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I am submitting herewith a thesis written by Robert A. Coughlin entitled "Developing a Case for the Implementation of a Control Augmentation System to Improve Flying Qualities in the EA-6B Aircraft." I have examined the final copy of this thesis for form and content and recommend that it be accepted in partial fulfillment of the requirements for the degree of Master of Science, with a major in Aviation Systems.

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DEVELOPING A CASE FOR THE IMPLEMENTATION OF A CONTROL AUGMENTATION SYSTEM TO IMPROVE FLYING QUALITIES IN THE EA-6B AIRCRAFT

A Thesis

Presented for the

Master of Science

Degree

The University of Tennessee, Knoxville

Robert A. Coughlin

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DEDICATION

This thesis is dedicated to my wife,

Jill Marie Coughlin,

whose support has been instrumental over the last thirteen years. Additionally, I would like to dedicate this thesis to my Brother-in-Law,

Ronald Bixman,

who passed away shortly after the this thesis was completed. At every important crossroad of my life, he was there to be a part of it. He was always asking what was next, and always encouraged me to never settle for "just enough". He is an inspiring man who I learned many things from both his life, and his death.

ACKNOWLEDGMENTS

My sincere appreciation goes out to all of the professional Naval Flight Officers and Naval Aviators whose inputs and feedback made this thesis possible. Their ideas, opinions, and knowledge assisted greatly in keeping the scope of this project within the bounds of reality.

I would also like to thank the instructors at the University of Tennessee Space Institute and the United States Naval Test Pilot School for their assistance in earning this degree.

Lastly, I would like to extend my gratitude to my children, Taylor, Sydney and Wyatt, and my wife Jill for their love and support throughout the entire experience. Their support and encouragement has meant the world to me.

ABSTRACT

The EA-6B aircraft was designed and built in the late 1960s by the Grumman Aerospace Corporation for the United States Navy and Marine Corps to be used as a tactical electronic warfare (EW) platform. High losses of U.S. attack aircraft to surface-to-air missiles (SAMs) in Southeast Asia led to the requirement for a carrier-based tactical aircraft capable of providing EW support in the form of electronic jamming in support of strike aircraft. The EA-6B became the aircraft that fulfilled the EW requirement. Forty years have passed since the introduction of the EA-6B and the demand for the tactical EW capability continues to increase. Meeting these requirements has taken a toll on the aircraft's wing fatigue life, and has produced a shortage of the aircraft's Automatic Flight Control System (AFCS) computer.

The replacement computer contracted to correct the current problem of the aging analog AFCS computer is an up-to-date digital flight control computer (Model EA-6B Aircraft Program, 1995). The implementation of this computer will allow greater processing power and should be used to the full extent of its capability. An up-to-date flight control computer will cost less to repair, be readily replaceable, and will improve mission readiness and safety of flight.

This thesis will build a case to use the additional processing power of the replacement digital flight control computer to improve flying qualities through a Control Augmentation System (CAS) that should have the effect in the following areas:

1. Limit accelerations during flight and thereby control the current problem of Fatigue Life Expenditure (FLE).
2. Improve flying qualities, specifically in the landing configuration with an emphasis to improve shipboard operations. With the current SAS the aircraft demonstrates less than satisfactory flying qualities in all axes (lateral, directional, and longitudinal) and configurations.
3. Limit Angle of Attack (AOA) in certain abnormal configurations, that are commonly encountered in the EA-6B, due to the aging airframe. The configuration that is particularly dangerous is the no slat-flaps down approach. This approach requires the pilot to maintain AOA precisely. The consequences of slowing one degree (equivalent to approximately 2 KIAS) below the recommended approach AOA will result in an uncontrollable pitch up (longitudinal instability) and subsequent departure from controlled flight (mandatory ejection criteria in this configuration) (EA-6B NATOPS Flight Manual, 1997).
4. Improve departure resistance by limiting angle of attack and controlling side-slip. Jamming pods are on all wing stations when the airplane is in the normal configuration. Departures that develop into spins frequently result in loss of the aircraft due to large inertia forces, caused by the weight of the stores carried on the wings stations, which can not be overcome by the aerodynamic forces required for recovery.

PREFACE

The information contained within this thesis with respect to control strategies was obtained from an historical research of control law implementation in previous aircraft. References to EA-6B handling qualities were obtained during actual flying qualities test flights performed by the author, and developmental test flights performed at the Naval Air Warfare Center Aircraft Division. The handling quality and stability flight tests were not performed specifically for the Control Augmentation System development effort but the data obtained during these tests were not affected. However, the flight control description, contained in Chapter 2 and appendix A, was developed in support of the control augmentation system development. The research, results and conclusions, and recommendations presented are the opinion of the author and should not be construed as an official position of the United States Department of Defense, the United States Navy, the Naval Air Systems Command, or the Northrop Grumman Corporation.

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LIST OF ABBREVIATIONS

AC	Alternating Current
ACLS	Automatic Carrier Landing System
ACT	Actuator
A/D	Alternating Current to Direct Current
ADC	Air Data Computer
ADI	Attitude Direction Indicator
AFC	Air Frames Change
AFCS	Automatic Flight Control System
AGL	Above Ground Level
AHRS	Attitude Heading Reference System
ALT	Altitude
ANC	Air Navigation Computer
AOA	Angle of Attack
APC	Approach Power Compensator
ARI	Attitude Reference Indicator
ASR	Assist Spin Recovery
ATTD	Attitude
CADT	Course Attitude Data Transmitter
CAINS	Carrier Aircraft Inertial Navigation System
CAS	Control Augmentation System
CR	Cruise Configuration
DC	Direct Current
DFCC	Digital Flight Control Computer
DFCS	Digital Flight Control System
DSDC	Digital Signal Data Converter
EADI	Electronic Attitude Direction Indicator
EA	Electronic Attack
ECMO	Electronic Countermeasures Officer

EFIS	Electronic Flight Instrumentation System
EGI	Embedded GPS/INS
EHSI	Electronic Horizontal Situation Indicator
EP	Electronic Protection
ES	Electronic Surveillance
EW	Electronic Warfare
FBW	Fly By Wire
FCC	Flight Control Computer
FLE	Fatigue Life Expenditure
FLT	Flight (hydraulics)
HDG	Heading
HQR	Handling Quality Rating
ICAP	Improved Capability
ICLS	Instrument Carrier Landing System
INS	Inertial Navigation System
IPI	Integrated Position Indicator
KIAS	Knots Indicated Airspeed
LAT	Lateral
LED	Leading Edge Down
LEU	Leading Edge Up
LONG	Longitudinal
MSL	Mean Sea Level
NATOPS	Naval Air Training and Operating Procedures
NWS	Nose Wheel Steering
PA	Power Approach
PIO	Pilot Induced Oscillation
PRI	Primary
RAT	Ram Air Turbine
ROE	Rules of Engagement

SAS	Stability Augmentation System
SAM	Surface-to-air Missile
SEAD	Suppression of Enemy Air Defense
SEC	Secondary
STAB AUG	Stability Augmentation
T/R	Transformer Rectifier
UHF	Ultra High Frequency
USAF	United States Air Force
VAC	Volts Alternating Current
VDC	Volts Direct Current
VERT REF	Vertical Reference

CHAPTER 1: INTRODUCTION

BACKGROUND

The EA-6B Prowler is the only airborne tactical electronic warfare (EW) platform flying today in support of the United States Armed Forces. Until 1998 the United States Air Force (USAF) operated a fleet of EF-111A Raven aircraft that was also capable of providing tactical EW support. With the retirement of the EF-111A, the capabilities of the aging EA-6B have become more important than ever before. The retirement of the EF-111A and the increased global instability has created the largest demand for EW support in our history. This demand has taken a toll on the airplane and the components that are required for basic airworthiness. Specifically, Fatigue Life Expenditure (FLE) with respect to the center wing section has exceeded initial design requirements which will result in the majority of airframes reaching 100% FLE by 2005. The problem of limited fatigue life has been compounded by Congress demanding that the EA-6B maintain responsibility for the EW mission until 2015. Additionally, the current flight control system computer has recently come under great scrutiny due to the high cost of repair, shortages of replacement units, and a recent rash of non-pilot-initiated flight control movements which may have been a causal factor in the recent loss of an aircraft. An effort to improve aircraft reliability and flight safety has been focused on replacing the current analog flight control computer with an up to date digital Flight Control Computer (FCC) (Model EA-6B Aircraft Program, 1995).

This thesis will build a case to implement a Control Augmentation System (CAS) which will provide the capability to control FLE, improve mission readiness and safety, and improve the aircraft's flying qualities.

STATEMENT OF THE PROBLEM

The importance of this aircraft has resulted in it being added to the highly distinguished list of "National Assets". Current Rules Of Engagement (ROE) require the EA-6B not only to be airborne, but it must be established on station before any other aircraft may proceed into an area where a SAM battery may exist. To fully understand the need for the CAS, the problem statement must be broken down into four areas, namely, mission readiness, safety, flying qualities improvement, and FLE.

Mission readiness has been impacted by the failure of the current ANC, and has been identified as the number one impact on EA-6B mission readiness (COMVAQWINGPAC, DTG 271819 Z Sep 00). Without the EA-6B airborne, no other aircraft today are allowed to fly in an area of hostility or where the possibility of hostility exists. To emphasize this point, "no aircraft" implies all aircraft from all U. S. services (Army, Navy, Airforce, Marines) that are involved in any mission. Additionally, "no aircraft" implies all NATO aircraft that are flying in support of U. S. operations. Therefore, mission readiness has a much larger implication when it is known that no aircraft can attack targets without the EA-6B airborne. The U. S. fighter-attack aircraft total inventory consists of approximately 5,000 aircraft. NATO forces aircraft total approximately 2,000. The current inventory of the EA-6B, a mere 100, is required to support any one of 7,000 strike aircraft. The numbers speak for themselves and the new flight control computer is needed to ensure the mission readiness of the entire force.

Safety must be addressed with respect to AFCS due to an increase in the amount of reports of non-pilot induced flight control inputs. The author has actually been at the controls of an EA-6B when the aircraft began rolling right and left lateral stick inputs did not affect the right roll. Control of the aircraft was regained by actuating the AFCS emergency disconnect to remove the AFCS from the control loop. The seriousness of the any aircraft responding differently than commanded does not need further explanation as to what the outcome may be, but it is hypothesized that the recent loss of an EA-6B may have been caused by this very problem. A well developed CAS is needed to eliminate aircraft attrition caused by AFCS malfunctions.

Flying qualities improvement is a hard case to sell when the aircraft has been flying this way for forty years. A case can be built for CAS as a flying qualities improvement project by saying that the CAS will reduce the time to train new aviators. The EA-6B maintains the highest percentage of student disqualification during initial carrier qualifications (Lucas J. B., 2000). An attempt to correct this problem was initiated by sending only these pilots that had the best landing grades during the advanced jet training phase to become EA-6B fleet replacement pilots . However, the disqualification numbers still remain high. This high disqualification rate has required that 40% of the training a new EA-6B pilot receives is focused on landing the airplane. The carrier qualification stage of training where the student does nothing but land the airplane requires 2 long months out of the total of 6 months of training. Additionally, if the student disqualifies at the carrier the time to finish this training is normally doubled, resulting in 6 more months of training with the hope that the extra training will prevent the student from disqualifying a second time. The author has performed a flying qualities evaluation and the results are unsatisfactory for the training and the EW mission. Additionally, the initial developmental test report also cited unsatisfactory flying quality results (Luecke M. A., 1971). The CAS will improve flying qualities and thereby reduce the time and cost to train student Naval Aviators. The emphasis of this thesis is to build a case that a CAS system is needed to correct flying quality deficiencies that produce level II flying qualities.

The problem of limited wing fatigue life is a direct result of Congress not appropriating money for the development of a replacement platform. In other words, the design fatigue life for an airplane that was supposed to be in service for 30 years has fulfilled the original design objective. The result of no replacement airplane in development brings the need of operating this aircraft for 25 years more than the original design requirement. In an attempt to achieve this new objective, administrative load factor limitations have been placed on the airplane which are less than the design load factor limitations. The EA-6B FLE problem has become such an important issue that the pilots flying the airplane are very cautious not to exceed the administrative limits, so cautious that in the opinion of the author, pilots are not training effectively and as a result are more susceptible to the threats which require dynamic flight maneuvers to defeat.

The CAS will allow flight control strategies to be implemented that will allow limiting the load factor that the pilot can command. This will allow pilots to train more effectively and control the problem of FLE at the same time.

MISSION OF THE EA-6B AIRCRAFT

The mission of the EA-6B aircraft is to provide airborne tactical EW in support of air or ground operations with the purpose of denying, degrading, or destroying enemy radar and communications systems. However, this mission description does not explain what the airplane has to be capable of to be effective. It is the author's opinion that the EA-6B community and the people responsible for the acquisition of weapons systems to accomplish this mission have not taken the airplanes' capabilities into account when planning for the future. The emphasis has always been on improving the mission systems of the aircraft, not improving the aircraft. Accomplishing this mission requires the EA-6B to be well within range of Surface to Air Missiles (SAMs), and enemy fighter aircraft. As a result of these threats, the pilot must train to or actually defend against these threats. This requires dynamic maneuvering, and elevated load factors and angles of attack. Additionally, the airplane is operated in the most demanding environment that an aircraft can encounter during the landing on an aircraft carrier. The task of landing aboard an aircraft carrier results in pilot workload so high that any pilot compensation for poor handling qualities is unacceptable. With a reduced capability of defending against threats or landing the aircraft after the mission, the overall mission capability of this platform is reduced.

CHAPTER 2: REVIEW OF THE LITERATURE

GENERAL

The invention of digital data transmission across data busses and the incorporation of sophisticated high speed computers into avionics architecture has provided the opportunity for new methods of controlling aircraft. However, these "new methods" of controlling aircraft are not immune to failure and subsequent redesign due to surprises found during actual flight test. As flight control systems have become more complex, the open-loop aircraft response have been found to be completely different than the closed loop system, which contains the pilot as an operator. With the possibility of differences between the open loop and pilot in the loop response following control law development produces risk which must be understood. As a result of these differences of aircraft response caused by the pilot in the loop, extensive studies and writings have been completed in the area of Handling Qualities Ratings (HQR). A well developed HQR scale is needed to accurately identify flying quality deficiencies. The HQR rating scale that the author is most familiar with and will be discussed herein is the Cooper-Harper rating scale. A description of a CAS, various control strategies, and what mechanization may be used to implement the control strategy will be discussed. Finally, a complete flight control description of the EA-6B has been developed and contained in appendix A to provide a starting point for control system engineers to help with developing a plan for a CAS implementation. A portion of the flight control system description will be presented in this section to show that the CAS may be implemented by using the SAS actuators and the series control system already in place.

EVOLUTION OF AIRCRAFT AUGMENTATION

Early in the historical development of aircraft, when a design exhibited stability or control deficiencies, attempts were made to correct them by modifying the external shape or size of aircraft components, such as tails, wings and control surfaces, or by adding control system "gadgetry" such as bob-weights, springs, and dampers. This philosophy continued to the later part of World War II when the technologies needed to augment the aircraft's basic dynamics were developed sufficiently to allow safe implementation. As an example, various forms of the flying wing from Northrop's YB-49 had some deficient flying qualities which decreased its effectiveness as a bombing platform. One of the augmentation devices used in the development of the YB-49 included fully powered, irreversible control surfaces. However, this presented more problems by eliminating the feedback to the pilot of the actual surface movement. To provide this aerodynamic feel to the pilot, an artificial feel system was incorporated which changed its characteristics with airspeed. This was accomplished by means of a Q-bellows system where dynamic pressure expanded or contracted a bellows which in turn moved a lever that altered command gearing. The end result was a change of control feel to the aircraft response to give the pilot a feel of the changing airspeed (Neal T. P. and R. E. Smith 1970). The lesson to be learned here is that the Q-bellows system was developed as a result to fix the problem that fully powered irreversible control system presented with respect to pilot feel. This "gadget" is more complicated than a bob-weight, but just as the Q-bellows system, the bob-weight is usually incorporated well after the aircraft has flown and is primarily used to change the control feel to correct or improve a deficient flying quality.

These early augmentation concepts were extended to the development of the Stability Augmentation System (SAS), which is currently the only augmentation implemented in the EA-6B. Early attempts at jet fighter designs, which exploited the power of the jet engine, displayed many undesirable handling quality characteristics that had to be overcome by pilot skill, minimization of exposure (i.e. accelerating quickly through the transonic region), or avoidance of certain flight conditions (i.e. high angle of attack). The inability of the pilot to avoid these characteristics sometimes resulted in the loss of both the pilot and the aircraft. To improve the aircraft's flight characteristics without significantly altering the external shape of the aircraft, powered control surfaces were employed to allow the modification of the aircraft's motions without depending entirely on the pilot. This augmentation required the addition of sensors to detect aircraft motion, and the use of these measurements to alter the surface's movement without the pilot's perception. These systems, in the form of pitch and yaw dampers, were the earliest forms of the SAS employed on some of the Century series fighters, the F-4, F-5, A-4 and still in use today in the EA-6B (Hutchinson, K. T. 1999).

The SAS ultimately proved to be of limited utility due to its low authority with respect to the pilot's commands. This necessitated the development of higher authority systems, such as CAS, which not only augmented the basic aircraft's stability through the use of feed-backs to the surfaces, but augmented the pilot's inputs to improve his ability to command the aircraft. The CAS was very successful in the F-15, and YF-17, however; they did require a complex mechanical system. A natural evolution of this design was to replace the mechanical systems with electrical hardware. This led to the development of the Fly-By-Wire (FBW) implementation of control systems. By replacing the mechanical linkages with electrical signals, a reduction in weight, size and complexity was achieved. In addition, the power of these high authority control systems provided a large amount of flexibility to alter the basic characteristics of the aircraft.

CONTROL AUGMENTATION PROJECT CASE STUDY

This section cites several examples of highly augmented aircraft designs which experienced handling qualities deficiencies early in their flight test programs. The author's intent of citing these examples is to provide examples of instances in which aircraft have exhibited flight control system design deficiencies, and hopefully illustrate the fact that handling quality problems have not been isolated events nor is their existence intuitive to designers during the developmental phase. Flight control design and the accompanying handling characteristics are fairly mundane topics, and it is often high performance and state-of-the-art avionics breakthroughs that sell aircraft. For the purpose of this discussion, the following examples have shown that most new designs have experienced incidents or accidents caused by unforeseen deficiencies in the closed loop design. These examples were chosen randomly to emphasize the risk that is associated with modifying the EA-6B's control system characteristics to achieve a change in the resultant aircraft closed loop dynamics.

EFFECT OF TIME DELAY

Time delay between a pilot input and the aircraft response is a very insidious characteristic that can be disastrous. This characteristic is insidious due the small amount of time delay that is required to produce adverse handling qualities. Time delays as small as 100 milliseconds have resulted in PIOs during high gain tasks such as in-flight-refueling and carrier landings (Tischler M. B. 1990). Time delays are not only found in augmented or FBW systems, they may also be found in purely mechanical systems where actuator response is sluggish. Additionally, effects of time delay can be pilot dependent due to the technique (i.e. aggressiveness) used during the task.

The YF-17 was Northrop's entry into the United States Air Force (USAF) light weight fighter competition in the early 1970's. The YF-17 pitch control system consisted of a mechanical connection between the stick and the horizontal tail actuator. In addition to the mechanical linkage, a CAS actuator was also used to augment the aircraft's response by commanding an electrical signal to the horizontal tail. This CAS actuator had a limited authority when compared to the authority of the control stick. An electrical measurement of the pitch stick position was used to augment the feedback commands, and these combined electrical signals were sent to the CAS actuators. The open loop response to a step input as shown below in figure 1 would indicate that this arrangement would be sufficient.

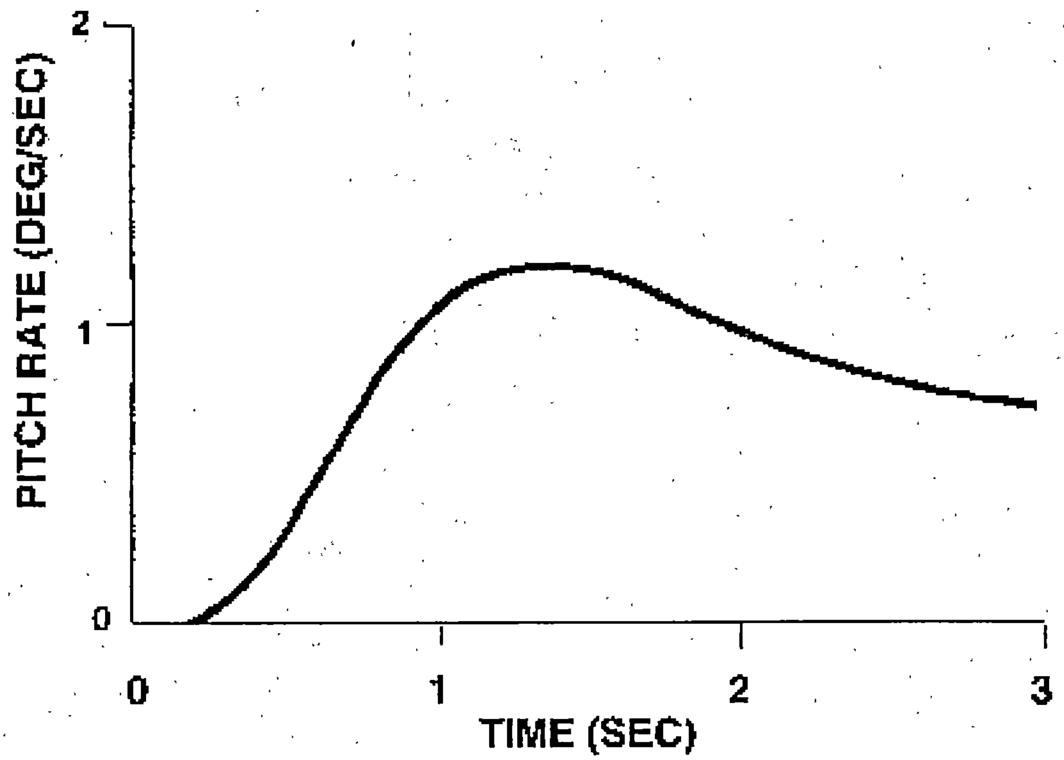

Figure 1: YF-17 Open Loop Pitch Rate Response

Source: Hutchinson, K. T., Augmented Aircraft Handling Qualities, Buffalo New York, Flight Research Group, Veridian Engineering, 1999.

However, by examination of figure 2 it can seen that the closed loop response was much different and continued to get worse as the aircraft proceeded closer to the ground during landing. This arrangement required the addition of prefilters into the pilot's command path in order to modify the aircraft's resultant response. The prefilter used in the YF-17 produced an excessive time delay of 200 milliseconds between a step input and the pitch rate response. Prior to the first flight this configuration was evaluated during approaches and landings in the USAF NT-33A in-flight simulator. Cooper-Harper ratings of 9 and 10 (10 being uncontrollable) were obtained and Pilot-Induced-Oscillations (PIO) were experienced by all pilots. Following this event the prefilter was removed and replaced

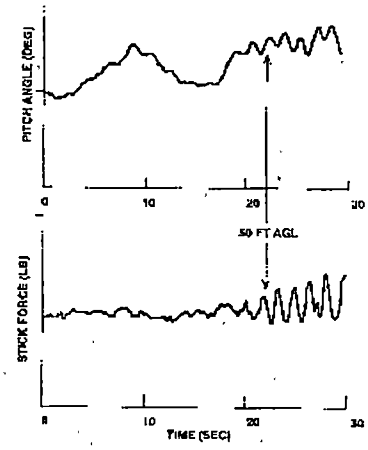

Figure 2: YF-17 Closed Loop Pitch Rate Response

Source: Hutchinson, K. T. (1999), Augmented Aircraft Handling Qualities, Buffalo New York, Flight Research Group, Veridian Engineering.

with another less complex filter. As a result the time delay was significantly reduced and Cooper-Harper pilot ratings were reduced to 2.

Similarly, the F/A-18 was a derivative of the original YF-17 aircraft, and developed in response to the U.S. Navy's Naval Advanced Combat Fighter (NACF) requirement. This aircraft was a joint development effort by McAir and Northrop. As part of the development concept it was decided to incorporate fly-by wire technology, and to further push this technology to include the use of a digital processor, rather than an analog computer, as was used in the YF-16. At this time, experience with digital FBW systems was limited, and the F/A-18 development laid the ground work for subsequent digital systems. However, early versions of the F/A-18 were plagued with flying quality problems during high gain tasks such as in-flight refueling and carrier landings which were a direct result of time delay. The contributing factor that led to the time delays were again caused by pre-filters in the control stick command path (Hutchinson, K. T. 1999).

Finally, the space shuttle flight control system was developed using a time history boundary that placed an upper and lower limit on the vehicles pitch rate response. This criterion allowed considerable time delay (200 milliseconds) in pitch rate response which was employed to satisfy a pitch overshoot requirement spelled out in the flying quality specification. This is a prime example of design engineers not understanding the fundamentals of the closed loop response. This time delay contributed to the Pilot Induced Oscillation (PIO) that occurred on the 5th flight which nearly resulted in the loss of the prototype (Hutchinson, K. T. 1999).

EFFECT OF RATE LIMITING

When a control actuator is commanded to move faster than it is capable of moving, that component is said to be moving at its rate limit. Augmented aircraft are more vulnerable to rate limiting problems due to the fact that the surfaces are being driven harder. They are no longer being commanded by the pilot alone, but are also being commanded by feed-backs and FCCs.

The JAS-39 Grippen is the latest Swedish fighter developed by Saab-Scania Aircraft Company. This aircraft is a delta winged aircraft with a closely coupled canard. The development of the flight control system occurred exclusively during ground simulation. As a result, the control of the resulting system used for early flights required considerable pilot compensation. Several problems existed with the first flight version of the control laws, and including, the use of low rate limits in the pilot's command path to prevent the pilot from demanding more rate than the feed-backs required; considerable lateral motion resulting from lateral gusts, an excessively high augmented roll mode time constant and short period responses, and the locking out of pilot commands in one axis when the other axis demanded a control surface rate exceeding the capability of the actuators. The last feature was the major contributing factor to an accident that destroyed the first prototype, as shown below in figure 3. As a result of a rate limited surface during the sixth flight the pilot's lateral control was cut out and resulted in loss of roll control and subsequent out of control flight into the terrain.

Additionally, the YF-22 was destroyed during the later stages of the demonstration/validation flight test program due to a combination of rate limit and pitch command gain being abruptly changed with the movement of the landing gear. The incident began when the pilot retracted the landing gear, which increased the stick sensitivity and activated the thrust vectoring. During this occurrence close to the ground, the horizontal tail surfaces began to move at their rate limit. The occurrence of the rate limiting and the large increase in pitch command sensitivity caused by the gain change and thrust vectoring combined to cause the pilot to couple with the motions of the aircraft and perpetuate the rate limiting of the control surfaces as shown below in figure 4. The pilot interpreted the resulting oscillations as a failure of the flight control system and elected to let the aircraft to pancake onto the runway (Hutchinson, 1999).

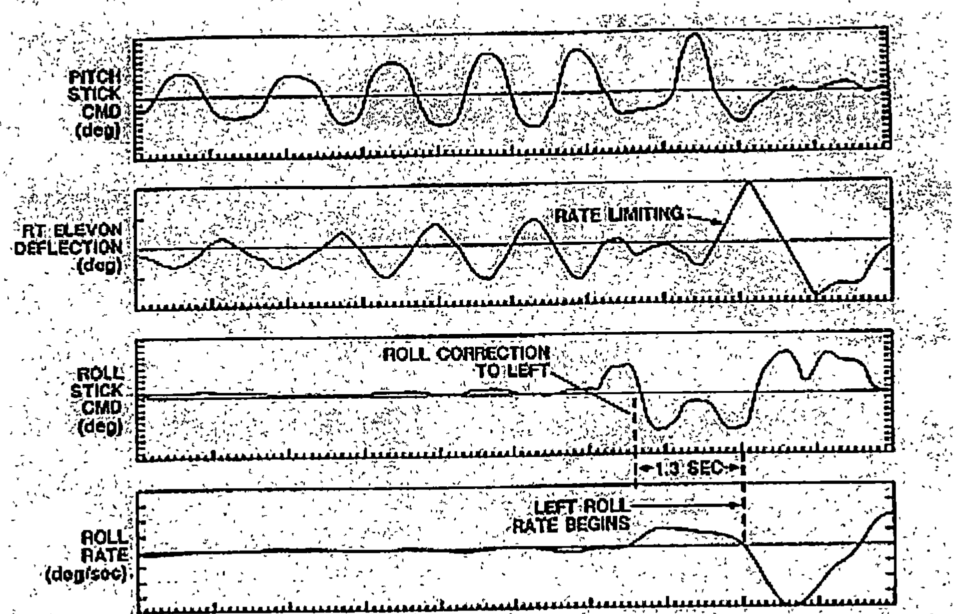

Figure 3: Time History of the Loss of Control of the Grippen

Source: Hutchinson, K. T., Augmented Aircraft Handling Qualities, Buffalo New York, Flight Research Group, Veridian Engineering, (1999).

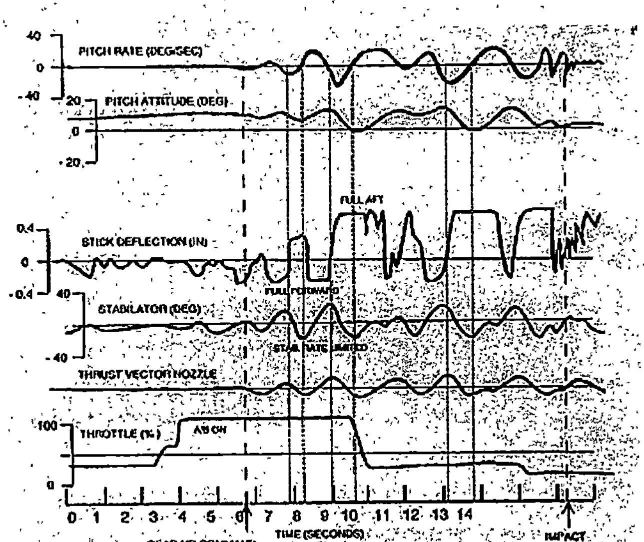

Figure 4: Time History of YF-22 Loss of Control Event

Source: Hutchinson, K. T., Augmented Aircraft Handling Qualities, Buffalo New York, Flight Research Group, Veridian Engineering (1999).

CONTROL AUGMENTATION SYSTEM

The Control Augmentation System was the natural follow-on to the Stability Augmentation System (SAS) during the development of augmented flight control systems. The main reason for the development of the CAS was due to the development of less stable (i.e. more maneuverable) aircraft where the augmented system required more authority to produce the desired aircraft dynamics. The EA-6B has a large stability margin and is maneuverable, however not nearly as nimble as a current day fighter, so why implement a CAS in the EA-6B?

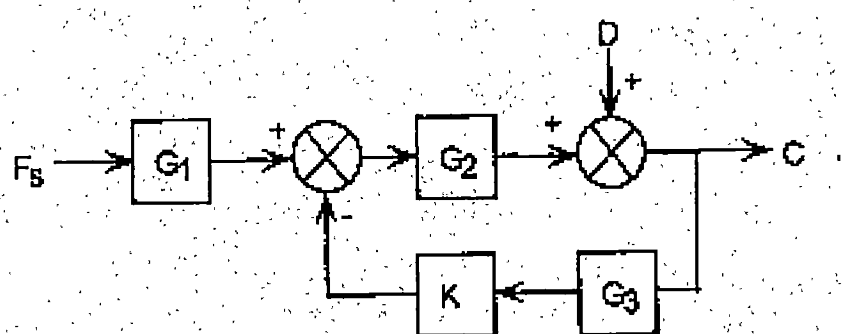

Figure 5: Stability Augmentation System Block Diagram

Source: Hoh R. H and Mitchell D. G., A Functional Requirement for the Flight Control System, Bristol Pennsylvania, Taylor and Francis Inc, (1996).

