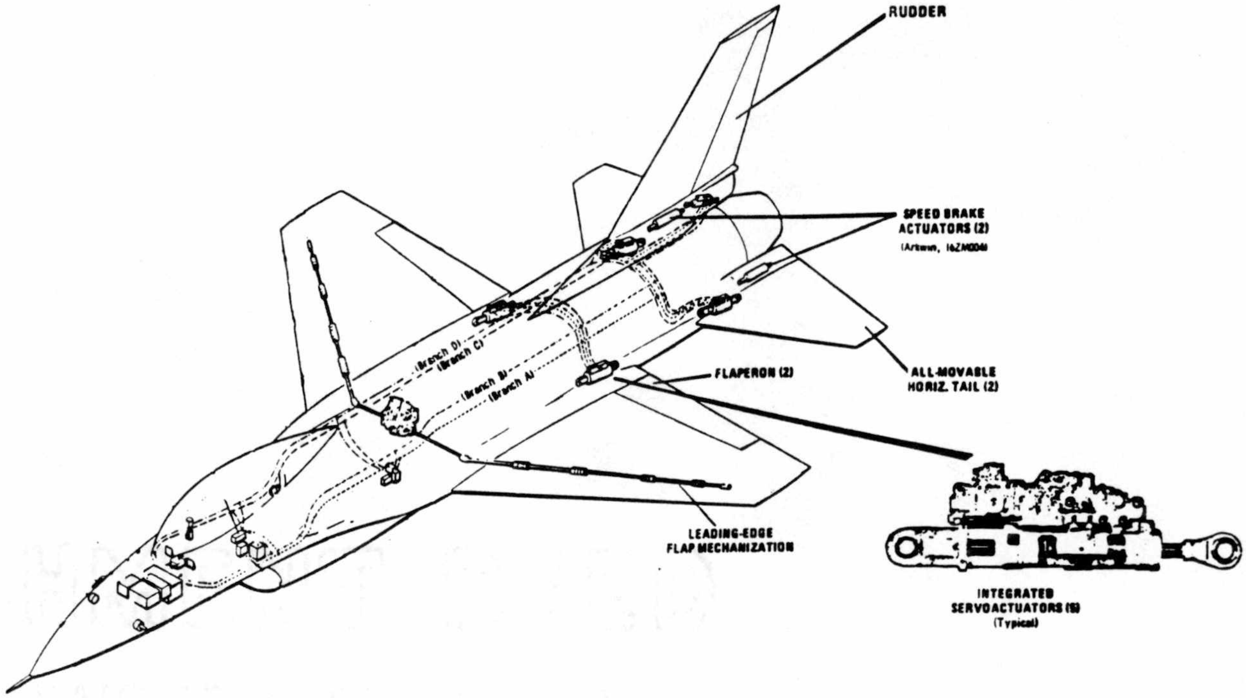
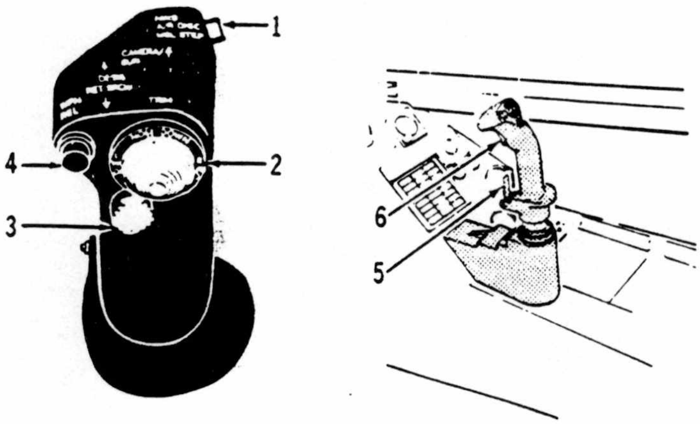
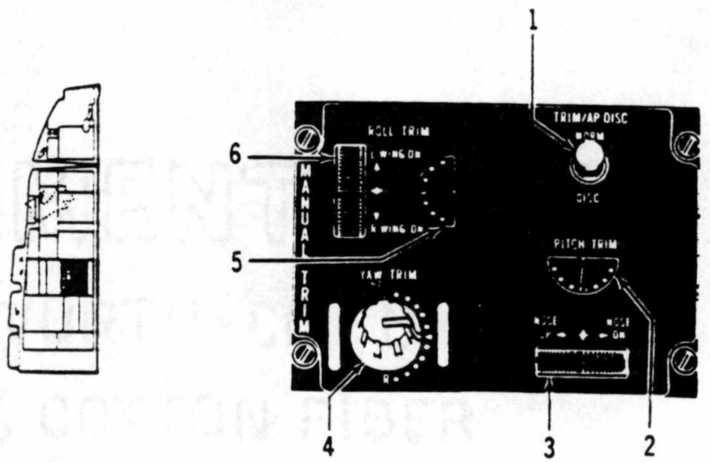


Figure 1
FLIGHT CONTROL SURFACES



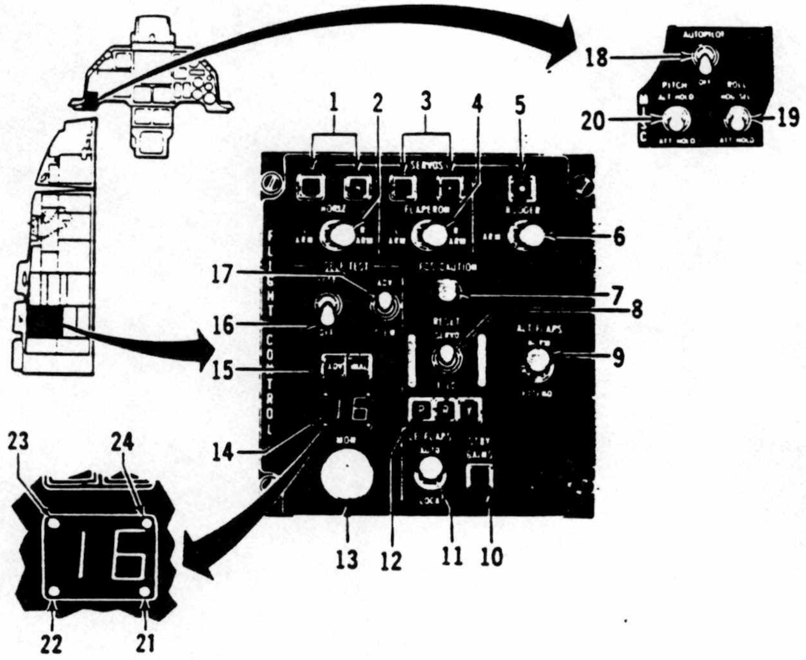
1. NWS A/R DISC MSL STEP Button
2. TRIM Switch (4-Way, Momentary)
3. **BLOCK 10** DESIG RET SRCH Switch (2-Way, Momentary)
   BLOCK 15 DSG RS DO DS Switch (4-Way, Momentary)
4. WPN REL Button
5. Paddle Switch
6. CAMERA/GUN Trigger (2-Position)

Figure 2
CONTROL STICK



1. TRIM/AP DISC Switch
2. PITCH TRIM Indicator
3. PITCH TRIM Wheel
4. YAW TRIM Knob
5. ROLL TRIM Indicator
6. ROLL TRIM Wheel

Figure 3
MANUAL TRIM PANEL



1. HORIZ Tail Servo Status Light
2. HORIZ Tail Servo Arm Switch
3. FLAPERON Servo Status Light
4. FLAPERON Servo Arm Switch
5. RUDDER Servo Status Light
6. RUDDER Servo Arm Switch
7. FCS CAUTION RESET Button
8. SERVO ELEC RESET Switch
9. ALT FLAPS Switch (Lever Lock)
10. STBY GAINS Light
11. LE FLAPS Switch
12. Pitch, Roll, and Yaw Status Lights
13. Test Monitor Receptacle
14. Test Number
15. Test ADV MAL Light
16. SELF TEST Switch
17. ADV SLEW Switch
18. AUTOPILOT Switch
19. Autopilot ROLL Switch
20. Autopilot PITCH Switch
21. FLCC Data Word Dot Light
22. ECA Data Word Dot Light
23. Spare
24. Rate Gyro Speed Detect Dot Light

