FLY-BY-WIRE
FLIGHT CONTROL SYSTEMS

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Attn: FDCL.

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Introduction

The purpose of this paper is to provide the reader with an introduction to fly-by-wire and an outline of state-of-the-art fly-by-wire techniques. An outline of the philosophy of fly-by-wire flight control systems is given, the evolution of fly-by-wire is discussed, the advantages of fly-by-wire over mechanical systems are listed, current fly-by-wire techniques are outlined, and a brief review of the Air Force Flight Dynamics Laboratory proposed in-house and contracted fly-by-wire development programs is given.

The Philosophy of Fly-by-Wire

Before discussing fly-by-wire, it is important to understand what is meant by the term "fly-by-wire". Two other terms, "electrical primary flight control system" and "pseudo fly-by-wire", are often used in discussions of fly-by-wire and therefore also require definition. The following definitions of these three terms apply throughout this paper and have been generally accepted by the Air Force Flight Dynamics Laboratory.

Electrical Primary Flight Control System (EPFCS) - A flight control system mechanization wherein the pilot's control commands are transmitted to the moment or force producer only via electrical wires.

Fly-by-Wire - A fly-by-wire flight control system is an electrical primary flight control system employing feedback such that vehicle motion is the controlled parameter.

Pseudo Fly-by-Wire - A fly-by-wire flight control system with a normally disengaged mechanical backup.

Fly-by-wire, that is, the complete replacement of the mechanical linkages between the pilot's stick and the control surface actuators by electrical signal wires, offers a convenient and logical solution to many of the control system problems associated with modern high performance aircraft and aerospace vehicles. However, there exists a strong reluctance on the part of both pilots and flight control system designers to remove all flight control cables and mechanical linkages and rely solely on electrical signals and electronic devices. Nor is this reluctance unreasonable. Since the Wright Brothers first flew at Kittyhawk in 1903, there has been some form of direct mechanical linkage between the pilot and the control surfaces or control surface actuators. The successful use of such systems has resulted in the growth of a sense of security toward mechanical control linkages which now tends to inhibit fly-by-wire development. "Security is a mechanical flight control system" quips Snoopy as he pursues the Red Barron (Figure 1). Yet in today's high performance aircraft,