To answer this question the simplified control system block diagrams of a SAS and a CAS, and the output due to pilot input transfer functions may be compared. Figure 5 represents a simplified SAS block diagram. Where C is the closed loop response of the aircraft, F_S represents the pilot's input, D is an external disturbance to the aircraft, K represents the feed back gain, G_1 represents any stick gearing and dynamics, G_2

represents the dynamics of the main power actuator, control surface and aircraft, and G_3 represents a combination of any feedback compensation. The feed back gain (K) can be fixed as in simple dampers or can vary with flight conditions such as a Q-Bellows system as described earlier. The characteristic of this block diagram the author would like to bring attention to is the output due to pilot input transfer function as shown below in equation (1).

$$\frac{C}{Fs} = \frac{G_1 G_2}{1 + K G_2 G_3} \qquad \text{(Equation 1)}$$

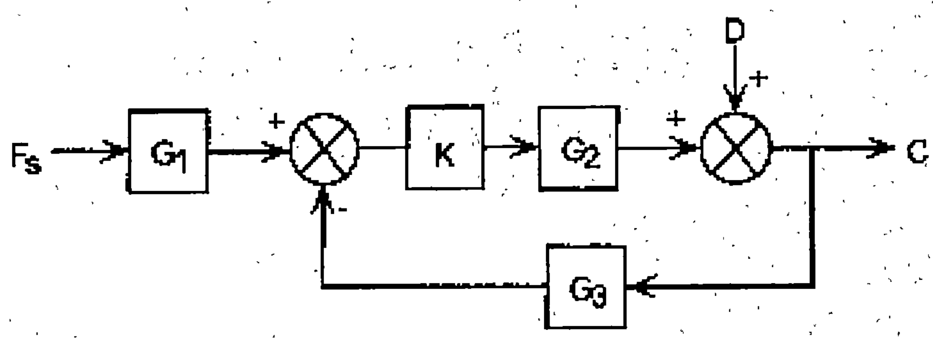

Figure 6: Control Augmentation System Block Diagram

Source: Hoh R. H and Mitchell D. G., A Functional Requirement for the Flight Control System, Bristol Pennsylvania, Taylor and Francis Inc, (1996).

If the value of gain K were increased, the transfer function would decrease, which means the system would respond less to the pilot's inputs as the gain is increased. The CAS can be represented by the block diagram as shown below in figure 6. In this figure, C is the closed loop response, D is the external disturbance, K represents the error feedback gain, G_1 represents the stick gearing and dynamics as well as any feedforward dynamic compensation, G_2 represents the actuator, surface and aircraft dynamics, additional compensation (i.e. lead/lag filters) necessary for the error signal, and G_3 represents the combination of the sensor dynamics. The output due to pilot input transfer function is shown below in equation 2.

$$\frac{C}{Fs} = \frac{KG_1G_2}{1 + KG_2G_3} \qquad \text{(Equation 2)}$$

Again, increasing the value of K, now results in the transfer function approaching a value of G_1/G_3, not zero. The result is a masking of the aircraft's dynamics so the resulting response looks more like the combination of the control system characteristics. The advantage of the CAS over the SAS is that the error feedback gains must be kept small in the SAS system or the pilot's command authority will be reduced. While CAS gains can be increased to desired levels. The fact that the feedback gains must be kept small in the SAS is obvious when examining the EA-6B flight control system. In the longitudinal control axis alone there are four bob-weights at various locations, three feel spring mechanisms, and an eddy current damper. It is the author's opinion that during the design of this airplane, a control system engineer could not have known ahead of time that all these "gadgets" would be needed, and therefore they were implemented due to reports of less than desirable handling qualities. These "gadgets" help to change the control feel which then causes the pilot to shape the control input, but the handling qualities that brought these "gadgets" into the control system are still there, just less noticeable (Tischler M. B., 1990).

CONTROL LAW STRATEGIES

There are many control strategies utilized on aircraft equipped with a CAS. Some longitudinal examples include Angle of Attack (AOA) command, rate command, attitude command, and "g" command or load factor command. In general these systems use a combination of feed-backs and control law mechanization to achieve the desired dynamic characteristics.

Control law strategies each have their advantages and disadvantages. For example, the F/A-18 is a "g" command system in the cruise configuration (CR). A characteristic of this system is an auto trim system where changes in airspeed result in no change in pitch trim (i.e. zero stick position corresponds to 1 g). An obvious advantage of this system is that the pilot can begin an acceleration to supersonic speeds to effect an air to air intercept without giving any attention to trimming the aircraft. This will allow the pilot to have more spare capacity to perform other tasks such as arming weapons and working the radar. An obvious disadvantage to this type of control strategy is that the airspeed is free to wander without producing any cues to the pilot. In the F/A-18 this problem is compounded by the fact that the cockpit is extremely quiet. Because of this negative characteristic of apparent neutral longitudinal stability the F/A-18 has control logic that results in an apparent positive longitudinal static stability gradient above 12 degrees AOA. This characteristic prevents a stall from occurring if the pilot was distracted and the aircraft was allowed to continue to decelerate. The basic control strategies chosen by the author to be implemented in the EA-6B will be discussed in chapter 5 and will be implemented through the use of feedback and proportional plus integral control.

FEED-BACK CONTROL

Feed-back control modifies the aircraft dynamics through the measurements of the aircraft's motion. Using these measurements, commands to the actuator are determined which then directs the control surface to move appropriately to augment the resultant motion. The original aircraft dynamics, described by its frequency, damping ratio, or time constant, can be changed to new values by feeding back an appropriate aircraft response through a gain to the control surface. For example, AOA feed-back can be used to stabilize an unstable aircraft by feeding AOA back to the elevator or horizontal stabilizer (Neal T. P. and R. E. Smith, 1970). In the case of the EA-6B, with the engine thrust line below the center of gravity, changes in power result in pitch attitude changes which ultimately correspond to changes in AOA and an apparent longitudinal instability. Using feed-back control can generate better HQR's by using AOA feed-back to help with maintaining a desired AOA.

PROPORTIONAL PLUS INTEGRAL CONTROL

The use of a proportional plus integral control is commonly used to force the aircraft's response to follow and maintain a specific level of response to an input. To better understand the effect of the proportional plus integral control the output of the integral

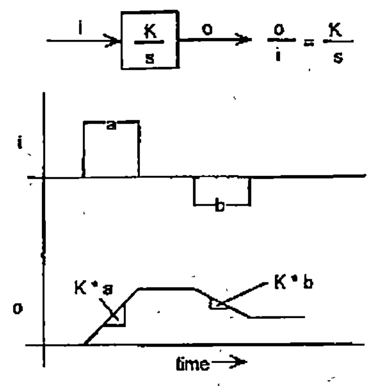

Figure 7: Block Diagram, Input, and Resulting Output of an Integral Control System

Source: Hutchinson, K. T. (1999), Augmented Aircraft Handling Qualities, Buffalo New York, Flight Research Group, Veridian Engineering.

function is shown below in figure 7. First, it is shown from the output that if the area of input "b" is not equal and opposite to the input "a", then a steady output from the integrator will result. Second, the slopes of the output will be equal to the value of the integrator gain (K) times the input magnitude.

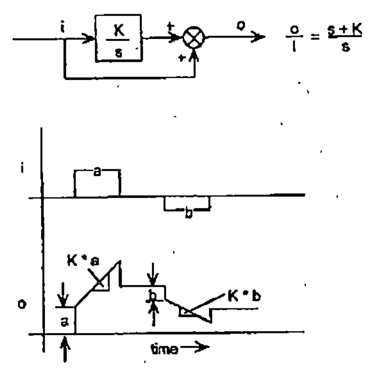

Figure 8: Block Diagram, Input, and Resulting Output of a Proportional Plus Integral Control System

Source: Hutchinson, K. T. (1999), Augmented Aircraft Handling Qualities, Buffalo New York, Flight Research Group, Veridian Engineering.

Figure 8 represents the effect of adding a proportional path in parallel with the integrator shown below. Given the same type of input, it is shown that as long as the input is applied, the output consists of an output immediately proportional to the input magnitude and is a result of the proportional gearing. Over time the output ramps up at a rate proportional to the gain used by the integral path. This proportional plus integral control arrangement is useful in the implementation of control systems where the designer wants the vehicle to follow the input to the system with no steady state error. For example, if the pilot commands a load factor of 2 in a load factor command system, the result will cause the aircraft to move to the load factor of 2, and remain at that level and will not be affected by external disturbances. Figure 9 below, represents a block diagram of a proportional plus integral controller.

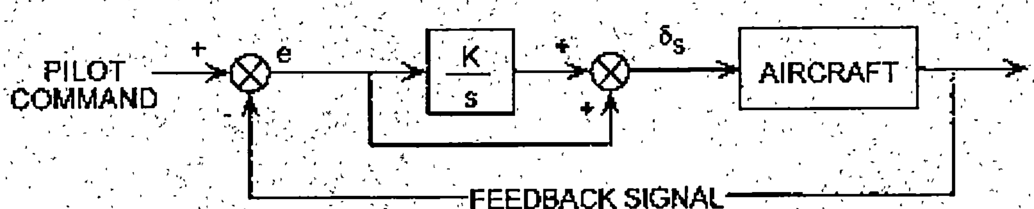

Figure 9: Proportional Plus Integral Control System

Source: Hoh R. H and Mitchell D. G (1996), A Functional Requirement for the Flight Control System, Bristol Pennsylvania, Taylor and Francis Inc.

The error quantity "e" is the value that results when the feedback signal is subtracted from the pilot input. If this signal were normal acceleration, the closed loop system would result in an aircraft that would appear to the pilot to respond with a steady normal acceleration for a constant value input. When the input is removed the load factor becomes 1 and remains constant.

HANDLING QUALITIES

"During flight profiles where zero indicated airspeed resulted, a very objectionable and unpredictable aircraft response in pitch and roll frequently occurred and normal accelerations varied between 4 and negative 3. The loss of EA-6B BuNo 156480 prevented completion of the Navy's evaluation of the post stall gyration and spin characteristics testing" (Luecke M. A., 1971). Is this a flying quality deficiency? Not in the EA-6B. The above flight characteristic is expected following a maneuver that results in an indication of zero airspeed. What about in the V-22 Osprey? Absolutely this is a handling quality deficiency in the V-22. Zero airspeed maneuvers are the fundamental flight characteristic that the airplane was designed to perform. In other words, handling quality ratings can only be assigned to a task if there is a task that is required for the aircraft to accomplish the mission it was designed for.

A task may be as simple as holding a heading or as complex as the entire approach and landing phase of flight. Even a specific task can vary by the amount of precision required. For example, on Combat Air Patrol (CAP) it is only necessary to hold a general heading, but on short final of an instrument approach, very tight heading control is required. Tasks are described as "high gain" when great precision is required as is during a carrier approach and landing.

A handling quality evaluation deals strictly with the closed loop characteristic (i.e. Pilot-in-the-Loop). Open loop characteristics should be observed and commented on during the evaluation to gain more information about the aircraft response. Additionally, the open loop characteristic may help understand why the pilot was doing what he/she was to compensate for the aircraft response. The characteristic responses of an aircraft (dynamic modes of motion) are only one variable in the loop. Control feel characteristics and pilot technique are also very important. The control system characteristics such as movement, forces, friction, and free-play, have an obvious effect on the pilot's opinion of the flying qualities (USNTPS Flight Test Manual 103, 1997). Less obvious is the effect of variations in the evaluation pilot's aggressiveness. Individual pilot smoothness and aggressiveness may also affect overall handling qualities ratings. In studies of highly augmented control systems, it was found that pilot ratings varied over a large range, depending on how aggressively the task was flown. It is important to evaluate the aircraft using inputs representative of those that are required to perform the mission. A pilot with recent experience in the particular mission is usually the best judge of what those inputs should be (Cooper, G.E. and Harper, R.P 1969).

When evaluating aircraft handling qualities, two key factors must be considered, performance and workload. Performance is the precision of aircraft control attained by the pilot. Workload is the amount of effort and attention, both physical and mental, the pilot must provide to attain that level of performance. Pilot compensation is the measure of increased workload a pilot must supply to maintain a level of performance due to less favorable or deficient handling qualities. The total workload is the sum of workload due to compensation and workload due to the task (Cooper, G.E. and Harper, R.P 1969). The Cooper-Harper rating scale shown below in figure 10 requires the pilot to answer the yes-no questions on the left side of the scale, working from the bottom up.

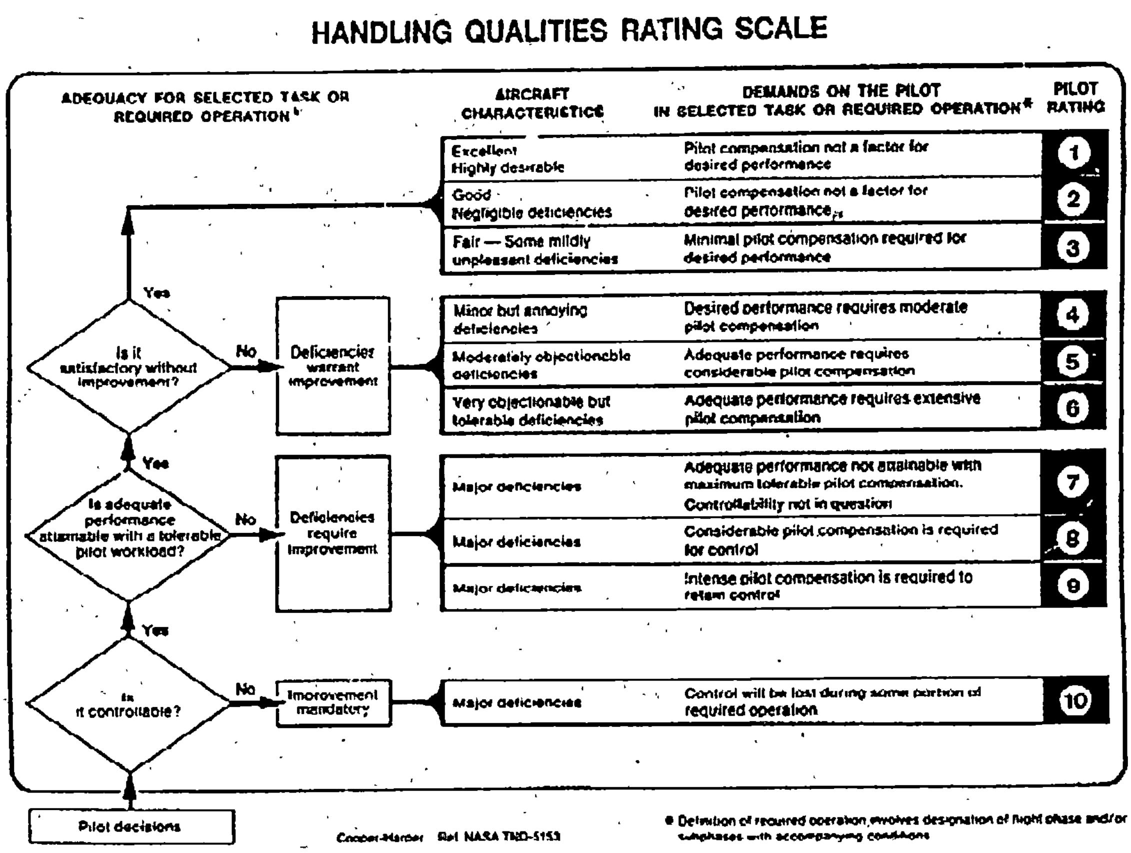

Figure 10: Cooper-Harper Handling Qualities Rating Scale

Source: Cooper, G.E. and Harper, R.P, Use of Pilot Rating in the Evaluation of Aircraft Handling Qualities, NASA Technical Note D-5153, Moffett Field, California, Ames Research Center, 1969 .

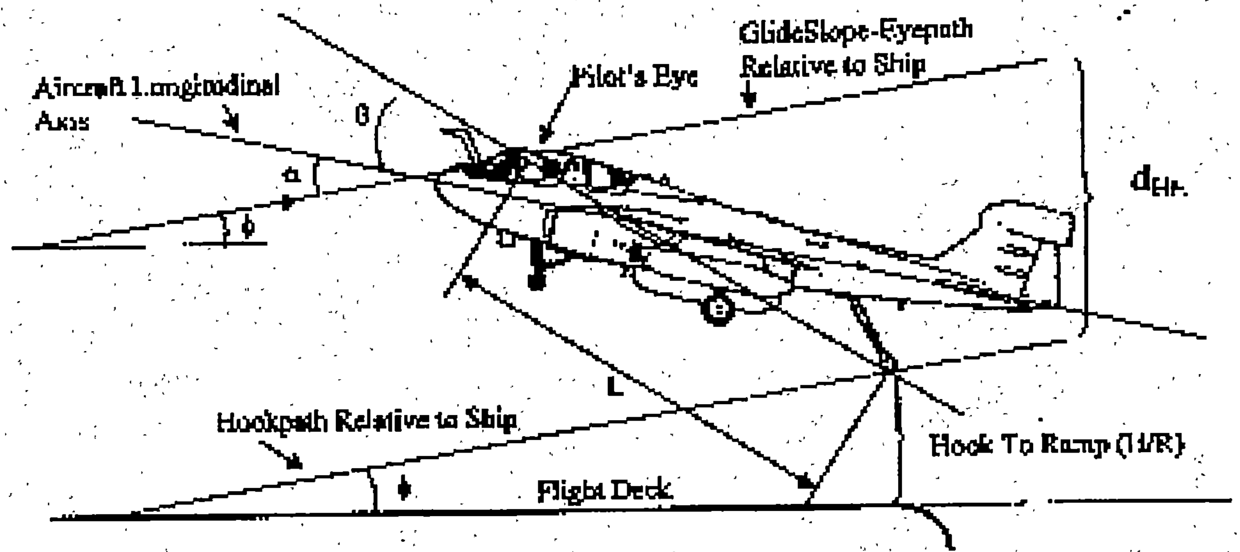

Figure 11: Aircraft Hook to Eye (H/E) Value

Source: "Aircraft Recovery Bulletin No. 10-10D For All Arresting Gear and Optical Landing Systems", Patuxent River, Maryland, Naval Air Systems Command Headquarters, 18 Nov 1997 (NAVAIRWARCENACDIVLKE-4.8.10.4B).

To answer these questions effectively, a set of task tolerances must be agreed upon. For example, landing an airplane on an aircraft carrier requires very precise angle of attack control for the following reason. The approach angle of attack corresponds to the correct hook-to-eye distance between the aircraft tailhook and the pilot's eye as shown below in figure 11. This allows the pilot to fly the glideslope that is intended to fly. The difference of one degree angle of attack in the EA-6B can change actual hook touch down point by as much as 40 feet (i.e. 1 deg nose down = landing 40 feet long and vice versa). So the task of AOA maintenance may have tolerances of +/- ½ degree desired and +/- 1 degree as adequate due to the arresting gear cables being separated by 40 feet.

After performing the task the test pilot will use the Cooper-Harper rating scale to answer the questions on the left side of the scale, working from the bottom up. This will lead to a general category of handling qualities. If the task tolerances were developed with realistic, and mission representative values, the results of the test should lead the test pilot to the handling quality rating category without question. The test pilot must then decide on the final numerical rating using the three descriptions. The area of a handling quality evaluation that is open to pilot opinion is between deficiencies that warrant improvement (HQRs' 4, 5, or 6) and those that require improvement (HQRs' 7, 8, or 9). For example, the test pilot may be able to perform a task within adequate tolerance levels however, he/she may decide that the workload to obtain this performance is not tolerable. This example would put the handling quality deficiency into the category that requires vice warrants improvement.

DESCRIPTION OF THE EA-6B FLIGHT CONTROL SYSTEM

GENERAL

A complete description of the EA-6B's flight control system is needed and was developed by the author to determine the capabilities and restrictions that the current flight control system will place on the CAS implementation. The main emphasis of the following control system description is to show that the current flight control system configuration will readily allow the implementation of a CAS without major modification of the current flight control system. The EA-6B flight control system will allow the implementation of a CAS through the series path of control to control flight control surface movement. The series control system allows manual flight control inputs to be

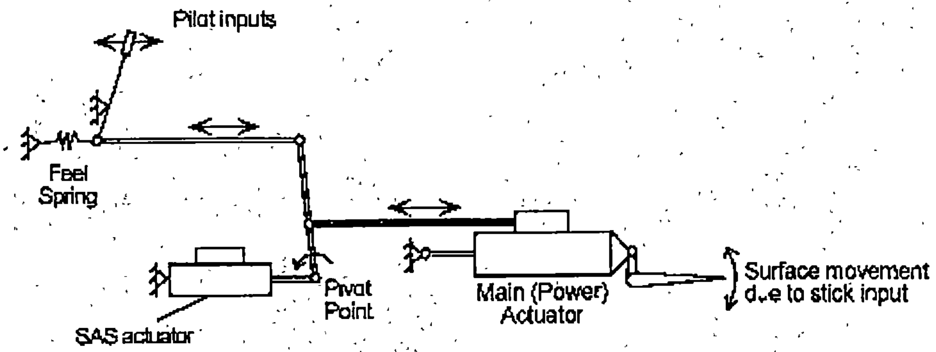

Figure 12: Pilot Inputs in a Series Control System

Source: Organizational Maintenance Manual, Principles of Operation, Automatic Flight Control System AN/ASW-41, NAVAIR 01-85ADC-2-27.1.1, Change 5, 15 February 1995.

made by the pilot and inputs to be made by the stability augmentation actuator as seen below in figure 12. By examining the figure, it is shown that when the pilot moves the control stick, these movements are transferred to the valve of the main actuator by pivoting the walking beam about the SAS actuator attach point. Additionally, this control system arrangement allows the stability augmentation actuator to make inputs that are commanded by the flight control computer through control laws to shape the resulting aircraft motion. Figure 13 illustrates the effect when the stability augmentation actuator is commanded by the FCC.

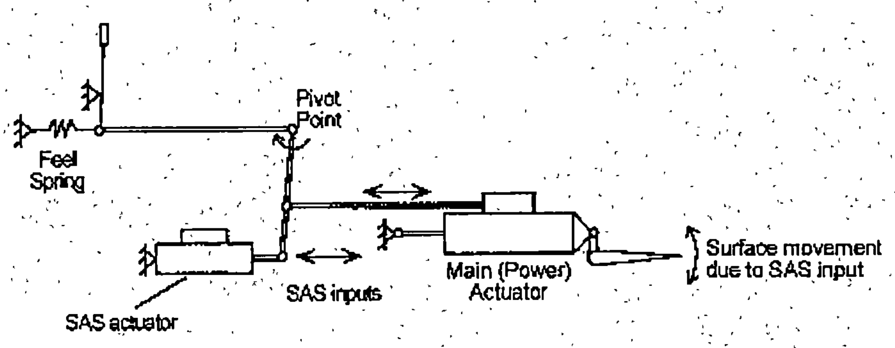

Figure 13: SAS Actuator Inputs in a Series Control System

Source: Organizational Maintenance Manual, Principles of Operation, Automatic Flight Control System AN/ASW-41, NAVAIR 01-85ADC-2-27.1.1, Change 5, 15 February 1995.

In this situation the walking beam now pivots about the pilot's attach point; these movements are transferred to the valve of the main actuator to move the control surface, but not the control stick. This is a simplified example, however the following partial flight control description shows that the EA-6B's current flight control system contains a series path in all axes which will allow the implementation of the CAS.

The EA-6B, depicted in figure 14, is a four-place, twin-engine, all weather, electronic warfare airplane manufactured by Northrop Grumman Aerospace Corporation which incorporates a hydraulically powered, irreversible flight control system.

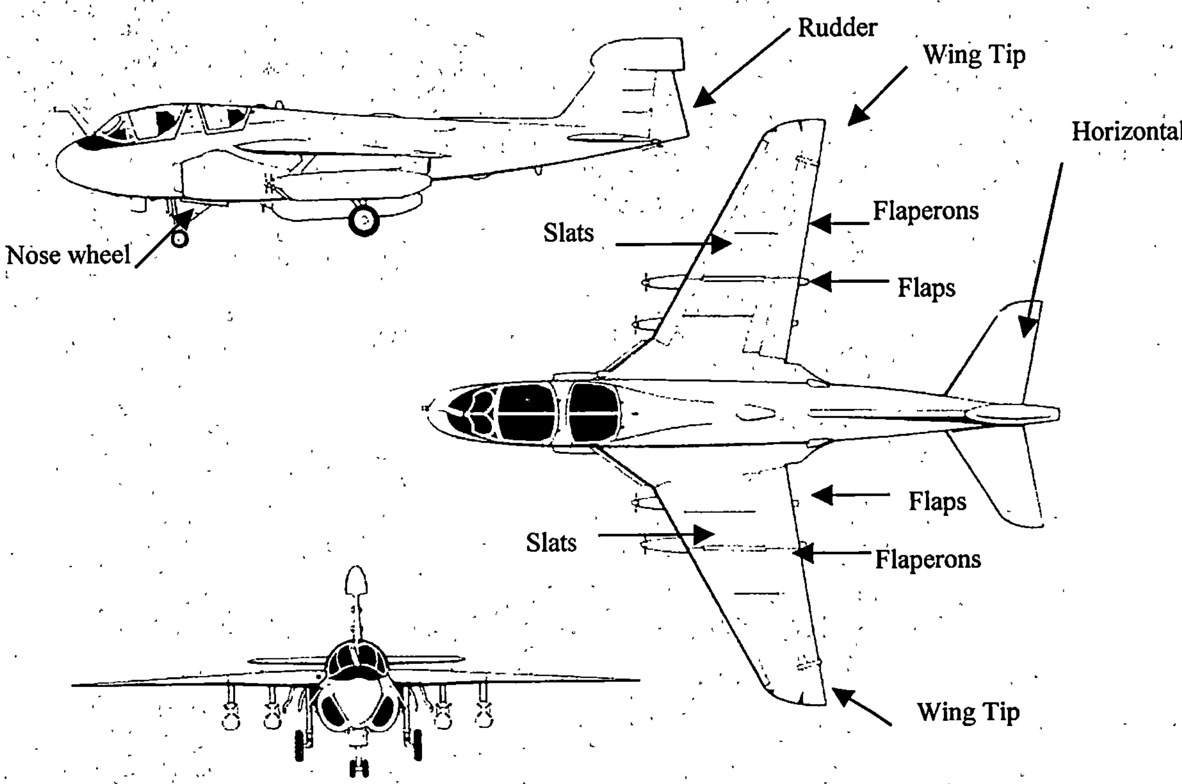

Figure 14: EA-6B AIRPLANE

Source: EA-6B NATOPS Flight Manual, NAVAIR 01-85ADC-1, Interim change No. 72, dated 15 July 1997.

The primary flight control surfaces, also depicted in figure 14, consist of a horizontal stabilizer, rudder and upper wing surface spoilers (flaperons). Cockpit controls and displays are presented in figures B-1 and B-2. A conventional center stick and rudder pedals are linked directly to the respective control surface actuators by a system of push-rods, linkages and bell-cranks to provide pitch, roll and yaw control. The stabilizer actuator, the flaperon auto-pilot actuator, and the rudder actuator consist of electro-hydraulic tandem power control cylinders which direct control surface movement in response to mechanical commands from the pilot or the stability augmentation system. Control deflection characteristics are shown in table C-1. Electrical commands from the Automatic Flight Control System (AFCS) or a combination of electrical AFCS commands and mechanical pilot inputs may be used to command the aircraft's flight control surfaces. Electrical flight control system requirements are shown in table C-2. The lateral control system also includes four mechanical tandem type flaperon hydraulic actuators (two in each wing) which receive mechanical-only lateral inputs through the flaperon auto-pilot actuator and command the appropriate flaperon deflection. Control stick and rudder pedal inputs position an input slide valve on the respective actuator to command the associated control surface deflection, and a proportional follow-up linkage holds the control surface fixed relative to the pilot's control deflection. Each control surface actuator is powered by 3000 psi hydraulic pressure from two independent systems, the flight and the combined primary systems. A more complete description of the EA-6B hydraulic system is presented in appendix E. In addition, extended rudder throw and stabilizer travel are provided for takeoff, landing and spin recovery whenever the flaps are extended or the ASSIST SPIN RECOV switch is actuated (EA-6B NATOPS Flight Manual, 1997).

ARTIFICIAL FEEL

Artificial feel is provided in all three axes through the use of mechanical devices. The longitudinal system incorporates a double-acting bungee (two-gradient feel spring), a g-sensing positive bob-weight, pitch-acceleration-sensing positive and two negative bob-weights, and a leaf spring and eddy-current damper. Control feel in the lateral axis is provided by a double-acting bungee and an eddy-current damper. A cam and roller device loaded by two extension springs provides the artificial feel in the directional axis. Flight control system schematics for each axis are presented in figures B-3 through B-5 (EA-6B NATOPS Flight Manual, 1997).

AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS)

The automatic Flight Control System (AFCS) is an electromechanical system designed to provide automatic control and damping about the airplane's pitch, roll and yaw axes. The AFCS incorporates electromechanical error sensing, coupling, and amplifying stages to provide two-axis stability-augmentation (pitch and yaw), three-axis attitude control and automatic flight-path control. The AFCS consists of an AFCS servo system and the airplane flight control system and is designed to provide electrohydraulic and mechanical coupling to deflect each control surface. The AFCS servo system is designed to provide electromechanical hydraulic power amplification of shaped error or correction signals for each airplane control axis. Each axis flight control system (i.e. hydraulic power actuator and control linkage) is proportionally driven by the output of the respective AFCS servo signals or through linkages from the pilot's manual inputs. The AFCS operating modes are manually selected via the auto-pilot controller depicted in figure B-1, and provide varying degrees of airplane control.

1. Manual mode (Auto-pilot switch OFF): No stability augmentation is provided, and pilot control displacements are the only inputs to the control surface actuators to command surface deflection.
2. Stability-Augmentation (STAB AUG) mode: Designed to automatically damp out oscillations in the pitch and yaw axes and provide turn coordination by positioning the stabilizer and rudder proportionately to airplane pitch, roll, and yaw rates.
3. Auto-pilot (AUTO) mode: The basic hands-off operating mode of the AFCS designed to provide attitude hold (pitch, bank and heading hold), altitude hold, Mach hold, and the capability to couple the AFCS to Automatic Carrier Landing System (ACLS) pitch and roll commands through the AN/ASW-25 data link. The control stick incorporates auto-pilot disengage switches in the column that cause the AFCS to revert to the STAB AUG mode whenever the stick is grasped. When the stick is released, the AUTO mode resumes operation.

An AFCS and an AFCS / AIR DATA CMPTR circuit breaker are located on the aft main circuit breaker panel and can secure electrical power to the Air Navigation Computer (ANC), Stabilizer Trim Actuator, Lateral and Normal Accelerometer transmitters, Air Data Computer (ADC), and Auto-pilot controller. In addition, the yellow and black barber poled AFCS emergency disconnect switch located on a junction box immediately below and to the right of the control stick grip (figure B-1) disconnects electrical power from all AFCS circuits when momentarily actuated. AFCS emergency disconnect switch actuation also returns all switches and pushbuttons on the auto-pilot control panel to the OFF position (NAVAIR 01-85ADC-2-14, Change 3, 1 December 1990).