Figure 4
FLIGHT CONTROL PANEL

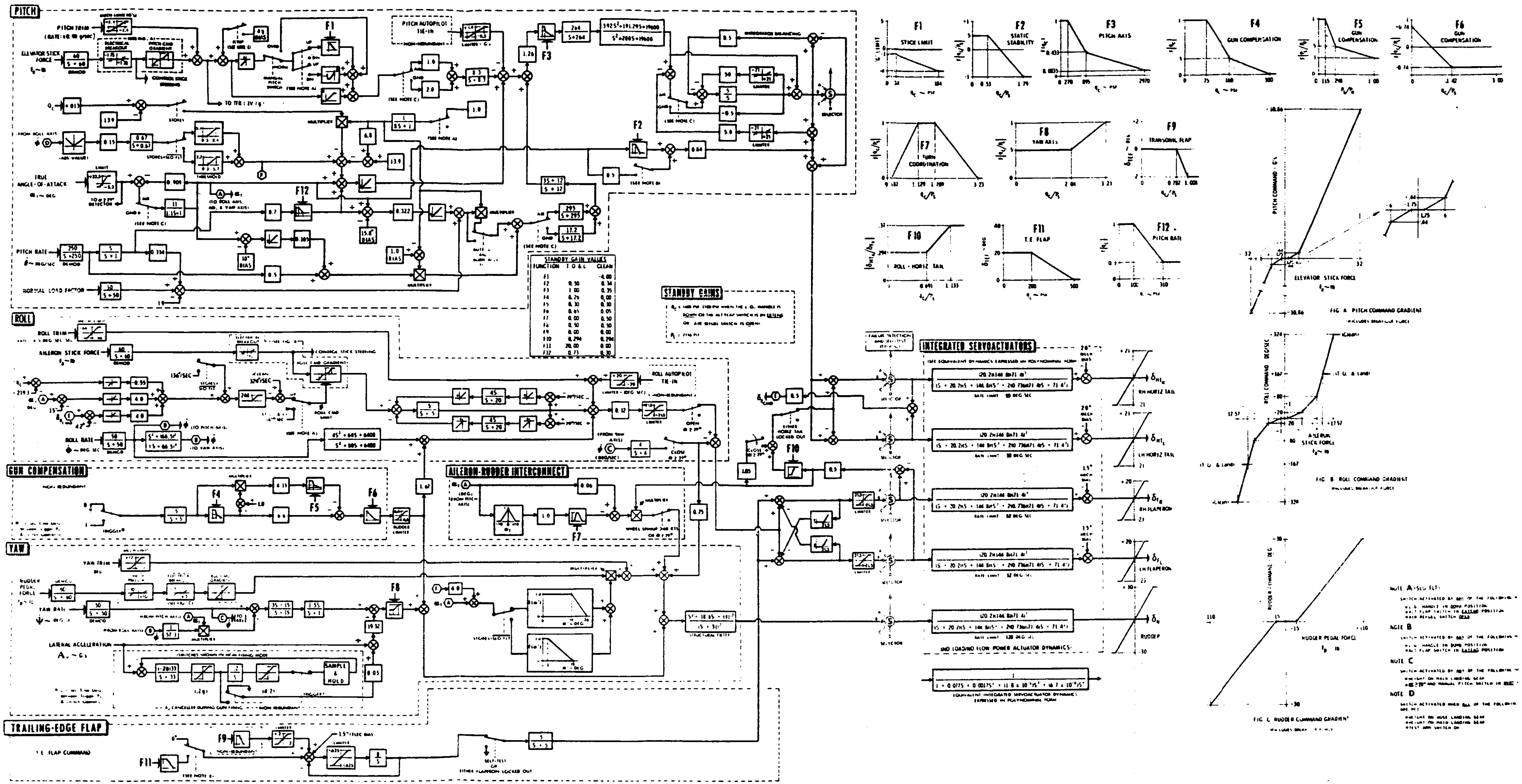


Figure 5
FLIGHT CONTROL FUNCTIONAL BLOCK DIAGRAM

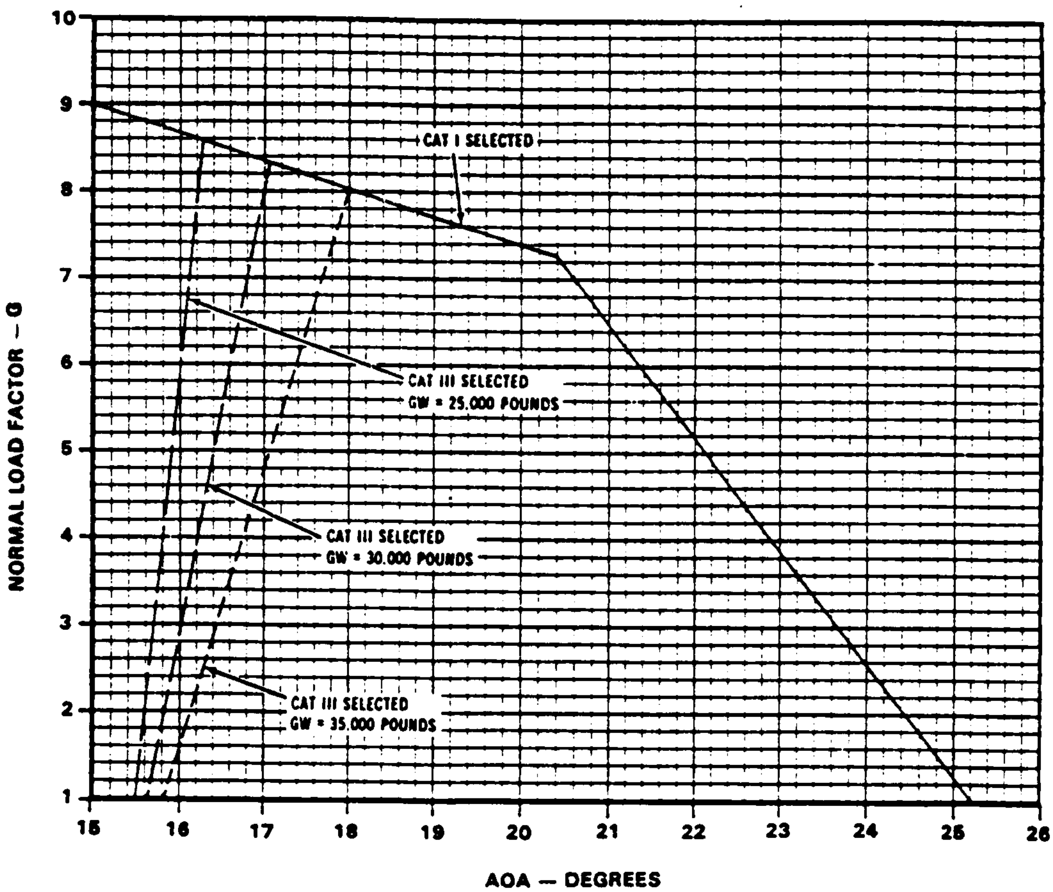


Figure 6

ANGLE OF ATTACK/LOAD FACTOR LIMITER BOUNDARIES

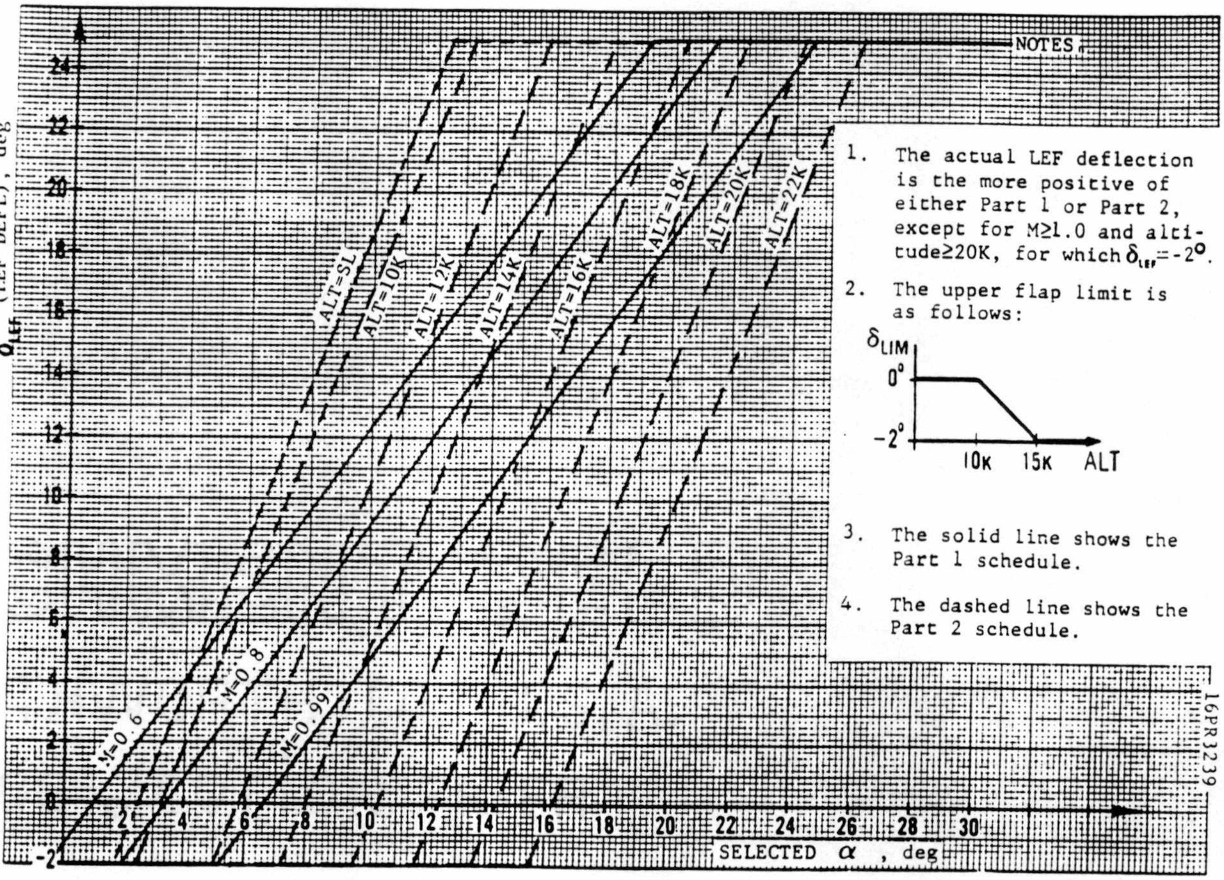


Figure 7

LEADING EDGE FLAP EXTENSION SCHEDULE

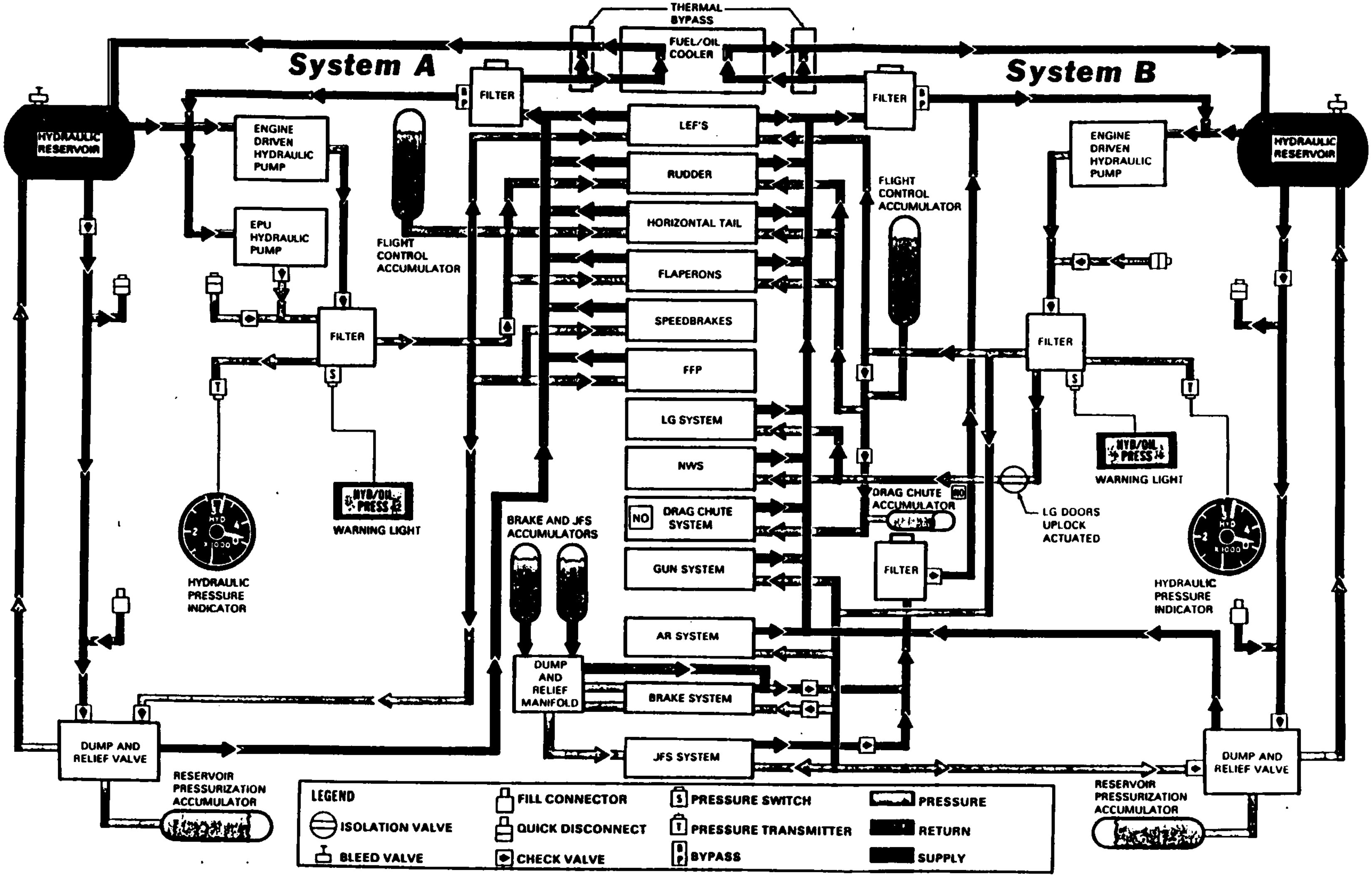
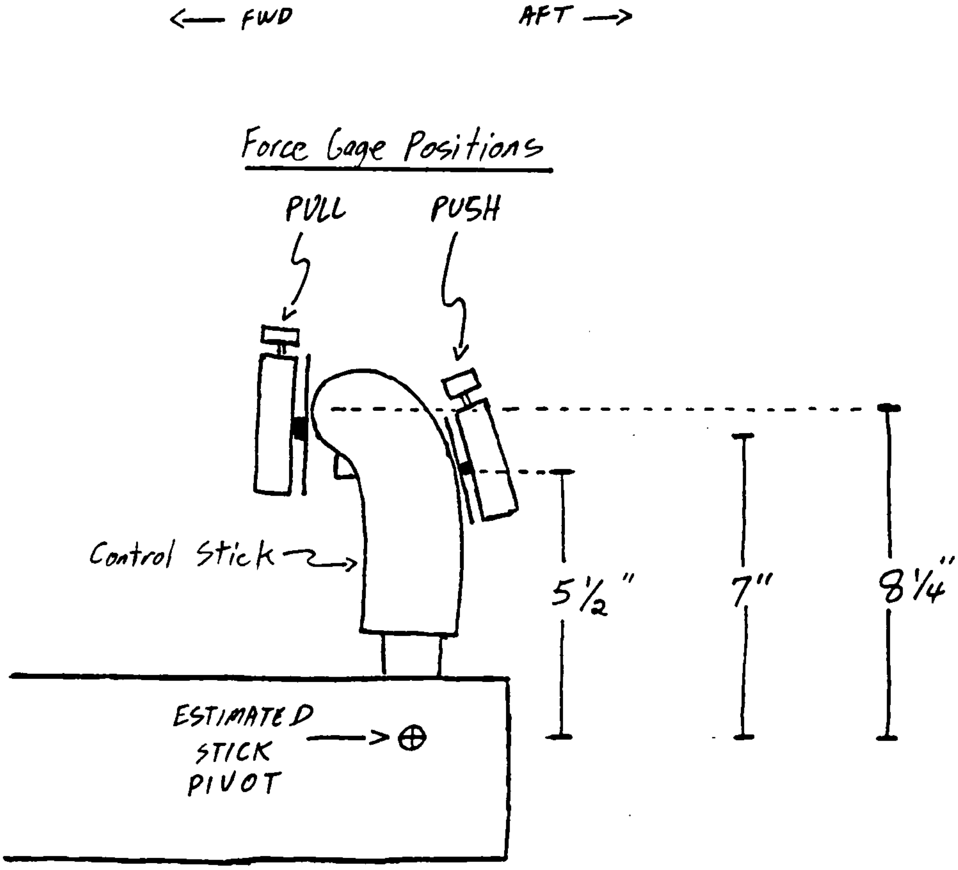


Figure 8

HYDRAULIC POWER SYSTEM



Measured forces were scaled to "normalize" for a 7" lever arm; i.e.

$$F_{s_{PULL_{TRUE}}} = F_{s_{PULL_{MEASURED}}} \times \frac{8\,1/4}{7}$$

$$F_{s_{PUSH_{TRUE}}} = F_{s_{PUSH_{MEASURED}}} \times \frac{5\,1/2}{7}$$

Figure 9

LONGITUDINAL STICK FORCE MEASUREMENT

APPENDIX B

TABLES

Table I
F-16B TEST CONFIGURATIONS

Configuration	Landing Gear	Flap (1) LE/TE	Refuel Door	Speed Brakes	Thrust
Takeoff (TO1)	Down	Auto/ Auto	Closed	Closed	MIL
Takeoff (TO2)	Down	Auto/ Auto	Closed	Closed	MAT (2)
Cruise (CR)	Up	Auto/ Auto	Closed	Closed	PLF (3)
In Flight Refuel (IFR)	Up	Auto/ Auto	Open	Closed	PLF (3)
Combat (CO)	Up	Auto/ Auto	Closed	Closed	MAT (2)
Power (P)	Up	Auto/ Auto	Closed	Closed	MIL
Glide (G)	Up	Auto/ Auto	Closed	Closed	IDLE
Decel (DEC)	Up	Auto/ Auto	Closed	Open	IDLE
Dive (D)	Up	Auto/ Auto	Closed	Closed	A/R
Power Approach (PA)	Down	Auto/ Auto	Closed	Closed	PNA (4)
Land (L)	Down	Auto/ Auto	Closed	Open	IDLE

(1) Leading edge flaps programmed as a function of mach number, AOA, and altitude when auto was selected. Trailing edge flaps (flaperons) also followed an airspeed/mach number program in auto with the landing gear handle up; when the landing gear handle was down, they deflected 20 degrees downward.
(2) Maximum Augmented Thrust (MAT).
(3) Power for Level Flight at trim airspeed (PLF).
(4) Power for Normal Approach (PNA).

Table II
F-16B TEST CONDITIONS

Test	Configuration	Altitude (ft MSL)	Airspeed (KIAS)
Human Engineering(1)	All	All	All
Ground Handling	TX	GND (1)	0 - 32
Mechanical Characteristics	TX P	GND 15 - 25,000	0 390