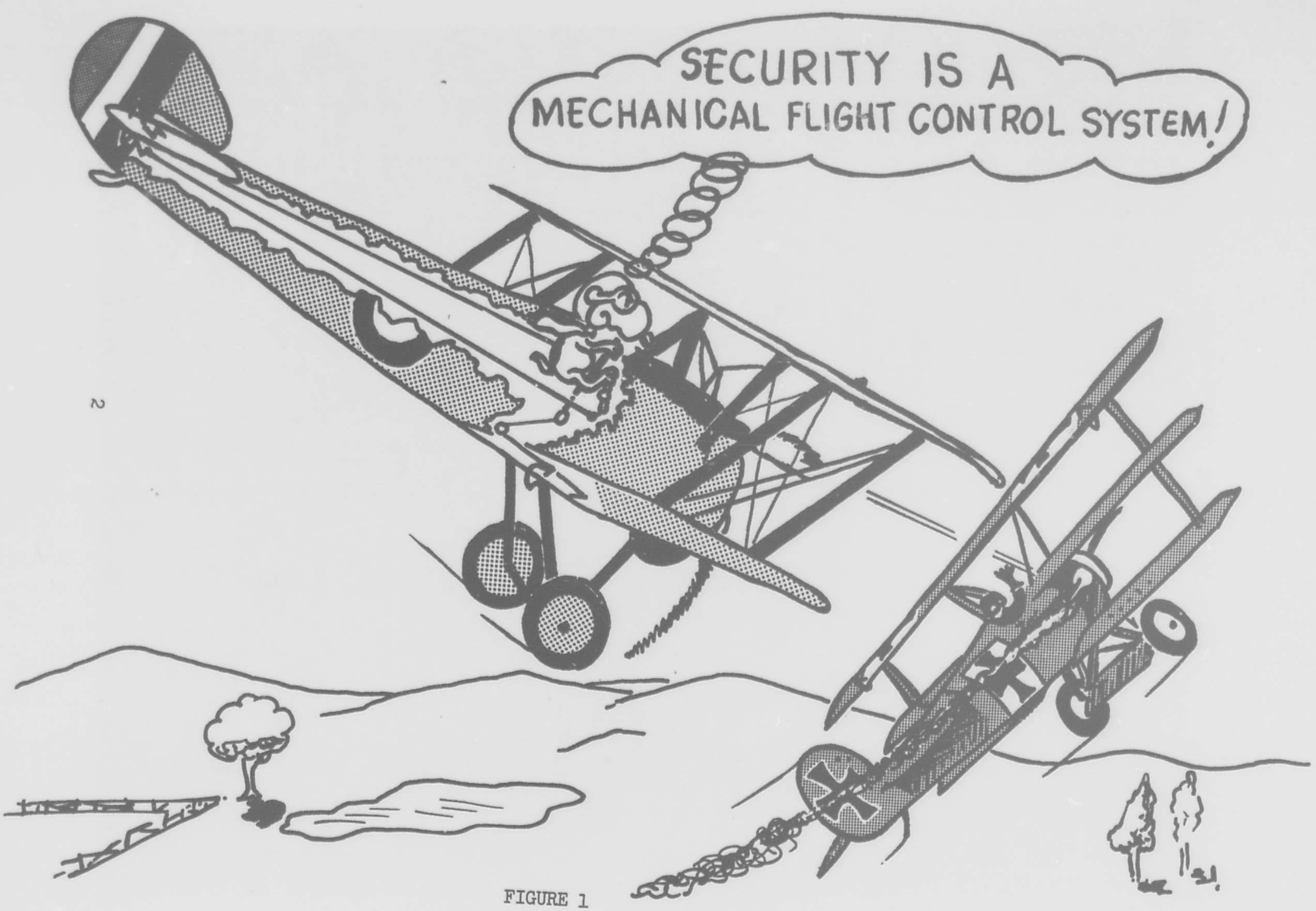


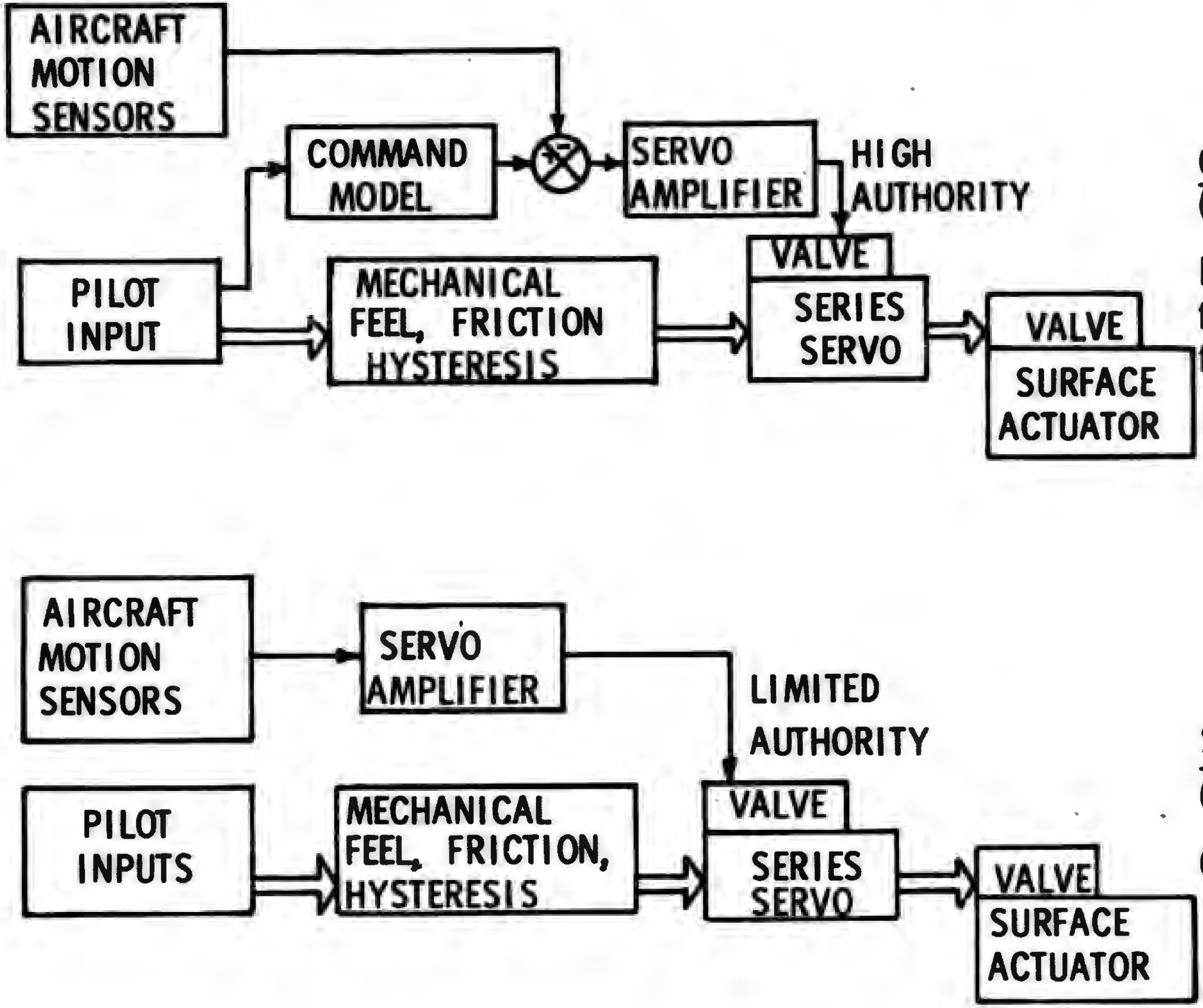
FIGURE 1

security is definitely not a mechanical control system. Instead, security is having a reliable Stability Augmentation System (SAS). For without SAS many of our high performance aircraft are only marginally stable and are, in fact, uncontrollable over a significantly large part of their operational flight envelope. Classic examples of these are the B-58 and F-111, both of which have severe stability problems if they lose their SAS and must, therefore, rely on electronic devices (black boxes) for the successful completion of a mission. In such cases, an effort has been made to obtain some of the advantages of fly-by-wire without losing the "security" of a mechanical system with the result that many of the disadvantages of the mechanical system are retained. The state-of-the-art in electronic circuits and redundancy techniques has now antiquated this approach. It is now possible to talk realistically about building a pure fly-by-wire flight control system that is <u>more</u> reliable than its mechanical counterpart. Until it is actually done, however, and successfully demonstrated in flight tests, the Missourian in many of us will prevail and the security stigma associated with mechanical control systems will predominate. Our fly-by-wire effort is orientated towards fulfilling this need.

The Evolution of Fly-by-Wire

The concept of fly-by-wire is not something which sprung up over night, but rather it evolved slowly through the years as aircraft flight control system requirements changed. With progressive increases in aircraft size and speed, power-boosted control quickly became a requirement in order to enable the pilot to utilize the full maneuver capability of the aircraft. Hydraulic boost, where a hydraulic actuator is connected in parallel to add to the pilot's force on the control cables, is still used on many aircraft; for example, the B-47, T-33, 707 rudder, and 727 elevators and ailerons. Shortly after World War II, fully powered controls came into being. Here the control cables from the pilot's stick are attached directly to the spool of the servo valve on the actuator and are in no way physically connected to the control surface. Feel is introduced into the system artificially with springs, dash pots, bob weights, and in some cases "q" bellows. This artificial feel, while not required in moving the control surfaces, is needed to give the pilot the proper handling qualities characteristics for control of the aircraft. Hence, although the pilot has no direct physical connection with the control surfaces, the artificial feel system gives him the impression that he has. Examples of aircraft using fully powered controls are the F-86, F-4C, F-104, F-105, and 727 rudder. One of the primary reasons for using fully powered control is that in the transonic region the forces on the surfaces vary greatly and are highly nonlinear. The resulting stick forces with direct mechanical connection to the control surfaces were unacceptable from a handling qualities point of view. Fully powered controls are inherently irreversible and thus unaffected by nonlinearities in the

COMPARISON OF STABILITY & CONTROL AUGMENTATION



CONTROL AUGMENTATION
(B-70, F-111, A-7A)

Remove mechanical link from Pilot to Series Servo for "Fly-by-Wire"

STABILITY AUGMENTATION
(F4, Century Series, C-141)

Often called "Damper"

FIGURE 2

transonic region, allowing the artificial feel system to be designed to give smooth transition from subsonic to supersonic flight.

As aircraft continued to increase in size and performance, it became necessary to add stability augmentation to assist the pilot in his control task. Stability augmentation systems (SAS), having very limited authority, were added in series with the normal flight control system. For some aircraft in certain flight regimes, however, the proper functioning of the SAS was required for the very survival of the aircraft. The success of SAS led to the introduction of CAS, Control Augmentation System. A control augmentation system has an electrical system operating in parallel with the mechanical control system. The electrical system predominates by virtue of its high gain and servo authority and performs essentially as a fly-by-wire system. The F-111 and B-70 aircraft have command augmentation flight control systems. The step from CAS to pseudo fly-by-wire is a small one and involves declutching the mechanical system when it is not in use. Examples of aircraft employing pseudo fly-by-wire flight control systems are the XV-4, the Concorde, and the X-15 when being flown on side stick. To get a fly-by-wire system from a pseudo fly-by-wire system, one needs only to remove the mechanical flight control system entirely. Fly-by-wire flight control systems are currently used in space vehicles such as Mercury, Gemini, and Apollo. Figure 2 illustrates the SAS to CAS to FBW evolution.

Need for and Advantages of Fly-by-Wire

The flight control systems of yesteryear, which consisted of relatively simple direct mechanical linkages, cables, and feel springs, can no longer meet the demands of advanced aircraft control system requirements. The flight control designer has been forced to replace the simple manual control system with complex nonlinear linkages, mixing assemblies, power actuation devices, and active artificial feel systems containing literally hundreds of different parts and interconnections. In his struggle to meet rigid performance and environmental requirements (such as immunity to aircraft structural changes due to flexing and thermal expansion) the designer has been confined by the requirements for low weight and high reliability. Hence, a compromise is forced and the full potential of many aircraft is never realized because of the resulting control system limitations. The degree of complexity to which flight control designers have had to go in their effort to solve these problems is best illustrated by an examination of Figure 3 which depicts a portion of the flight control system of the F-111 aircraft. You will note that the system is made up of a great number of relatively heavy push rods, bell cranks, and other linkages with a total of one hundred and fourteen bearing points. Each bearing point represents a source of friction and a possible failure point. Nor is the complexity of the F-111 FCS example shown unique. The B-70 flight control system is even more complex but would require such a large foldout to display that one might say it is

PITCH/ROLL MECHANICAL CONTROL SYSTEM

(F-111)