HORIZONTAL STABILIZER ACTUATOR

The horizontal stabilizer actuator, located at the top center of the fuselage near the stabilizer leading edge, consists of a power valve shuttle, two tandem-mounted power pistons, an electrohydraulic servo valve, a parallel-mode solenoid valve and a series-mode solenoid valve as depicted in figure 15.

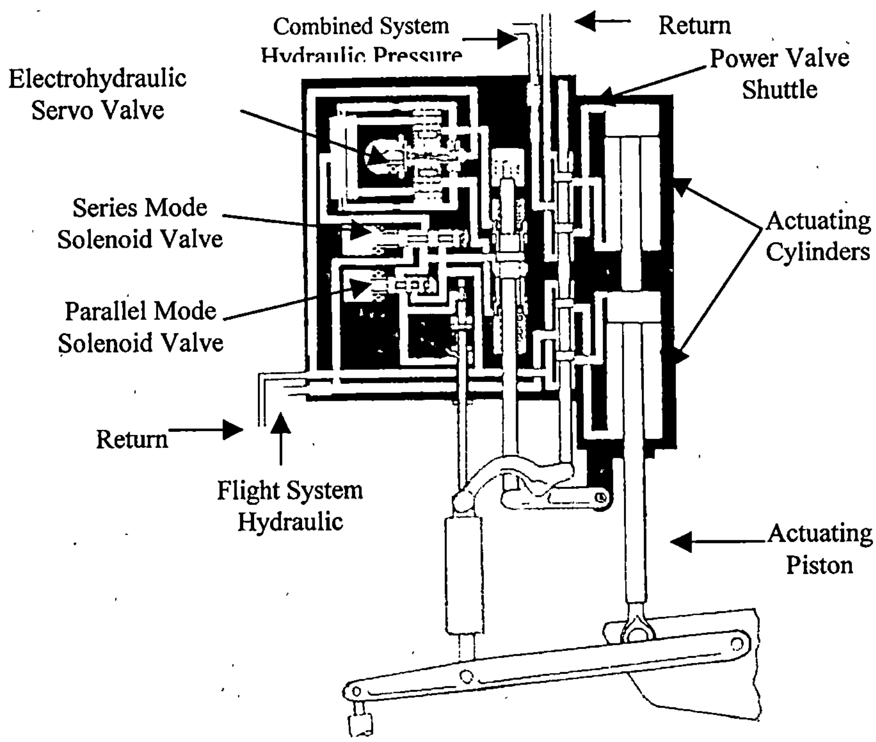

Figure 15: STABILIZER ACTUATOR

Source: Organizational Maintenance Manual, Stabilizer Control System, Description and Principles of Operation, NAVAIR 01-85ADC-2-4, Change 14, 1 April 1996.

The horizontal stabilizer actuator is unique in reference to the primary flight control actuators in that it has provisions for a manual mode, a parallel and a series mode of operation associated with the AFCS.

1. Manual mode of operation (AFCS OFF): Designed such that pilot inputs alone control the power valve shuttle which directs pressure to the actuator pistons. The hydraulic pressure moves the pistons and the stabilizer, to which they are mechanically linked. For constant-velocity stick inputs, the stabilizer is designed to move at a constant speed. When the stick input stops, the power valve shuttle returns to neutral and the stabilizer movement stops.
2. Series mode of operation (STAB AUG or AUTO engaged): Input signals from the AFCS are designed to be used independently or combined with manual pilot inputs to control stabilizer movement. The STAB AUG and AUTO modes automatically damp oscillations about the pitch axis by programming stabilizer deflections proportionately in response to pitch rates. This programmed movement of the control surface is not transmitted to the control stick. The series-mode solenoid valve is energized from the AFCS auto-pilot controller, porting flight hydraulic system pressure to the electrohydraulic servo valve. The servo valve directs movement of the power valve shuttle, which produces a corresponding stabilizer deflection relative to DC electrical signals from the AFCS amplifier or manual longitudinal stick inputs. When operating in the series mode, control surface displacement is not reflected at the control stick. In manual

and stability-augmentation (STAB AUG) flight modes, the control stick applies a mechanical input to the stabilizer hydraulic actuator's input slide valve to generate control surface movement. The control surface deflection is held in a fixed relationship to the pilot's control deflection by a proportional follow-up linkage. Two hydraulic check valves located at both the combined and flight hydraulic system pressure ports on the stabilizer actuator permit free flow to the actuator but prevent back-flow.

3. <u>Parallel mode of operation (ACLS engaged):</u> Designed such that input signals from the AFCS alone control stabilizer movement. Both series-mode and parallel-mode solenoid valves are energized. Flight hydraulic system pressure is ported to the electro-hydraulic servo valve, which drives the power valve shuttle relative to DC electrical signals from the AFCS. In the parallel mode, the control stick and linkages follow the motion of the control surface. To override the AFCS in this mode, a longitudinal stick force of 24 lbs is required to overpower the lockout actuator. Rate switches are incorporated to sense and prevent a hard-over failure, and if necessary, revert the actuator back to the series mode of operation. A hard-over signal, if not corrected immediately, could cause an uncontrollable condition in the longitudinal control system response.

A stabilizer shift mechanism mechanically programs the stabilizer position with flap deflection, and is designed to minimize the effects of flap trim changes, provide extended stabilizer throws when the flaps are extended, and provide maximum control and reduce the possibility of overstress under all flight conditions. The normal travel of the horizontal stabilizer is 1.5° leading edge up to 9.6° leading edge down. When the flaps are extended or ASSIST SPIN RECOVERY is selected, the stabilizer travel increases to 1.5° leading edge up to 24° leading edge down, as illustrated in table C-1. In both configurations, the longitudinal control stick deflection ranges from 4.0 inches forward to 7.8 inches aft. When the stabilizer shifts to extended throws, the stabilizer position indicator, located on the integrated position indicator (IPI) on the left side of the pilot's instrument panel (figure B-2), presents a symbol of a stabilizer. The word CLEAN is displayed on the stabilizer position indicator when normal stabilizer travel is available. The stabilizer position indicator is powered by the essential 28 VDC bus through the CAUTION LT circuit breaker on the forward left circuit breaker panel. Restricted stabilizer travel at low speed may result in inadequate pitch control and extended stabilizer travel at high speed may cause structural failure. If the horizontal stabilizer disconnects, the stabilizer may deflect full leading edge up, full leading edge down, or float free in an intermediate position (NAVAIR 01-85ADC-2-4, Change 14, 1996).

FLAPERON AUTO-PILOT ACTUATOR

The flaperon auto-pilot actuator depicted in figure 16 is a hydraulic servo unit designed to operate in two modes; manual mode or series mode (associated with the AFCS). The actuator is designed to transmit only mechanical inputs to the flaperon hydraulic actuators in each wing.

1. Manual mode of operation (AFCS OFF):The solenoid valve is de-energized and no hydraulic fluid is ported to any portion of the actuator. The series-link cylinder contains two spring-loaded pistons designed to clamp the two halves of the series-link rod together to form a rigid link. Lateral control stick inputs are transferred directly through the input lever and the series link to the output lever.
2. Series mode of operation (AFCS engaged – STAB AUG, AUTO, or ACLS):

The flaperon auto-pilot actuator receives input signals from the AFCS, which are used independently or combined with manual control stick inputs to direct flaperon movement. AFCS signals drive the actuator piston rod and output lever, commanding the appropriate flaperon deflection. Manual pilot inputs generate an electrical signal based on the relative motion between the input and output levers. This manual pilot input signal is combined with the AFCS signal and converted to a mechanical movement, which in turn, actuates the flaperons via the flaperon hydraulic actuators (NAVAIR 01-85ADC-2-4, Change 14, 1 April 1996).

FLAPERON HYDRAULIC ACTUATORS

Two identical, purely hydro-mechanical, flaperon actuators are mounted in each wing and directly control the flaperon movement in response to mechanical inputs. A schematic of a flaperon hydraulic actuator is presented in figure 17. Pilot or AFCS generated mechanical commands are introduced through a load relief bungee and direct hydraulic pressure to two tandem-mounted power pistons.

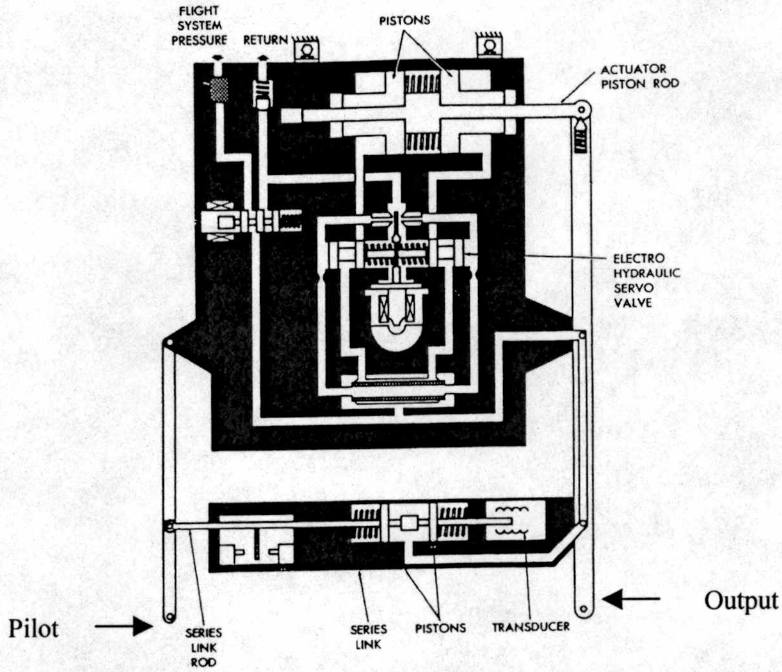

Figure 16: FLAPERON AUTO-PILOT ACTUATOR

Source: Organizational Maintenance Manual, Flaperon Control System, Description and Principles of Operation, NAVAIR 01-85ADC-2-4, Change 14, 1 April 1996.

The power pistons are mechanically connected to the flaperons and are powered by both flight and primary combined hydraulic pressure to ensure complete operating capability is retained if either system fails.

In manual and stability-augmentation (STAB AUG) flight modes, the control stick applies a mechanical input to the flaperon hydraulic actuator's input slide valve to generate control surface movement. The control surface deflection is held in a fixed relationship to the pilot's control deflection by a proportional follow-up linkage. With bank angle between 5° and 60°. Automatic heading hold is inhibited when the ACLS mode switch is placed in the HDG OFF position, and the auto-pilot will hold any bank

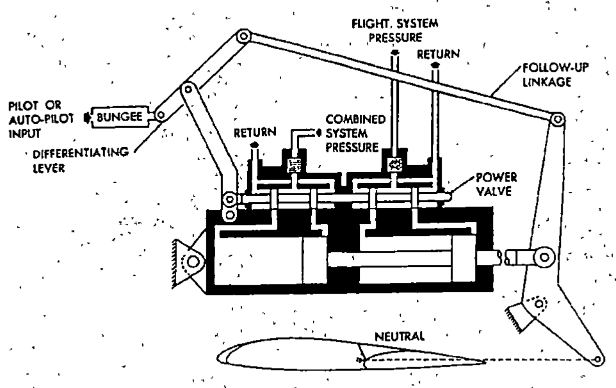

Figure 17: FLAPERON HYDRAULIC ACTUATOR

Source: Organizational Maintenance Manual, Flaperon Control System, Description and Principles of Operation, NAVAIR 01-85ADC-2-4, Change 14, 1 April 1996.

angle between 0° and 60°.AUTO mode selected, the AFCS provides heading hold or will maintain any selected A ball-lock detent permits disengagement of the flaperon auto-pilot actuator in the event of an actuator malfunction. The AFCS may be manually overridden by applying a lateral stick force of 25 lbs (NAVAIR 01-85ADC-2-4, Change 14, 1 April 1996).

RUDDER ACTUATOR

The tandem-type hydraulic rudder actuator is powered by both the flight and primary combined hydraulic systems. Displacement of the rudder pedals positions an input slide valve, which directs hydraulic pressure to the appropriate side of two tandem-mounted power pistons, which generate the corresponding rudder surface deflection. A proportional follow-up linkage maintains the rudder position relative to the pilot's pedal deflection. The rudder actuator, depicted in figure 18, is designed to operate in two modes; manual mode or series mode (associated with the AFCS).

1. Manual mode of operation (AFCS OFF): Pilot input alone controls the power valve, directing hydraulic pressure to the two tandem-mounted power pistons which are mechanically linked to the rudder. For constant-velocity pilot inputs, the rudder deflects at a constant speed. When the pilot input stops, the power valve shuttle returns to neutral and the rudder stops.

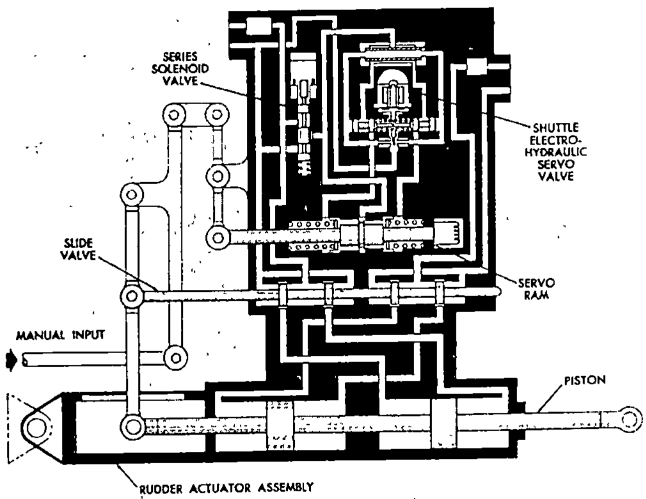

Figure 18: RUDDER ACTUATOR

Source: Organizational Maintenance Manual, Rudder Control System, Description and Principles of Operation, NAVAIR 01-85ADC-2-4, Change 13, 1 February 1995.

2. Series mode of operation (AFCS engaged – STAB AUG, AUTO, or ACLS): The series-mode solenoid valve is energized from the AUTO PILOT controller located on the center console (figure A-1). Input signals from the AFCS may then be used independently or combined with mechanical control stick inputs to command rudder movement (NAVAIR 01-85ADC-2-4, Change 13, 1 February 1995).

SECONDARY FLIGHT CONTROLS

Secondary flight controls, depicted in figure 14, consist of DC electrically controlled, hydraulically actuated semi-Fowler type, slotted, trailing edge flaps and leading edge slats. Wing tip speed brakes are AC electrically controlled, hydraulically actuated and consist of a set of split trailing-edge surfaces at the tip of each wing, outboard of the flaps. The nose wheel steering (NWS) system provides directional control and shimmy damping through a DC electrically controlled nose wheel steering button on the control stick, as depicted in figure B-1. The wheel brakes are mechanically controlled by toe pressure and incorporate a DC electrically controlled anti-skid system. The NWS and wheel brake systems are hydraulically actuated and designed to provide directional control while the anti-skid system is designed to increased high-speed braking effectiveness. The primary and secondary flight control surface deflection limits are presented in table C-1. Electrical power for the secondary flight controls and trim system is provided by any AC generator and transformer-rectifier (T/R) combination, or an emergency ram air turbine (RAT) driven AC generator and emergency transformer-rectifier. The primary and secondary flight control system electrical power requirements are outlined in tables C-1 and C-2, and a more detailed description of the EA-6B electrical system is provided in appendix E. Additional discussion of the secondary flight control systems is presented in appendix F (EA-6B NATOPS Flight Manual 1998).

SUMMARY

The evolution of aircraft augmentation has occurred over a time period of only sixty years. During this period the technology has progressed from powered flight control surfaces used to overcome high dynamic pressure during high speed flight to controlling unstable aircraft and producing handling qualities in these aircraft which are better than ever. However, the great advances and successes in the area of augmented flight control systems have not come without failure, and have resulted in the loss of aircraft and in some instances loss of life. The reason for this is simple. The control system design may be very different when the complex variable of the pilot is used to close the loop. The author feels it is important to understand this limitation and the risks that it poses.

The CAS was discussed and compared to the SAS to demonstrate the advantages of using such a system in the EA-6B. Control strategies to implement the CAS in the EA-6B are many but in the author's opinion the problems with the EA-6B's flying qualities can be corrected using two basic types of control. The proportional plus integral control will provide a path to implement a normal acceleration command system in the CR configuration and should be implemented to reduce FLE which will result in more efficient pilot training, and improve various other flying qualities issues that will be discussed later. In the PA configuration the use of feedback control can be used to further stabilize the EA-6B in the longitudinal and lateral/directional axis to eliminate unwanted aircraft motions. Additionally, the EA-6B's flight control system was discussed to help understand limitations and capabilities just through examination. The control system as it is designed does allow the implementation of the CAS through the use of the stability augmentation actuators and the series control system already in place in all control axis.

Finally, handling qualities and how they are rated using the Cooper-Harper rating scale were discussed. Handling quality evaluations and ratings are very important and are required to accurately determine which tasks result in deficient handling qualities. Even more important than the HQR for the task is a thorough and accurate description of the compensation required to maintain a predetermined level of performance. The compensation is the key that will help determine what aircraft response or control system characteristic may be causing the unwanted aircraft motions and this information will help to form control laws to correct the deficient flying quality.

CHAPTER 3: METHODOLOGY

GENERAL

This section will discuss the author's methodology used to develop a case to implement a CAS into the EA-6B to improve flying qualities. First, a flying qualities evaluation was conducted and Cooper-Harper ratings were given for each task evaluated. The flying qualities evaluation consisted of both shore-based and shipboard flight tests where 22.0 flight hours were flown and 70 shipboard approaches and landings were made. This evaluation was not conducted to determine flying quality deficiencies to be corrected by a CAS, however the data obtained from this test did not affect the results. Following the flying quality evaluation, pilot compensation comments were reviewed and based on this information a plan for a stability evaluation in both the longitudinal and lateral-directional axis was developed. Additionally, a review of the flying qualities and stability flight test report preformed on the prototype in 1971 was conducted (Luecke, 1971). The results of the stability evaluation were used to support or discount the author's theories of what EA-6B stability characteristics may be adversely affecting the task in question. The longitudinal and lateral/directional stability evaluation was performed in accordance with the procedures outlined in the Stability and Control Flight Test Manual 103 (USNTPS Flight Test Manual 103, 1997). Ultimately a good understanding of the task, what makes the task difficult to maintain at a desired performance level, and identifying the stability characteristics that may be influencing this adverse characteristic will be all the information needed to form a plan to shape the aircraft's dynamic response with a CAS.

HANDLING QUALITIES EVALUATION

The first step used to develop a strategy to execute a handling qualities evaluation was to determine what tasks the pilot must perform to accomplish the mission for which the aircraft was designed. This was a very easy task due to the author being a EA-6B pilot who already has a good understanding of both the tasks and tolerances required to perform the mission. The tasks and tolerances used for the handling qualities evaluation are shown below in table 1.

Table 1: TASKS AND TOLERANCES

Phase	Task	Desired/Adequate Tolerance
Cruise Configuration		
Level Flight	Altitude Control	+/- 50 feet/ +/- 100 feet
Maneuvering Flight	G Control	+/- 0 g / +/- .5 g
Power Approach		
Gear and Flap Transition	Altitude Control	+/- 50 feet/ +/- 100 feet
Downwind	Airspeed Control	±2 kts / ±5 kts
	Heading Control	±2° / ±5°
	Altitude Control	±50 ft / ±75 ft
Base	AOA Control	±½ unit / ±1 unit
	Roll Attitude	±1° / ±3°
Final	Heading Control	±1° / ±2°
	Centerline Capture	+/- 10 feet
	AOA Control	±½ unit / ±1 unit
	Glideslope Capture	±½ ball / ±1 ball
Touchdown	Centerline Capture	±5 ft / ±10 ft
Waveoff/Bolter	AOA Control	±½ unit / ±1 unit
Approaches at Altitude	Rate of Descent Control	±50 fpm / ±100 fpm
	Heading Control	±1° / ±3°
	AOA Control	±½ unit / ±1 unit

Upon completion of the handling quality evaluation the Cooper-Harper HQR's and compensation required to perform a task were used by the author to focus on particular longitudinal and lateral-directional stability characteristics. Additionally, research of previous stability tests were used to form an opinion of which stability characteristics may be generating less than desired handling qualities.

LONGITUDINAL STABILITY EVALUATION

Tests that are addressed in the "USNTPS Fixed Wing Stability and Control Theory and Flight Test Techniques Manual," (USN Test Pilot School, 1997) were conducted in accordance with the procedures therein. The data obtained from this evaluation was qualitative in nature. For example, the non-maneuvering static stability test was performed as if data were going to be taken, however only trends of data were recorded (i.e. the variation in stick force with respect to airspeed was shallow indicating apparent weak longitudinal stability). The main goal of this evaluation was to uncover the major stability contributors that produced HQR's which must be corrected. This information was then used to better understand or formulate hypotheses of why certain tasks produce higher pilot workload. The goal was to provide enough information to help control law developers pinpoint areas to concentrate on.

NON-MANEUVERING STATIC STABILITY

Non-maneuvering static stability, as indicated by the variation of longitudinal stick force with airspeed, was qualitatively measured in the PA configuration. The stick forces during this test were estimated. Evaluating the gradient of the static longitudinal stability was performed by trimming the aircraft to the desired airspeed, and without re-trimming the control stick was displaced small increments both forward and aft to stabilize the aircraft at a new airspeed. Following a longitudinal control stick displacement and stabilized at the new airspeed, the control position was held and the stick force was noted. This test was repeated to the airspeed which activated stall warning and to 30 knots faster than the desired trim airspeed. Finally, the aircraft was returned to the trim airspeed and the open loop response was observed to assess phugoid and gust sensitivity characteristics.

MANEUVERING LONGITUDINAL STATIC STABILITY

Maneuvering static stability, as indicated by the variation of longitudinal stick force with normal load factor, was qualitatively measured in the CR configuration. Again stick forces were estimated and the qualitative impression of the maneuvering static stability gradient was recorded. To begin the test the aircraft was trimmed to the desired test airspeed. Because the thrust line was below the center of gravity, the airplane had to be trimmed at this airspeed with the throttles at full thrust. Trimming the aircraft at full thrust corrected the trim position of the control stick for the thrust effects, or additional nose up pitching moment caused by the effects of thrust. The maneuvering static stability test was conducted by increasing the throttles to full thrust, rolling the aircraft to a desired bank angle and adjusting the attitude and normal acceleration to maintain the desired airspeed. When the airspeed and load factor were stable the longitudinal stick force was noted. This test was repeated in ½ load factor increments up to limit load factor of 5.5. Dynamic maneuvering stability or sudden pulls was evaluated at each load factor before moving on to the next stabilized point. The dynamic maneuvering stability test was conducted by abruptly inputting a longitudinal step input to achieve the desired load factor. The resulting longitudinal stick force was noted.

SHORT PERIOD RESPONSE AND DAMPING

The short period response as indicated by the variation of normal acceleration with time was measured in the cruise configuration following a sinusoidal longitudinal control input. Prior to performing the test a continuous sinusoidal stick input beginning with a low frequency input and increasing to high frequency input was performed. The frequency that produced the largest aircraft longitudinal response during sinusoidal stick pumping was recorded as the aircraft's best response frequency. With this information the test was performed with a sinusoidal stick input at the best response frequency followed by releasing the control stick at the trim position and then recording the normal acceleration overshoots (above and below a load factor of 1) and the time between them. The time between the overshoots describes the half period of the aircraft's observed natural frequency response and the number of overshoots give an estimate how well the short period response is damped.

$$\zeta = \frac{\ln\left[\frac{D_1}{D_2}\right]}{\sqrt{\pi^2 + \left(\ln\left[\frac{D_1}{D_2}\right]\right)}} \qquad \text{(Equation 3)}$$

The equation used by the author to quantify the damping ratio (ζ) is shown in equation 3, where D_1 and D_2 are the magnitude of the load factor overshoot above and below the load factor of 1 for a second order system (USNTPS Class Notes, 1992).

PHUGOID PERIOD AND DAMPING

The phugoid characteristic as indicated by the variation of airspeed with time was measured in both the PA and CR configurations. The phugoid was measured by first trimming the aircraft to the desired airspeed. With the aircraft stabilized at the trim airspeed, the aircraft was decelerated with an aft longitudinal input to achieve an airspeed slower than the trim speed. Once stabilized at this new airspeed the control stick was placed in the trim position and the open loop response of the aircraft was observed. Specifically, the period, number of overshoots, and magnitude of overshoots were all recorded. The magnitude of overshoot was determined by recording the peak airspeed before the motion of the aircraft reversed. The period was measured from the time of zero vertical velocity at the peak slow airspeed to zero vertical velocity at the peak fast airspeed. The time between peaks defines the phugoid half period. The phugoid damping ratio was obtained by using the magnitudes of peak airspeed deviations above and below the initial condition trim airspeed using equation 3.

FLIGHT PATH STABILITY

The flight path stability as indicated by the variation in airspeed with flight path angle was measured in the PA configuration. The test was performed by stabilizing the aircraft at the final approach airspeed with trim and power setting and then holding the appropriate longitudinal input to achieve the new desired airspeed. With the aircraft stabilized at this out of trim airspeed the vertical velocity change for a period of 30 seconds was recorded. The test was performed in 5 knot increments to 15 knots above trim airspeed and 10 knots below trim airspeed. The flight path angle (γ) is described by equation 4 where ROD is the observed rate of decent and TAS is the test true airspeed (USNTPS Class Notes, 1992).

$$\gamma = Sin^{-1}\left(\frac{ROD}{TAS}\right) \qquad \text{(Equation 4)}$$

LATERAL/DIRECTIONAL STABILITY EVALUATION

The lateral/directional evaluation is much more complex than the longitudinal evaluation because the lateral/directional motions of the aircraft are coupled in two axes (roll and yaw) while the longitudinal motion is restricted to a single axis (pitch). It is this inter-relationship or coupling that makes it very difficult to conclude from the aircraft motion on the effect of a single stability derivative. As a result of this last statement there may be an attempt of improving roll control sensitivity (roll rate per pound of lateral stick force) because the pilot complains of sluggish aircraft response to lateral inputs. However, the sluggish roll response may have really been caused by adverse yaw which produced a side slip, and caused the dihedral force to oppose the commanded roll. The lateral-directional stability evaluation was conducted without a side-slip indicator. Due to the complexity of the lateral-directional stability problem, the test should be repeated with side-slip indication.

ADVERSE YAW

Adverse yaw as indicated by a perceived yaw away from the direction of roll was evaluated in the CR and PA configurations. The spoiler configuration used for roll control on the EA-6B led the author to believe that the aircraft would possess pro-verse yaw due to the drag caused by the spoiler during roll commands. Before the actual test was conducted the SAS was turned off and the open loop response of the aircraft was observed. With the SAS off, the aircraft most definitely exhibited adverse yaw following a roll command. In fact the adverse yaw was such that heading control within 5 degrees was impossible. With the SAS on, the test was conducted by inputting an abrupt step roll command, while visually observing the aircraft's yaw motion with respect to a visual reference point on the ground, and the heading indicator.

DIHEDRAL EFFECT

Dihedral effect as indicated by increasing lateral stick force with increasing side-slip was evaluated in the PA configuration. The test was conducted by placing the aircraft in a Steady Heading Side Slip (SHSS) and increasing side-slip through rudder pedal input in one quarter increments. As the rudder input was increased, the lateral stick input was increased to maintain the steady heading condition. This is an indication of positive dihedral effect.

CHAPTER 4: RESULTS AND CONCLUSIONS

LONGITUDINAL FLYING QUALITIES EVALUATION

LOAD FACTOR CAPTURE RESULTS

Load factor capture was evaluated in configuration CR during break turns, simulated 30 degree dive weapon delivery profiles, and aerobatics. Targeting a load factor onset rate of approximately 4 per second during aggressive maneuvering, consistently produced overshoots of the target by 1.5 and would have resulted in overstressing the aicrcraft if a visual scan of the "g- meter" was not maintained. Initial stick forces following the longitudinal input were very heavy as compared to the steady state stick force at the desired load factor. Steady state longitudinal stick force gradients of approximately 6 lbs per unit of load factor did not prevent normal acceleration capture overshoots, in fact they seemed to produce oscillations around the target load factor caused by the required control shaping inputs to compensate for the changing stick force as the load factor increased. The high steady state stick force per load factor was objectionable in that holding 30 pounds of stick force at a load factor of four was fatiguing and resulted in poor load factor maintenance for extended periods of time. Load factor capture within ± 1 during aggressive maneuvering was not possible (HQR-7). The imprecise load factor capture during a defensive break turn in a visual engagement or during SAM defensive maneuvering will result in aircraft structural damage. Additionally, imprecise load factor capture will not allow the aircrew to adequately train to aggressively maneuver the aircraft, for fear that the maneuver will result in aircraft overstress.

Analysis of Imprecise Load Factor Capture

The results of the dynamic and static longitudinal stability evaluation conducted by the author resulted in the following observations:

1) Sudden pulls resulted in substantially higher stick forces than stabilized points at the same load factors (350 KIAS and 10,000 ft MSL).
2) Sudden pulls resulted in somewhat larger control stick deflections than stabilized points at the same load factors (350 KIAS and 10,000 ft MSL).
3) Short period response was characterized as being lightly damped (350 KIAS and 10,000 ft MSL).

Review of the flight test data and previous flight test results led the author to make the following conclusions as to why load factor capture in the EA-6B is imprecise:

1) The fact that the sudden pull longitudinal stick force at the target load factor is substantially higher than the longitudinal stick force at the stabilized condition has the following effect: during aggressive maneuvering following an abrupt longitudinal input, the resulting load factor is substantially lower (more than 2 times) than the steady state load factor with the same longitudinal stick force. As shown in figure 19 a sudden pull longitudinal input while targeting a load factor of 3 results in 30 lbs of stick force. However, during stabilized maneuvering stability tests, a load factor of 5 only results in 25 lbs of

longitudinal stick force. There are two effects that this characteristic of varying control feel will have. First, if a sudden longitudinal input of 30 lbs was input, the pilot will observe an initial load factor of 3. The load factor will be maintained by maintaining the same control feel or stick force. Doing so will result in the steady state load factor being almost twice the initial load factor. This characteristic of varying of longitudinal stick force, and poor load factor maintenance due to the high forces was observed by the author during the handling quality evaluation. The second effect of higher sudden pull longitudinal stick force than the steady state longitudinal stick force results in the pilot observing the load factor response of the aircraft during the abrupt input and then adjusting the input. The adjusting input would require an increase in aft longitudinal input because the resulting load factor did not reach the desired value. This control shaping or additional longitudinal input will definitely result in a much higher than desired steady state load factor if not closely monitored by the pilot.

2) The same argument can be made for the variation of control stick deflection between the sudden and steady state longitudinal inputs. Sudden longitudinal inputs resulted in larger stick deflection than the steady state stick deflection at the same load factor. If the pilot inputs a step longitudinal input, as would be made during an aggressive defensive maneuver, and the input was held at that

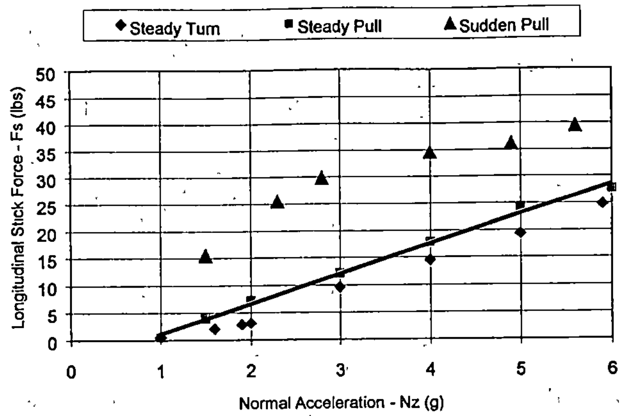

Figure 19: EA-6B BUNO 156478 MANEUVERING LONGITUDINAL STABILITY IN CONFIGURATION CRUISE

Source: Luecke M. A, ENG/EA-6B-RTR-71-001, Naval Air Warfare Test Center Aircraft Division, Patuxent River, Maryland, 1971.

position, the final steady state load factor would again be almost 2 times the initial (Luecke, M. A., 1971).