Longitudinal Flying Qualities	CO, P CR PA	15,000 6 - 41,000 6 - 10,000	500 - 720 100 - 410 120 - 150
Lateral-Directional Flying Qualities	CO, P CR PA	15,000 6 - 41,000 6 - 10,000	500 - 720 100 - 410 120 - 150
Stall-Departure Characteristics	CR PA	13 - 17,000 7 - 10,000	105 - 250 95 - 140
Takeoff and Landing Flying Qualities	TO1, TO2, P, PA, L	GND - 3,800	0 - 175
Trimmability and Mission Flying Qualities	ALL	GND - 41,300	0 - 525
Takeoff and Landing Performance	TO1, TO2	GND - 3,000	0 - 570
Climb and Descent Performance	P, GL CO	3 - 41,000 3 - 25,000	202 - 560
Level Flight Accel Performance	CO CO	4,800 15,800	180 - 560 150 - 720
Aircraft Systems	ALL	ALL	ALL

(1) Human engineering evaluation conducted during day ground test and during all flight evaluations.
(2) Field elevation 1775 MSL.
GND - Ground.

Table III

DUTCH ROLL CHARACTERISTICS

Configuration	Pressure Altitude (ft)	Airspeed (KIAS)	Gross Weight (lb)	Center of Gravity (% mac)	ζ_d		ω_{n_d} (rad/sec)		ϕ/β	$\omega_{n_d}^2 \lvert\phi/\beta\rvert$	$\zeta_d \omega_{n_d}$	
					EST (1)	SPEC MIN	EST (1)	SPEC MIN	(1)	$(rad/sec)^2$	EST (1)	SPEC MIN
Cruise	14,000	125 (2)	20,400	36.6	0.4	0.4	1 3/4	1.0	3	9	0.7	0.35
Power Approach	8,500	140	18,600	33.4	0.3	0.08	1 3/4	1.0	3	9	0.5	0.15

(1) Data estimated using the following methods:

ζ_d

Cruise: Counted bank angle overshoots
Power approach: Calculated from bank angle decay

ω_{n_d}

Timed roll period and calculated using ζ_d

ϕ/β

Observed ellipse subscribed by wingtip

(2) 125 KIAS corresponded to 18 degrees AOA

APPENDIX C

WEIGHT AND BALANCE FORMS

WEIGHT AND BALANCE CLEARANCE FORM F – TACTICAL
(USE REVERSE FOR TRANSPORT MISSIONS)

FOR USE IN T.O. 1-1B-40, NAVAIR 01-1B-40, AND TM 55-405-9

DATE (YYMMDD)	10-9-87	AIRCRAFT TYPE	F-16B	FROM	[illegible]
MISSION		SERIAL NO.	78-0088	TO	LOCAL
HOME STATION	[illegible]	PILOT			

MOST FWD, MOST AFT, CORRECTIONS (Ref. 11)

COMPT	ITEM	CHANGES (+ or –) WEIGHT	INDEX OR MOM/
TOTAL WEIGHT ADDED		+	+
TOTAL WEIGHT REMOVED		–	–
NET DIFFERENCE (Ref 11)			