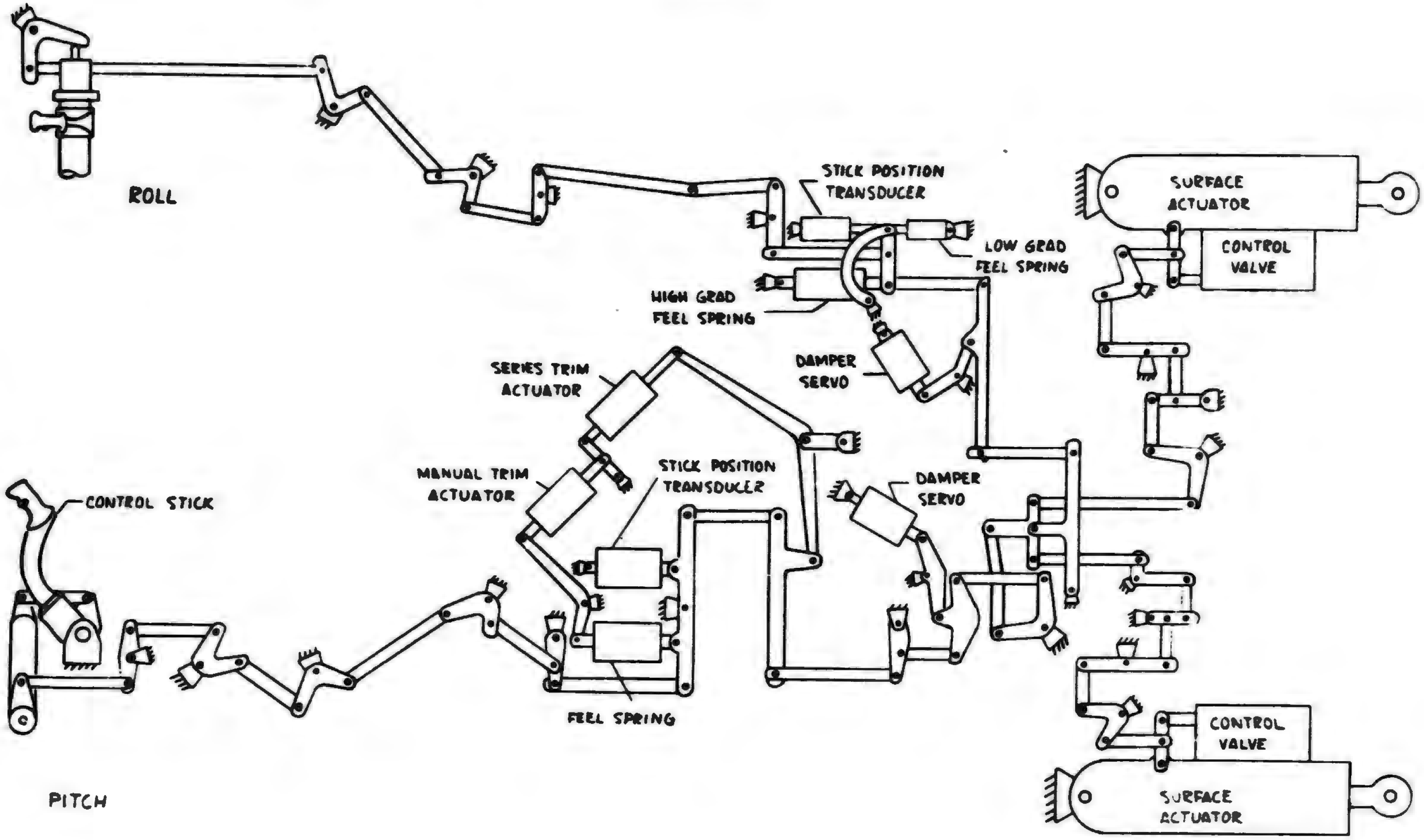


FIGURE 3

EQUIVALENT FLY-BY-WIRE SYSTEM
NON REDUNDANT

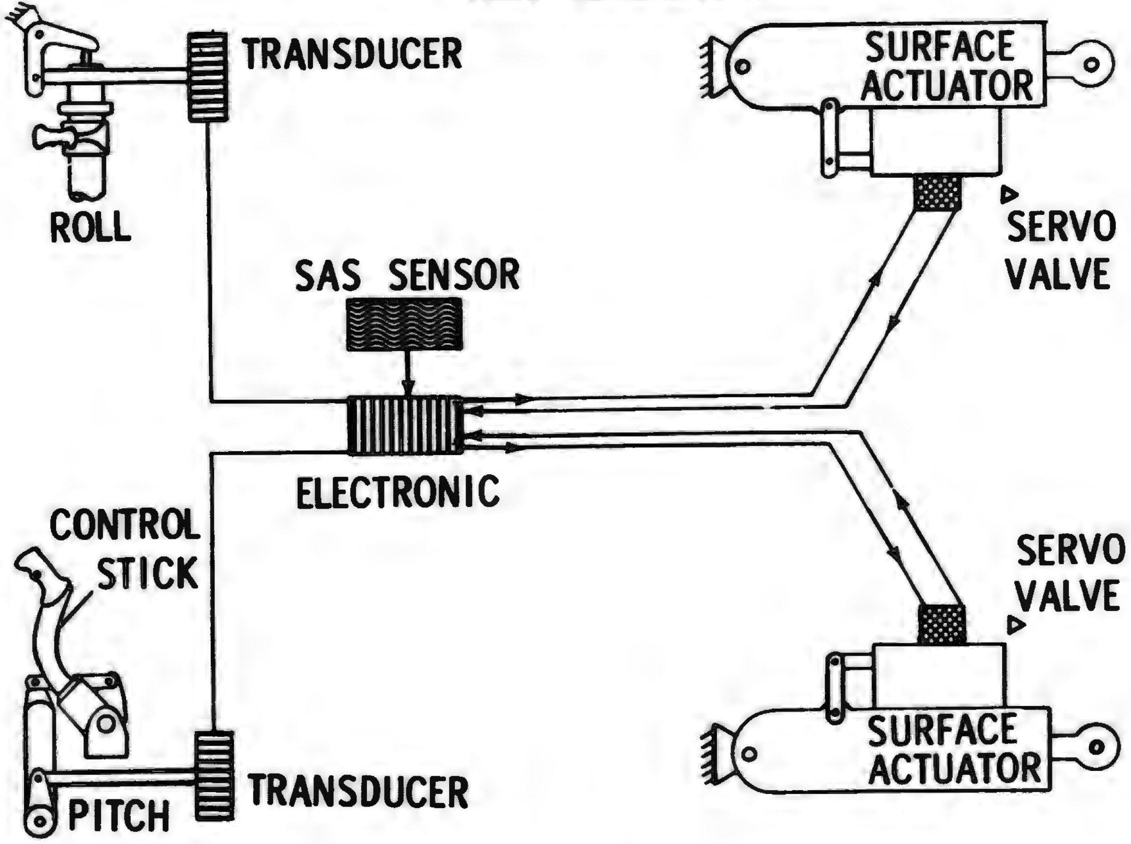


FIGURE 4

EQUIVALENT FLY-BY-WIRE SYSTEM FOR F-111
QUADRUPLY REDUNDANT

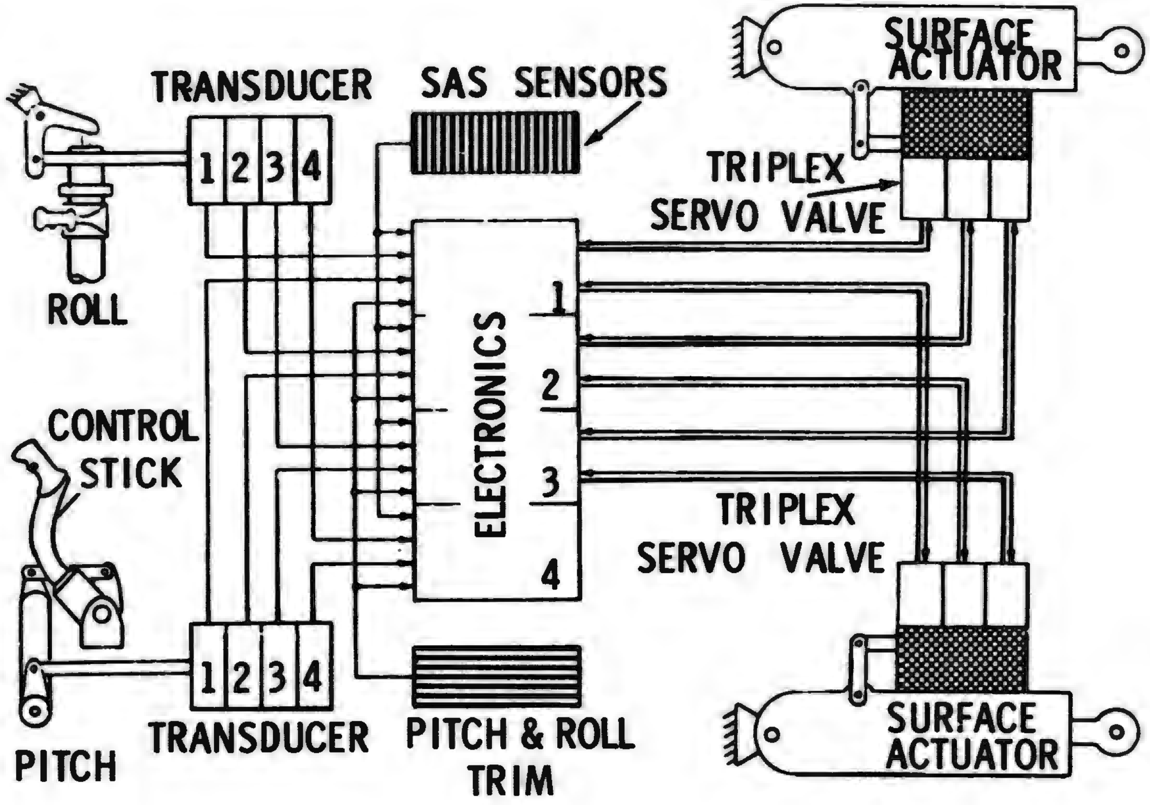


FIGURE 5

beyond the scope of this paper. Helicopter flight control systems are also enormously complex and their problems were multiplied several fold with the introduction of V/STOL aircraft. Figure 4 illustrates what a simple nonredundant fly-by-wire system for the F-111 might look like. This fly-by-wire system would do the same task as that complex mechanical system shown in Figure 3 and do it better. We would not suggest, however, that a simple nonredundant fly-by-wire system should ever be used in such an aircraft since its reliability would not be high enough for the mission requirement. A two-fail/operate or quadruply redundant fly-by-wire system, as shown in Figure 5, could meet all the requirements of the F-111 including reliability. This additional redundancy can be added to the simple fly-by-wire system with little complexity or weight penalty and with a significant increase in reliability.

Many of the advantages of a fly-by-wire flight control system over the conventional mechanical flight control system are self evident. Some, however, tend to become obscured by misinformation, skepticism, inertia, prejudice, or just plain ignorance. Below are listed some of the advantages of fly-by-wire with supporting facts and figures where available and applicable.

a. Design and Installation Savings - The design and installation manhour savings that can be realized by using fly-by-wire are fairly self evident. Cable tension, routing, and maintenance accessibility are only a few of the many problems which are virtually eliminated by fly-by-wire design. North American Aviation estimated that of the approximately 100,000 manhours spent on the design and installation of the flight control system of the XB-70, 90,000 manhours could have been saved had fly-by-wire been used. This represents a 90% reduction in manhours and would significantly affect the cost of the end item.

b. Weight Savings - The weight saving that can be realized by using fly-by-wire is very significant when considered as a percentage of the flight control system weight. For example, on the F-111 a 58.4% reduction in flight control system weight (277 lbs) could have been realized by using fly-by-wire. For an aircraft of this type where weight is at a premium, this savings would be very significant. A less significant, but nevertheless important, savings could be realized on aircraft of the C-5A type. Douglas estimated that on the DC-10 a reduction of weight of 21% (230 lbs) could have been realized by using fly-by-wire. A very significant savings in weight can also be achieved by using fly-by-wire on helicopters. Vertol estimates that on the CH-46 and CH-47 helicopters a savings of 77% (385 lbs) and 86% (718 lbs) , respectively, could have been realized by using fly-by-wire.

c. Volume Savings - The volume savings realized by using fly-by-wire is particularly significant on high performance aircraft such as the F-111. Sperry Phoenix estimated that a reduction in volume of

1469 cubic inches could have been effected on the F-111A/B by using fly-by-wire.

d. Increase in Flight Control System Reliability - Using a two-fail/operate approach, a net increase in system reliability can be realized. It should be noted that besides increasing the reliability number that can be applied to the flight control system, this degree of redundancy also takes care of unpredictable failures due to such things as maintenance oversights, an act of nature, or an inflight crew or passenger action, all of which are normally catastrophic failures with the conventional mechanical system. The probability of flight control system failure for a one-hour flight as derived from the maintenance records of the Civil Aeronautics Board and the Federal Aviation Agency for the period of 1952 to 1959 is 2.3 X 10^{-7}. (Ref 3)This value establishes the nominal reliability criteria for fly-by-wire. Using a two-fail/operate or quadruple redundancy approach where two of four channels must operate for system success, this criterion can be met with a single channel failure rate per hour of 2.7 X 10^{-3}. This is well within the current state-of-the-art.