3) Finally, the characteristic of the short period response (figure 20) being lightly damped, where the calculated damping ratio (ζ) determined from equation 4 to be 0.372, did cause load factor overshoots and most likely contributed to the poor load factor maintenance. As a result of the lightly damped short period response, imprecise load factor capture will occur when inputs are made at the aircraft's best response frequency (USNTPS Class Notes, 1997).

LOAD FACTOR CAPTURE CONCLUSIONS

In summary, precise load factor capture is required for the EA-6B to be able to perform its mission effectively.

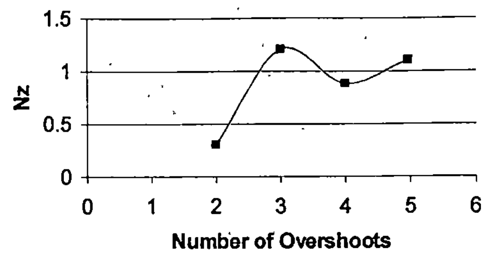

Figure 20: EA-6B BUNO 163892 DYNAMIC LONGITUDINAL STABILITY

Precise load factor capture is needed to prevent aircraft overstress during a SAM launch or any defensive maneuver where the pilot's main concern is evading the incoming missile or enemy. If load factor capture is imprecise, this situation will most likely result in an aircraft overstress and possible structural damage. Additionally, imprecise load factor capture will reduce the effectiveness of pilot training in that the pilot must be more aware of aircraft load factor than the training objective at hand. The handling quality deficiency of imprecise load factor capture is well known by all who fly the EA-6B. This deficient handling quality has resulted in the EA-6B community limiting the allowable load factor below the flight manual limit by a factor of 1.5. This administrative load factor limit was placed on the aircraft due to the high number of aircraft overstresses that occurred during normal training missions (Breslauer M. D., 1999). Looking back at the flight control description, it is not surprising that the stick forces during aggressive maneuvering vary so much. A review of the longitudinal control system feel shows a double-acting bungee (two-gradient feel spring), a g-sensing positive bob-weight, pitch-acceleration-sensing positive and two negative bob-weights, a leaf spring, and an eddy-current damper. It is the author's opinion that all this gadgetry was used to mask the poor flying qualities in the longitudinal axis.

A possible solution to correct the flying quality deficiency of imprecise load factor capture would be to implement a CAS. Implementation of the CAS and use of a control loop that feeds back normal acceleration with a proportion plus integral control mechanization would allow the implementation of a load factor command system in the CR configuration. This system could be implemented in many ways, however there must be something added to the current flight control system that will either measure the control stick position or force. With a control stick position sensor, the load factor command system could be implemented where a control stick deflection would correspond to a load factor. Additionally, the control loop could provide logic to limit load factor to prevent the pilot from overstressing the aircraft. The fact of changing control feel during aggressive maneuvering however will still exist if all the control system gadgetry remains in place. Correcting the problem of imprecise load factor capture in the area where an overstress is most likely to occur, will not be affected by this. Finally, the lightly damped short period response would be corrected as a result of the load factor command system being implemented. The FCC will continuously monitor the normal acceleration feed-back and automatically adjust the horizontal tail input. This would provide an increase in the short period damping.

ALTITUDE MAINTENACE RESULTS

Altitude maintenance was evaluated in configuration CR at 25,000 ft MSL and 0.7 IMN while performing normal mission duties such as monitoring navigation and radar. The longitudinal stick forces were trimmed to zero with the aircraft stabilized at the desired indicated Mach number. While performing the normal mission duties, altitude deviations of +/- 200 feet were routinely observed. Maintaining altitude within +/- 100 feet required small (approximately ½ inch) longitudinal inputs approximately every 30 seconds (HQR-4). Altitude deviations will prevent the pilot from performing normal mission duties resulting in reduced mission effectiveness.

Analysis of Altitude Deviations

Review of the flight test data obtained by the author and previous flight test results led to the following conclusions about why altitude maintenance in the EA-6B is imprecise:

1. The phugoid period and damping observed during the stability evaluation was characterized as being very lightly damped (7 overshoots) with a period of approximately 60 seconds, as shown in figure 21. This stability characteristic is most likely the major contributor of poor altitude maintenance handling quality deficiency. With a phugoid period of approximately 60 seconds, the peak altitude deviations would occur at approximately 30 second intervals, as seen during the handling qualities altitude maintenance task evaluation. Additionally, once the correction has been input the phugoid period will begin again following the next external or pilot induced disturbance.

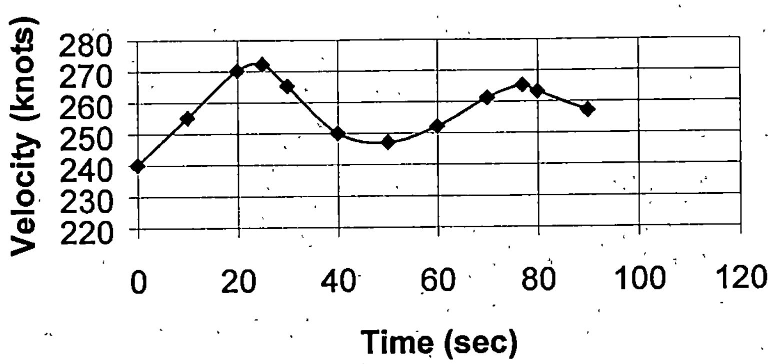

Figure 21: EA-6B BUNO 163892 PHUGOID IN CONFIGURATION CRUISE

2. Flight control system friction could also be another contributing factor, because friction in the longitudinal control system will result in poor control surface centering due to the friction holding the control linkages in a position other than desired. The EA-6B longitudinal control system does have friction of approximately ¾ of a pound.

ALTITUDE MAINTENANCE CONCLUSIONS

Altitude deviations during normal mission tasks are more of a nuisance than a flying quality deficiency, however this will be corrected as a result of incorporating a CAS and implementing a load factor command system in the CR configuration. A load factor command system characteristic results in phugoid suppression due to the system maintaining a load factor of 1 in the neutral stick free position. However, due to control system friction, the neutral stick free position may change between control inputs due to the non-absolute centering of the control stick caused by friction. A possible solution to compensate for the poor control stick centering would be to incorporate a control stick force sensor. Zero control stick force would command the flight control surfaces to the neutral stick free trim position and would negate the poor centering due to friction.

AOA MAINTENANCE DURING APPROACH RESULTS

AOA maintenance during approach was evaluated in the power approach configuration during approaches to landing at the field and to the carrier. The importance of evaluating this task during shipboard approaches is due to the disturbance of airflow around the ship just prior to the aircraft coming aboard. This disturbance is known as the "burble" and can be thought of as an area of turbulence or disrupted free stream airflow. During field approaches, the aircraft was trimmed to the appropriate AOA during the wings level downwind phase. During the approach turn phase the AOA seemed to wander +/- 1 degree without pilot input (open loop response). Attempting to maintain the desired AOA require continuous small longitudinal inputs of approximately ½ inch and 1 pound. The AOA wandering continued during the wings level final approach phase. During shipboard approaches the same AOA wandering characteristic existed, however at the "in-close" position (approximately 1,000 feet aft of the ship) AOA excursions seemed to be more abrupt and the variations increased in magnitude (+/- 2 degrees) without pilot input. Attempting to correct the abrupt AOA variations resulted in larger than desired glide-slope deviations. Maintaining AOA within +/- 1 unit was not possible within a tolerable pilot workload (HQR-7). Imprecise AOA maintenance resulted in poor glide-slope maintenance and, in the author's opinion, created a workload level that was too high for the task of landing on the carrier. This characteristic will result in poor boarding rates, increased wave-off, and possible aircraft ramp strikes.

Analysis of Imprecise AOA Maintenance

Review of the flight test data obtained by the author and previous flight test results led to the following conclusions as to why AOA maintenance by the EA-6B is imprecise:

1) The longitudinal static stability test conducted revealed that the non-maneuvering static stability gradient of the aircraft in this configuration is neutral. The change in aft longitudinal stick force between the initial condition trim speed and stall warning was only 1 and ¾ pounds and the stick force did not change between 115 and 100 KIAS. Previous EA-6B prototype testing also revealed this longitudinal stability characteristic, as shown below in figure 22. The apparently neutral longitudinal static stability gradient is definitely a stability characteristic contributing to the imprecise AOA maintenance. This apparently neutral longitudinal static stability characteristic is amplified during shipboard approaches when flying through the "burble", where the external disturbances are amplified and produce large abrupt AOA excursions.
2) Finally, the longitudinal control system friction in the test aircraft was as much as ¾ of a pound. Flight control system friction could also be another contributing factor, because friction in the longitudinal control system will result in poor control surface

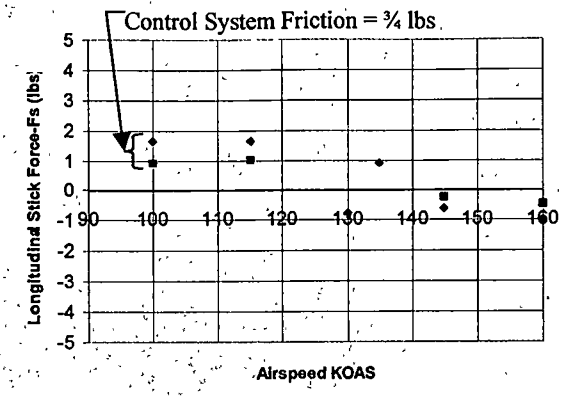

Figure 22: EA-6B BUNO 156478 STATIC LONGITUDINAL STABILITY IN CONFIGURATION POWER APPROACH

Source: Luecke M. A., ENG/EA-6B-RTR-71-001, Naval Air Warfare Test Center Aircraft Division, Patuxent River, Maryland, 1971.

centering due to the friction holding the control linkages in a position other than that desired, resulting in a change in trim airspeed.

AOA MAINTENANCE EVALUATION CONCLUSIONS

Precise AOA maintenance is required for many reasons which include proper hook to eye distance, proper hook touch down geometry, and proper arresting gear engaging speed. Additionally, easy AOA maintenance is required because landing aboard an aircraft carrier is such an intense pilot workload task. The approach to the carrier is also complicated by the ship centerline is continuously moving, caused by both heavy seas and the geometry of the angle deck with respect to the forward translation of the ship. Additionally, glide-slope control is imperative for obvious reasons, and this task is complicated due to AOA deviations caused by external disturbances (i.e. gusts and

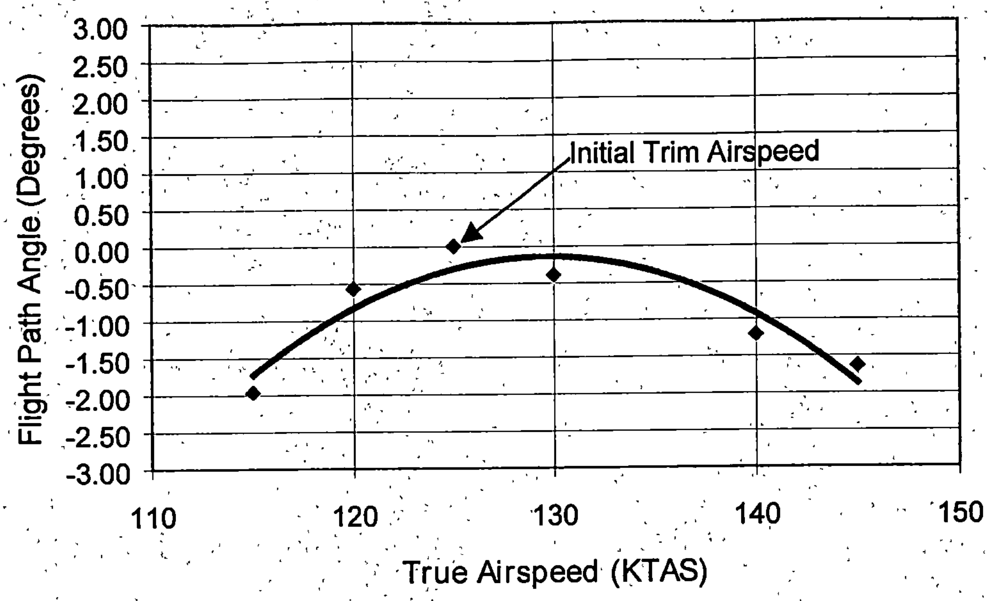

Figure 23: EA-6B BUNO 163892 FLIGHT PATH STABILITY PA CONFIGURATION

"burble"), poor control feel caused by a very shallow gradient of apparent longitudinal stability, and a large amount of control system friction. An evaluation of the flight path stability is shown below in figure 23. The flight path stability is not a contributing factor to poor AOA maintenance, however the characteristic of the EA-6B's flight path stability shows that imprecise AOA maintenance will result in poor glide-slope control. Examining figure 23, indicates that any airspeed deviations from approach speed results in the aircraft rate of decent increasing. A change in approach speed is directly related to a change in approach AOA. The deficient AOA maintenance flying quality results in poor glide slope control because all AOA deviations correspond to a change in airspeed and thereby cause an increased rate of decent. This characteristic increases the propensity for the aircraft to deviate below the intended glide-slope and, as stated previously, may have disastrous results and is most definitely a contributing factor to poor landing performance by student Naval Aviators.

Imprecise AOA maintenance is an unacceptable flying quality deficiency for the carrier landing task. In the author's opinion this is most definitely a contributing factor to the high disqualification rates for student aviators under training. Again the CAS can be used to correct this problem by artificially increasing the static stability through a feed back control strategy where AOA is fed back to the horizontal stabilizer. With this control strategy implemented, the control loop will continuously command the horizontal stabilizer to maintain the desired AOA. This type of control strategy would also compensate for the turbulence encountered when flying through the "burble". However, as with correcting the imprecise load factor capture and altitude maintenance, implementing an AOA feed back loop will not solve the problem in its entirety. That is, feeding back AOA will not stabilize the airplane due to the shallow longitudinal stability control stick force gradient. The open loop response of the aircraft will appear stable because the commanded AOA will be maintained by the control loop, however, with the pilot in the loop, small longitudinal stick inputs will not be noticed by the pilot due to the shallow longitudinal stability stick force gradient. This will result in an unintentional commanded AOA which will be different than the desired. As with the load factor command system, stick sensors, either force or position will be needed to determine the commanded AOA, and due to the high control system friction, the author recommends using force sensors. Control stick force sensors will correct the problem of control system friction by commanding AOA on a schedule with the longitudinal stick force. Force sensors will correct the poor control stick centering due to friction by the FCC commanding the trim AOA when the longitudinal stick force is zero, regardless of the control stick position. Finally, the shallow apparent longitudinal stability gradient could also be corrected by scheduling AOA with longitudinal stick force. This will allow control law developers to change the apparent longitudinal stability gradient.

AOA feedback will also allow an avenue to limit the AOA in various configurations. As stated in the abstract, abnormal configuration approaches do occur. The configuration that is most dangerous is the no slat-flaps down configuration. The final approach AOA of 13 degrees is required to minimize the approach speed, however if the aircraft is allowed to decelerate one degree below its approach speed (approximately 1 ½ knot), the airplane becomes longitudinally unstable and uncontrollable. The nose will violently pitch up and cannot be controlled with forward longitudinal control inputs. With the CAS and AOA command system implemented in the power approach configuration, logic can be used to identify this configuration and limit the allowable AOA command from the pilot's input.

ALTITUDE MAINTENANCE DURING CONFIGURATION CHANGE RESULTS

Altitude maintenance was evaluated during configuration changes in the normal landing pattern and during instrument approaches. The airplane is normally configured from the cruise configuration to the power approach configuration at an airspeed of approximately 250 knots. The importance of configuring the aircraft at this airspeed is due to its high stall speed and the low load factor that is available (load factor of 2 is available at 45,000 pounds) in the cruise configuration at normal landing gross weights. The normal configuration change is accomplished by lowering the landing gear and placing the flaps to the full down position. The initial aircraft response is to "balloon", as indicated by a positive vertical velocity of approximately 500 feet per minute. The initial "ballooning" tendency can be compensated for by a forward longitudinal stick input of approximately 5 pounds. Immediately following the forward longitudinal input the vertical velocity builds up very rapidly in the negative direction. The open loop response (controls remain free) results in a negative vertical velocity of over 4,000 feet per minute and a nose down pitch attitude of 30 degrees. The open loop response observation was terminated due to safety, however, the vertical velocity would have continued to increase in the same nose down direction. During the closed loop evaluation following the initial "ballooning" correction, aft longitudinal stick forces built to approximately 30 pounds and altitude deviations in excess of +/- 200 feet were routinely observed. Compensating for the large change in the longitudinal stick force during the configuration change by trimming did not correct the problem due to the slow trim response. During instrument approaches under actual instrument conditions the configuration change was very disorienting. Altitude maintenance within +/- 200 feet during configuration changes from cruise to power approach configuration was not possible with a tolerable pilot workload (HQR-7). Abrupt altitude deviations during configuration changes will cause pilot disorientation and subsequent vertigo during actual instrument condition approaches and may result in subsequent flight into the terrain.

Analysis of Abrupt Altitude Deviations With Configuration Change

Review of the flight test data obtained by the author and the EA-6B flight control description led to the following conclusions about why abrupt altitude deviations occur during configuration changes in the EA-6B:

1) A stabilizer shift mechanism mechanically programs the stabilizer position with flap deflection, and is designed to minimize the effects of flap trim changes, to provide extended stabilizer throws when the flaps are extended, and to provide maximum control and reduce the possibility of overstress under all flight conditions. In the author's opinion, large longitudinal trim changes, and excessive nose down pitch attitudes are caused by improper gearing of the stabilizer shift mechanism which leaves the aircraft in an out-of-trim condition following the configuration change.
2) The abrupt altitude deviations are most likely being caused by the large variation of longitudinal stick force following a configuration change from cruise to power approach. During the initial gear and flap transition a small amount of forward input is required to compensate for the initial increase in lift or "ballooning" caused by the flap transition. Immediately following the small forward input a very large aft input was required and built so rapidly that the trim system could not trim out forces quickly enough. The initial "ballooning" tendency is most likely caused by the initial increase in lift caused by the transition of the flaps and slats. The large nose down

pitching moment is most likely caused by the flaps extending, which produces a large nose down pitching moment.

ALTITUDE MAINTENANCE EVALUATION CONCLUSIONS

Altitude maintenance during configuration changes must be easy during high pilot workload tasks such as flying an instrument approach during actual instrument conditions. The USNTPS specification for aircraft flying qualities specifically states that pitch trim changes in excess of 10 pounds are unsatisfactory. The effect of a large pitch trim change during reconfiguring the aircraft is complicated during formation approaches. The author has seen many formation instrument approaches and during the configuration change of the formation the large pitch trim changes frequently result in both aircraft fighting to maintain altitude and doing so 180 degrees out of phase with each other. This situation has resulted in near aircraft collisions and caused the wingman to become detached from the lead aircraft. This is a very dangerous situation due to both aircraft being so close following a lost sight situation.

A possible solution to correct this deficiency might be to maintain the load factor command system, that was recommended for the cruise configuration, through the flight control logic until the flaps have made the full transition. Additionally, an AOA blend could be implemented that would command an AOA at some predetermined airspeed that would correspond to the zero or nearly zero longitudinal trim position. This control system logic would then recognize that the configuration change occurred normally and then flow directly into the AOA command system to improve AOA maintenance, as previously discussed.

LATERAL DIRECTIONAL FLYING QUALITIES EVALUATION

CENTERLINE MAINTENANCE RESULTS

Centerline maintenance was evaluated in the power approach configuration during landings at the field and aboard the ship. The importance of evaluating this task aboard an aircraft carrier is due to two characteristics that are not found at the field. The first is the fact that the center-line of the angled deck is continuously moving as the ship moves forward. Secondly, the task of landing an aircraft aboard a carrier is a much higher gain task than a field landing, due to the increased precision of the task. These two characteristics alone make the task difficult and deficient flying qualities will just compound an already complicated task. During field landings, centerline maintenance was somewhat difficult due to an appearance of a hesitation of the aircraft to make small bank angle changes during centerline corrections. Due to this hesitation, many times the correction to centerline was over controlled, because too much angle of bank was commanded. This resulted in an overshoot of the centerline and the same over correction back to centerline. The result was continuous roll commands all the way to touch down. During ship-board approaches, the lateral inputs doubled in frequency and the workload of centerline maintenance was so high that scanning of the glide-slope and AOA were degraded. Centerline maintenance within +/- 10 feet was not possible with a tolerable pilot workload (HQR-7). Imprecise centerline maintenance results in a degraded scan of AOA and glide-slope deviations which are likely to result in frequent wave-off and possible ramp strikes.

Analysis of Imprecise Centerline Maintenance

The lateral/directional stability evaluation revealed the following characteristics:

1) The airplane exhibited an apparently positive dihedral effect during steady heading side-slip maneuvers. This characteristic was indicated by a linear increase of the lateral stick force into the side-slip.
2) During abrupt lateral step inputs of approximately 1 inch, apparent adverse yaw was indicated by the aircraft yawing away from the direction of roll. This was observed by monitoring both the heading indicator and a ground reference.

Review of the flight test data obtained by the author and previous flight test results (Luecke M. A., 1971) led to following conclusion about why centerline maintenance in the EA-6B is imprecise:

1) The adverse yaw characteristic combined with the positive apparent dihedral effect is most likely the cause of the initial bank angle hesitation noted during the handling qualities evaluation. The adverse yaw may have been caused by the increase drag on the high wing, thereby producing a side-slip. The side-slip causes the apparent positive dihedral effect to counter the commanded roll. This is most likely the cause of over controlling centerline corrections. The initial input does not produce the

desired roll command, and the hesitation results in an additional roll command and causes the over-control correction.

CENTERLINE MAINTENANCE EVALUATION CONCLUSIONS

As explained earlier, the lateral/directional stability problem is a difficult one to analyze. However, the CAS provides various ways to correct this deficient flying quality. The problem may be as simple as feeding back side-slip to the rudder to better coordinate the turn. Many augmented airplanes incorporate an Aileron Rudder Interconnect (ARI) and the use of the FCC to schedule rudder input with lateral stick input in different amounts for varying airspeed and configurations. As with the previous longitudinal examples, additional sensors may be needed to provide the necessary information to the FCC.

Finally, no departure or high angle of attack testing was conducted by the author, but if side-slip can be minimized, the aircraft could maintain a full stall without the fear of departure or spin. It is well documented that when the wing of an aircraft is stalled, there must be a force to begin the rolling and yawing motion required to develop a spin. Additionally, it is well documented that every EA-6B that has entered into a spin has been lost. The inability to recover the aircraft from a spin is thought to be caused by the large yaw inertia due the heavy jamming pods being loaded on the outboard wing stations. Further testing will be required to complete this objective, but it should be noted that at high enough angles of attack, the rudder may become less effective and may not be suitable to accomplish this last objective. A possible solution, if the rudder effectiveness becomes limited, would be to use the CAS to limit the amount of AOA that may be commanded by the pilot. This will prevent the situation where rudder ineffectiveness might occur.

SUMMARY

The current AFCS computer in the EA-6B is going to be replaced due to maintainability and reliability issues with the AFCS computer. Replacing this computer will definitely correct these problems. Because of the increased processing power of the new FCC and because the current flight control system is mechanized with a series path, a CAS can be implemented without major aircraft modifications. Flight control law development will have to be completed to provide the aircraft with the basic function of stability augmentation that exists currently, as a result of replacing the AFCS computer.

The deficient flying qualities have been well documented by the fleet training squadron, original prototype testing, and the most recent flying quality evaluation conducted by the author. Changing the dynamics of the airplane to correct the deficient flying qualities will require additional flight control law development and possibly additional sensors. The result of this effort will improve the flying qualities of the aircraft, produce an easier aircraft to fly, and ultimately lead to an increase of its mission effectiveness. Improved flying qualities will most likely reduce the time to train student Naval Aviators and subsequently reduce the cost of training. Improved flying qualities will also enhance aircrew training with the implementation of the control strategies recommended by the author. Finally, such an improved system is likely to reduce aircraft losses due to spin entry since every EA-6B that has entered into a spin to date has been lost. The cost of implementing a CAS capability has been estimated at 80 million dollars for the development, test, and implementation into the entire fleet of EA-6Bs' (Appendix G). The cost of replacing one EA-6B is approximately 75 million dollars. The cost of replacing an aircrew is priceless.

CHAPTER 5: RECOMMENDATIONS

Based upon the research performed during the course of this thesis, the author concludes that the technology to incorporate a CAS to correct the deficient flying qualities in the EA-6B exists. Based upon the extensive personal EA-6B flight experience (fleet pilot and test pilot) of the author, and the documentation of the EA-6B's deficient flying qualities, the CAS should be incorporated in the EA-6B with the replacement of the current AFCS computer and should include control law development to improve flying qualities. Specific recommendations are as follows:

1) Investigate the feasibility of incorporating a CAS in the current EA-6B configuration. It would be required to:
    a) Define EA-6B stability augmentation actuator limitations
    b) Determine the types of control stick sensors required in a CAS to correct the deficient flying qualities discussed in chapter 4
    c) Determine the effects of control system friction on the choice of control stick sensors chosen to correct deficient flying qualities
    d) Define EA-6B flight control actuator limitations
2) Investigate the feasibility of the development of the control laws to necessary to correct the deficient flying qualities described in chapter 4.
3) Investigate the feasibility of developing control laws to limit the AOA in certain abnormal configurations.

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APPENDICES

DESCRIPTION OF THE EA-6B FLIGHT CONTROL SYSTEM

GENERAL

The EA-6B, depicted in figure A-1, is a four-place, twin-engine, all weather, electronic warfare airplane manufactured by Northrop Grumman Aerospace Corporation which incorporates a hydraulically powered, irreversible flight control system.

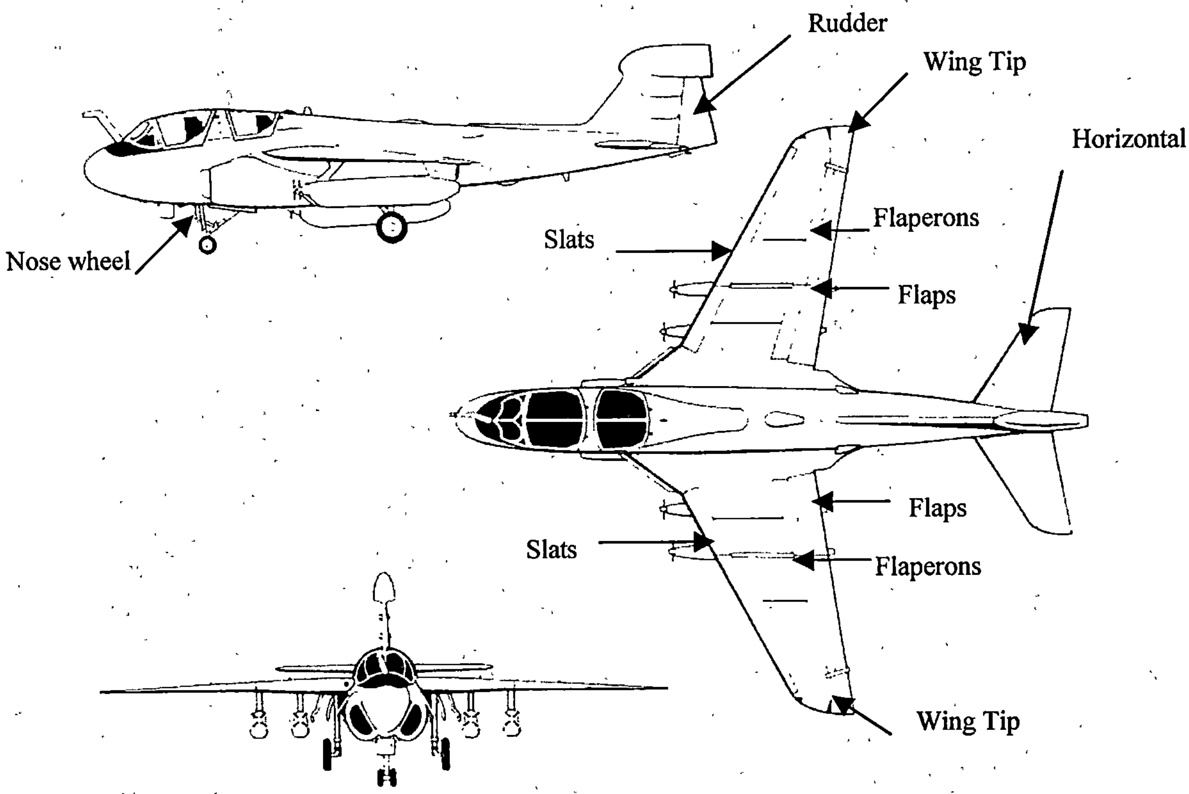


Figure A-1: EA-6B AIRPLANE

The primary flight control surfaces, also depicted in figure A-1, consist of a horizontal stabilizer, rudder and upper wing surface spoilers (flaperons). Cockpit controls and displays are presented in figures B-1 and B-2. A conventional center stick and rudder pedals are linked directly to the respective control surface actuators by a system of pushrods, linkages and bellcranks to provide pitch, roll and yaw control. The stabilizer actuator, the flaperon auto-pilot actuator, and the rudder actuator consist of electrohydraulic tandem power control cylinders which direct control surface movement in response to mechanical commands from the pilot. Control deflection characteristics are shown in table C-1. Electrical commands from the Automatic Flight Control System (AFCS), or a combination of electrical AFCS commands and mechanical pilot inputs. Electrical flight control system requirements are shown in table C-2. The lateral control system also includes four mechanical tandem type flaperon hydraulic actuators (two in each wing) which receive mechanical-only lateral inputs through the flaperon auto-pilot actuator and command the appropriate flaperon deflection. Control stick and rudder pedal inputs position an input slide valve on the respective actuator to command the associated control surface deflection, and a proportional followup linkage holds the control surface fixed relative to the pilot's control deflection. Each control surface actuator is powered by 3000 psi hydraulic pressure from two independent systems; the flight and the combined primary systems. A more complete description of the EA-6B hydraulic system is presented in appendix D. In addition, extended rudder throw and stabilizer travel are provided for takeoff, landing and spin recovery whenever the flaps are extended or the ASSIST SPIN RECOV switch is actuated.

ARTIFICIAL FEEL

Artifical feel is provided in all three axes through the use of mechanical devices. The longitudinal system incorporates a double-acting bungee (two-gradient feel spring), a g-sensing positive bob-weight, pitch-acceleration-sensing positive and two negative bob-weights, and a leaf spring and eddy-current damper. Control feel in the lateral axis is provided by a double-acting bungee and an eddy-current damper. A cam and roller device loaded by two extension springs provides the artificial feel in the directional axis. Flight control system schematics for each axis are presented in figures B-3 through B-5.

TRIM SYSTEM

Trim controls depicted in figure A-1 drive irreversible electromechanical trim actuators in each axis. The longitudinal and lateral trim is actuated through a conventional five-position switch located on the control stick grip which is spring-loaded to the off (center) position. Longitudinal trim varies the neutral position of a feel bungee at two-speeds; stabilizer displacement at a rate of approximately 1° per second during manual and stability augmentation (STAB AUG) operation and approximately 1/20° per second during auto-pilot (AUTO) operations. Lateral trim repositions the flaperon feel device to a new no-load position. With the AUTO mode selected, the manual trim button on the control stick grip is deactivated, and the longitudinal and lateral trim functions are performed automatically. Directional trim is controlled by a switch located on the pilot's left console and is available in all operating modes. The rudder trim switch provides directional trim by varying the neutral position of a mechanical linkage. The stabilizer and rudder trim gages are located on the pilot's lower instrument panel, as depicted in figure B-2. The flaperon trim indicator is mounted on the forward side of the control stick grip adapter as presented in figure B-1. A LAT/LONG TRIM ACT circuit breaker and a RUDDER TRIM circuit breaker located on the forward main circuit breaker panel in the forward cockpit provide a means of interrupting electrical power to the respective trim systems.

AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS)

The automatic Flight Control System (AFCS) is an electromechanical system designed to provide automatic control and damping about the airplane's pitch, roll and yaw axes. The AFCS incorporates electromechanical error sensing, coupling, and amplifying stages to provide two-axis stability-augmentation (pitch and yaw), three-axis attitude control and automatic flight-path control. The AFCS consists of an AFCS servo system and the airplane flight control system and is designed to provide electrohydraulic and mechanical coupling to deflect each control surface. The AFCS servo system is designed to provide electromechanical hydraulic power amplification of shaped error or correction signals for each airplane control axis. Each axis flight control system (i.e. hydraulic power actuator and control linkage) is proportionally driven by the output of the respective AFCS servo signals or through linkages from the pilot's manual inputs. The AFCS operating modes are manually selected via the auto-pilot controller depicted in figure B-1, and provide varying degrees of airplane control.

1. <u>Manual mode (Auto-pilot switch OFF)</u>: No stability augmentation is provided, and pilot control displacements are the only inputs to the control surface actuators to command surface deflection.
2. <u>Stability-Augmentation (STAB AUG) mode</u>: Designed to automatically damp out oscillations in the pitch and yaw axes and provide turn coordination by positioning the stabilizer and rudder proportionately to airplane pitch, roll, and yaw rates.

3. Auto-pilot (AUTO) mode: The basic hands-off operating mode of the AFCS designed to provide attitude hold (pitch, bank and heading hold), altitude hold, Mach hold, and the capability to couple the AFCS to Automatic Carrier Landing System (ACLS) pitch and roll commands through the AN/ASW-25 data link. The control stick incorporates auto-pilot disengage switches in the column that cause the AFCS to revert to the STAB AUG mode whenever the stick is grasped. When the stick is released, the AUTO mode resumes operation.

An AFCS and an AFCS / AIR DATA CMPTR circuit breaker are located on the aft main circuit breaker panel and can secure electrical power to the Air Navigation Computer (ANC), Stabilizer Trim Actuator, Lateral and Normal Accelerometer transmitters, Air Data Computer (ADC), and Auto-pilot controller. In addition, the yellow and black barber poled AFCS emergency disconnect switch located on a junction box immediately below and to the right of the control stick grip (figure B-1) disconnects electrical power from all AFCS circuits when momentarily actuated. AFCS emergency disconnect switch actuation also returns all switches and pushbuttons on the auto-pilot control panel to the OFF position.

SECONDARY FLIGHT CONTROLS

Secondary flight controls, depicted in figure A-1, consist of DC electrically controlled, hydraulically actuated semi-Fowler type, slotted, trailing edge flaps and leading edge slats. Wing tip speed brakes are AC electrically controlled, hydraulically actuated and consist of a set of split trailing-edge surfaces at the tip of each wing, outboard of the flaps. The nose wheel steering (NWS) system provides directional control and shimmy damping through a DC electrically controlled nose wheel steering button on the control stick, as depicted in figure B-1. The wheel brakes are mechanically controlled by toe pressure and incorporate a DC electrically controlled anti-skid system. The NWS and wheel brake systems are hydraulically actuated and designed to provide directional control while the anti-skid system is designed to increased high-speed braking effectiveness. The primary and secondary flight control surface deflection limits are presented in table C-1. Electrical power for the secondary flight controls and trim system is provided by any AC generator and transformer-rectifier (T/R) combination, or an emergency ram air turbine (RAT) driven AC generator and emergency transformer-rectifier. The primary and secondary flight control system electrical power requirements are outlined in tables C-1 and C-2, and a more detailed description of the EA-6B electrical system is provided in appendix E. Additional discussion of the secondary flight control systems is presented in appendix F.

PRIMARY FLIGHT CONTROL SYSTEM

<u>LONGITUDINAL CONTROL SYSTEM</u>

Longitudinal control is provided by an electrohydraulic mechanical control system. A schematic of the longitudinal flight control system is presented in figure B-3.

Horizontal Stabilizer Control System

The horizontal stabilizer control system is an irreversible mechanical system powered by an electrohydraulic tandem power control cylinder and consists of a control stick linked to the stabilizer actuator through a system of linkages, pushrods and bellcranks.

The stabilizer is the all-movable type consisting of interconnected right and left halves that rotate as a unit in response to fore and aft manual control stick inputs and to AFCS electrical signals introduced at the stabilizer actuator. Both flight and primary combined hydraulic system pressure drives the power pistons to command horizontal stabilizer movement. The longitudinal control system also incorporates a nonlinear stick-to-stabilizer gearing ratio in both the flaps up and flaps down configurations, as depicted in figure A-2, which is designed to provide increased stick motion per unit stabilizer deflection at the more sensitive areas of maneuver control.

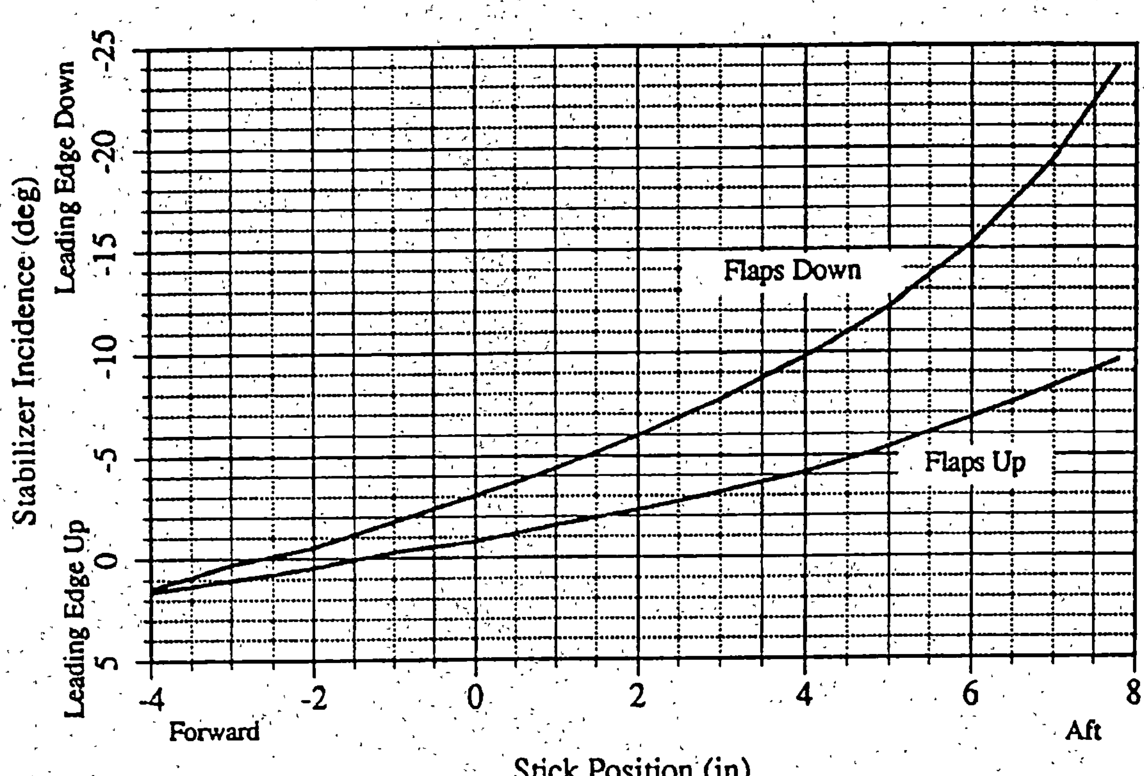


Figure A-2: EA-6B LONGITUDINAL STICK-TO-STABILIZER GEARING

Artificial longitudinal control feel is provided by a double-acting bungee, a g-sensing positive bob-weight, pitch-acceleration-sensing positive and negative bob-weights, and a leaf spring and eddy-current damper.

Horizontal Stabilizer Actuator

The horizontal stabilizer actuator, located at the top center of the fuselage near the stabilizer leading edge, consists of a power valve shuttle, two tandem-mounted power pistons, an electrohydraulic servo valve, a parallel-mode solenoid valve and a series-mode solenoid valve as depicted in figure A-3.

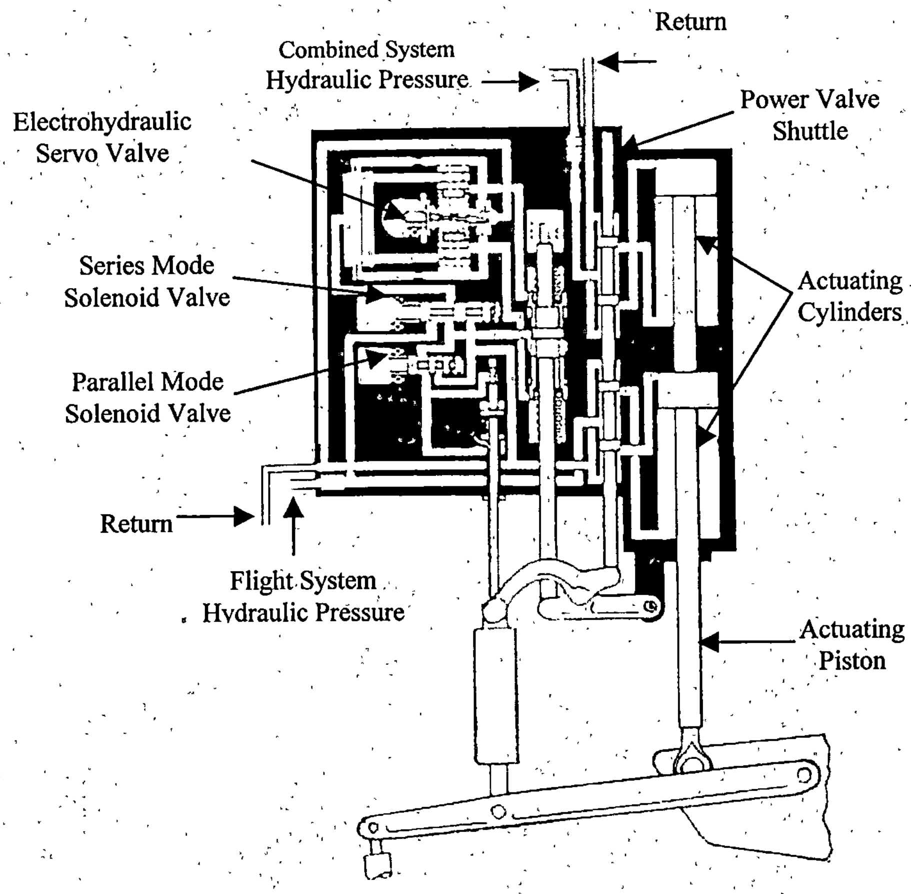


Figure A-3: STABILIZER ACTUATOR

The horizontal stabilizer actuator is unique in reference to the primary flight control actuators in that it has provisions for a manual mode, and a parallel and series mode of operation associated with the AFCS.

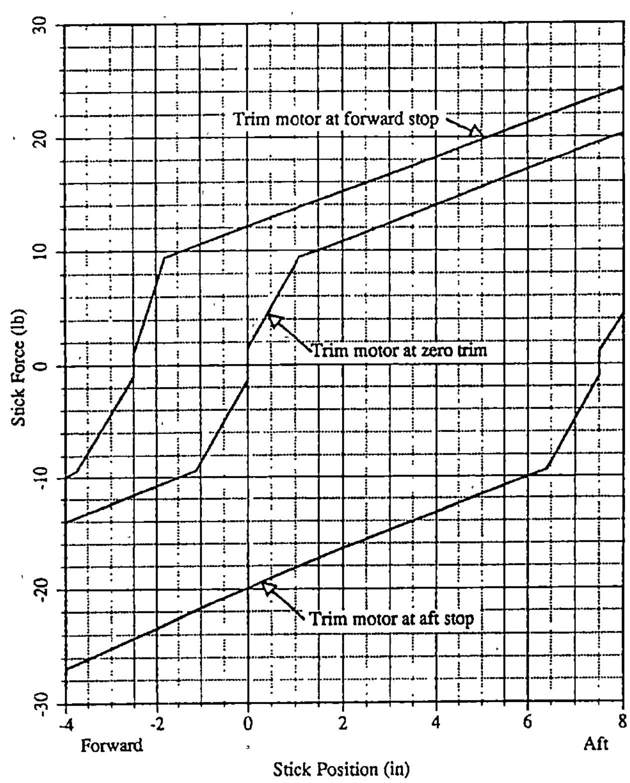
3. <u>Manual mode of operation (AFCS OFF):</u> Designed such that pilot inputs alone control the power valve shuttle which directs pressure to the actuator pistons. The hydraulic pressure moves the pistons and the stabilizer to which they are mechanically linked. For constant-velocity stick inputs, the stabilizer is designed to move at a constant speed. When the stick input stops, the power valve shuttle returns to neutral and the stabilizer movement stops.
4. <u>Series mode of operation (STAB AUG or AUTO engaged):</u> Input signals from the AFCS are designed to be used independently or combined with manual pilot inputs to control stabilizer movement. The STAB AUG and AUTO modes automatically damp oscillations about the pitch axis by programming stabilizer deflections proportionately in response to pitch rates. This programmed movement of the control surface is not transmitted to the control stick. The series-mode solenoid valve is energized from the AFCS auto-pilot controller, porting flight hydraulic system pressure to the electrohydraulic servo valve. The servo valve directs movement of the power valve shuttle, which produces a corresponding stabilizer deflection relative to DC electrical signals from the AFCS amplifier or manual longitudinal stick inputs. When operating in the series mode, control surface displacement is not reflected at the control stick. In manual and stability-augmentation (STAB AUG) flight modes, the control stick applies a mechanical input to the stabilizer hydraulic actuator's input slide valve to generate control surface movement. The control surface deflection is held in a fixed relationship to the pilot's control deflection by a proportional follow-up linkage. Two hydraulic check valves located at both the combined and flight hydraulic system pressure ports on the stabilizer actuator, permit free flow to the actuator but prevent backflow.

5. <u>Parallel mode of operation (ACLS engaged):</u> Designed such that input signals from the AFCS alone control stabilizer movement. Both series-mode and parallel-mode solenoid valves are energized. Flight hydraulic system pressure is ported to the electrohydraulic servo valve, which drives the power valve shuttle relative to DC electrical signals from the AFCS. In the parallel mode, the control stick and linkages follow the motion of the control surface. To override the AFCS in this mode, a longitudinal stick force of 24 lbs is required to overpower the lockout actuator. Rate switches are incorporated to sense and prevent a hardover failure, and if necessary, revert the actuator back to the series mode of operation. A hardover signal, if not corrected immediately, could cause an uncontrollable condition in the longitudinal control system response.

A stabilizer shift mechanism mechanically programs the stabilizer position with flap deflection, and is designed to minimize the effects of flap trim changes, provide extended stabilizer throws when the flaps are extended, provide maximum control and reduce the possibility of overstress under all flight conditions. The normal travel of the horizontal stabilizer is 1.5° leading edge up to 9.6° leading edge down. When the flaps are extended or ASSIST SPIN RECOVERY is selected, the stabilizer travel increases to 1.5° leading edge up to 24° leading edge down, as illustrated in figure 2. In both configurations, the longitudinal control stick deflection ranges from 4.0 inches forward to 7.8 inches aft. When the stabilizer shifts to extended throws, the stabilizer position indicator, located on the integrated position indicator (IPI) on the left side of the pilot's instrument panel (figure B-2), presents a symbol of a stabilizer. The word CLEAN is displayed on the stabilizer position indicator when normal stabilizer travel is available. The stabilizer position indicator is powered by the essential 28 VDC bus through the CAUTION LT circuit breaker on the forward left circuit breaker panel. Restricted stabilizer travel at low speed may result in inadequate pitch control and extended stabilizer travel at high speed may cause structural failure. If the horizontal stabilizer disconnects, the stabilizer may deflect full leading edge up, full leading edge down, or float free in an intermediate position.

CONTROL SYSTEM FORCES

The stabilizer control system is irreversible and incorporates conventional artificial feel to simulate inflight stick loads through a double-acting bungee (two-gradient feel spring), positive and negative bob-weights, and a leaf spring and eddy-current damper. The control stick incorporates auto-pilot disengage switches in the column, so the stick grip moves to a slight degree independent of the control stick. The variation of longitudinal stick force with stick deflection at various trim settings is illustrated in figure A-4.



FigureA-4: EA-6B LONGITUDINAL CONTROL FEEL

Double-Acting Bungee

The double-acting bungee (two-gradient feel spring) is designed to provide three characteristics:

1. Stick force proportional to stick deflection.
2. A greater breakout force gradient (approximately 7.16 lbs/in) near stick neutral to ensure reasonable speed stability about trim.
3. A lower gradient in the more forward and aft stick positions (approximately 1.61 lb/in. beyond ±1.15 inches of stick travel from neutral).

The longitudinal breakout force required to initiate control surface movement at all attainable flight conditions is designed at 1.1 lbs, and the force required to actuate the control stick auto-pilot disengage switches is designed to be 1.1±0.2 lbs.

Bob-weights

The g-sensing positive bob-weight is located at the base of the control stick on the control linkage aft of the pilot's seat and is designed to provide a maneuvering force gradient of approximately 3.15 lb/g. The designed maneuvering gradients for the system, using the EA-6B's limit load factor of 5.5, are shown in table A-1. Pitch-acceleration-sensing positive and negative bob-weights, both forward and aft of the center of gravity, are designed to provide dynamic pitch damping for the stabilizer control system and a pitch acceleration sensing coefficient, B_{nps}, of approximately 15.1 ft-lb/rad/sec^2.

Table A-1: EA-6B MANEUVER GRADIENTS (1)

Gradient	MIL-F-8785C Limits (lb/g)	EA-6B Maneuver Gradient (lb/g)
Maximum Gradient	$(240/(n/\alpha))^* \leq Fs/n \leq (56/(n_L-1))$ * to a limit of 28.0	12.45
Minimum Gradient	The higher of $(21/(n_L-1))$ and 3.0	4.25

Note: (1) Data from reference 2, Grumman Aerospace Corporation Report, ENG/EA6B RPT-92-001, April 1992.

A vertical bob-weight, located at the sloping bulkhead, is designed to compensate for variations in stick force per g with stick position introduced by imbalances in the linkage assembly and the balance weight at fuselage station 524. Another balance weight of 4.2 lb, located at fuselage station 534.0, is combined with the aft control system linkage and designed to de-couple the control system from the 18 Hz stabilizer bending mode by reducing overall gain of the linkage motion per unit fuselage motion.

Leaf Spring and Eddy-Current Damper

The leaf spring and eddy-current damper provides stick rate damping designed to minimize the effects of coupling between control system and airplane natural frequencies. The damper assembly is designed to generate lead to the control system to provide the required stick force compensation to the stabilizer for the short period mode. The damping assembly consists of a linear feel spring grounded through an eddy current damper and is designed to generate damping forces proportional to the control stick velocity (approximately 0.456 lb/in/sec). The eddy-current damper, located on the cockpit floor forward of the control stick, is a magnetic-mechanical device designed to produce a resisting torque proportional to the stick velocity. If the damper locks, the system can be overridden, but if the damper breaks free (free-wheels), the velocity of fore and aft stick inputs is unrestricted. Abrupt longitudinal stick inputs can disengage the eddy-current damper, however, moving the stick fore and aft through the detent should reset the damper.

LONGITUDINAL TRIM SYSTEM

Longitudinal trim is provided by forward and aft movement of the conventional five-position trim switch located on the control stick, as depicted in figure B-1. The trim switch is 115 VAC electrically operated and spring-loaded to the off (center) position. For airplanes incorporating AFC 745, the trim switch provides inputs to a trim relay box which requires 28 VDC to energize the associated nose up or nose down relays. The trim switch controls a two-speed, irreversible, electromechanical stabilizer trim actuator which is mechanically linked (in series) to and varies the neutral position of the stabilizer control system artificial feel bungee. The stabilizer trim actuator also receives inputs from the AFCS. The actuator consists essentially of one high-speed and one low-speed, 115 VAC, 400 Hz, single-phase motor, a gearbox, and a threaded power screw. The two motors drive through a planetary gear system to obtain a two-speed rate of power screw motion; high speed for manual trimming and low speed for automatic trimming. For manual operation (AFCS OFF or STAB AUG ON), longitudinal trim is designed to provide a stabilizer displacement rate of approximately 1° per sec. However, during auto-pilot (AUTO) operations, the trim switch on the control stick is deactivated and longitudinal trim functions are performed automatically at a decreased stabilizer displacement rate of approximately 1/20° per sec. The trim limits with flaps UP are 3/4° leading edge up to 9° leading edge down, and 1/2° leading edge up to 22° leading edge down with flaps DOWN. In the event of a trim switch malfunction, occasional use of the AFCS AUTO mode will assist the pilot in overcoming excessive stick forces resulting from attitude, configuration, or airspeed changes. The stabilizer trim gage, located on the pilot's lower instrument panel as depicted in figure B-2, indicates horizontal stabilizer trim position from 3 units nose down to 12 units nose up. The letter N on the trim gage represents the neutral position of the stabilizer surface (3° leading edge down) in the flaps down configuration. The trim pointer is positioned by a DC motor, which is driven by a signal generated by the stabilizer position transmitter. The LAT/LONG TRIM ACT circuit breaker located on the forward main circuit breaker panel in the forward cockpit provides a means of interrupting electrical power to the longitudinal trim system.

STABILIZER SHIFT MECHANISM

The stabilizer shift mechanism consists of a cable and drum assembly mounted on a shaft extending from the flap gearbox in the right wing root. Stabilizer travel is increased for takeoff, landing and spin recovery whenever the flaps are extended by the normal or emergency method, or when the ASSIST SPIN RECOV switch is actuated. As the flaps extend to the 20° or 30° position, the drum rotates and the cable unwinds, increasing the horizontal stabilizer travel to 1.5° leading edge up to 24° leading edge down and extending the rudder throw to ±23°. The flap gearbox is designed to handle loads up to the breaking strength of the cables; however, if the stabilizer shift cable fails, cycling the flaps/slats in either direction may result in a split flap condition due to possible entanglement of the cable with the flap drive linkage. The ASSIST SPIN RECOV switch allows selection of extended stabilizer throw for maximum control surface deflection without lowering the flaps. The guarded switch, located on the forward right edge of the throttle quadrant (figure B-1), electrically energizes a selector valve to port primary combined hydraulic fluid to a shift cylinder. The cylinder automatically shifts the stabilizer and rudder control surfaces into the extended throw authority. Operation of the

ASSIST SPIN RECOVERY system with the flaps down can result in damage that will prevent reselection of cruise control throws. In addition, operation of the ASSIST SPIN RECOV switch results in a nose-up pitch (hands off) which may overstress the aircraft if actuated during accelerated maneuvers, or when operating at minimum airspeeds, may pitch the aircraft up into stall or heavy buffet. An assist spin recovery battery has been incorporated into Block 89 airplanes to provide emergency DC power to the assist spin recovery circuit and the yaw rate indicator when no other DC source is available.

LATERAL CONTROL SYSTEM

Lateral control is provided by an electrohydraulic mechanical control system. A schematic of the lateral flight control system is presented in figure B-4.

FLAPERON CONTROL SYSTEM

The flaperon control system is an irreversible mechanical system powered by left and right flaperon hydraulic actuators through an electrohydraulic tandem-type flaperon auto-pilot actuator. The flaperon control system consists of a control stick linked to the flaperon auto-pilot actuator and flaperon hydraulic actuators through a system of linkages, pushrods and bellcranks. The left and right flaperon tandem-type hydraulic actuators are powered by either the flight or primary combined hydraulic system pressure, and translate lateral stick movement into flaperon control surface deflection. There are two flaperon surfaces on each wing separated at the wing fold, and each flaperon surface is deflected by its respective flaperon hydraulic actuator. The flaperon auto-pilot actuator is powered by the flight hydraulic system only and responds to mechanical inputs as well as electrical commands from the AFCS to direct left and right flaperon movement through the flaperon hydraulic actuators. Stick movement left of neutral results in upward deflection of the left flaperons while the right flaperons remains flush with the wing, and vice versa. The maximum flaperon deflection is 51° except as limited by aerodynamic hinge moments and available actuator power. Full flaperon deflection corresponds to ±4.5 inches of lateral control stick displacement, which is limited by stop-bolts attached to the pedestal. The lateral stick-to-flaperon gearing ratio shown in figure A-5 is designed to be essentially linear relative to stick position, and lateral artificial feel is provided via a double-acting bungee, and an eddy-current damper.

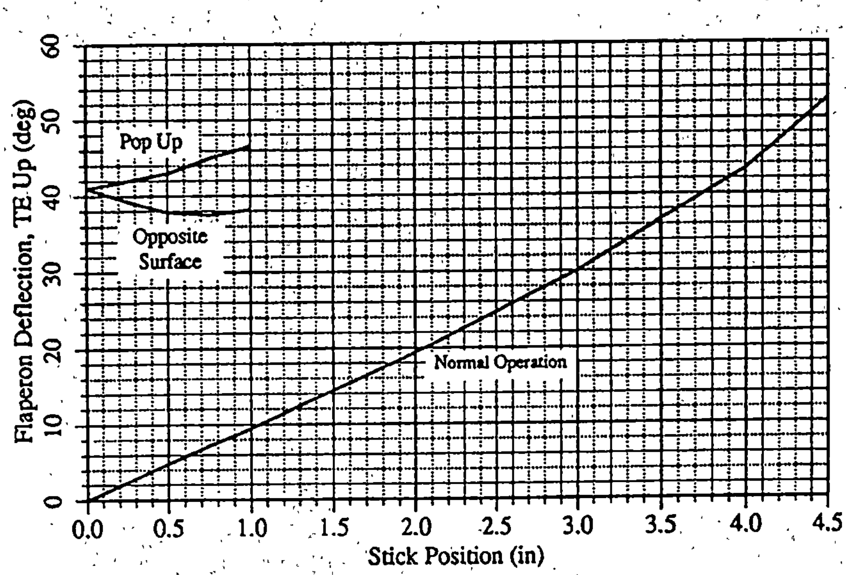


Figure A-5: EA-6B LATERAL STICK-TO-FLAPERON GEARING

Flaperon Auto-pilot Actuator

The flaperon auto-pilot actuator depicted in figure A-6 is a hydraulic servo unit designed to operate in two modes; manual mode or series mode (associated with the AFCS).

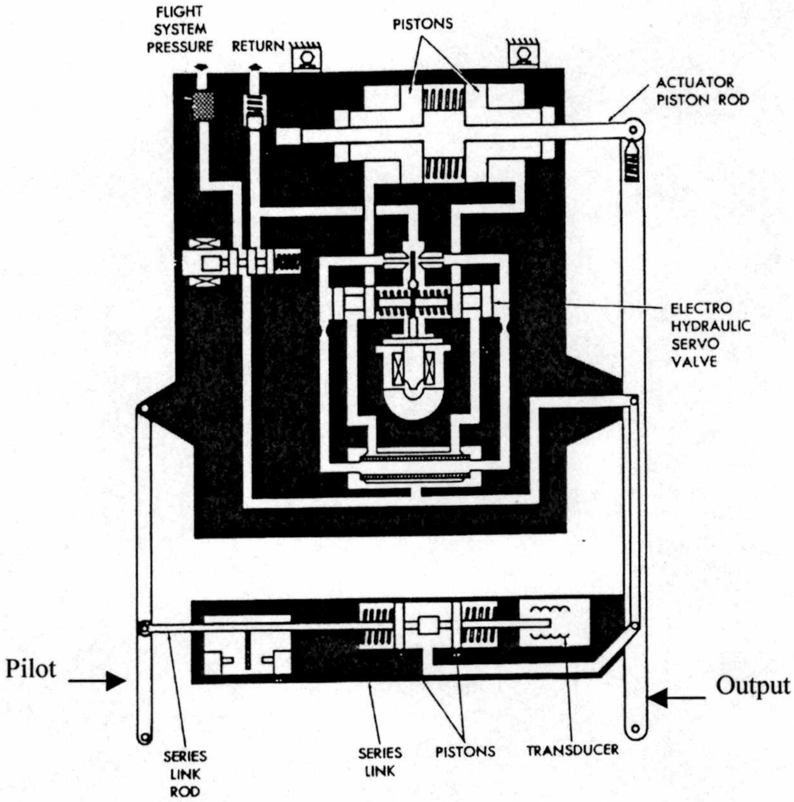


Figure A-6: FLAPERON AUTO-PILOT ACTUATOR

The actuator is designed to transmit only mechanical inputs to the flaperon hydraulic actuators in each wing.

1. Manual mode of operation (AFCS OFF): The solenoid valve is deenergized and no hydraulic fluid is ported to any portion of the actuator. The series-link cylinder contains two spring-loaded pistons designed to clamp the two halves of the series-link rod together to form a rigid link. Lateral control stick inputs are transferred directly through the input lever and the series link to the output lever.

2. Series mode of operation (AFCS engaged – STAB AUG, AUTO, or ACLS): The flaperon auto-pilot actuator receives input signals from the AFCS, which are used independently or combined with manual control stick inputs to direct flaperon movement.

AFCS signals drive the actuator piston rod and output lever, commanding the appropriate flaperon deflection. Manual pilot inputs generate an electrical signal based on the relative motion between the input and output levers. This manual pilot input signal is combined with the AFCS signal and converted to a mechanical movement, which in turn, actuates the flaperons via the flaperon hydraulic actuators.

Flaperon Hydraulic Actuators

Two identical, purely hydro-mechanical, flaperon actuators are mounted in each wing and directly control the flaperon movement in response to mechanical inputs. A schematic of a flaperon hydraulic actuator is presented in figure A-7.

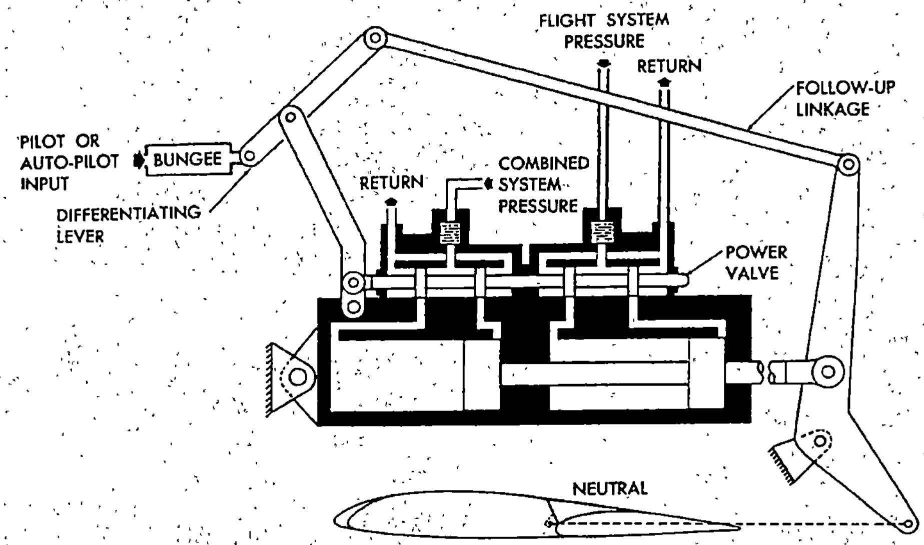


Figure A-7: FLAPERON HYDRAULIC ACTUATOR

Pilot or AFCS generated mechanical commands are introduced through a load relief bungee and direct hydraulic pressure to two tandem-mounted power pistons. The power pistons are mechanically connected to the flaperons and are powered by both flight and primary combined hydraulic pressure to ensure complete operating capability is retained if either system fails.