REMARKS

- SMALL TAIL
- ALL FCR IN

LOADING A

REF	ITEM	WEIGHT	MOM/ 100
1	BASIC AIRCRAFT (From Chart C)	16661	53348
2	OIL (3.1 Gal)	24	90
3	DISTRIBUTION OF LOAD — CREW: COMPT / NO. / WEIGHT / CARGO/MISC		
	C / 1 / 215 / PILOT	215	316
	C / 1 / 215 / PILOT	215	336
4	OPERATING WEIGHT	17115	54070
5	AMMO		
6	BOMBS, MISSILES, ETC.		
7	FUEL — FULL INTERNAL @ 6.5#/GAL	5785	19709
8	MISC. VARIABLES		
9	TAKEOFF CONDITION (uncorrected)		
10	TAKEOFF C.G. IN % M A C OR IN		
11	CORRECTIONS (If required)		
12	TAKEOFF CONDITION (corrected)	22900	73779
13	TAKEOFF C.G. IN % M A C	36.12% 68.0 IN	
14	TAKEOFF FUEL	-5785	-19709
	LESS EXPENDABLES		
15	ESTIMATED LANDING FUEL	1000	3532
16	ESTIMATED LANDING CONDITION	18115	57652
17	ESTIMATED LANDING C.G. IN % M A C	33.23% 62.0 IN	

LIMITATIONS

GROSS WT TAKEOFF (lb)	35400#	GROSS WT LANDING (lb)	35400#
PERMISSABLE C.G. TAKEOFF (% M A C or in)	FORWARD 23.6%	AFT 33.5%	
PERMISSABLE C.G. LANDING (% M A C or in)	FORWARD 19.6%	AFT 38.5%	

COMPUTED BY SIGNATURE: CB Jensen

WEIGHT AND BALANCE AUTHORITY SIGNATURE: CB Jensen

PILOT SIGNATURE:

DD FORM 365-4, 87 JAN — REPLACES DD FORM 365F, SEP 84 WHICH WILL BE USED.

LOADING "A" WEIGHT AND BALANCE

WEIGHT AND BALANCE CLEARANCE FORM F – TACTICAL
(USE REVERSE FOR TRANSPORT MISSIONS)

FOR USE IN T.O. 1-1B-40, NAVAIR 01-1B-40, AND TM 55-405-9

DATE (YYMMDD)	11-9-87	AIRCRAFT TYPE	F-16B	FROM	EAFB
HOME STATION	EAFB	MISSION		SERIAL NO	78-0081
TO	LOCAL	PILOT			

MOST FWD MOST AFT CORRECTIONS (Ref 11) COMPT	ITEM	CHANGES (+ or –) WEIGHT	INDEX OR MOM.
TOTAL WEIGHT ADDED		+	+
TOTAL WEIGHT REMOVED		–	–
NET DIFFERENCE (Ref 11)			

REMARKS

- SMALL TAIL
- ALL ECR-11V

LOADING B

REF	ITEM	WEIGHT	INDEX OR MOM. /100
1	BASIC AIRCRAFT (From Chart C)	16615	53150
2	OIL (3.1 Gal)	24	90
3	DISTRIBUTION OF LOAD — COMPT / CREW NO. / WEIGHT / CARGO MISC		
	C 1 215 PILOT	215	296
	C 1 215 PILOT	215	436
	1/9 2 AIM-9L/P	390	1123
	3/7 2 WEAP PYLONS	521	1820
	3/7 2 SUU-20 S	480	1626
4	OPERATING WEIGHT	18460	58823
5	AMMO		
6	BOMBS, MISSILES, ETC. — 12 BDU-33 D/B's STA 3/7	276	939
7	FUEL — FULL INTERNAL @ 6.5#/GAL	5785	19709
8	MISC. VARIABLES		
9	TAKEOFF CONDITION (uncorrected)		
10	TAKEOFF C.G. IN % MAC OR IN		
11	CORRECTIONS (if required)		
12	TAKEOFF CONDITION (corrected)	24521	79471
13	TAKEOFF C.G. IN % MAC	37.53%	62.2 IN
14	TAKEOFF FUEL		
	LESS EXPENDABLES	–5785	–19709
		–276	–939
15	ESTIMATED LANDING FUEL	1000	3582
16	ESTIMATED LANDING CONDITION	19460	62403
17	ESTIMATED LANDING C.G. % MAC	35.02%	62 IN

LIMITATIONS

GROSS WT TAKEOFF (lb)	35400#	GROSS WT LANDING (lb)	35400#
PERMISSIBLE C.G. TAKEOFF (% MAC or in)	FORWARD 25.4%	AFT 40.5%	
PERMISSIBLE C.G. LANDING (% MAC or in)	FORWARD 19.6%	AFT 40.5%	

COMPUTED BY SIGNATURE: C B Jensen

WEIGHT AND BALANCE AUTHORITY SIGNATURE: C B Jensen

PILOT SIGNATURE:

DD FORM 365-4, 22 JAN — REPLACES DD FORM 365F, SEP 54, WHICH WILL BE USED.

LOADING "B" WEIGHT AND BALANCE

APPENDIX D

COOPER-HARPER HANDLING QUALITIES RATING SCALE

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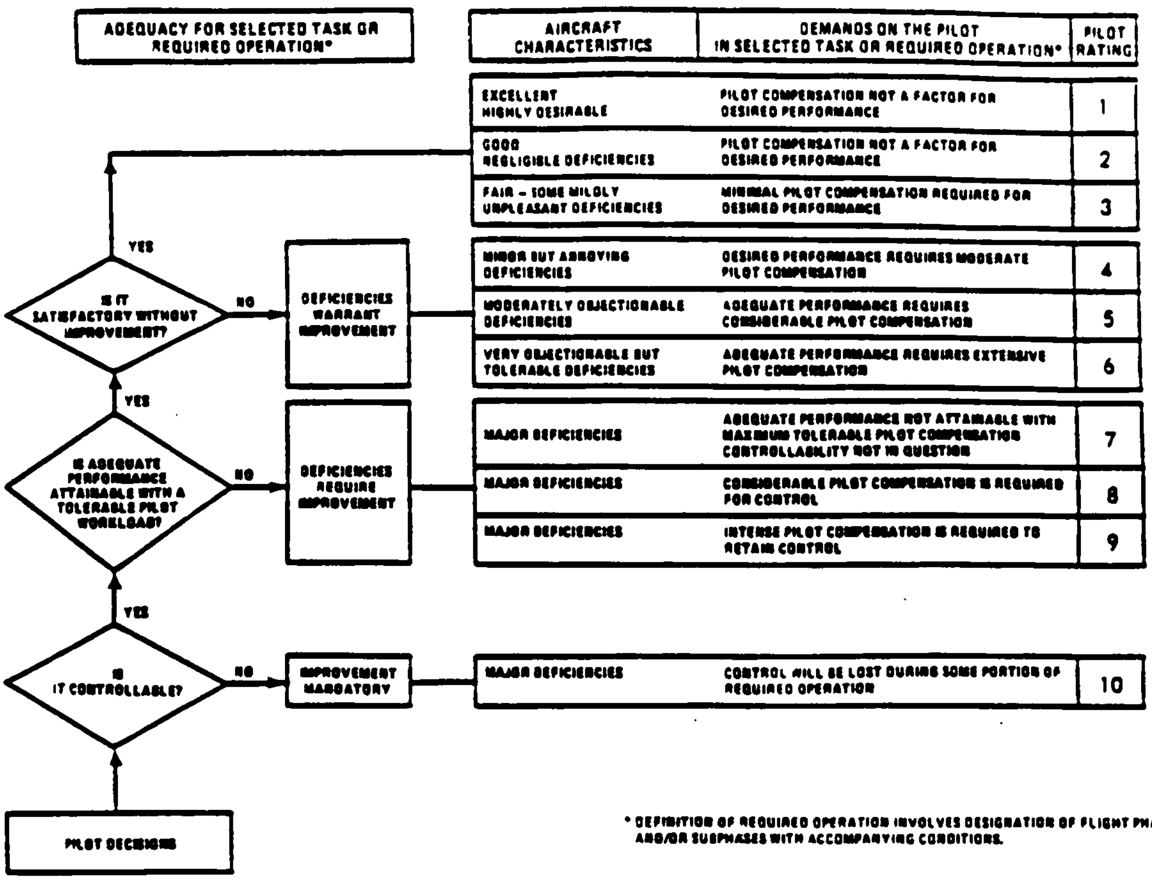


* DEFINITION OF REQUIRED OPERATION INVOLVES DESIGNATION OF FLIGHT PHASE AND/OR SUBPHASES WITH ACCOMPANYING CONDITIONS.