e. Reduction in Maintenance Manhours - Contrary to popular belief, a fly-by-wire system could actually result in a reduction in control system maintenance manhours. Modular packaging would permit the rapid repair of failed units, and the expensive and time consuming rerigging which must now be periodically done, would be eliminated. Built in test equipment (BITE) would quickly detect and isolate failed components and these could then be replaced by new ones. An increase in the frequency of failure occurrence would be expected, but the ease of failure isolation and repair would more than offset this increase. Vertol estimates that for complex control systems such as are used in helicopters and other V/STOL aircraft, a reduction in control system maintenance manhours of as much as 80% or more could be realized by using fly-by-wire.

f. Decrease in Vulnerability - The flexibility of routing which is inherent in a fly-by-wire system would permit redundant control channels and electronics to be separated radially and longitudinally throughout the aircraft so that complete control system failure due to any one bullet or projectile would be impossible unless it was of such a nature as to destroy the integrity of the airframe anyway. Vertol estimates that with no increase in armour they could realize a 50% reduction in vulnerability for their helicopters by using fly-by-wire. Minimum control system vulnerability could be achieved by using fly-by-wire in conjunction with self contained hydraulic actuator packages. These servo actuator packages, currently being studied by the AF Flight Dynamics Laboratory, would be comprised of redundant electrically driven motor-hydraulic pump-accumulator units completely contained within the actuator housing. With careful design of this housing, the parasitic armour required to protect these servo actuator packages from small arms fire would be minimized. Figure 6 illustrates how a redundant fly-by-wire flight control system in conjunction with such servo actuator packages could make the flight control system of an aircraft, such as the F4C shown here, virtually invulnerable to small arms fire except in the case of direct multiple hits.

g. Improved Aircraft Handling Qualities - Mechanical control system nonlinearities such as stiction, friction, and hysteresis are eliminated by using fly-by-wire. The aircraft response versus feel (stick force) can be readily adjusted to meet pilot desires at all flight conditions. A recently completed study (Reference 9) which applied linear optimal control theory to the problem of control blending (phasing) and attitude stabilization and control for VTOL aircraft, and which used the Bell X-22A ducted-propeller VTOL aircraft as the application vehicle emphasizes the improvement in handling qualities with fly-by-wire. During this study, Bell Aerosystems test pilots on the X-22A project evaluated, using Cooper rating, X-22A transition controller designs on a six degree of freedom flight simulator. Comparative flight tests were done using the present X-22A control system, an optimal controller tied into the existing mechanical blender, and finally, an optimal controller using fly-by-wire. From the results of these evaluations, as shown in Table I, it is evident that a very significant improvement in handling qualities can be gained by using fly-by-wire on VTOL aircraft.