In manual and stability-augmentation (STAB AUG) flight modes, the control stick applies a mechanical input to the flaperon hydraulic actuator's input slide valve to generate control surface movement. The control surface deflection is held in a fixed relationship to the pilot's control deflection by a proportional follow-up linkage. With AUTO mode selected, the AFCS provides heading hold or will maintain any selected bank angle between 5° and 60°. Automatic heading hold is inhibited when the ACLS mode switch is placed in the HDG OFF position, and the auto-pilot will hold any bank angle between 0° and 60°. A ball-lock detent permits disengagement of the flaperon auto-pilot actuator in the event of an actuator malfunction. The AFCS may be manually overridden by applying a lateral stick force of 25 lbs.

CONTROL SYSTEM FORCES

The flaperon control system is irreversible and incorporates a conventional artificial feel system using a double-acting bungee and an eddy-current damper.

Double-Acting Bungee

The double-acting, single gradient spring bungee is designed to provide an initial lateral preload and a linear force-gradient over the displacement range of the stick as depicted in figure A-8. The lateral bungee is similar to the longitudinal bungee but produces a lower magnitude force. The designed lateral breakout force is 1.1 lb.

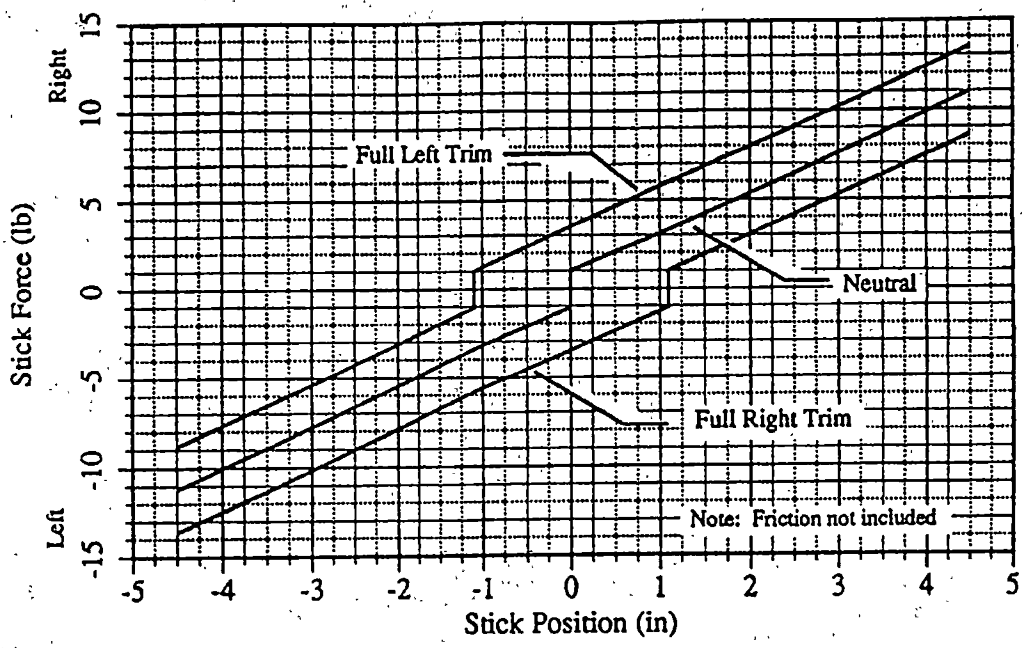


Figure A-8: EA-6B LATERAL CONTROL FEEL

Eddy-Current Damper

The eddy-current damper is designed to provide lateral stick rate damping similar to the longitudinal damper.

LATERAL TRIM SYSTEM

Lateral trim is provided by left and right movement of the conventional five-position trim switch located on the control stick. The trim switch is 115 VAC electrically operated and spring-loaded to the off (center) position. The trim switch controls an irreversible electromechanical linear-type actuator, which repositions the flaperon feel bungee to a new no-load position. Total flaperon deflection through use of the trim switch is 9° up at a rate of approximately 1.2° per second. The lateral trim indicator, located on the forward base of the control stick as depicted in figure B-1, is calibrated in three units left and right of zero (neutral) and is mechanically operated. During auto-pilot (AUTO) operations, the trim switch on the control stick is deactivated and lateral trim functions are performed automatically. The LAT/LONG TRIM ACT circuit breaker located on the forward main circuit breaker panel in the forward cockpit provides a means of interrupting electrical power to the lateral trim system. On airplanes incorporating AFC 745, a trim relay box assembly is installed in the trim circuitry between the trim switch and the trim actuator and is designed to reduce the electrical shock hazard to the pilot. The trim relay box assembly requires 28 VDC power to energize the right or left wing down relays and complete the circuit supplying 115 VAC power to the trim actuator.

FLAPERON POPUP SYSTEM

The flaperon popup system is selectable by the pilot through the flaperon popup ARM-OFF switch located on the pilot's left console as depicted in figure B-1. The system is designed to reduce lift and shorten ground roll after landing by commanding symmetric deployment of the flaperons. With ARM selected, the flaperons are driven to the 41° trailing edge up position by combined secondary hydraulic pressure when the left weight-on-wheels switch is actuated and the throttles are reduced to idle. The system also requires the wings to be spread and locked for the flaperon popup to function. With the flaperons in the popup position, lateral control stick movement is limited to ±1 inch from neutral as a small amount of asymmetric deflection is retained for control in crosswinds.

DIRECTIONAL CONTROL SYSTEM

Directional control is provided by an electro-hydraulic mechanical control system. A schematic of the directional flight control system is presented in figure B-5.

RUDDER CONTROL SYSTEM

The conventional rudder control system is an irreversible mechanical system powered by an electro-hydraulic tandem-type rudder actuator. The rudder pedals are linked directly to the control surface actuator through a system of linkages, pushrods and cables to provide yaw control. A rudder pedal adjustment handle, depicted in figure B-1, permits the pedals to be adjusted in 1 inch increments from 4 inches forward to 5 inches aft from neutral. The rudder actuator responds to mechanical commands from the pilot via rudder pedal displacements and electrical commands from the AFCS to control rudder deflection. The directional control system is designed to provide a linear gearing of rudder deflection with pedal displacement, as depicted in figure A-9.

Rudder Actuator

The tandem-type hydraulic rudder actuator is powered by both the flight and primary combined hydraulic systems. Displacement of the rudder pedals positions an input slide valve, which directs hydraulic pressure to the appropriate side of two tandem-mounted power pistons, which generate the corresponding rudder surface deflection. A proportional follow-up linkage maintains the rudder position relative to the pilot's pedal deflection. The rudder actuator, depicted in figure A-10, is designed to operate in two modes; manual mode or series mode (associated with the AFCS).

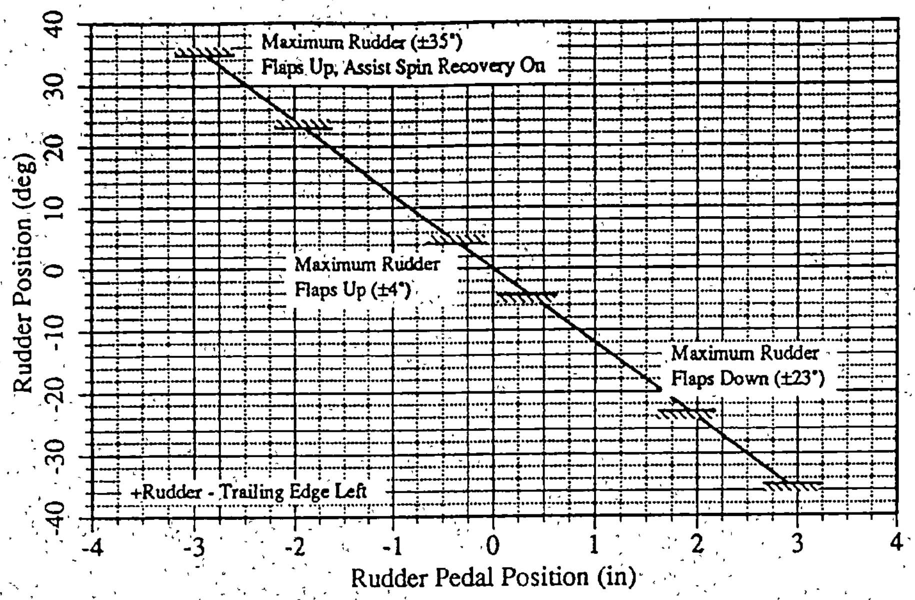


Figure A-9: EA-6B RUDDER PEDAL-TO-RUDDER GEARING

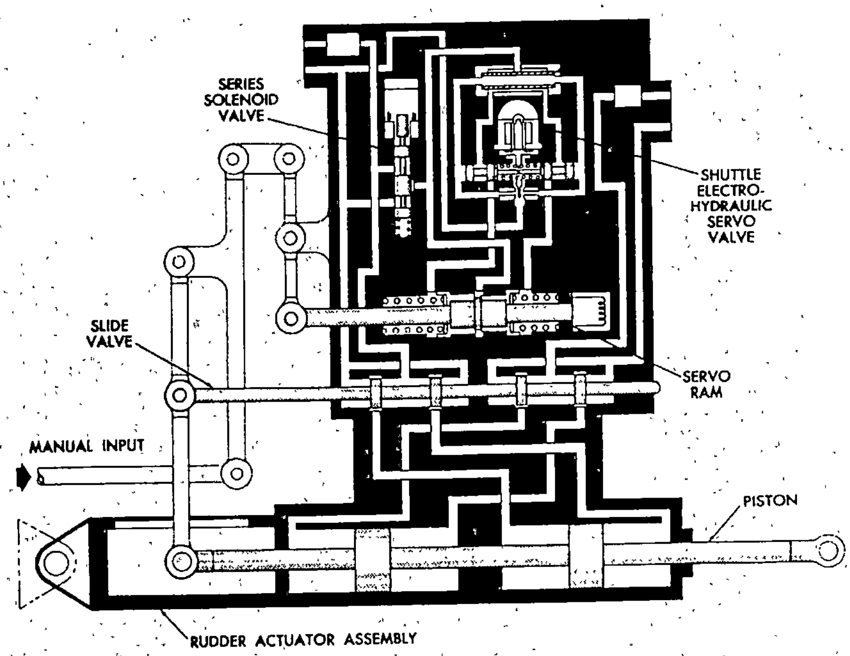


Figure A-10: RUDDER ACTUATOR

1. Manual mode of operation (AFCS OFF): Pilot input alone controls the power valve, directing hydraulic pressure to the two tandem-mounted power pistons which are mechanically linked to the rudder. For constant-velocity pilot inputs, the rudder deflects at a constant speed. When the pilot input stops, the power valve shuttle returns to neutral and the rudder stops
2. Series mode of operation (AFCS engaged – STAB AUG, AUTO, or ACLS): The series-mode solenoid valve is energized from the AUTO PILOT controller located on the center console (figure A-1). Input signals from the AFCS may then be used independently or combined with mechanical control stick inputs to command rudder movement.

23° Rudder Stop Actuator

The 23° rudder stop actuator is a single-acting actuator requiring normal 3000 psi primary combined hydraulic system pressure. The actuator is designed to restrict rudder movement to 23° left or right of neutral with flaps down and weight off wheels. The rudder is mechanically shifted by a cam stop mechanism that is actuated in turn by the stabilizer shift cable as the flaps deflect, providing extended rudder throws for landing. On deck with weight on wheels, the 23° rudder stop system is de-energized and the stops are removed. In this situation, if the flaps are extended, rudder deflection ± 35° is available.

ASSIST SPIN RECOVERY SYSTEM

In the clean configuration, extended rudder throw of ±35° is available through actuation of the ASSIST SPIN RECOV switch, which electrically energizes a selector valve to port primary combined hydraulic fluid to a shift cylinder. The cylinder automatically shifts the stabilizer and rudder control surfaces into the extended throw authority. Selecting the ASSIST SPIN RECOVERY switch ON deactivates the directional stability augmentation system authority, thus making full rudder deflection available for spin recovery, regardless of the STAB AUG switch position. The RUD THRO caution light on the caution lights panel illuminates whenever rudder throw exceeds 4° in the flaps up configuration. Rudder surface travel limits and corresponding flap position are illustrated in figure A-9 and summarized in table A-2.

Table A-2: RUDDER SURFACE TRAVEL LIMITS

Flap Position / Stab Aug Mode	Rudder travel from neutral	Rudder Pedal Displacement
Flaps Up	± 4°	± 0.3 inches
Flaps Down (20° or 30°)	± 23°	± 1.9 inches
ASSIST SPIN RECOVERY ON	± 35°	± 2.9 inches
Flaps Up and AUTO Mode	± 4°	N/A
Flaps Down (20° or 30°) and AUTO Mode	± 10°	N/A
Flaps Down (20° or 30°) and Weight on Wheels	± 35°	± 2.9 inches

Directional artificial feel is provided by a mechanical linkage unit consisting of a cam and roller device loaded by two extension springs.

The rudder interconnect system (stability augmentation) in the AFCS is designed to correct for adverse yaw while rolling the aircraft, and for an out-of-trim condition in steady turns. An acceleration device is used to sense roll, adverse yaw, and long-term lateral acceleration. Yaw oscillations are automatically damped by rudder applications proportional to the roll and yaw rates. In the STAB AUG and AUTO modes, the AFCS rudder corrections are fed automatically to eliminate adverse yaw and coordinate turns. The movement of the rudder is not transmitted back to the rudder pedals, and the only evidence of augmentation is improved aircraft stability.

CONTROL SYSTEM FORCES

The directional control system is irreversible and incorporates a cam and roller device loaded by two extension springs that produce resistive forces at the rudder pedals proportional to pedal displacement. The mechanism is designed to provide a decreasing force gradient with pedal deflection as depicted in figure A-11. The designed directional breakout force is approximately 12.0 lbs.

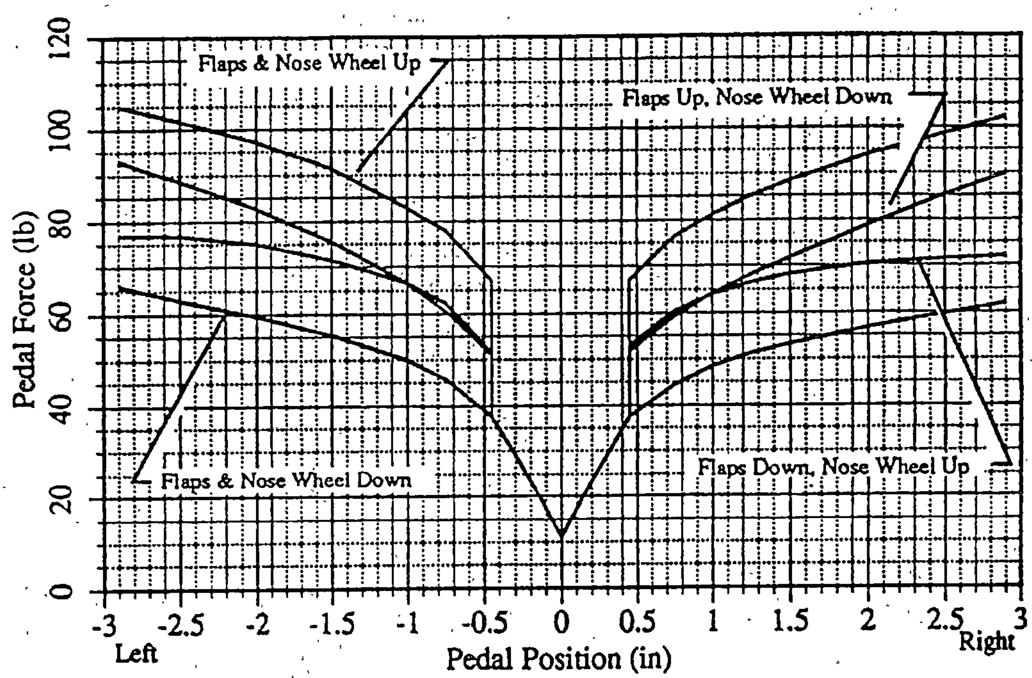


Figure A-11: EA-6B DIRECTIONAL CONTROL FEEL

DIRECTIONAL TRIM SYSTEM

Directional trim is provided by a rudder trim switch located on the pilot's left console which controls an irreversible, electromechanical linear-type actuator that varies the neutral position of the mechanical linkage. The trim switch, depicted in figure B-1, is 115 VAC electrically operated and spring-loaded to the off (center) position. Full trim corresponds to rudder travel of 4° either side of neutral in the flaps up configuration, and 6° either side of neutral in the flaps down configuration. Manual rudder trim is available in both AFCS modes of operation (STAB AUG and AUTO). The rudder trim gage, located on the pilot's lower instrument panel as depicted in figure B-2, indicates rudder position from 4 units left to 4 units right. A DC electrical signal generated by a position transmitter drives the pointer on the indicator. The RUDDER TRIM circuit breaker located on the forward main circuit breaker panel in the forward cockpit provides a means of interrupting electrical power to the directional trim system.

AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS)

There are five operating modes of the AFCS:

1. Stability Augmentation
2. Attitude Hold
3. Altitude Hold
4. Mach Hold
5. Automatic Carrier Landing System (ACLS)

Each mode is manually selected by the pilot via the AUTO PILOT controller located on the center console as depicted in figure B-1. The AUTO PILOT controller consists of three solenoid-held ON-OFF, ACLS-NORM-HDG OFF, and AUTO-STAB AUG toggle switches and two solenoid-held ALT and MACH square pushbutton switches. Electrical signals corresponding to the selected mode of operation are transmitted to the Air Navigation Computer (ANC), and AFCS interlocks are set to prevent selection of incompatible functions. All AFCS modes are integrated into the primary flight control system. A spin recovery mode is also manually selectable by the pilot through the ASSIST SPIN RECOVERY switch located on the throttle quadrant, as depicted in figure B-1.

GENERAL

The AFCS is an electromechanical system designed to provide two-axis stability augmentation, three-axis attitude control, and Mach or altitude hold. Airplane motion is detected by error sensors in each axis which send electrical control signals to the ANC. The AFCS consists of positioning-type servosystems, the ANC, a rate gyroscope assembly, and normal and lateral accelerometers.

AIR NAVIGATION COMPUTER (ANC)

The ANC is an electronic and electromechanical analog signal processor which couples, shapes and amplifies stability augmentation and automatic flight control signals to actuate the rudder, flaperon auto-pilot and stabilizer actuators and the stabilizer trim actuator. The computer is designed to accept pitch, roll and yaw rate signals, acceleration signals, and control surface/actuator position signals from the AFCS and provide output signals to the appropriate actuators to command control surface movement. The ANC also receives inputs from auxiliary subsystems such as the Attitude-Heading Reference System (AHRS), Course Attitude Data Transmitter (CADT), and Air Data Computer (ADC).

RATE GYROSCOPE

The rate gyroscope, consisting of a miniature rate gyroscope in each axis, is designed to detect and measure airplane pitch, roll and yaw rates and supplies this rate data to the ANC for 3-axis damping. The rates of change sensed by the rate gyroscope provide error signals to the ANC.

NORMAL AND LATERAL ACCELEROMETERS

The normal and lateral accelerometer transmitters are designed to generate an acceleration error signal proportional to the vertical and lateral acceleration of the airplane, respectively. The normal acceleration error signal is processed in the ANC with altitude and Mach error signals to produce an altitude or Mach hold vertical path damping signal which is supplied to the stabilizer and flaperon auto-pilot actuators. The normal acceleration rate signal is also used in the ANC as the g-command reference signal. The lateral acceleration error signal is used for turn coordination and to compensate for adverse yaw. Yaw and roll rate signals from the gyroscope coupled with the lateral acceleration error signal from the lateral accelerometer transmitter are used by the ANC to generate a control signal to hold or command a change in the rudder position. Pitch and roll rate signals from the rate gyroscope are used by the ANC to generate control signals to hold or command a change in the stabilizer and flaperon positions.

Magnetic heading, roll angle, and pitch angle are also processed in the ANC to provide the hold or actuate signals used by the stabilizer or flaperon auto-pilot actuators. The control stick includes sensing switches, which are actuated by stick motion to command the airplane's control surfaces via the AFCS mechanical linkage. Pitch and roll control forces applied to the control stick are transmitted to the airplane's control linkage via a transducer. A linear position transducer also transmits the stabilizer position signal to the ANC and AFCS when the stabilizer actuator is operating in the parallel mode.

Automatic disengagement of all AFCS modes except Stability Augmentation occurs when the pitch and roll outputs provided by the CAINS and AHRS do not agree with each other as well as the attitude driving the Attitude Direction Indicator (ADI). If any pitch and roll attitude comparison does not agree, a disengagement input is sent to the AUTO PILOT controller, the ADI OFF FLAG is displayed and the ATTD REF light illuminates on the Caution Lights panel. AFCS modes can be manually disengaged using the AUTO-PILOT controller, the AFCS emergency disconnect switch, control stick force inputs, and the AFCS and AFCS / AIR DATA CMPTR circuit breakers located on the aft main circuit breaker panel. In the event of uncommanded control inputs after the auto-pilot controller ON-OFF switch has been secured, electrical power from the auto-pilot must be removed by pulling the AFCS circuit breakers or deploying the emergency RAT and securing the generators.

STABILITY AUGMENTATION MODE

Stability augmentation is selected on the AUTO PILOT controller by placing the ON-OFF switch to ON with the AUTO-STAB AUG switch in STAB AUG. In this mode, the pilot flies the airplane using control stick and rudder pedal inputs while the AFCS provides pitch damping, yaw damping, and turn coordination. The two-axis damper is designed to automatically reduce airplane motion in the pitch and yaw axes by positioning the stabilizer and rudder proportionately to airplane pitch, roll and yaw rates, respectively. In this mode, the stabilizer is actuated proportionately in response to pitch rates and the rudder is positioned proportionately in response to roll and yaw rates. Pitch, roll and yaw rate signals from the rate gyroscope are processed with signals generated in the ANC to provide the signals to control the surface positions of the airplane via the electro-hydraulic flight control actuators. Turn coordination is accomplished by automatically positioning the rudder through the rudder actuator in response to airplane lateral acceleration.

ATTITUDE HOLD MODE

Attitude hold mode is the basic automatic pilot operating mode of the AFCS and is engaged by setting the AUTO-STAB AUG switch to AUTO on the AUTO PILOT controller with the ON-OFF switch ON. The AUTO mode is designed to remain engaged only if the pitch and roll modules are properly aligned with the attitude reference selected by the vertical reference (VERT REF) switch and the AC operating voltage of the AFCS is within limits. In this mode, the AFCS is designed to provide three-axis damping and attitude hold. The yaw axis is controlled in the same manner as described in the stability augmentation mode. The pitch axis is controlled by maintaining the reference pitch attitude, which is limited from 25° nose up to 60° nose down, while providing continuous automatic pitch trim. Pitch trim attitude is achieved by commanding stabilizer positioning proportional to the pitch attitude rate and displacement signal from an established reference attitude. The pitch attitude rate is derived from a pitch rate gyro in the rate gyroscope and the displacement signal is generated from the external navigational aids system. These signals are used to position the stabilizer via the stabilizer actuator and position transducer. Automatic pitch trim is accomplished by repositioning the cockpit control column and linkage to establish the trimmed stabilizer position. The AFCS senses the sustained electrically held surface position and repositions the cockpit control column and linkage through the electromechanical stabilizer trim actuator, thereby eliminating the requirement for a sustained electrical input. If the airplanes pitch attitude exceeds the 25° nose up or 60° nose down limit, the AFCS automatically returns the airplane to the nearest pitch limit. In the roll axis, the AFCS operates in roll attitude or heading hold mode. Roll attitude hold is accomplished in the same manner as pitch attitude hold, where the flaperons are positioned as a function of the roll rate signal from the rate gyroscope and the heading error signal from the external navigational aids system. For the AFCS to assume the roll attitude hold configuration, the bank angle must be between 5° and 60° with the ACLS-NORM-HDG OFF switch in the NORM position. If the bank angle is less than 5°, the airplane is automatically leveled and the heading hold mode is engaged. The reference heading is maintained by commanding bank angle as a function of heading error. The heading error signal is generated in the external navigational aids system. Positioning of the flaperons is accomplished by the flaperon auto-pilot actuator, which positions the flaperon hydraulic actuators through the AFCS linkage. With the ACLS-NORM-HDG OFF switch in the HDG OFF position, the AFCS maintains any bank angle between 0° and 60°, in the same manner described above. If the airplane's roll attitude exceeds 60° angle of bank in NORM or HDG OFF, the AFCS automatically returns the airplane to the 60° roll attitude limit. Longitudinal or lateral control stick forces of 1.1±0.2 lbs actuate the stick disengage switches which revert the AFCS to the STAB AUG mode, during which manual trim may be accomplished. When the control stick force is released, the AUTO mode is re-engaged and the AFCS maintains a constant pitch attitude and a constant bank angle or heading depending on the existing bank angle at the time.

There is no automatic disengagement in the AUTO mode upon failure of the attitude reference system input. Therefore, manual disengagement of the AUTO mode is required if an OFF FLAG appears on the ADI or EADI. However, on airplanes incorporating AFC 778, if the Digital Signal Data Converter (DSDC) fails and primary is selected as the attitude reference, the AFCS will disengage in AUTO, the EADI ATTITUDE FAIL will be displayed, the ATTD REF caution light will illuminate and the master caution light will flash. In addition, the AUTO mode should be secured prior to switching into or out of primary (PRI) or Secondary (SEC) on the VERT REF switch to prevent abrupt or unexpected attitude changes during a change in the reference signals to the AFCS.

ALTITUDE HOLD MODE

Altitude hold mode can be manually selected by the pilot only after selecting AUTO with the AUTO-STAB AUG switch to satisfy the AFCS interlock functions. The altitude hold mode is engaged by pressing the ALT pushbutton on the AUTO PILOT controller. The ALT and MACH pushbuttons are mutually exclusive and selection of the ALT pushbutton automatically disengages Mach hold, if activated. Airplane static and impact pressure from the pitot-static system and temperature from the total-temperature probe are processed in the Air Data Computer (ADC), which controls the altitude hold function. Clutches in the ADC are activated by the altitude hold pushbutton to establish the reference altitude. Automatic (barometric) altitude hold is accomplished by commanding pitch attitude proportional to displacement and rate deviations from the reference altitude. Also, sustained altitude errors are corrected by integration of the displacement errors. The displacement signal from the external navigational aids system, along with the normal acceleration rate signal from the normal accelerometer transmitter and the integrated displacement signal are combined and used to command airplane pitch attitude to reduce pressure altitude errors. These error signals are sent from the ADC to the AFCS, which commands the stabilizer actuator to position the stabilizer. Again, control stick forces of 1.1±0.2 lbs interrupt the altitude hold function, but the altitude hold mode is re-engaged when the control stick forces are released.

MACH HOLD MODE

Mach hold mode can be manually selected by the pilot only after selecting AUTO with the AUTO-STAB AUG switch to satisfy the AFCS interlock functions. The Mach hold mode is engaged by pressing the MACH pushbutton on the AUTO PILOT controller. The ALT and MACH pushbuttons are mutually exclusive and selection of the MACH pushbutton automatically disengages altitude hold, if activated. The Mach hold function is also controlled through the ADC by processing static and impact pressure from the pitot-static system and temperature from the total-temperature probe. The Mach hold pushbutton activates clutches in the ADC to establish the reference Mach number. Mach hold is then accomplished by commanding airplane pitch attitude to reduce Mach error in the same manner as altitude error is reduced in the altitude hold mode. The displacement signal is supplied to the AFCS from the external navigational aids system. Sustained Mach errors are corrected by integration of the displacement error, and vertical path damping is provided through integration of the normal accelerometer transmitter. Again, control stick forces of 1.1±0.2 lbs interrupt the Mach hold function, and the Mach hold mode is re-engaged when the control stick forces are released.

AUTOMATIC CARRIER LANDING SYSTEM (ACLS) MODE

The ACLS is designed to provide an automatic coupled AFCS approach capability by interfacing the ASW-25 Data Link with the AFCS and the Approach Power Compensator (APC). The ACLS mode can be manually selected by the pilot only after selecting AUTO with the AUTO-STAB AUG switch to satisfy the AFCS interlock functions. The ACLS mode is selected by placing the ACLS-NORM-HDG OFF switch to ACLS on the AUTO PILOT controller, as depicted in figure B-1. In the ACLS mode, aircraft carrier-based radars transmit glidepath data to the AFCS via the AN/ASW-25 Data Link. If the switch does not hold in the ACLS position, it indicates that not all of the pre-engagement requirements are met. The ACLS controls and displays are illustrated in figures B-6 and B-7.

The ASW-25 data link provides the capability to receive signals from the SPN-42 / 46 system, decode them, provide control signals to the AFCS, provide position signals to the ADI / EADI and ARI, and display system status on the ACLS discrete message indicator (DMI) lights illustrated in figure B-7. Normal data link operation requires the data link power ON, the proper UHF data link frequency dialed into the thumbwheels on the pilot's instrument panel, and a known address set in the decoder box located in the forward right equipment bay. All ACLS and Instrument Carrier Landing System (ICLS) controls are presented in figure B-6.

The ACLS is a closed-loop system designed to use pulse-coded KU-band information and Ka-band interrogation from the aircraft carrier to obtain and/or maintain glidepath. The carrier's landing radar system, AN/SPN-42 or 46, tracks the airplane and the radar computer compares present airplane position with the optimum approach profile. The airplane position is then corrected to the desired glidepath using commands from the Navy Tactical Data System transmitted over UHF data link to the airplane's ASW-25 data link receiver. The ASW-25 data link receiver then directs pitch and roll commands to the AFCS. Upon selection of ACLS, the shipboard generated pitch and roll commands are transmitted to the Air Navigation Computer (ANC) via the AN/ASW-25 Data Link, and the stabilizer actuator enters the parallel mode of operation. With ACLS mode COUPLED, shipboard generated pitch and roll commands are transmitted to the ANC via the data link which in turn commands the appropriate control surface deflection to effect the proper aircraft response. Stabilizer movements are commanded by input signals from the AFCS, and the corresponding stabilizer motion is transmitted through mechanical linkages to the control stick. In this mode, the auto-pilot input signal is designed to command a maximum stabilizer travel of ±7.8°.

The ASW-25 data link also transmits ACLS discrete messages, which are displayed on the pilot's instrument panel, as depicted in figure B-7. These messages provide information relative to the progress of the approach, as well as azimuth and glideslope deviations displayed by crossbars on the pilot's ADI / EADI and ARI. When an operable AN/SPN-41 is available on the aircraft carrier, independent heading and glideslope signals are displayed on the ADI / EADI and ARI, providing manual ICLS approach capability through the AN/ARA-63 receiver. The ACLS DATA LINK – MON switch, located on the pilot's instrument panel (figure B-6), allows SPN-41 or SPN-42 information to be selected on either the ADI / EADI or ARI and does not affect ACLS control if switched during an approach.