APPENDIX E

DATA PLOTS

METHOD: GROUND CONTROL SWEEP DATA: HAND HELD

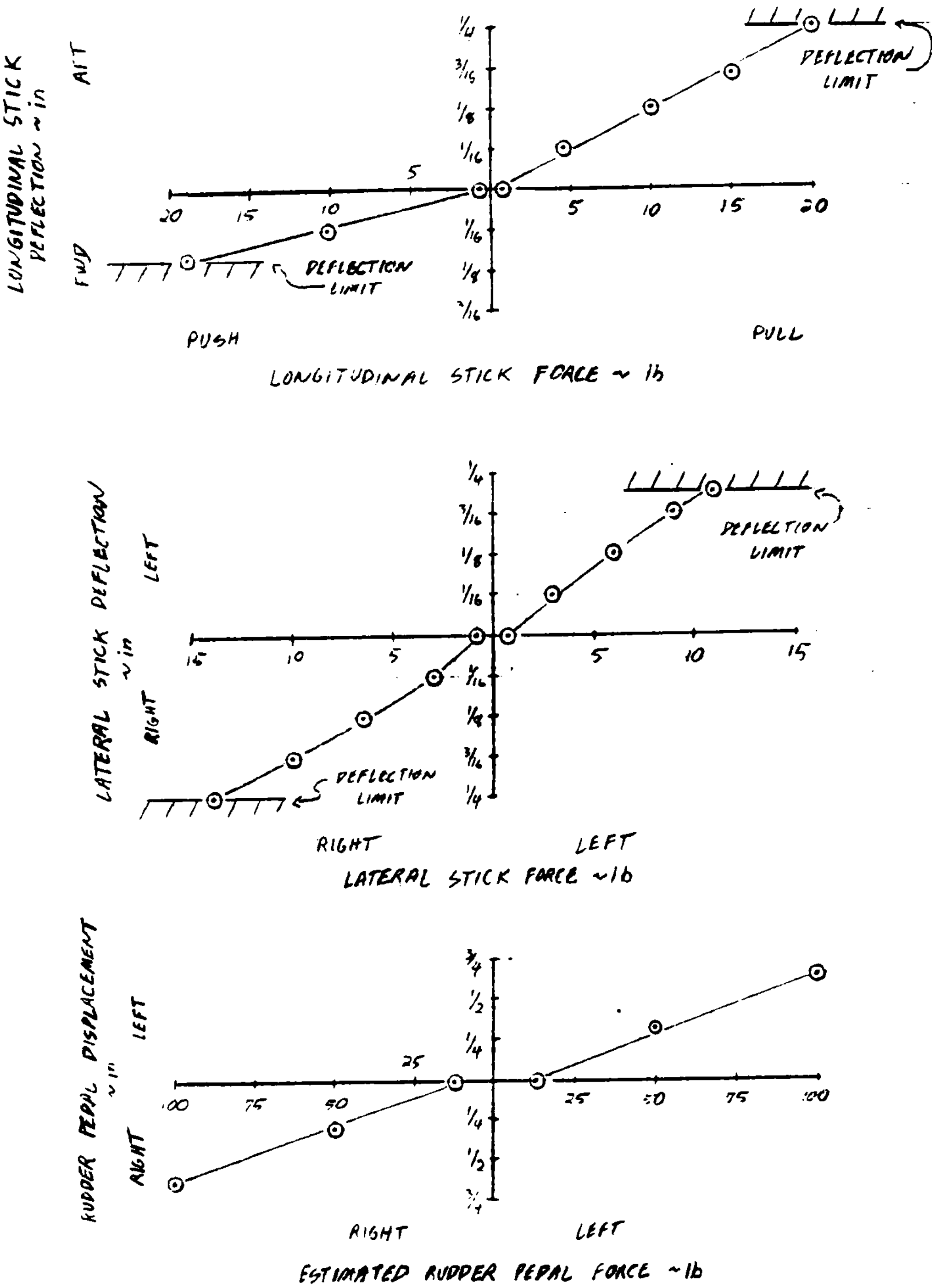


Figure 1
MECHANICAL CHARACTERISTICS

CONFIGURATION: CRUISE
LOADING: DESIGNATION A
DATA: HAND HELD
METHOD: SLOW START PHUGOID

PRESSURE ALTITUDE: 15,000 ft
TRIM AIRSPEED/AOA: 120 KIAS/19 deg
GROSS WEIGHT: 20,100 lb
CENTER OF GRAVITY: 36.8% MAC

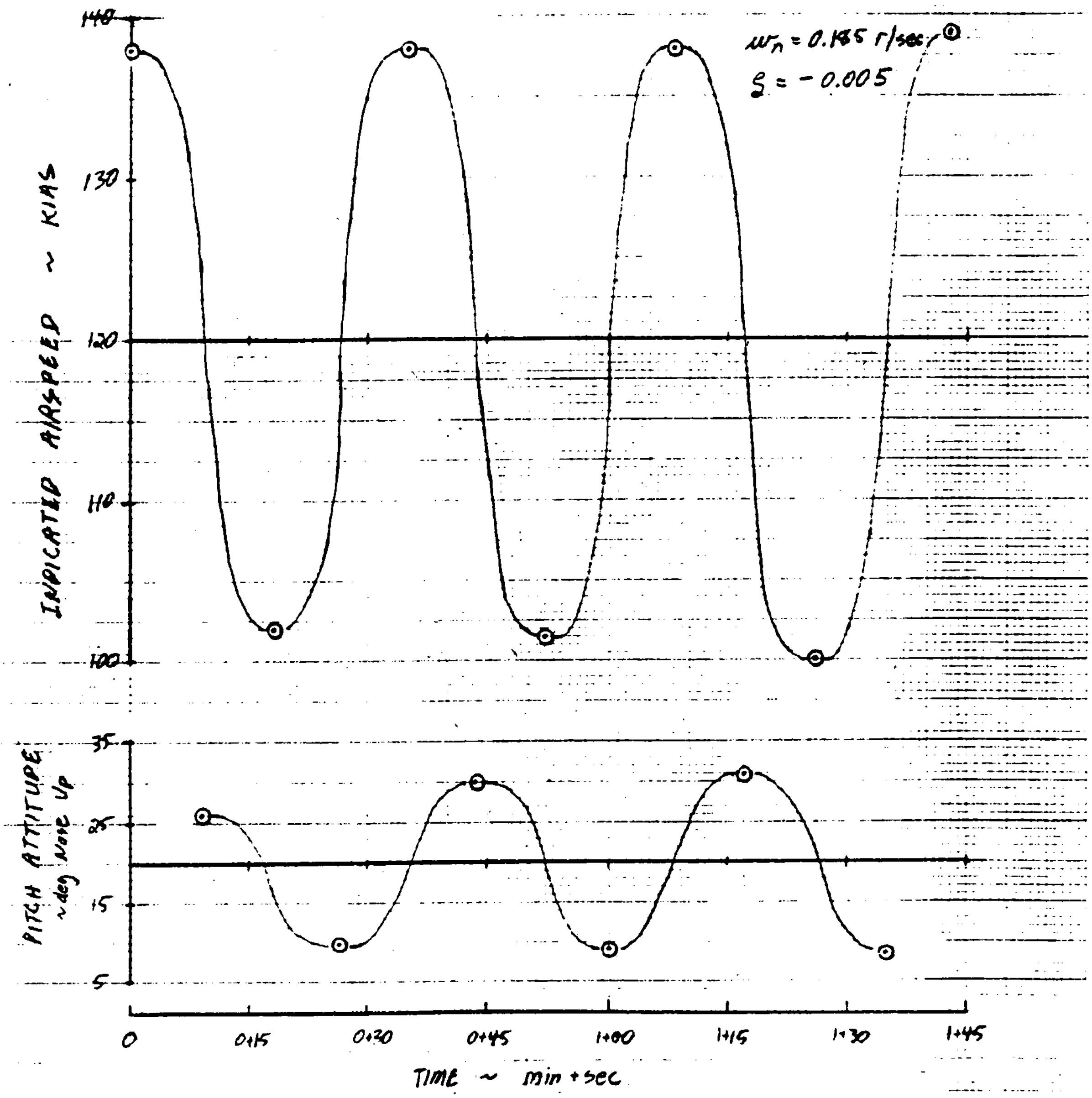
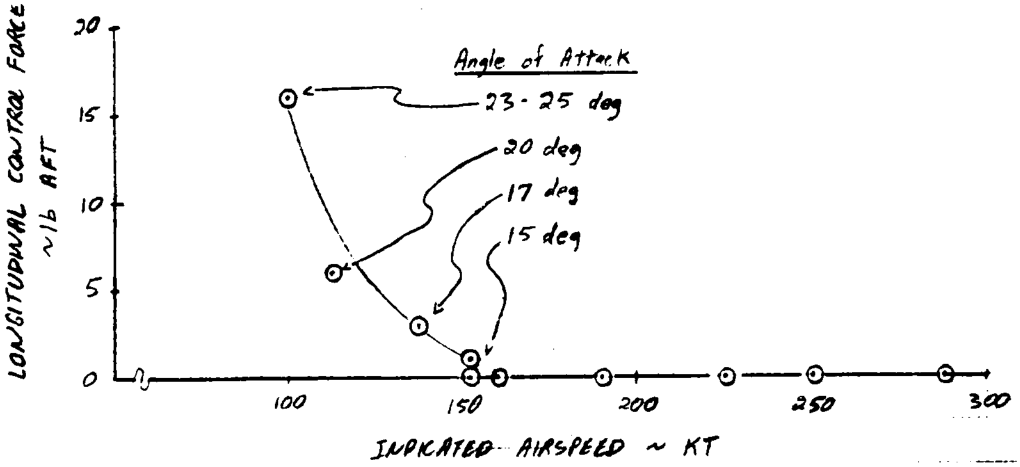


Figure 2
LONG PERIOD OSCILLATION

CONFIGURATION: CRUISE
LOADING: DESIGNATION A
DATA: HAND HELD
METHOD: STABILIZED POINT

PRESSURE ALTITUDE: 15,000 ft
TRIM AIRSPEED: 190 KIAS
AVERAGE GROSS WEIGHT: 20,400 lb
AVERAGE CG: 36.7%



CONFIGURATION: POWER APPROACH
LOADING: DESIGNATION A
DATA: HAND HELD
METHOD STABILIZED POINT

PRESSURE ALTITUDE: 15,000 ft
TRIM AIRSPEED: 135 KIAS
AVERAGE GROSS WEIGHT: 19,100 lb
AVERAGE CG: 36.3% MAC

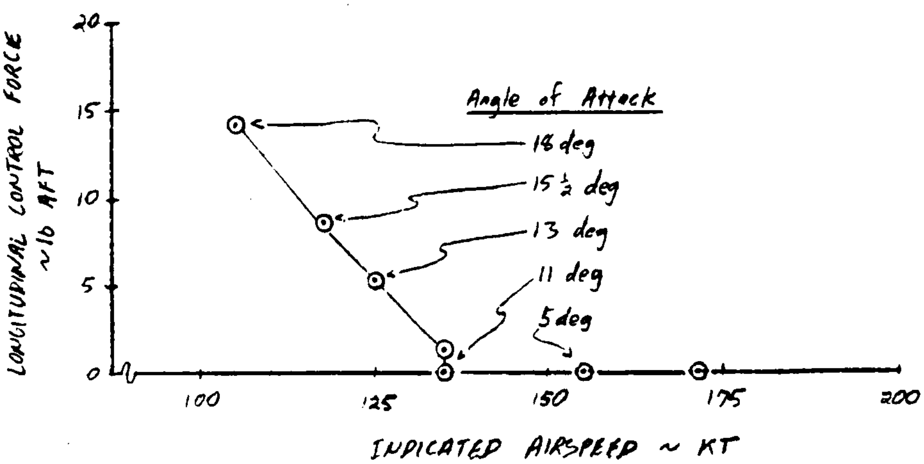


Figure 3
NON-MANEUVERING STATIC STABILITY

CONFIGURATION: POWER
LOADING: DESIGNATION A

PRESSURE ALTITUDE: 15,000 ft
DATA: HAND HELD

SYMBOL	METHOD	KIAS	AVG GROSS WT	CG (MAC)
○	STEADY TURN, PUSHOVER	400	20,400 lb	35 %
△	SUDDEN PULL			
□	STEADY TURN	250	19,800 lb	36.8 %