TABLE I

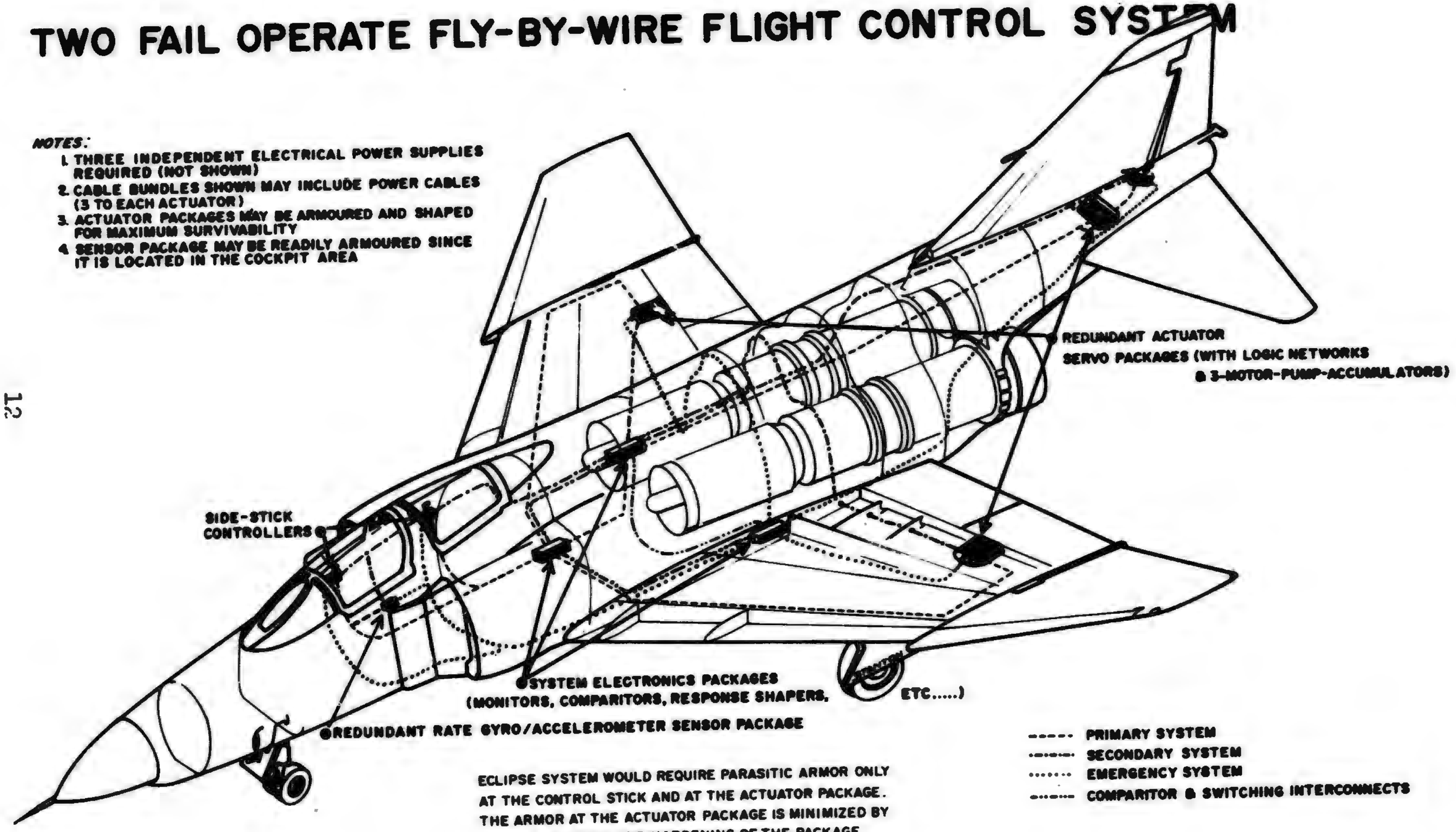
Pilot Cooper Ratings for the Several

X-22A Transition Controller Configurations

Control System Configuration	Pitch		Roll		Yaw	
	Hover	Transition	Hover	Transition	Hover	Transition
Optimal Controller With Fly-by-Wire	2.0	2.25	2.0	2.5	2.5	2.5
Optimal Controller With Existing Mechanical Blender	2.75	3.0	3.0	3.0	3.25	3.25
Present Control System	3.0	3.5	3.5	3.5	3.5	3.75

F-4C

TWO FAIL OPERATE FLY-BY-WIRE FLIGHT CONTROL SYSTEM



NOTES:

1. THREE INDEPENDENT ELECTRICAL POWER SUPPLIES REQUIRED (NOT SHOWN)
2. CABLE BUNDLES SHOWN MAY INCLUDE POWER CABLES (3 TO EACH ACTUATOR)
3. ACTUATOR PACKAGES MAY BE ARMOURED AND SHAPED FOR MAXIMUM SURVIVABILITY
4. SENSOR PACKAGE MAY BE READILY ARMOURED SINCE IT IS LOCATED IN THE COCKPIT AREA

ECLIPSE SYSTEM WOULD REQUIRE PARASITIC ARMOR ONLY AT THE CONTROL STICK AND AT THE ACTUATOR PACKAGE. THE ARMOR AT THE ACTUATOR PACKAGE IS MINIMIZED BY PROPER SHAPING AND HARDENING OF THE PACKAGE ITSELF.

FIGURE 6

A convenient method for mechanizing the flight control system feel/response is the C* approach as proposed by Boeing. This approach, which has gained some popularity, uses a linear blend of normal acceleration, pitch rate, and pitch acceleration. C* is defined as:

$$C^* = k_1 \, n_z + k_2 \, \dot{\theta} + k_3 \, \ddot{\theta}$$

where

n_z = normal acceleration at the c.g., positive up

$\dot{\theta}$ = pitch rate

$\ddot{\theta}$ = pitch acceleration

If C* is defined in g's, we find that the units of k_2 are equivalent to a velocity divided by g and those of k_3 are equivalent to a length divided by g. By setting k_1 equal to 1, the equation may now be written:

$$C^* = n_z + \frac{U_{co}}{g} \dot{\theta} + \frac{L}{g} \ddot{\theta}$$

where

U_{co} = the cross over velocity (approximately equal to 400 ft/sec)

L = the distance between the linear accelerometer and the c.g. of the aircraft

The crossover velocity represents the point where the contribution of pitch rate and normal acceleration to the feel are equal. (This does not necessarily mean that this is the velocity at which the pilot attaches equal importance to them.) C* is a convenient approach for the mechanization of a feel system because of the ease with which $\dot{\theta}$ (pitch rate) and $\ddot{\theta}$ and n_z (pitch and normal accelerations respectively) can be measured. These variables are often already being sensed in the stability augmentation system. Figure 7 shows a typical fly-by-wire flight control system with C* feedback. Figure 8 illustrates one method of specifying a handling qualities criterion in terms of the C* mechanization. These boundaries were obtained by reducing the results of the handling qualities studies conducted by Cornell on the F-94 (Reference 2).