In the event of an excessive flight-path error, a waveoff signal from the ACLS disengages the ACLS control and reinstates airplane flight control to the AFCS. Automatic disengagement of the ACLS mode is designed to occur when auto-pilot command signals exceed normal operating inputs or when the stabilizer actuator servo ram receives a hardover signal causing the main ram to actuate a rate switch. Either automatic disengagement causes the AFCS to revert to the STAB AUG mode. On aircraft incorporating AFC 778, if the DSDC detects an internal or catastrophic failure, the AFCS is designed to disengage if in AUTO or ACLS. The ACLS mode is manually disengaged by applying a control stick force of 10 lbs in pitch or 7 lbs in roll, which cause the AFCS to revert to and remain in the STAB AUG mode. If the AFCS fails to revert to STAB AUG or the control stick fails to move with pilot pitch inputs, the ACLS mode can be disengaged and normal stick movement restored through application of rapid fore and aft stick inputs of approximately 50 lbs at the stick grip. The ACLS mode can also be manually disengaged using the AFCS emergency disconnect switch located below and to the right of the control stick grip.

APPROACH POWER COMPENSATOR (APC)

The approach power compensator (APC) is designed to maintain an airspeed that will result in a constant average AOA. The APC is engaged using the APC power switch located on the pilot's left console (figure B-1), and automatically sets the throttles at the calculated thrust required to maintain 17 units AOA, regardless of previous settings. The throttle setting is dependent upon airplane gross weight, bank angle, flight-path angle, and selected temperature setting. The APC uses accelerations measured perpendicular to the glidepath, AOA error, and stabilizer position inputs to control throttle movement. The APC system is also damped to prevent excessive throttle movements when flying through heavy turbulence. The system is designed to disengage when the throttle friction lock lever is moved forward out of the OFF position, STBY or OFF is selected with the APC power switch, or the weight-on-wheels switch is actuated. The APC is also designed to manually disengaged when 8-12 lbs of forward or aft force is applied to the throttles. If throttle control is not relinquished after attempting to disengage the APC by normal methods, a force of 35-55 lbs applied to the throttles manually overrides the system and damages the APC actuators.

SPIN RECOVERY MODE

The spin recovery mode is initiated by selecting ASSIST with the guarded ASSIST SPIN RECOVERY switch located on the throttle quadrant (figure B-1). The spin recovery system provides an alternate method of increasing stabilizer and rudder throws in the clean configuration. Normally (SPIN RECOVERY switch OFF), the stabilizer shift mechanism limits stabilizer travel from 1.5° leading edge up to 9.6° leading edge down and rudder displacement to ± 4°. Selecting ASSIST electrically energizes a selector valve, which ports primary combined hydraulic fluid to a shift cylinder. The cylinder automatically shifts the tail control surfaces into an extended throw of 1.5° leading edge up to 24° leading edge down for the stabilizer and ± 35° for the rudder. Activation of the spin recovery mode eliminates the directional stability augmentation authority, ensuring full rudder deflection is available for spin recovery regardless of the STAB AUG switch position. In Block 89 airplanes, an assist spin recovery battery provides an emergency DC power source to activate the assist spin recovery circuit and power the yaw rate indicator.

MODE DISENGAGEMENT

On airplanes 163884 and subsequent or airplanes modified with AFC 639, the attitude hold, altitude hold, Mach hold, and ACLS modes are designed to automatically disengage if the pitch and roll attitude data supplied by the Carrier Aircraft Inertial Navigation System (CAINS), synchro or digital, or AHRS does not agree with the attitude reference inputs being transmitted to the Attitude Direction Indicator (ADI). The primary and secondary pitch and roll outputs from the CAINS and AHRS, respectively, are formatted by the CADT to obtain separate X, Y, and Z synchro lines which drive the ADI and are sent to the A/D converter. Any discrepancy in pitch or roll data between the CAINS, AHRS, and the ADI source results in disengagement of the AFCS input between the ANC and the AUTO PILOT controller. Relay logic is also designed to raise the ADI OFF flag, illuminate the ATTD REF light and cause the MASTER CAUTION light to flash.

APPENDIX B

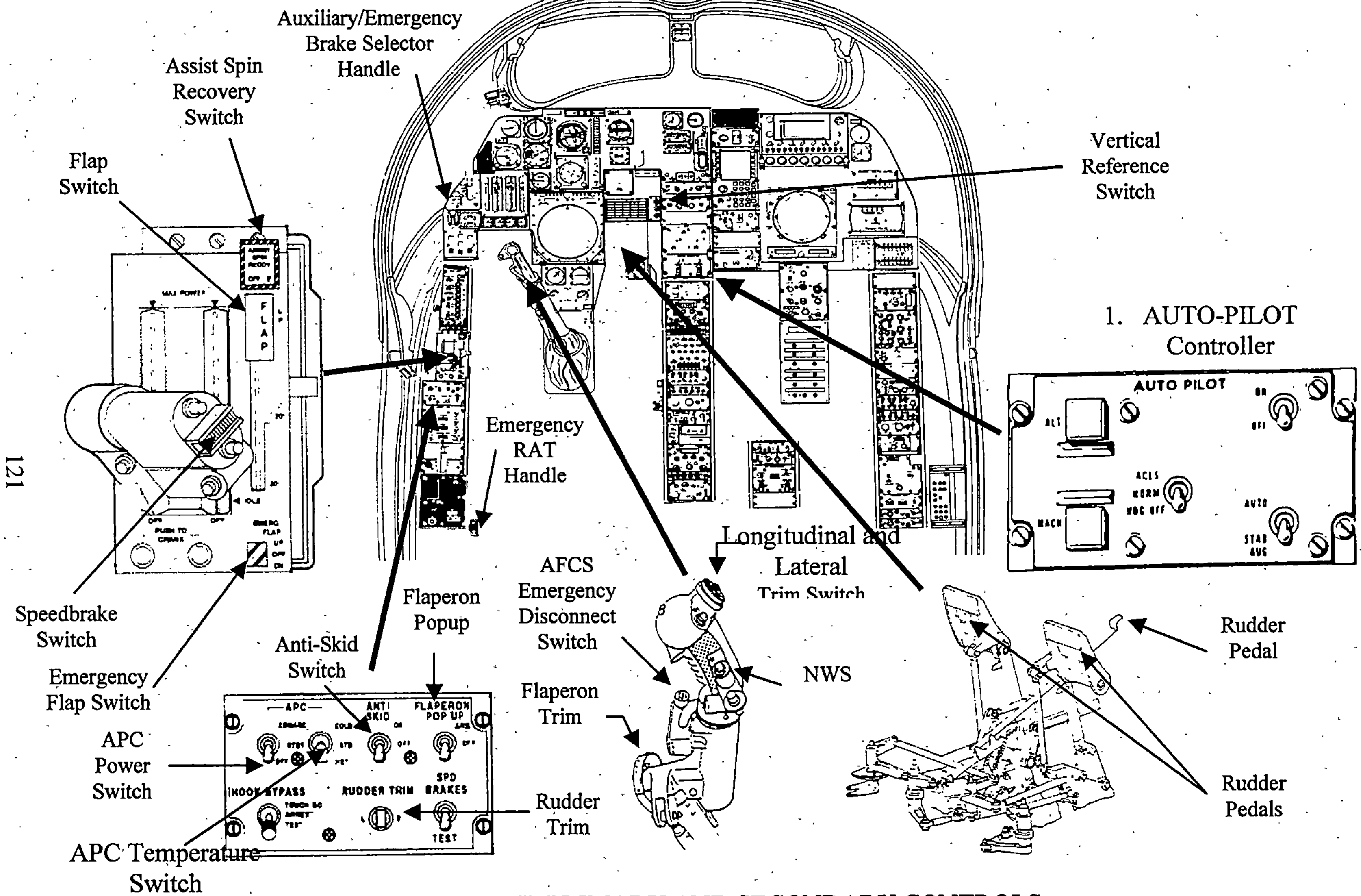


Figure B-1: EA-6B PRIMARY AND SECONDARY CONTROLS

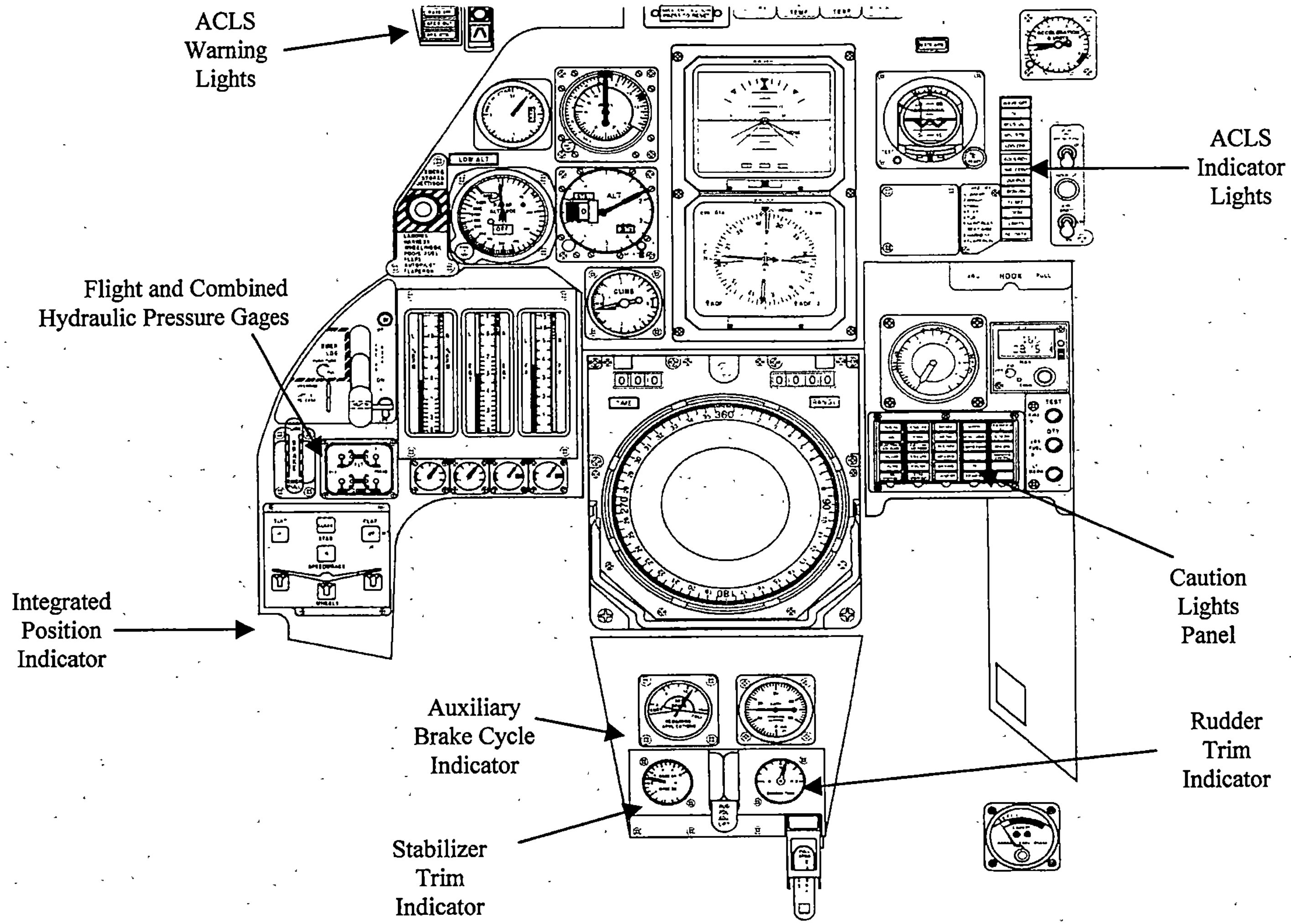


Figure B-2: EA-6B FLIGHT CONTROL SYSTEM DISPLAYS

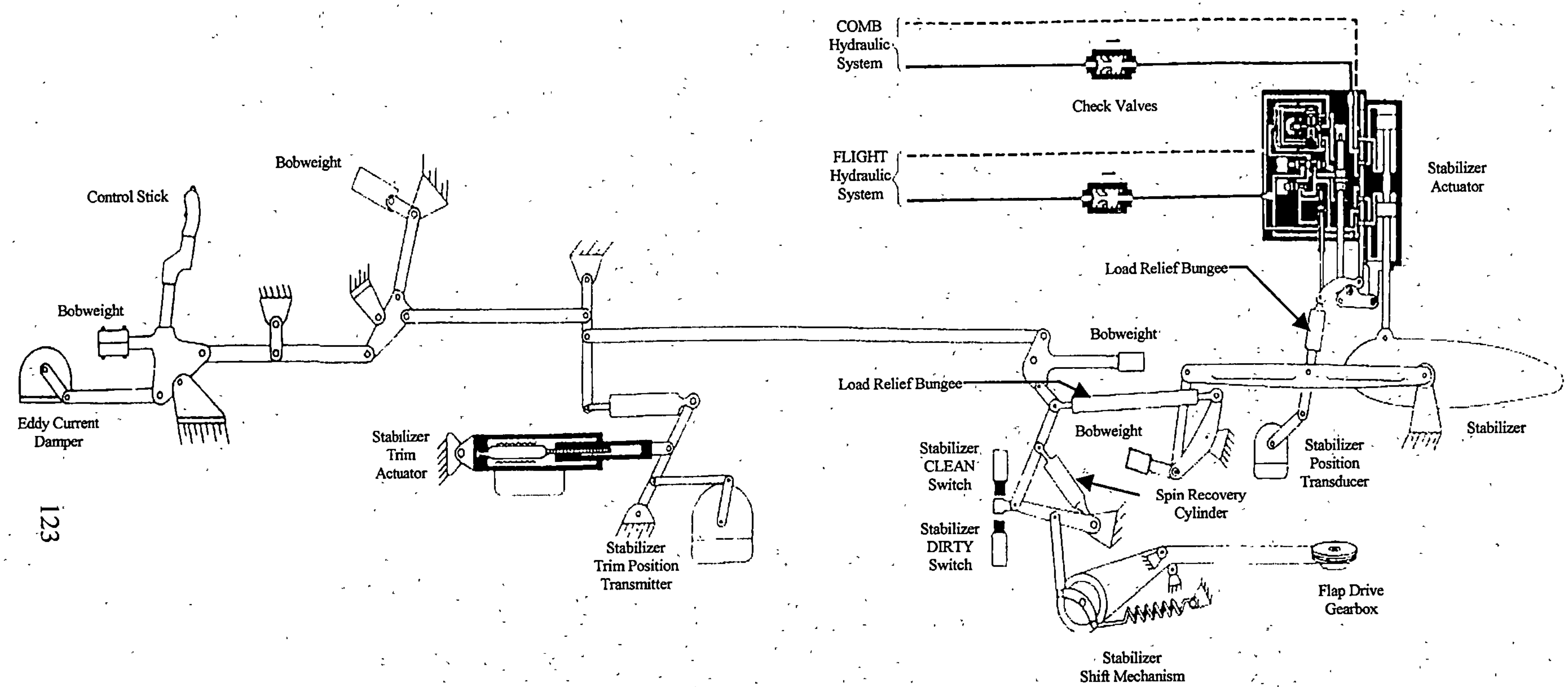


Figure B-3: EA-6B LONGITUDINAL FLIGHT CONTROL SYSTEM SCHEMATIC

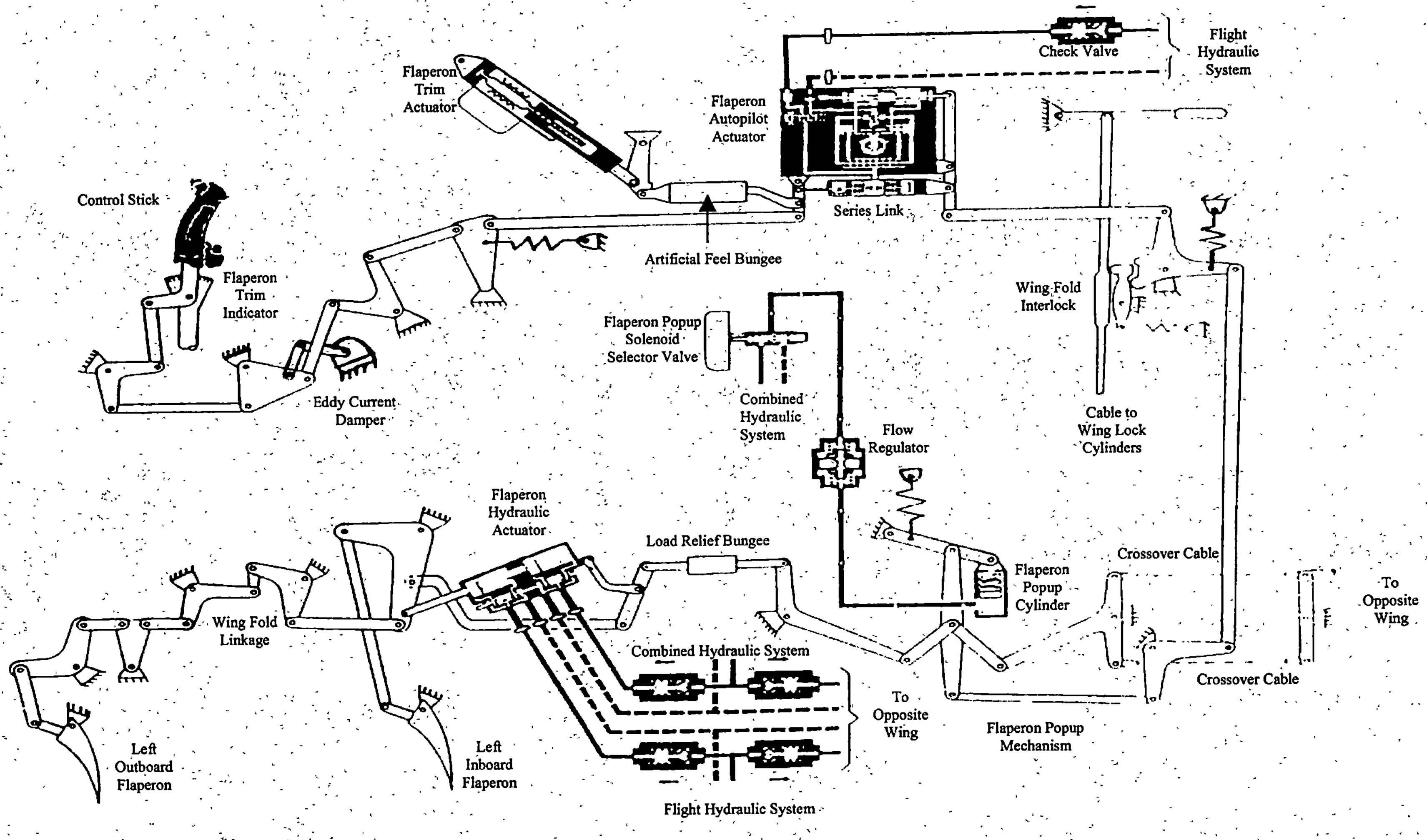


Figure B-4: EA-6B LATERAL FLIGHT CONTROL SYSTEM SCHEMATIC

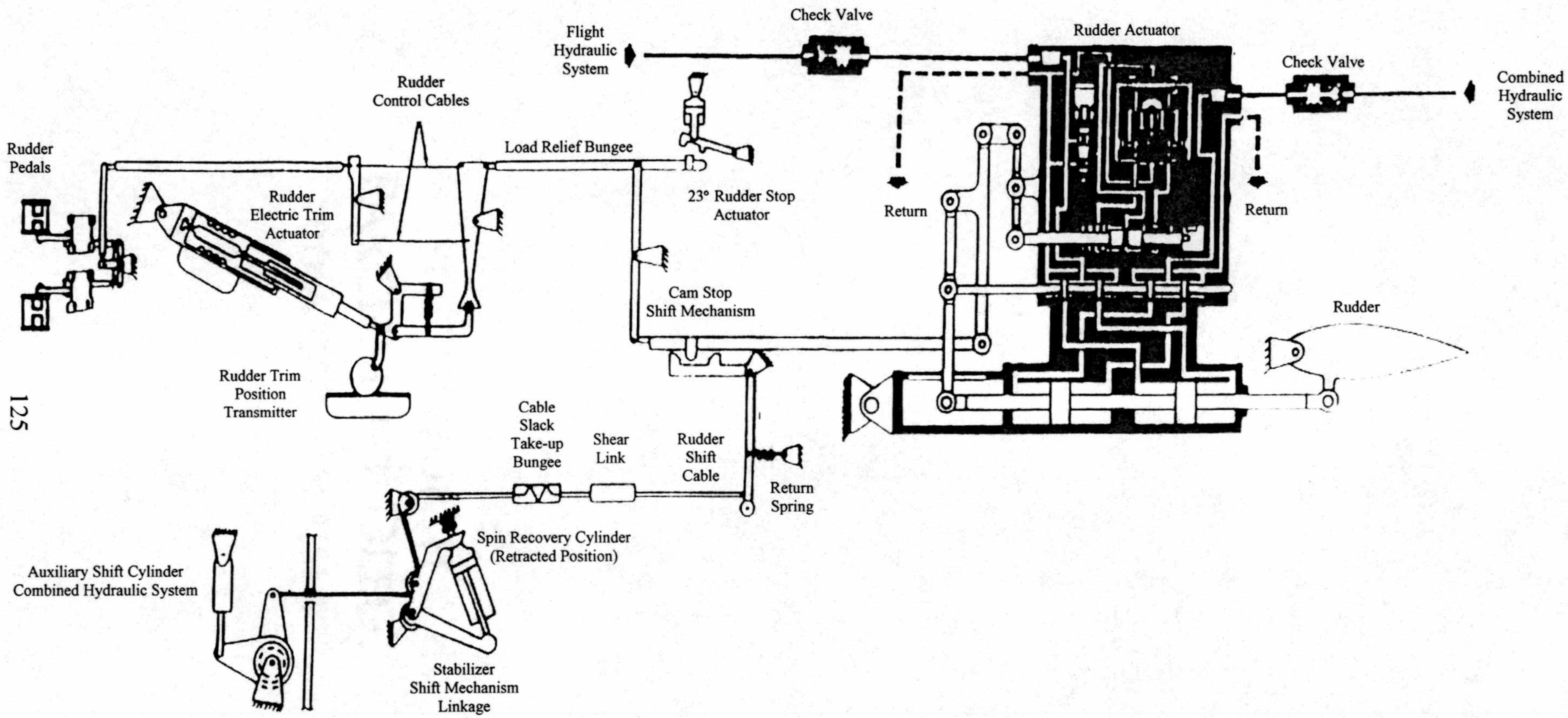


Figure B-5: EA-6B DIRECTIONAL FLIGHT CONTROL SYSTEM SCHEMATIC

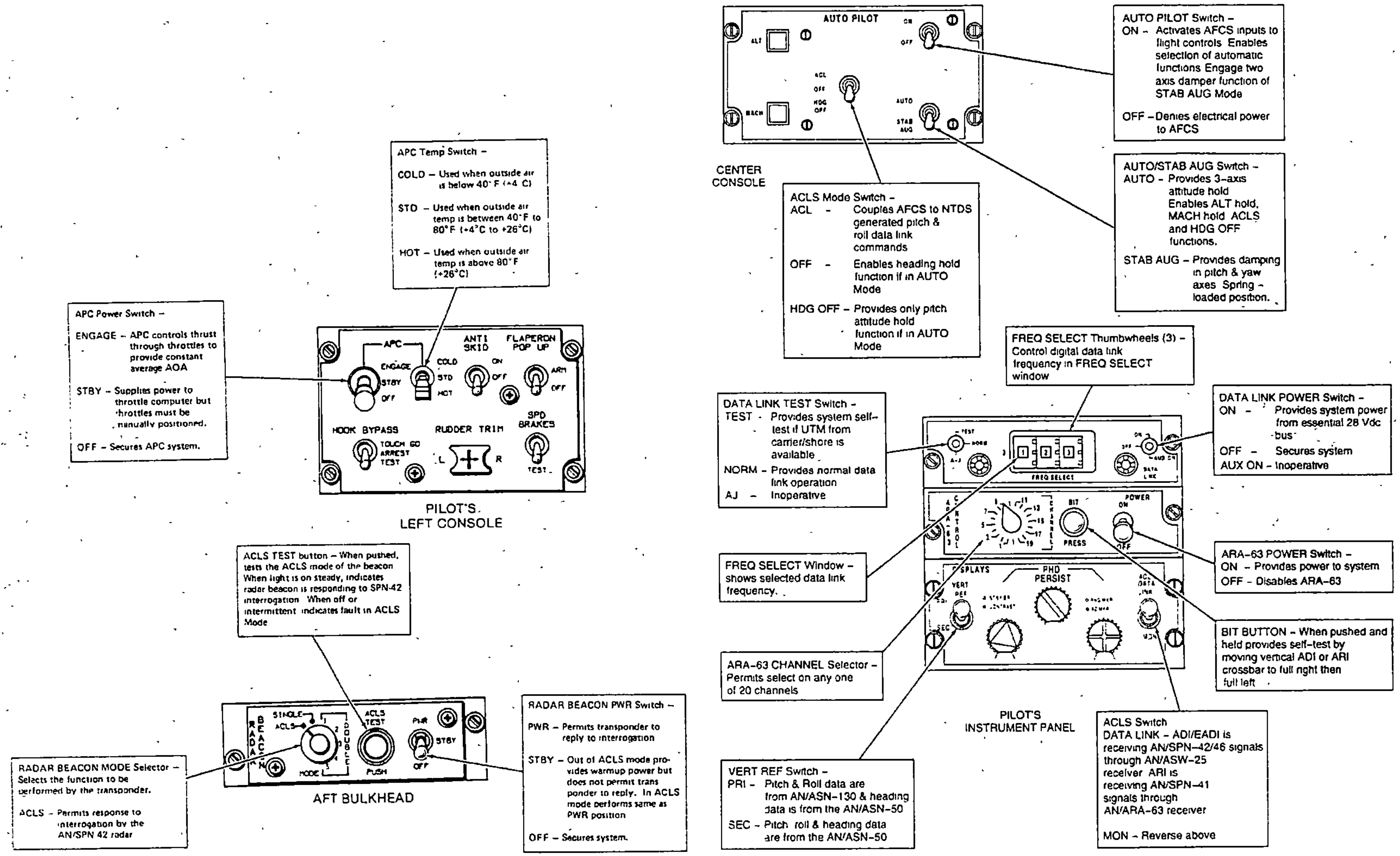


Figure B-6: EA-6B AUTOMATIC CARRIER LANDING SYSTEM CONTROLS

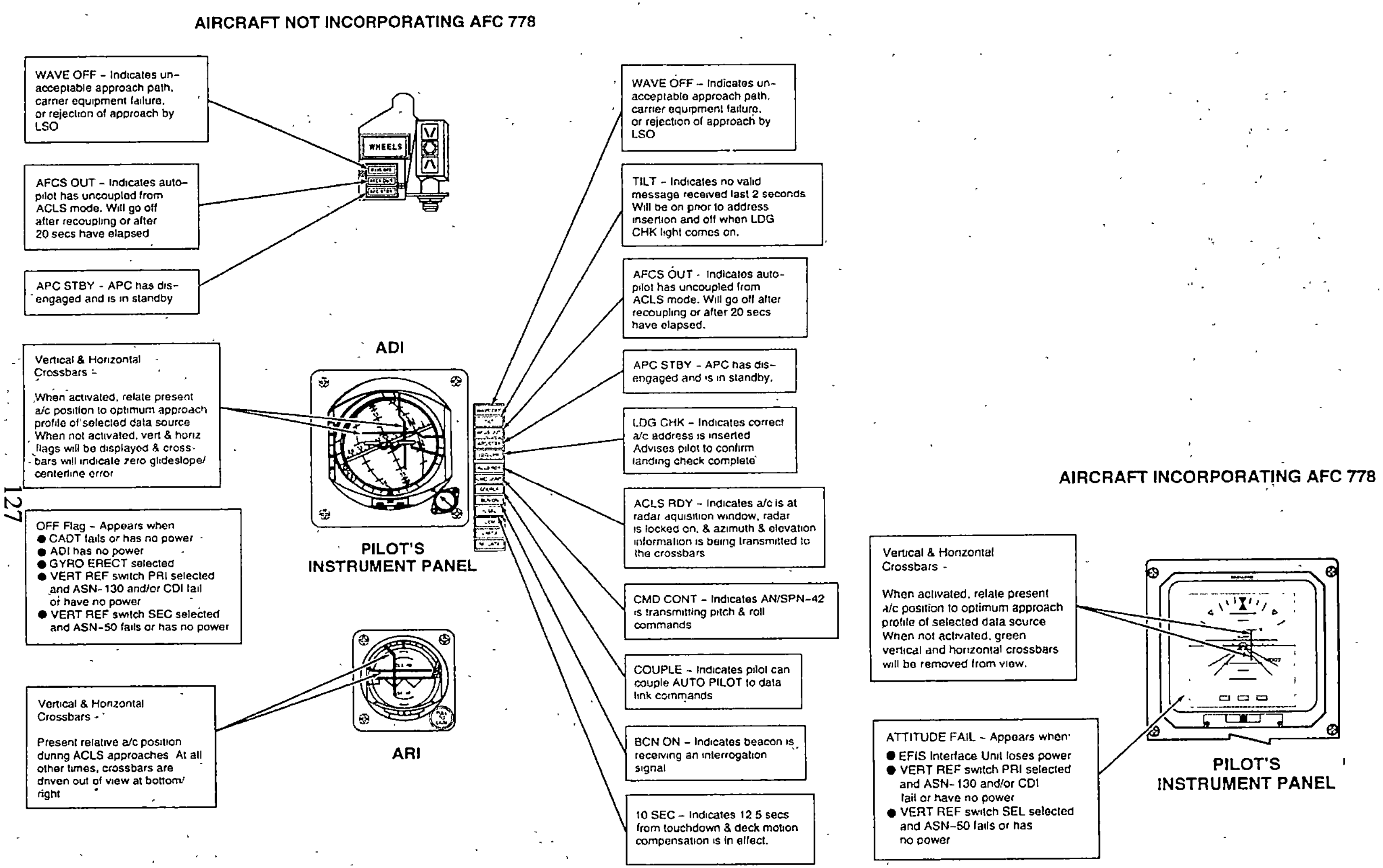


Figure B-7: EA-6B AUTOMATIC CARRIER LANDING SYSTEM DISPLAYS

APPENDIX C

Table C-1: EA-6B FLIGHT CONTROL SYSTEM CHARACTERISTICS

Control surface	Travel Limits (from neutral)	Control	Actuation
Horizontal Stabilizer Flaps Up Flaps Down or ASR-ON ACLS	 1.5° LEU to 9.6° LED 1.5° LEU to 24° LED ±7.8°	Mechanical / Electrical	Flight & Primary Combined Hydraulic
Horizontal Stabilizer Trim Flaps Up Flaps Down	 3/4° LEU to 9° LED 1/2° LEU to 22° LED	115VAC Electric 28VDC (AFC745)	115VAC Electric 28VDC (AFC745)
Flaperons	51° Up	Mechanical and Electrical	Flight & Primary Combined Hydraulic
Flaperon Trim	9° Up	115 VAC Electric	115 VAC Electric
Rudder Flaps Up Flaps Down ASR-ON Flaps Up in AUTO Flaps Down in AUTO	 ± 4° ± 23° ± 35° ± 4° ± 10°	Mechanical and Electrical	Flight & Primary Combined Hydraulic
Rudder Trim Flaps Up Flaps Down	 ± 4° ± 6°	115 VAC Electric	115 VAC Electric
Flaps Normal Emergency	20° or 30°	28VDC Electric 115VAC and 28VDC Electric	Secondary Combined Hydraulic
Slats	27°	28VDC Electric	Secondary Combined Hydraulic
Speedbrakes	60°	28VDC Electric	Primary Combined Hydraulic
Nose wheel Steering	± 60°	28VDC Electric	Secondary Combined Hydraulic
Flaperon Popup	41° Up	28VDC Electric	Secondary Combined Hydraulic
23° Rudder Stop	±23°	28VDC Electric	Primary Combined Hydraulic

Table C-2: EA-6B FLIGHT CONTROL SYSTEM ELECTRICAL REQUIREMENTS

Component	Control	Actuation
Stabilizer and Rudder Trim Indicators	28 VDC Electrical	28 VDC Electrical
Flaperon Trim Indicator	Mechanical	Mechanical
Emergency Flaps and Slats	115 VAC and 28 VDC Electrical	115 VAC and 28 VDC Electrical
Assist Spin Recovery	28 VDC or ASR Battery (Block 89)	Primary Combined Hydraulics
Anti-Skid	28 VDC Electrical	Secondary Combined Hydraulics
AFCS	115 VAC and 28 VDC Electrical	Flight and Primary Combined Hydraulics
AFCS Emergency Disconnect	Mechanical	Mechanical
Caution Lights Panel	28 VDC Electrical	28 VDC Electrical
Isolation Valve	28 VDC Electrical	Flight Hydraulics
Emergency RAT	Mechanical	Primary Combined Hydraulics and Pneumatic back-up
Integrated Position Indicator	28 VDC Electrical	28 VDC Electrical
Hydraulic Gages	26 VAC Electrical	26 VAC Electrical
Auxiliary Brakes	Mechanical	Auxiliary Hydraulics
Auxiliary Brake Cycle Indicator	Mechanical (Direct Reading Gage)	Pneumatic
ASW-25 Data Link	28 VDC Electrical	28 VDC Electrical
ACLS Warning and Indicator Lights	28 VDC Electrical	28 VDC Electrical

APPENDIX D

EA-6B HYDRAULIC POWER SUPPLY SYSTEM

The EA-6B hydraulic system, illustrated below in, figure D-1, is composed of two independent systems, the flight hydraulic system and the combined hydraulic system. Both hydraulic systems power the tandem-type flight control actuators, and either system alone is capable of delivering sufficient hydraulic pressure to the flight controls for maneuvering the aircraft throughout the flight envelope, except at high IMN where aerodynamic hinge moments and available actuator power become the limiting factors. Each system is powered by two 3000 psi engine-driven hydraulic pumps; one pump driven by each engine. The flight hydraulic system powers one side of the tandem-type flight control actuators, an isolation valve and the AFCS. The combined hydraulic system is separated into primary and secondary systems by the isolation valve designed to reduce the possibility of losing combined system pressure due to failure in any one of the utility system components. The primary combined hydraulic system powers the other side of the tandem-type flight control actuators, the emergency RAT, assist spin recovery system, wing tip speed brakes, and 23° rudder stops. The secondary combined hydraulic system provides power for operation of the flaps, slats, landing gear, flaperon popup, wingfold, wingfold locks, nose wheel steering, hook retract, strut lock and normal wheel brakes. Flight and combined hydraulic system bootstrap reservoirs, pressurized by the engine-driven pumps, supply fluid for each system. A wheel brake accumulator is incorporated into the combined secondary hydraulic system for operation of the auxiliary wheel brakes, the aft extensible equipment platform, and the radome. The auxiliary brake accumulator is charged using a hand pump or electrically-driven pump, both operated from either the cockpit or the nose wheel well. An auxiliary brakes gage, located on the lower center portion of the pilot's instrument panel (figure B-2), indicates the number of auxiliary brake applications available. Four hydraulic pressure gages are incorporated in a single unit located on the pilot's instrument panel (figure B-2) and are powered by the right main 26 VAC bus. The upper two side-by-side gages are identified as FLT and indicate the pressure output of the two engine-driven pumps for the flight hydraulic system. The lower two side-by-side gages are identified as COMB and indicate the pressure output of the two engine-driven pumps for the combined hydraulic system. A HYD SYS caution light on the caution lights panel, located on the lower center portion of the pilot's instrument panel (figure B-2), is designed to illuminate when either the flight or combined hydraulic system pressure falls below 1400 ± 100 psi. The HYD SYS caution light does not provide an indication of a single pump failure within either system.