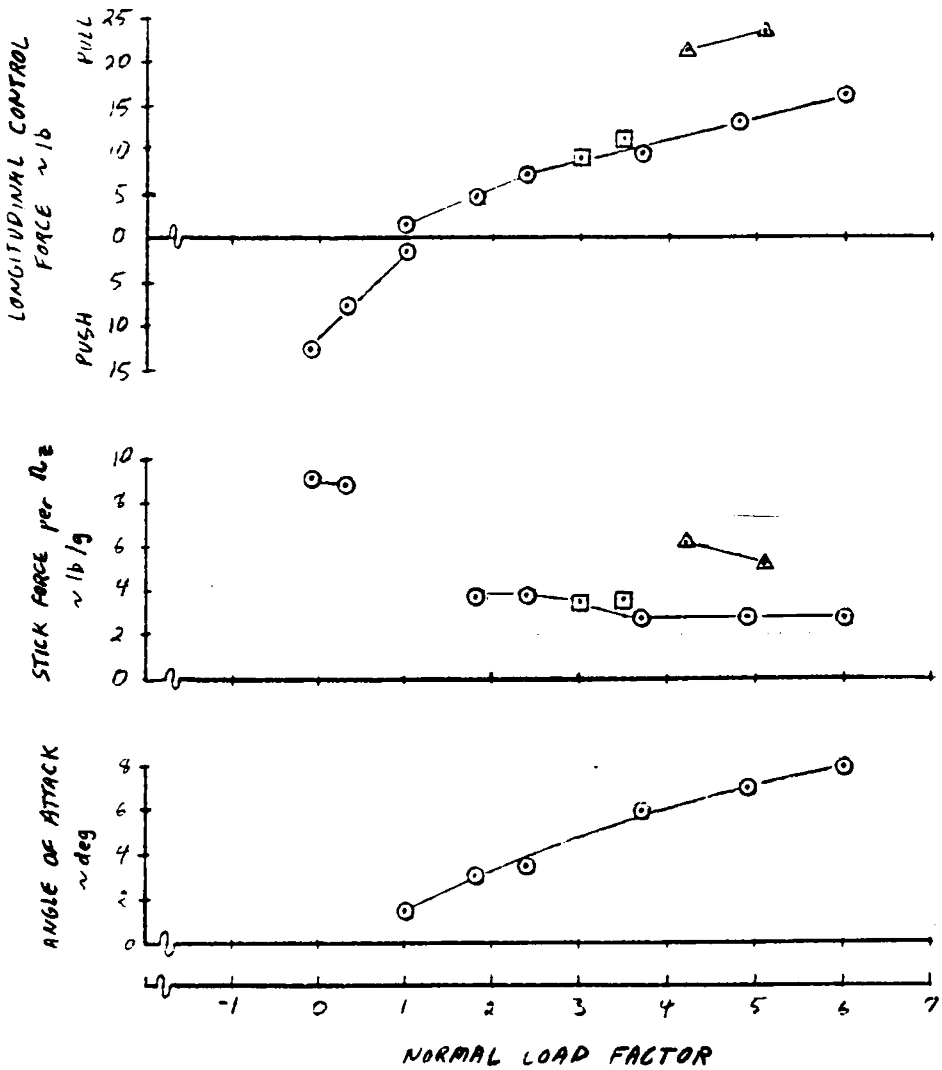


Figure 4
MANEUVERING STABILITY CHARACTERISTICS
(Configuration POWER)

CONFIGURATION: POWER
LOADING: DESIGNATION A
DATA: HAND HELD
METHOD: STEADY TURN
STEADY PUSH OVER

PRESSURE ALTITUDE: 8,000 ft
TRIM AIRSPEED: 150 KIAS
AVERAGE GROSS WEIGHT: 19,700 lb
AVERAGE CG: 36.8% MAC

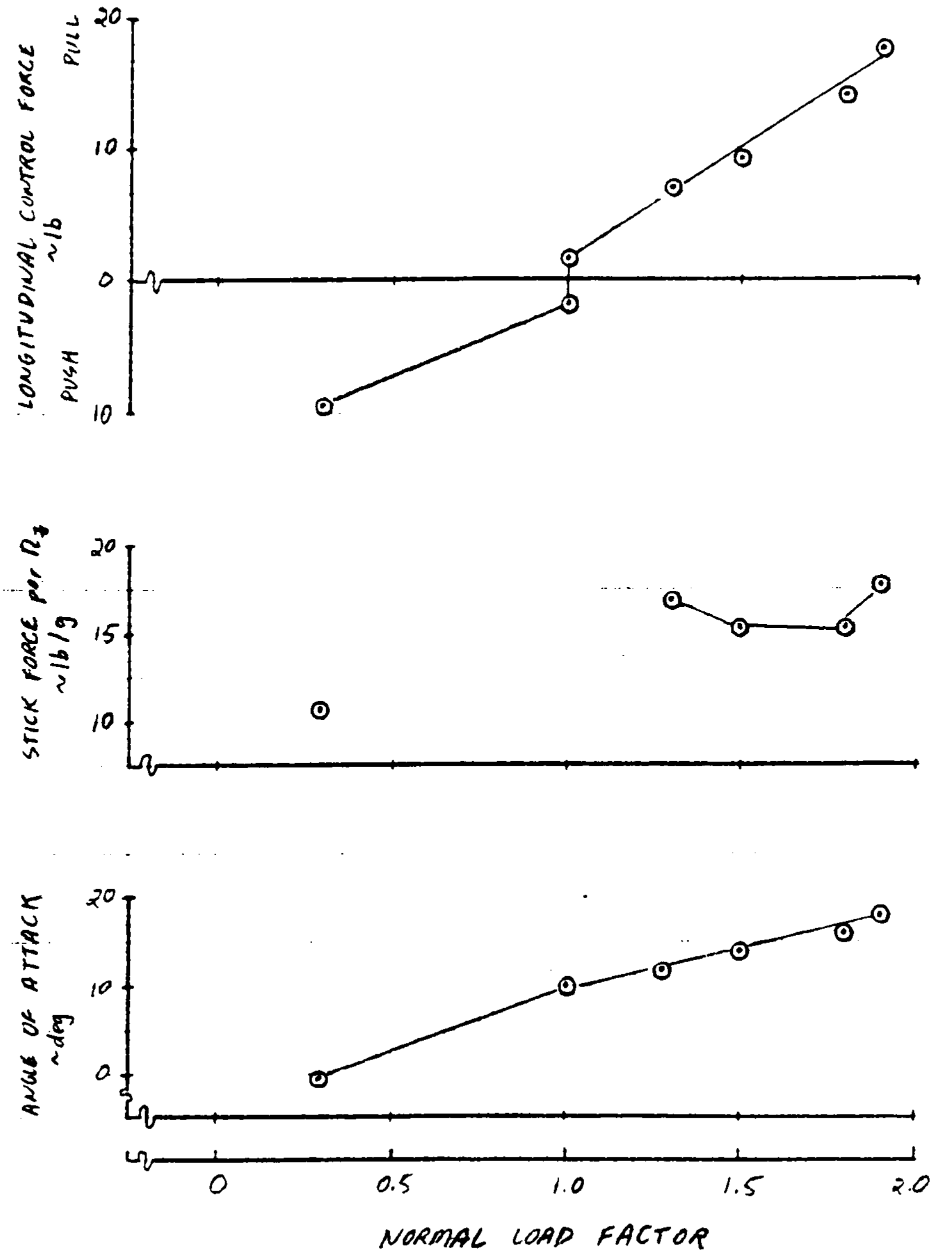


Figure 5
MANEUVERING STABILITY CHARACTERISTICS
(Configuration POWER APPROACH)

CONFIGURATION: CRUISE
LOADING: DESIGNATION A
DATA: HAND HELD
METHOD: STEADY HEADING SIDE SLIP

PRESSURE ALTITUDE: 20,000 ft
INDICATED AIRSPEED: 250 KIAS
GROSS WEIGHT: 20,300 lb
CENTER OF GRAVITY: 35% MAC

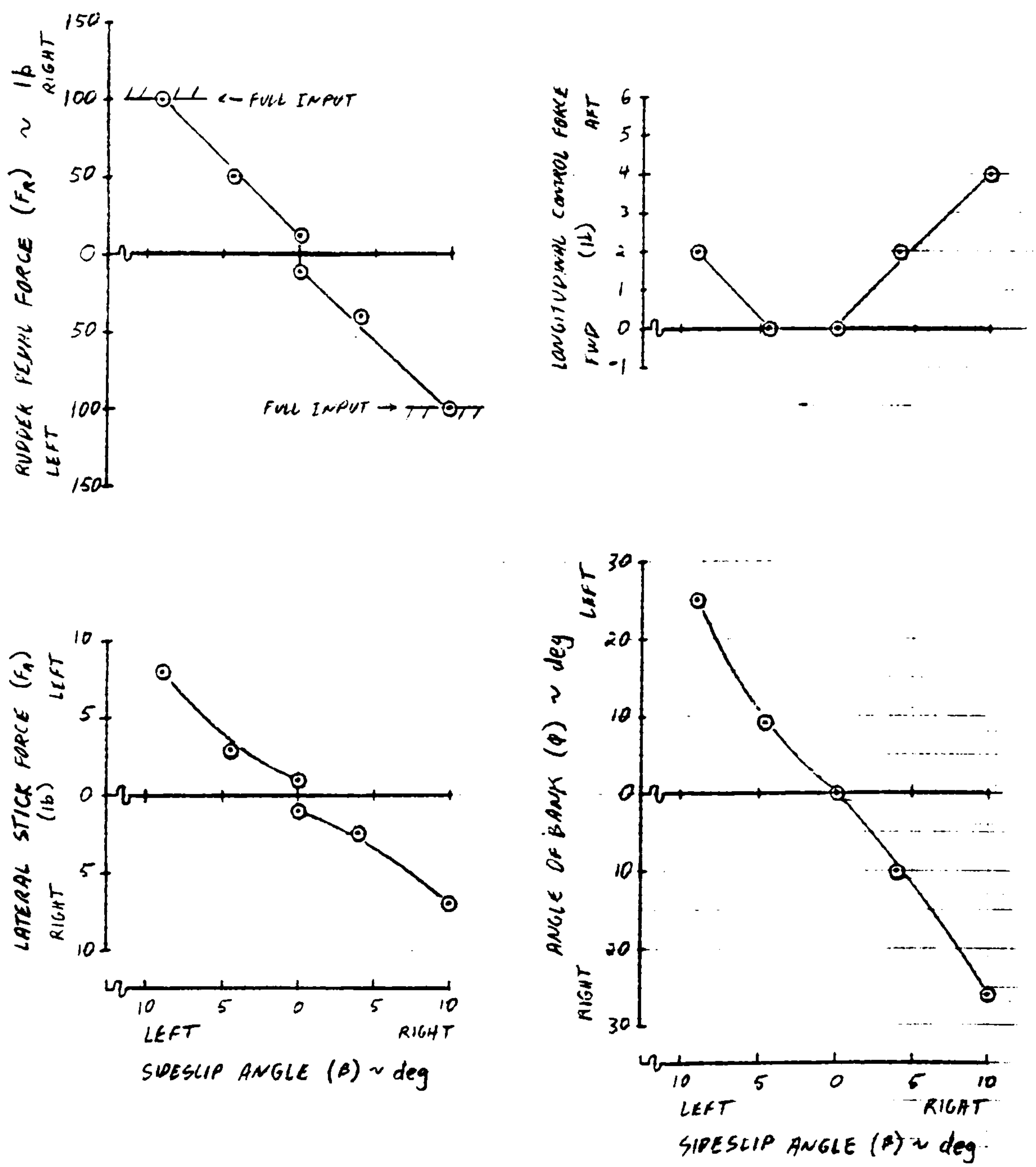


Figure 6
STEADY HEADING SIDE SLIP
(Configuration CRUISE)