h. Immunity to Aircraft Structural Changes Due to Flexing, Bending, Thermal Expansion, Etc. - Mechanical control systems are very sensitive to aircraft structural changes and great pains must be taken by the designer to try and minimize their effects. With fly-by-wire their effects are inherently eliminated. The fact that an SST will increase in length by approximately 7 to 12 inches due to aerodynamic heating

SIMPLIFIED BLOCK DIAGRAM OF FLY-BY-WIRE CONTROL SYSTEM-PITCH AXIS

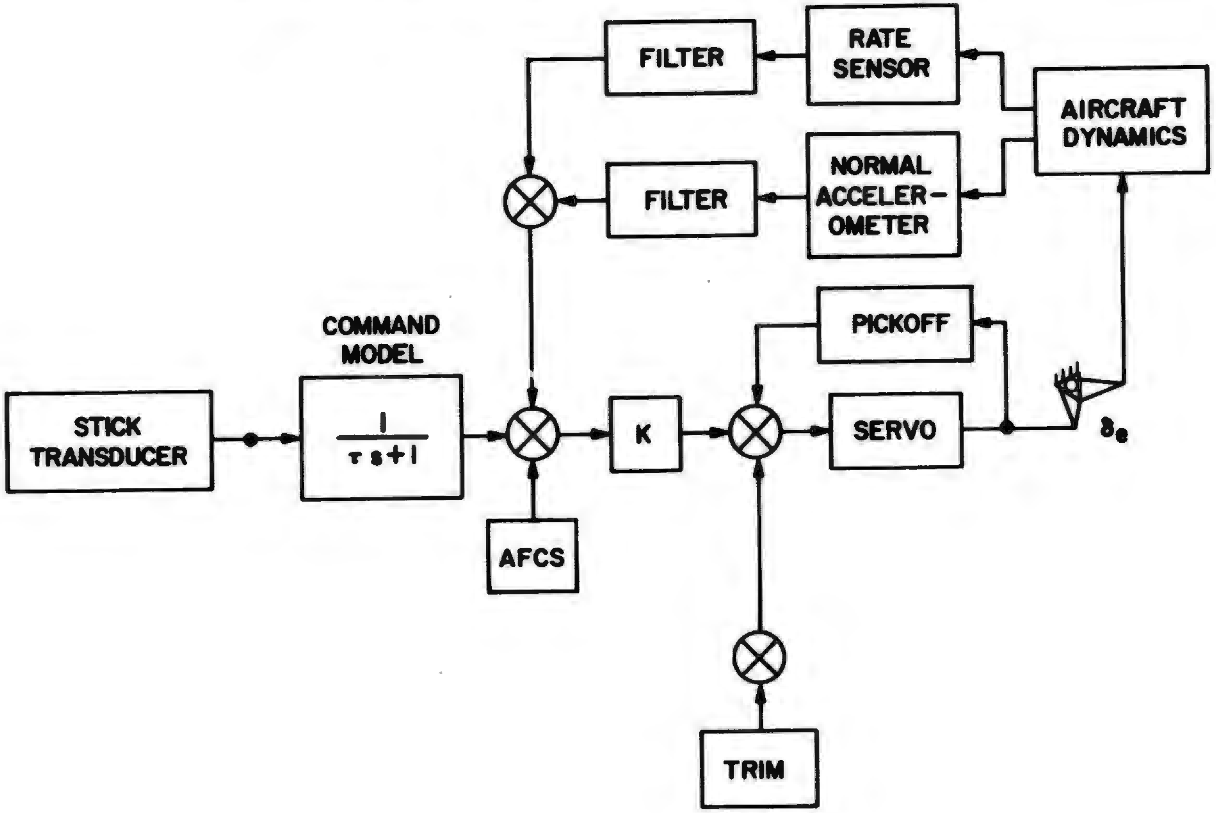
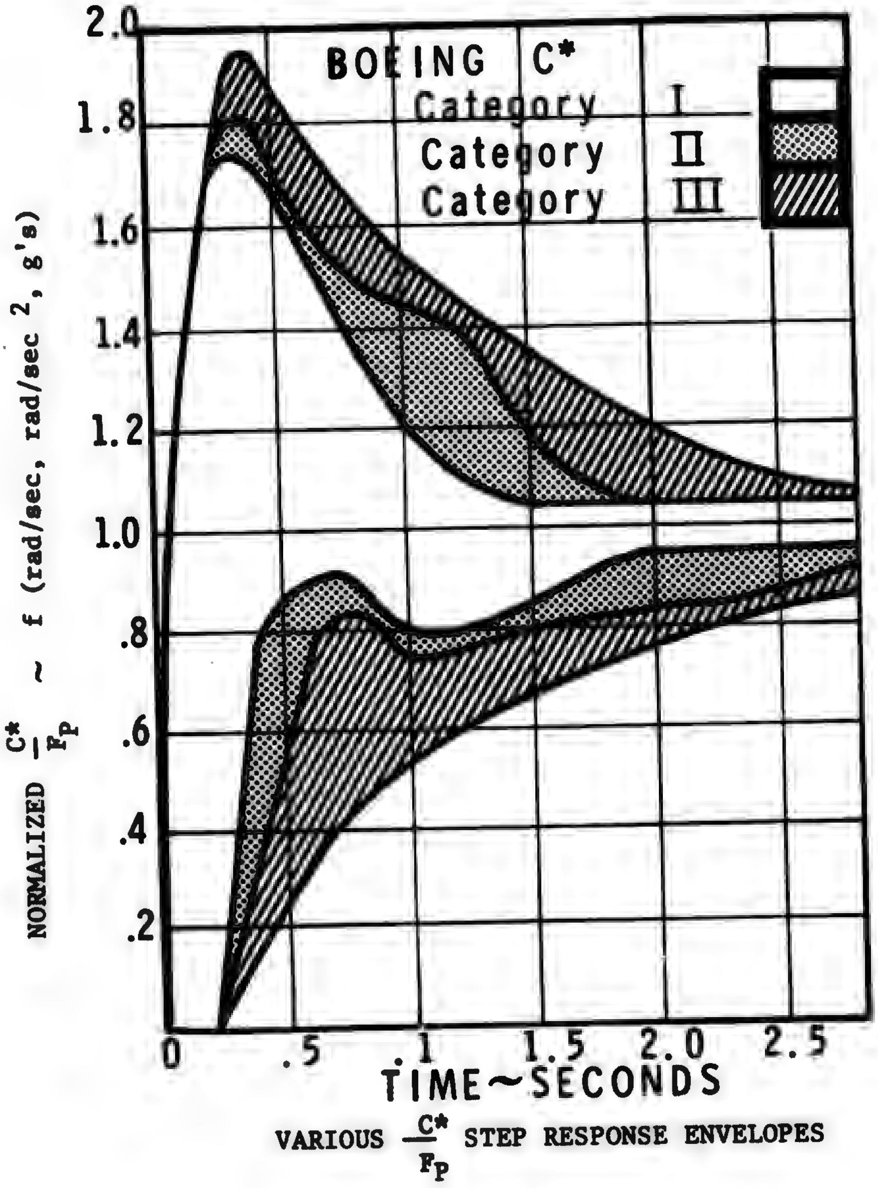


FIGURE 7



VARIOUS $\frac{C^*}{F_P}$ STEP RESPONSE ENVELOPES

FIGURE 8

is of little concern to the designer of a fly-by-wire control system. It is even possible to incorporate a form of structural mode stabilization into a fly-by-wire system to give the airframe an effective rigidity and thus increase its fatigue life. This concept is presently being investigated by the Air Force Flight Dynamics Laboratory on a B-52 aircraft at Boeing Wichita in a program called LAMS (Load Alleviation and Mode Stabilization). The initial goal of this program was to improve the structural fatigue life of the B-52 by 100%. The analysis effort on the test aircraft is complete and the results of the 35 degree of freedom computer simulation indicate that the sought improvement in structural fatigue life is generally met and in some cases greatly exceeded. The following table shows the improvement obtained at certain stress locations (Wing Stations, Body Stations, Stabilizer Butt Line, Fin Station) previously determined critical through Boeing B-52 fleet experience. Along with these benefits, a 20% improvement in rms acceleration at the pilot station was achieved.