The loss of a single hydraulic pump will be displayed on the respective hydraulic pressure indicator, but will not present any flight control degradation. However, failure of one pump may result in failure of the other pump in the same (FLT or COMB) system due to the common hydraulic fluid reservoir. In the event of a flight or combined hydraulic system failure, the remaining operating system will assume full demand for the stabilizer, rudder and flaperon actuators while the normal landing gear, normal flaps/slats, normal brakes, anti-skid, flaperon popup, and nose wheel steering systems will be inoperative. If the flight hydraulic system fails, the spring-loaded isolation valve will automatically cut off hydraulic pressure to the combined secondary hydraulic system, ensuring all available combined hydraulic pressure is supplied to the primary flight controls. However, loss of flight hydraulic system pressure will not automatically disengage the AFCS. Therefore, the AUTO PILOT ON-OFF switch located on the Auto Pilot controller on the center console (figure B-1) must be turned OFF to prevent uncommanded inputs to the flight control system caused by surges in pressure in the failed flight hydraulic system. A flight or combined hydraulic system failure requires use of emergency landing gear extension, emergency flaps/slats, and auxiliary brakes. The emergency flaps/slats are electrically controlled and actuated using the emergency flaps/slats switch located on the throttle quadrant inboard of the throttles as depicted in figure A-1. Auxiliary brakes are selected by rotating the brake selector handle, located on the lower left portion of the pilot's instrument panel (figure B-1), 90° clockwise. With auxiliary brakes selected, the brake selector valve ports brake accumulator hydraulic pressure to the brake control valves and the anti-skid system is de-energized. If electrical power is lost to the isolation valve with normal flight hydraulic pressure available, the isolation valve opens, energizing the combined secondary system and its associated utility systems. A complete hydraulic system failure renders all flight controls inoperative. During a dual engine failure or flameout (engines windmilling), a minimum airspeed of 140 KIAS is required to generate sufficient hydraulic pressure to maintain control of the aircraft.

APPENDIX E

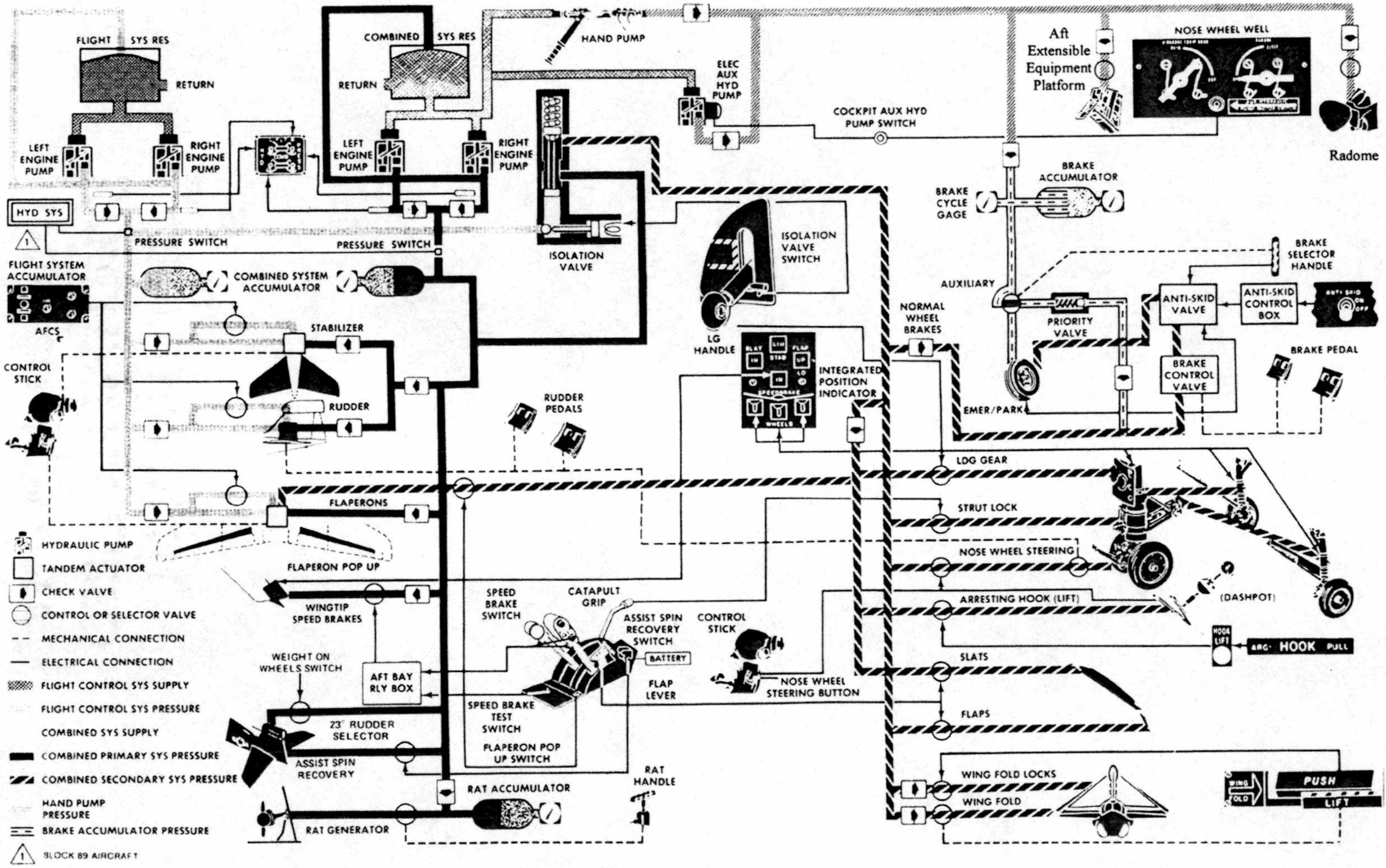


Figure D-1: EA-6B HYDRAULIC SYSTEM

EA-6B ELECTRICAL SYSTEM

The EA-6B electrical system, depicted below in, figure E-1, consists of two engine-driven 30 KVA 115/200 volt AC generators, two 100-ampere transformer-rectifiers, and a 24 volt DC, 0.33 ampere-hour battery. Emergency electrical power is provided by a ram air turbine (RAT) driven 2.5 KVA AC generator and 50-ampere emergency transformer-rectifier, and has a minimum operating airspeed of 110 KIAS. The emergency RAT is mechanically controlled via a handle located inboard of the pilot's left console, depicted in figure A-1, and hydraulically positioned into the airstream above the left wing. In the event of a combined hydraulic system failure, the RAT accumulator is designed to provide a single extension of the RAT pneumatically. The electrical power requirements for the secondary flight controls, AFCS, and trim systems are presented in table C-1 and C-2.

The electrical power requirements for the secondary flight controls, AFCS and trim systems can be met by any generator and transformer-rectifier combination through a series of AC contactors. In the event of a dual generator failure, all secondary flight control systems except speed brakes, nose wheel steering, flaperon popups, anti-skid, emergency flaps/slats, and AFCS will be available on emergency RAT power. In Block 89 airplanes, the assist spin recovery battery provides emergency DC power to the assist spin recovery circuit and the yaw rate indicator when no other DC source is available.

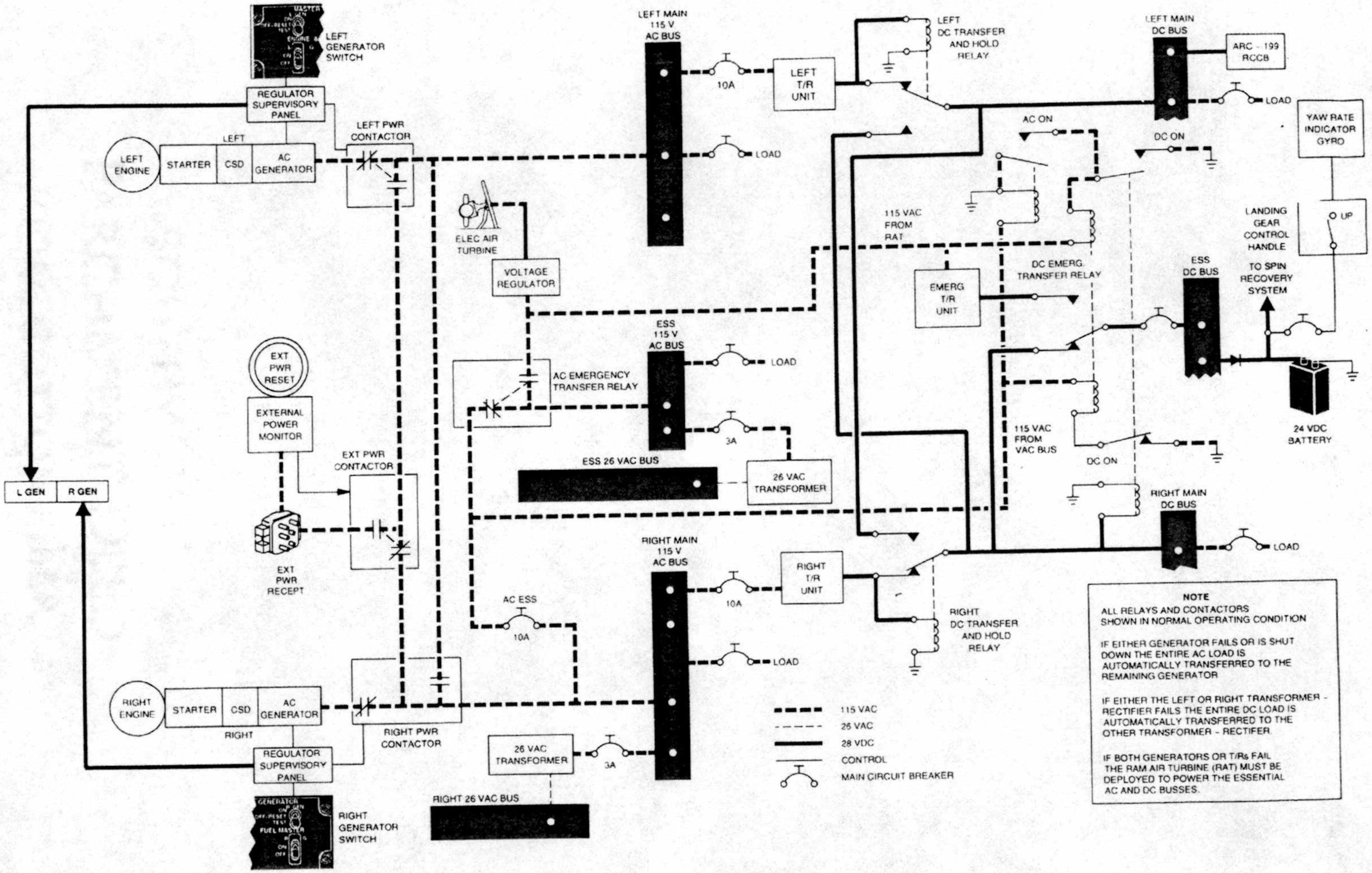


Figure E-1: EA-6B ELECTRICAL SYSTEM

APPENDIX F

SECONDARY FLIGHT CONTROL SYSTEM

WING FLAP AND SLAT SYSTEM

The semi-Fowler type, slotted, trailing edge wing flaps and leading edge slats are actuated by the flap selector lever located to the right of the throttles as depicted in figure B-1. The flaps and slats are DC electrically controlled and hydraulically actuated by the secondary combined hydraulic system. Both the flaps and slats are divided at the wing fold and can not be hydraulically actuated unless the wings are spread and locked and the wing lockpin switch is energized. Moving the flap selector lever actuates electrical switches that control the hydraulic motors and hydraulic brakes for the flaps and slats. The central flap gearbox located in the right wing root transmits torque through a series of spanwise torque shafts and offset gearboxes, driving a series of reversible screw jack actuators to extend and retract the flaps. The flaps are supported by a hinge and carriage assembly on rollers and guided on contour tracks in the wing trailing edge. At the same time, the central slat gearbox located in the left wing root extends or retracts the slats in a similar fashion. The slats are fully extended to 27° whenever the 20° or 30° flap position is selected and are guided by rollers in the wing leading edge. Flap and slat positions are indicated on the Integrated Position Indicator (IPI) on the pilot's instrument panel. The flap and slat gearboxes each have a hydraulic and an emergency AC electric-drive motor attached. An emergency flap switch, located on the throttle quadrant as depicted in figure A-1, energizes the 115 VAC emergency flap and slat motors for extension or retraction in the event of hydraulic motor failure or flight or combined hydraulic system failures. Four circuit breakers (c/b's) provide an additional means of securing electrical power to the flaps and slats; FLAP and SLATS c/b's on the forward left circuit breaker panel, and EMERG FLAP and EMERG SLAT c/b's on the pedestal forward of ECMO 1's position.

SPEEDBRAKE SYSTEM

Wing tip speedbrakes consist of a set of split trailing-edge surfaces at the tip of each wing as depicted in figure A-1. The speedbrakes require AC and DC power and are actuated by the primary combined hydraulic system. The speedbrake switch, located on the inboard face of the right throttle (figure B-1), permits selection of any intermediate position. Single tandem-type hydraulic cylinders in each wing tip deflect the surfaces via flow regulators to a maximum included angle of 120°. An electric synchronization system automatically retracts the speedbrakes if the surfaces become asymmetric by more than 8°. Speedbrake position is displayed on the IPI on the pilot's instrument panel as illustrated in figure B-2.

NOSE WHEEL STEERING (NWS) SYSTEM

The nose wheel steering system is designed to provide directional control and shimmy damping through a DC electrically controlled nose wheel steering button on the control stick grip, as illustrated in figure B-1. Nose wheel steering is controlled through rudder pedal deflection and is activated only when the weight-on-wheels switches on the right main landing gear are closed. The nose wheel steering system will power the lower strut barrel up to approximately 60° either side of centered using secondary combined hydraulic pressure and requires the NWS button to be depressed throughout system operation. Nose wheel steering is automatically engaged when the arresting hook is extended as soon as the right main gear weight-on-wheels switches are actuated. The relationship between rudder pedal deflection and nose wheel angle is presented in figure F-1.

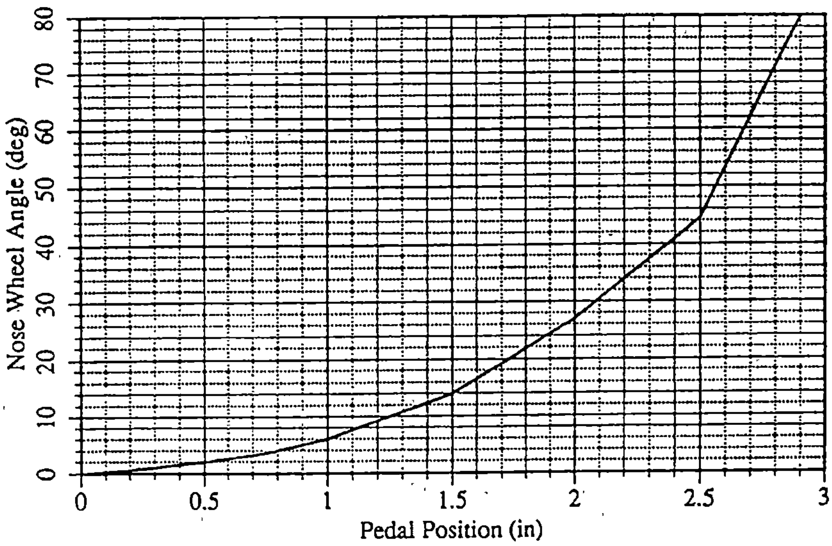


Figure F-1: EA-6B NWS SYSTEM PEDAL-TO-NOSE WHEEL GEARING

WHEELBRAKE CONTROL SYSTEM

The main wheel brakes are hydraulically powered multiple disk brakes designed to provide directional control and braking capability during ground operations, and incorporate an artificial feel system to simulate pedal forces. A brake selector handle, located on the left side of the pilot's instrument panel (figure B-1), allows selection of normal, auxiliary or emergency/parking brake operation. The normal brakes are powered by the secondary combined hydraulic system. The auxiliary and emergency/parking brakes are powered by a hydraulic brake accumulator that is charged by a hand-pump or via an electrically motor-driven hydraulic pump; both located to the right of the pilot's seat and in the nose wheel well. Accumulator pressure is displayed on an auxiliary brake cycle indicator (figure B-2), located on the pilot's lower instrument panel, in number of brake cycles remaining. The wheel brakes are actuated by conventional toe-operated pedals, which provide a direct input to the brake control valves which meter hydraulic pressure to the brakes.

ANTI-SKID CONTROL SYSTEM

The anti-skid system is used in conjunction with the normal brake system and is designed to increase high-speed braking effectiveness. The system is activated using the anti-skid switch located on the pilot's left console as depicted in figure B-1. The anti-skid system is DC electrically controlled and incorporates an ANTI-SKID light on the caution lights panel (figure B-2) which indicates failures within the anti-skid control box or a brake release signal that exceeds 3.5 seconds. The system is deactivated whenever the auxiliary brakes are selected or the parking brake is set. If the anti-skid system is inoperative, normal wheel brakes will still be available. Multiple failures within the anti-skid system can result in loss of braking capability for either or both main gear wheels with or without illumination of the ANTI-SKID light. Therefore, the anti-skid switch must be placed OFF should loss of braking occur.

APPENDIX G

EA-6B AFCS UPGRADE

The EA-6B AFCS (Automatic Flight Control System) provides the aircraft with Stability Augmentation (SAS) as well as standard Auto-pilot modes. These functions are accomplished through the use of the AN/ASW-41 Air Navigation Computer (ANC). The ANC uses position, attitude, and rate feedback from aircraft sensors to augment commands to the aero actuator surfaces.

EA-6B AFCS is considered the number one fleet readiness concern, with a MFHBF (Mean Flight Hours Between Failure) of 200.23 hours, as reported by a 10/4/00 LMDS query. Furthermore, it was voted the number one fleet safety concern by the 1999 and 2000 SSWG (System Safety Working Group). The ANC is a 40-year-old analog computer design. Many of the ANC components are obsolete, making support of the ANC increasingly difficult. The numerous failure modes associated with the ANC have resulted in uncommanded flight control inputs and are among the most compelling safety situations taxing aircrew. Many of the ANC components are obsolete, making support of the ANC increasingly difficult. Due to these factors, the reliability, maintainability and readiness of this item are expected to continue to degrade.

Based on information available from a previous program to update the EA-6B ANC[1] and information obtained during the F-14 AFCS retro-fit program[2], the Program Executive Office Tactical Aviation (PEO(T)) are aware of at least two possible replacements for the ANC; the Standard Automatic Flight Control System (SAFCS CP-1780/ASW-50) [1] and the Digital Flight Control System (DFCS AN/ASW-59)[2]. The 9/28/00 RFI (Request for Information) release seeking candidates to upgrade the EA-6B AFCS is serving as the vehicle to provide additional alternatives which will be considered.

The SAFCS was presented to PMA-234 management on 2/3/98 by Northrop Grumman to replace the existing AFCS in the EA-6B. BAE North America (previously Lockheed Martin) is the manufacturer of the SAFCS. The SAFCS is predicted to achieve a MTBF of 1306 hours. It is currently installed on E-2C aircraft and is supported by CASS. The SAFCS is comprised of a digital computer that uses a BIT (Built In Test) to simplify diagnostics and produce significant savings in labor. It also provides flexibility for future configuration modifications. SAFCS implementation has been flight demonstrated on the EA-6B. In 1998, Grumman presented the requirements for the implementation and demonstration phase (DEM/VAL Program) for Pelts just prior to the decision to fund for the scheduled 1.5-year retrofit program, which did not proceed. Many lab efforts have been completed for the SAFCS and many of the algorithms have been thoroughly scrubbed. However, not all of the SAFCS effort is applicable to the latest anticipated EA-6B requirements.

The F-14 DFCS is manufactured by BAE SYSTEMS, Rochester, UK. The digital design is capable of satisfying the existing ANC functionality. The F-14 DFCS is nearing completion of the production program, which will replace the legacy AFCS in the F-14 aircraft. The DFCS has achieved a MTBF of an excess of 1000 hours for the system. The F-14 DFCS consist of 3 Flight Control Computers (FCCs) and a Digital Control Panel (DCP). For an ANC upgrade, one FCC and a DCP will meet the requirements. The DFCS computer is composed of two independent channels that cross monitor each other. This provides a more direct and dependable means of detecting computer failures compared with the single channel approach of the SAFCS. The dual channel internal architecture of the DFCS provides the potential for integration with the dual Embedded GPS/INS (EGIs) on the EA-6B aircraft. Integration with the EGIs would eliminate the need for the existing, unreliable, 3-axis gyro unit and provide an increased level of system safety. The DFCS system designers have incorporated a robust BIT, which not only increases system integrity but also reduces maintenance costs by greatly assisting faultfinding by the maintainers.

Both the SAFCS and DFCS are considered to be COTS. However, some modifications for both units will be necessary to meet the unique platform requirements that are being proposed. The 2 units are approximately the same size and weight and share similar power and cooling requirements. The SAFCS software is written in JOVIAL and is documented against earlier software development standards and practices. The DFCS software is written in Ada and developed to the latest military software standards. For both systems, parts obsolescence has been a key design driver. The DFCS computers are modifications of the Eurofighter Flight Control Computers. The Eurofighter program should guarantee the availability of parts over its 15-year production program, which minimizes the impact of parts obsolescence.

Both options will require a level of non-recurring development. Because the SAFCS was integrated and flown in the EA-6B as part of the PELTS contract, its' non-recurring efforts are envisioned to be less than that of the DFCS. Against this, the anticipated unit price of the SAFCS is on the order of \$90K, whereas the anticipated unit price for a DFCS computer is on the order of \$50K. The SAFCS is a domestic product. The DFCS is manufactured in the UK. This enabled the program to offset its funding issues through the use of the Foreign Comparative Test (FCT) [5] program. Although the DFCS was manufactured in the UK, the integration effort was performed at NAS PAX River, MD, under the Integrated Product Team concept. DFCS has also supported combat operations over Kosovo.

Conclusion

In conclusion, the Standard Automatic Flight Control System (SAFCS) and the Digital Flight Control System (DFCS) are both potential candidates to replace the existing Air Nav Computer (ANC). It will be essential that all proposed replacements, including identifying the reliability improvements that could be made to the existing ANC, be evaluated and scrutinized for reliability and redundancy of the system. The goal is to provide the fleet with a safe and reliable system that will increase the confidence the pilots and maintainers have with the EA-6B flight control system, while significantly increasing aircraft availability, improving performance and decreasing operating costs

Specification Sheet
Standard Automatic Flight Control System (SAFCS CP-1780/ASW-50)
Manufacturer BAE North America (Formerly Lockheed Control Sys)
Cost for Program **Unknown**
Estimated Cost per unit $90 K
BIT Yes
Implementation EA-6B
MTBF Grumman presented 1306 for EA-6B.
MFH/VF 2000
Obsolescence Risk E-2C FST Reports Low Risk
Computer Language JOVIAL
Satisfy ANC requirements Yes
Unanticipated Problems ? Not a drop in replacement for E-2C.
Aging Sensors
E-2C required wiring change and AFC.
Install Time unknown
Support Costs Reasonable ? Yes

Digital Flight Control System (DFCS AN/ASW-59)

Manufacturer BAE U.K.
Cost $80M Total (732 FCCs/215 DCPs) (Note 1)
Estimated Cost per unit $50 K
BIT **Yes**
Implementation F-14
MTBF Greater than 1000 (Note 2)
Obsolescence Risk Minimal
Computer Language ADA
Satisfy requirements Yes
Unanticipated Problems Tailoring existing library functions/algorithms to implement the EA-6B control laws
Aging Sensors
Install Time 15 Man Weeks with Checkout (note 1)
Support Costs Reasonable? Yes

Note 1 This includes the non-recurring costs for developing a 3-computer solution, to meet a requirement of significantly greater functionality than the EA-6B.

Note 2 This is for a 3 computer (plus DCP) solution. A single computer would exhibit an MTBF of greater than 3000 hours.

References:

Model EA-6B Aircraft Program, Engineering and Logistics Transition Support (PELTS), Contract N00019-94-D-0005, Report No.: PELTS-ENG-19; EA-6B Technical Report for the Engineering Investigation into Incorporating the SAFCS in Fleet Aircraft (CDRL #E001), 7th June 1995.

Production Detail Specification for Digital Flight Control System for F-14 Weapon System, AS-5742, 1st September 1999.

The FCT (Foreign Comparative Test) is a congressionally mandated effort that supports U.S. policy of encouraging international armaments co-operation and helps reduce overall DOD acquisition costs by facilitating procurement of foreign NDI (Non Developmental Items)[4]. EA-6B FCT peculiar Risk and Reward considerations are not cited herein. However, some of the common platform FCT risks and rewards are identified below.

NDI is identified by Congress as:

Any item available in the commercial market
Any previously developed item in use by the US or cooperative foreign government
Any item of supply needing only minor modifications to meet DoD requirements

Risks

1. FCT projects are scrutinized at higher levels within the US Navy and OSD.
2. Mission performance trade-offs at using NDI.
3. Logistics Support
4. Product Modifications
5. Concern over continued product availability

Rewards

1. Higher level of scrutiny can be used to advantage by:
Showing vendor OSD support
Knowing funds will not be taken by financial raids
Having confidence that your program is specifically approved by Congress.
Federal Acquisition Streamlining Act has provisions for NDI to establish and maintain alternative supply sources.
2. Quick response to operator needs
3. Elimination or reduction of research and development costs.
4. Application of State-of-the-Art technology

5. Reduction of technology cost and schedule risks.
6. Cost could be mitigated since cost is regarded in DOD as an independent variable.
7. DoD 5000.1 supports FCT usage.

Legal Language

A J&A (Justification and Authorization) should be used as in F-14 Sole Source.
Milestone Decision Authority must consider cooperative opportunities.
DFARS subpart 225.872 waives the Buy American Act. Ref DFAR sub 225.
Life cycle cost, effectiveness and suitability should be used for all components.[3]

George R. Schneiter, Director, Strategic and Tactical Systems OUSD(A&T), Office of the Under Secretary of Defense Acquisition and Technology, Foreign Comparative Testing Program, Handbook 1999, [Magazine On Line]Foreward; available From Microsoft InternetExplorer:http://www.acq.osd.mil/sts/fct/site.htm/Internet

Maj. Stan L. Vanderwerf, USAF, "How to Use FCT (Foreign Comparative Test) in Your Program," Program Manager Magazine, [Magazine On Line] (Defense Systems Management College Press July-August 2000), p.10; available from Microsoft Internet Explorer: http://www.acq.osd.mil/sts/fct/site.htm/;Internet.

VITA

Robert A. Coughlin was born in Chicago, Illinois on April 21, 1966. He grew up in Cedar Lake, Indiana and graduated from Crown Point High School in May of 1984. He enlisted in the United States Navy in August of 1984. He served nine years as an Aviation Structural Mechanic before being selected by the Navy to participate in the Enlisted Commissioning Program. He received his Bachelor of Science degree in Aerospace Engineering (Magna Cum Laude) from Auburn University in March of 1993, and was commissioned an Ensign in the United States Navy. He reported to Pensacola, Florida to begin flight training as a Student Naval Aviator. After receiving his wings in August of 1995, he was selected to be an pilot in the EA-6B Prowler community. After completing initial EA-6B training at Naval Air Station Whidbey Island, Washington, he served an operational tour in VAQ-130 “Zappers”. During this tour, he deployed once to the Mediterranean in support of Operation Provide Comfort, followed by transition to the Persian Gulf in support to Operation Southern Watch. In December of 1998, he reported to the U.S. Naval Test Pilot School at Patuxent River Naval Air Station, Maryland, and graduated with class 116 in December 1999. He currently is serving as an EA-6B / FA-18 / X-31 developmental test project officer/test pilot at the Naval Strike Aircraft Test Squadron in Patuxent River, Maryland.