CONFIGURATION: POWER APPROACH
LOADING: DESIGNATION A
DATA: HAND HELD
METHOD: STEADY HEADING SIDE SLIP

PRESSURE ALTITUDE: 10,000 ft
INDICATED AIRSPEED: 137 KIAS
GROSS WEIGHT: 18,500 lb
CENTER OF GRAVITY: 33.5% MAC

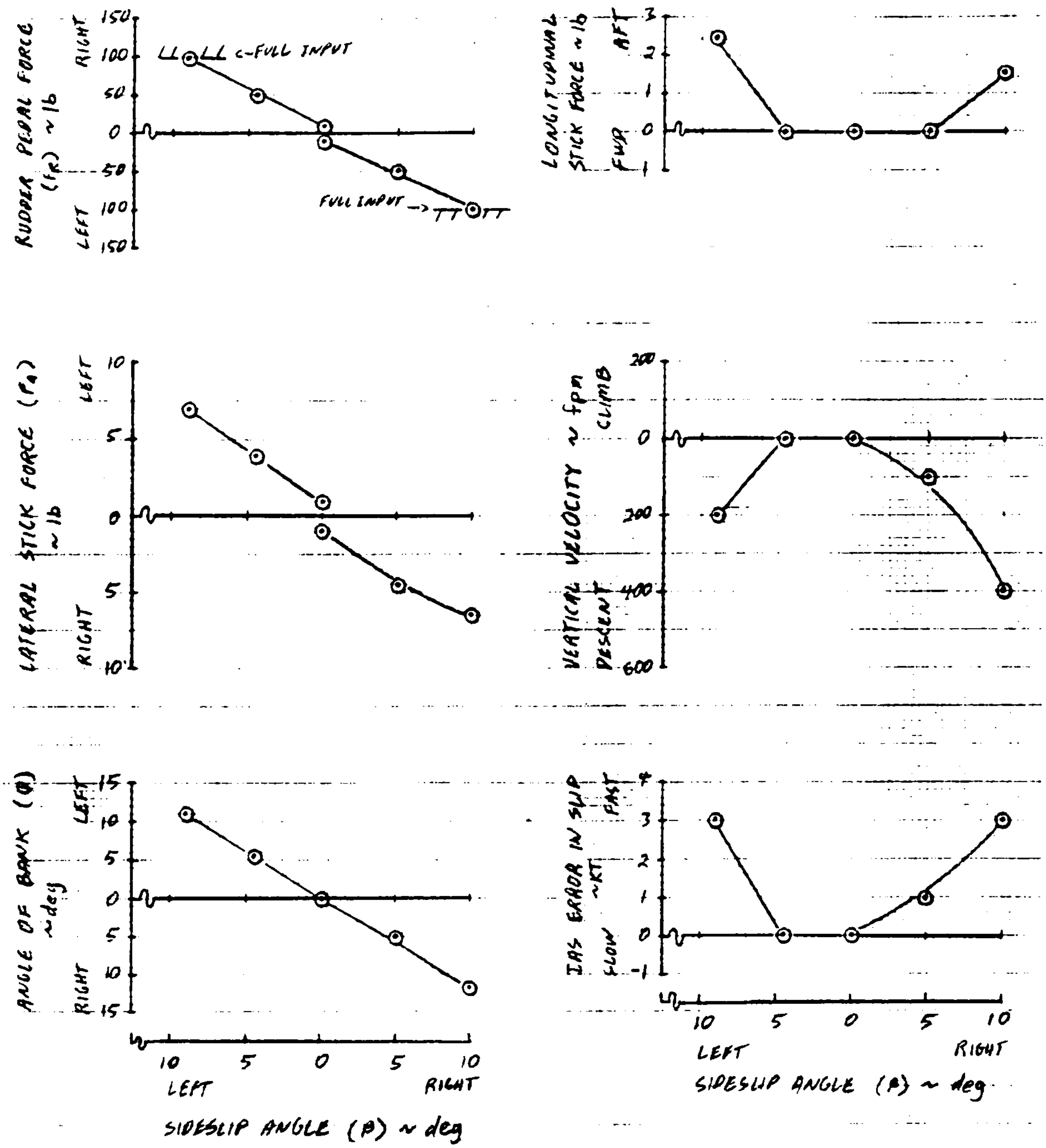


Figure 7
STEADY HEADING SIDE SLIP
(Configuration POWER APPROACH)

LOADING: DESIGNATION A METHOD: STEP FORCE INPUT
DATA: HAND HELD

SYMBOL	CONFIG.	CONTROL INPUT	GROSS WEIGHT	CENTER OF GRAVITY	PRESSURE ALTITUDE
○	CRUISE	FULL	18,600 to 21,700 lb	33.4 to 36.4 % MAC	15,000 ft
△	CRUISE	FULL	21,100 lb	34.1 % MAC	41,000 ft
□	CRUISE	PARTIAL	20,300 lb	35 % MAC	15,000 ft
⬡	POWER APPR.	FULL	18,700 lb	33.7 % MAC	9,000 ft

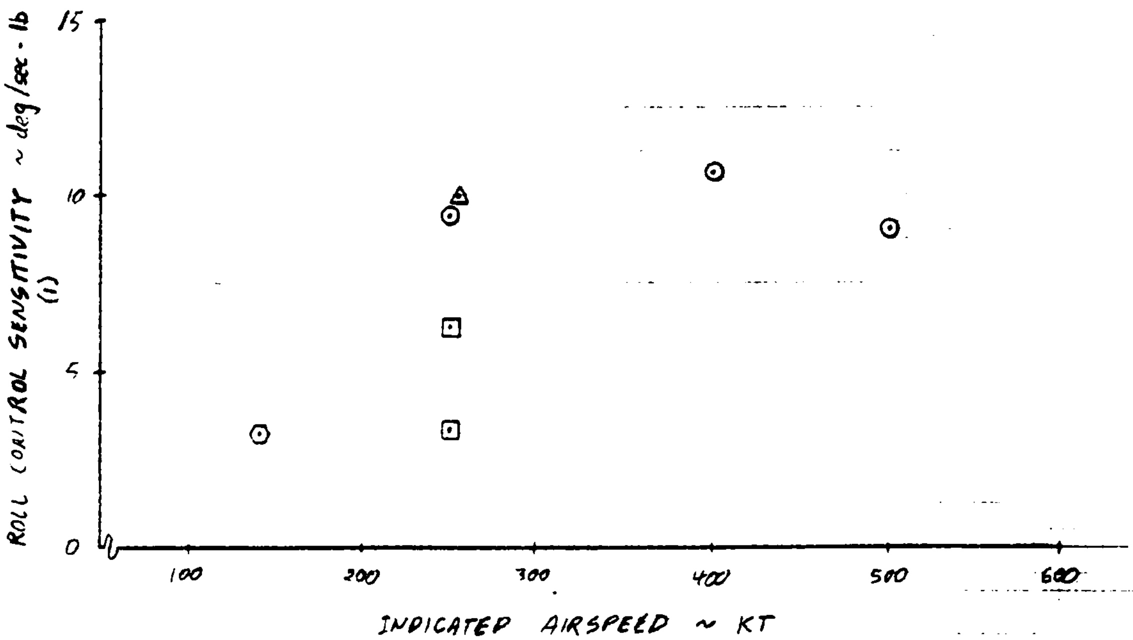
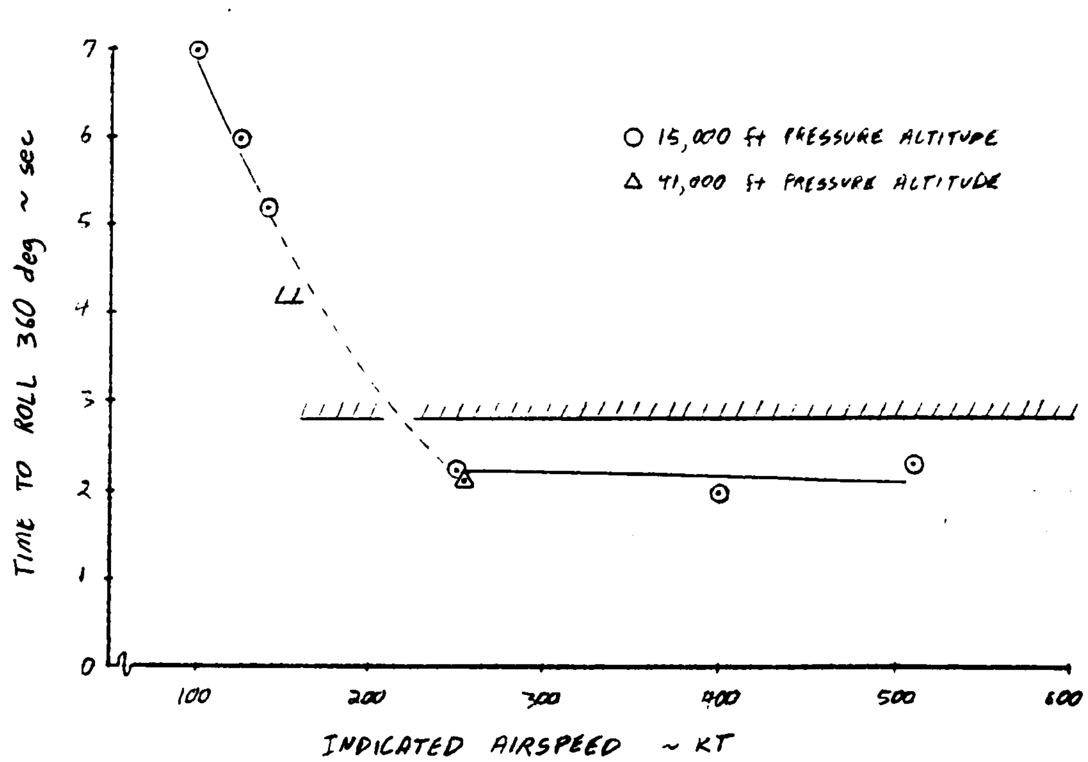