TABLE II

Fatigue Damage Per Flight Hour

Stress Location	WS 516	WS 899	BS 805	BS 1028	SBL 32	FS 135
Basic A/C	2.3×10^{-3}	2.2×10^{-3}	$.58 \times 10^{-3}$	$.78 \times 10^{-3}$	$.059 \times 10^{-3}$	1.0×10^{-3}
LAMS Control Sys	$.4 \times 10^{-3}$	1.2×10^{-3}	$.09 \times 10^{-3}$	$.2 \times 10^{-3}$	$.027 \times 10^{-3}$	$.17 \times 10^{-3}$
Factor of Improvement	5.75	1.83	6.45	3.9	2.18	5.9

i. Flexibility of Cockpit Layout - Fly-by-wire permits the use of side stick controllers and other forms of miniature controllers. Thus, the pilot's view of the instrument panel need no longer be obstructed by a large, center-mounted control stick. A variety of side stick and miniature controllers have been investigated and flown, and at present it would appear that the most acceptable type of side stick controller would be a displacement (rather than rigid) stick with a grip which is smaller than that currently being used. Another big advantage which comes from using a side stick controller with suitable arm rests or supports is the elimination of the effect of g's on the pilot's input to the stick.

j. Ease of Incorporation of Automatic Flight and Landing Systems - Fly-by-wire eliminates the requirement for series and parallel servo actuators and complex blending and mixing linkages for Automatic Flight Control Systems (AFCS) and Automatic Landing Systems (ALS) since all inputs are electrical and summing is done electronically forward of the servo power actuator. Furthermore, fly-by-wire has its own built-in stability augmentation system.

k. More Flexible to Design or Performances Changes - Fly-by-wire is affected very little by configuration or system performance changes which might be introduced during the production phase. This is true because of the inherent flexibility of fly-by-wire design, and the fact that sensed vehicle motion is the controlled variable rather than control surface position.

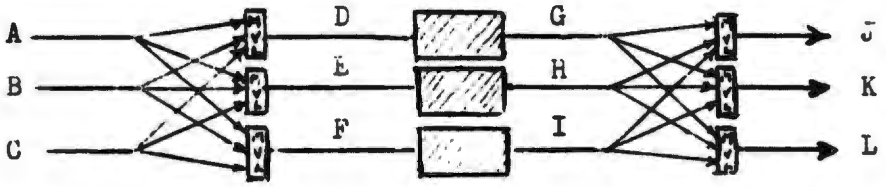
l. Decrease in Cost of Ownership - Each of the above factors contribute to an overall decrease in the total cost of ownership of aircraft with fly-by-wire flight control systems, a fact of major importance to both the Government and to commercial airlines.

Description of Fly-by-Wire Flight Control Systems

Degree of Redundancy

One of the first problems which faces the designer of a fly-by-wire flight control system, once he has decided to eliminate any form of mechanical backup, is the degree of redundancy to design into the system. Two factors must influence his decision: first, the reliability required, and second, the failure-mode philosophy, both of which are dependent on vehicle mission. Our studies to date have indicated, that for large advanced military or commercial aircraft, the most acceptable degree of redundancy would be one which would provide a two-fail/operate capability, thus allowing completion of mission after any single failure. This is sometimes referred to as quadruple redundancy and will accept two <u>like</u> failures in the system with little or no system performance degradation. On a third similar failure, the system goes to a neutral (soft) or preselected trim position. In this system, the pilot and/or flight engineer will have the option of bypassing each logic section and positively selecting each functional module throughout the control system. This will permit the pilot, after three like failures, to make full use of any system capabilities which might remain. A module which failed to produce acceptable performance for normal operation might thus be successfully used to get the aircraft home. This two-fail/operate philosophy is illustrated in Figure 9. Here three functional modules and an electrical model are used to give the quadruple redundancy. The number of comparators and their location in the system and the type of logic to be used (median select, majority vote, etc.) are optional. These and the degree of performance degradation allowable after a second failure would be determined at the time of system design. The advantage of using several sets of comparators and logic networks throughout the system as opposed to a single set at the output of each channel is obvious from this figure when it is noted that the hypothetical flight control system depicted here could withstand eight (8) distinct failures and still be fully operable provided no three failures were alike; i.e., one functional module of each set must be serviceable. The system, however, should not contain any "nodes" where all channels feed into a single comparator or black box, the failure of which would wipe out the entire control system. The comparators should also be redundant, a fact which Figure 9 does not depict. Further, if a comparator should fail, it should be detected and switched out like any other failure. This is not to say that redundant monitors are required to monitor the operation of the redundant comparators that monitor the operation of the redundant modules!! Careful design of the comparators should accomplish this end without resorting to the ridiculous. Consider, for example, the redundant comparators shown in Figure 10 which use Mid-Value-Logic (MVL) techniques. For simplicity, only three channels are shown.