Figure 8
ROLL SENSITIVITY

CONFIGURATION: CRUISE
LOADING: DESIGNATION A
DATA: HAND HELD

METHOD: FULL DEFLECTION STEP
GROSS WEIGHT: 15,600 to 21,700 lb
CENTER OF GRAVITY: 36.4% MAC



CONFIGURATION: POWER
LOADING: DESIGNATION A
DATA: HAND HELD
METHOD: FULL DEFLECTION STEP

PRESSURE ALTITUDE: 15,000 ft
INDICATED AIRSPEED: 400 KIAS
GROSS WEIGHT: 21,700 lb
CENTER OF GRAVITY: 35.6% MAC

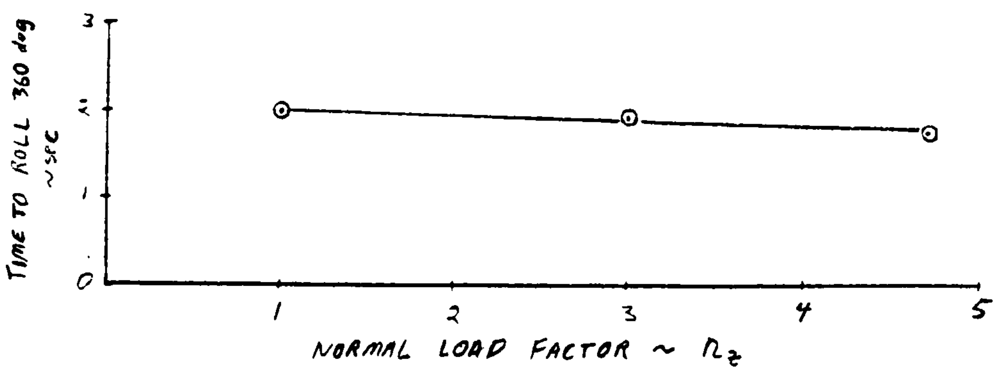


Figure 9
ROLL CONTROL EFFECTIVENESS

LOADING: DESIGNATION A
DATA: HAND HELD
METHOD: STEADY HEADING SIDE SLIP

GROSS WEIGHT: 18,500 to 20,400 lb
CG: 33.5 to 36.6% MAC
CONFIG. CR, 15,000 ft PRESS. ALT.
CONFIG. PA 9,000 ft PRESS. ALT.

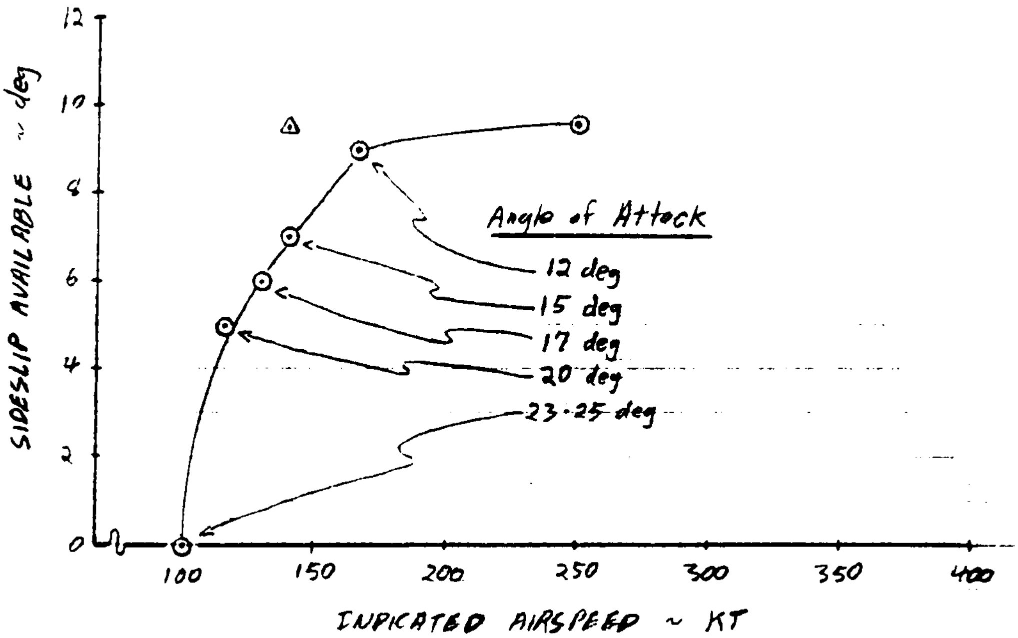


Figure 10
DIRECTIONAL CONTROL EFFECTIVENESS

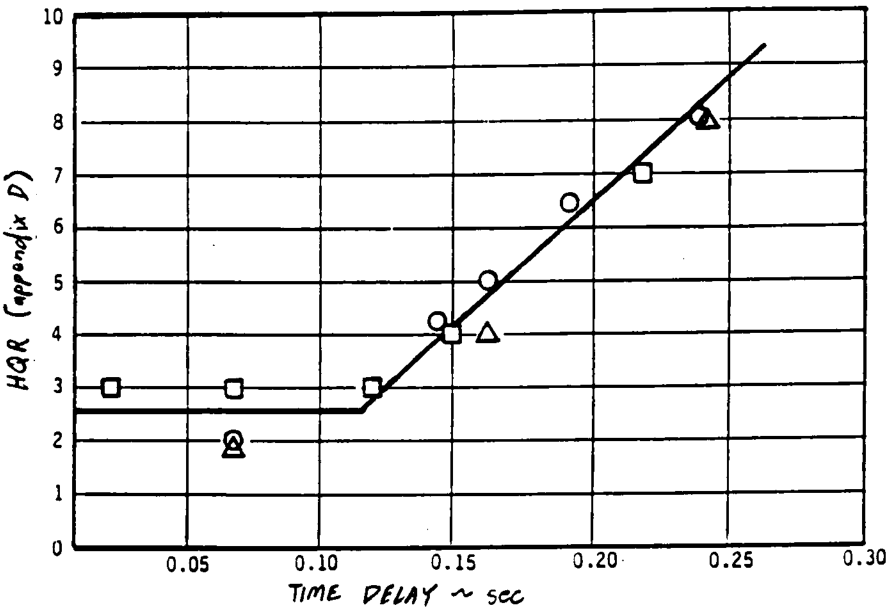


Figure 11
DELAY EFFECTS ON LONGITUDINAL FLYING QUALITIES DURING LANDING [2]

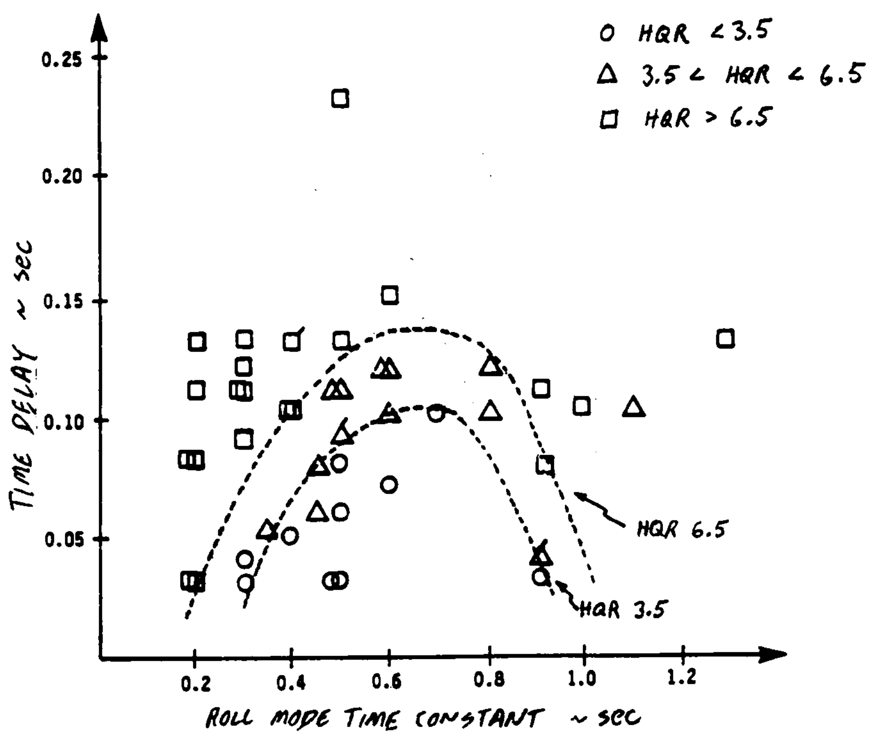


Figure 12
DELAY EFFECTS ON LATERAL FLYING QUALITIES DURING A/A TRACKING [2]

VITA

David Craig Dykhoff was born August 29, 1956 in Baltimore, Maryland. He grew up in West Virginia, where his parents still reside. Graduating in 1978 from Case Western Reserve University in Cleveland, Ohio with a Bachelor of Science in Electrical Engineering, he took a position with Boeing Aerospace in Seattle, Washington as a circuit design engineer.

After a short time with Boeing, Mr. Dykhoff joined the Navy and received a commission in 1980. Following flight training, he spent 18 months as a flight instructor in Kingsville, Texas. He was then assigned to Fighter Squadron 154 flying the F-14 Tomcat, during which time he made several deployments including an extended period in the Western Pacific and North Arabian Sea.

LT Dykhoff was then selected for USN Test Pilot School which he completed in December of 1987. He currently is stationed at the Naval Air Test Center where he evaluates the latest models of the F-14.