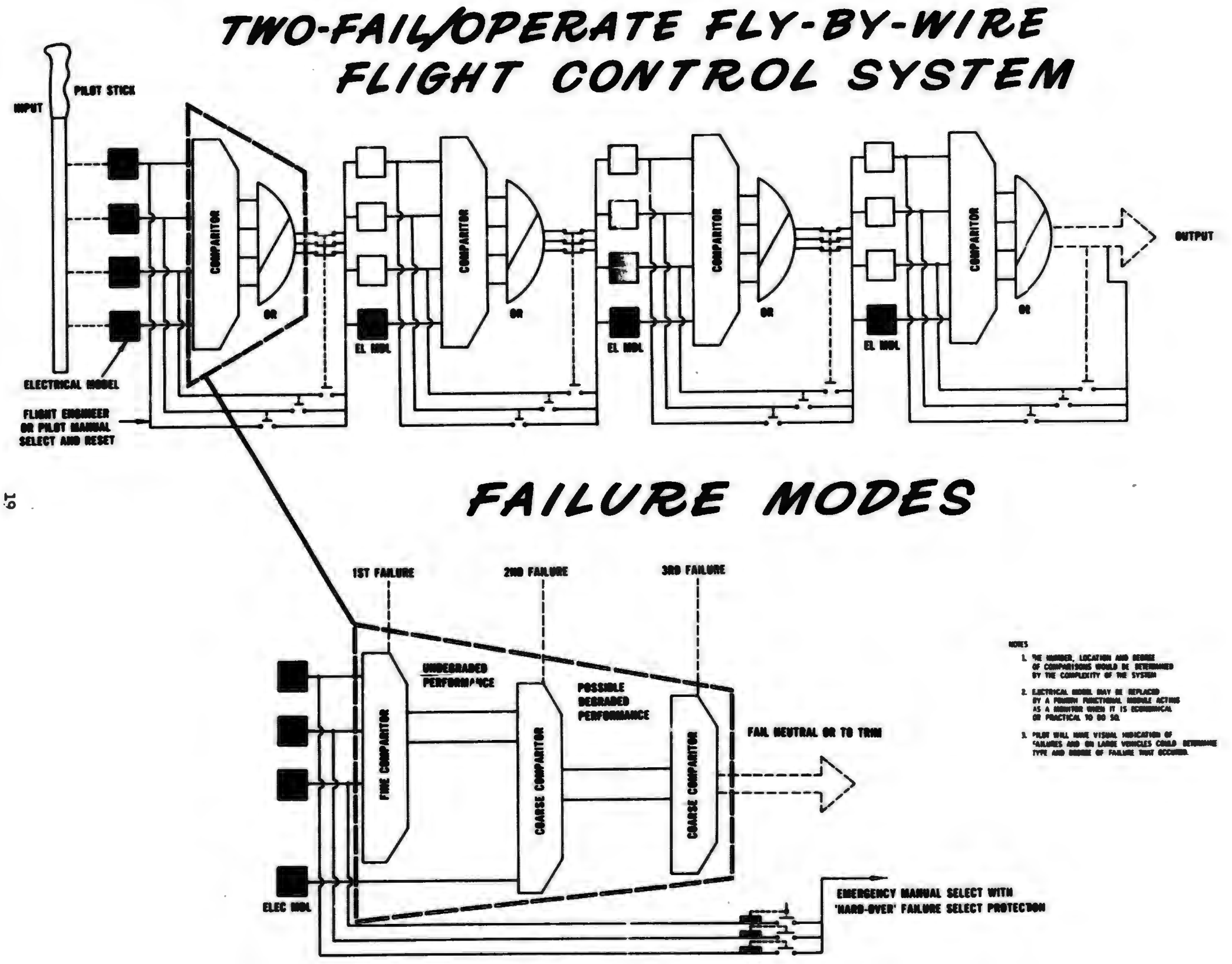
FIGURE 10



Where $A \neq B \neq C$; $D = E = F$; $G \neq H \neq I$; $J = K = L$

One of the things which makes a two-fail/operate or quadruply redundant scheme feasible is the current state-of-the-art of sensors, transducers, electronics, and servo actuators. Small, reliable, lightweight, and relatively low cost sensors and transducers are available as off-the-shelf hardware. The penalty which must be paid for using quadruple redundancy is thus minimized. Sensors of the future currently under development at the Air Force Flight Dynamics Laboratory promise to reduce this penalty even further. An example of this is the DART sensor (Figure 11) which uses rotating mercury to sense rate of rotation and linear acceleration in two axis. A quadruply redundant sensor package measuring pitch rate and normal acceleration could be contained in a package 6" X 3" X 3" weighing less than three pounds. The electronics field has miniaturized even more so. Microelectronic circuits enable mixing, blending, voting, and, in general, response shaping to be done on a relatively small number of easily replaced cards. Metal oxide semiconductor (MOS) techniques decrease the size of these components by several orders of magnitude. Using these techniques it is now possible to redundasize electronics at a functional module level with a resulting decrease in size, weight, and cost and a net increase in system reliability.

If there is a "fly in the ointment", so to speak, it is in the area of the servo actuator. Until recently very little research work had been done in the area of redundant servo actuators. This discussion will be confined to hydraulic servo actuators since they appear to be most suitable for aircraft requirements of the immediate future. That is not to say that pneumatic, electrical, or other types



NOTES

1. THE NUMBER, LOCATION AND DEGREE OF COMPARISONS WOULD BE DETERMINED BY THE COMPLEXITY OF THE SYSTEM
2. ELECTRICAL MODEL MAY BE REPLACED BY A FOURTH FUNCTIONAL MODULE ACTING AS A MONITOR WHEN IT IS ECONOMICAL OR PRACTICAL TO DO SO.
3. PILOT WILL HAVE VISUAL INDICATION OF FAILURES AND ON LARGE VEHICLES COULD DETERMINE TYPE AND DEGREE OF FAILURE THAT OCCURRED.

FIGURE 9

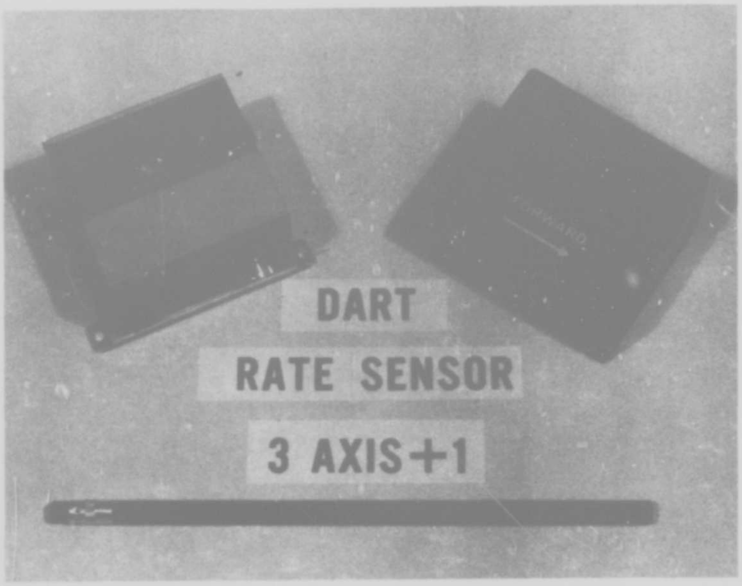


FIGURE 11 DART RATE SENSOR

of servo actuators will not be required or used on future aircraft. At present, redundant hydraulic actuators promise the most for the least amount of development. The redundant servo actuators currently under development fall conveniently into the following three categories: (a) electronic logic and switching, (b) fail passive with electronic logic, and (c) hydraulic logic and switching. Each of these techniques has specific advantages and disadvantages in comparison with one another. The first, electronic logic and switching, can make full use of the size and weight advantages to be gained by using MOS techniques. The use of well proven electronic logic techniques and the ease of fault detection and correction through the use of modular packaging also makes this method attractive. It does, however, require transformation from one power media to another; i.e., electrical to hydraulic, in order first to detect the fault and second to correct it. The result is an increase in switching time and a decrease in reliability. A typical solenoid valve operates in 20 to 25 milliseconds. To keep the switching time down to a reasonable value (say 50 milliseconds), it would be necessary for the detection and electronic logic to take place in 25 milliseconds or less. Such a reaction time, although demanding, is within the current state-of-the-art. The second technique, fail passive with electronic logic, has the advantage of no switching transients since all servo valves are operating and when one or more fail passively the remaining servo valves continue to drive the secondary actuator with negligible system performance degradation. This system has a further advantage of being able to supply two-fail/operate redundancy with three servo valves. The main disadvantage with this technique is that it is extremely difficult to design a 100% pure fail passive system. Consequently, it is necessary to include an electronic model as protection against a hardover failure even though the possibility of one occurring is remote. The third technique, hydraulic logic and switching, eliminates the power interface problem by performing all detection, logic, and switching functions in the hydraulic medium. The result is a decrease in switching time and an increase in reliability. Typical detection logic and switching times for such actuators are less then 10 milliseconds. These systems are sensitive to contamination and silting in the hydraulic fluid, but state-of-the-art filtering techniques can minimize this problem.

Two-Fail/Operate Redundant Servo Valve with Electronic Logic and Switching

A two-fail/operate hydraulic servo actuator using electronic logic and switching technique has been designed by the General Electric Company (Johnson City) under the sponsorship of the Air Force Flight Dynamics Laboratory and is described in technical report AFFDL-TR-67-17. An excerpt from this report describing the operation of this actuator follows:

> The actuator (see Figure 12) consists of three pistons and sleeves (cylinders) mounted in a common housing. The

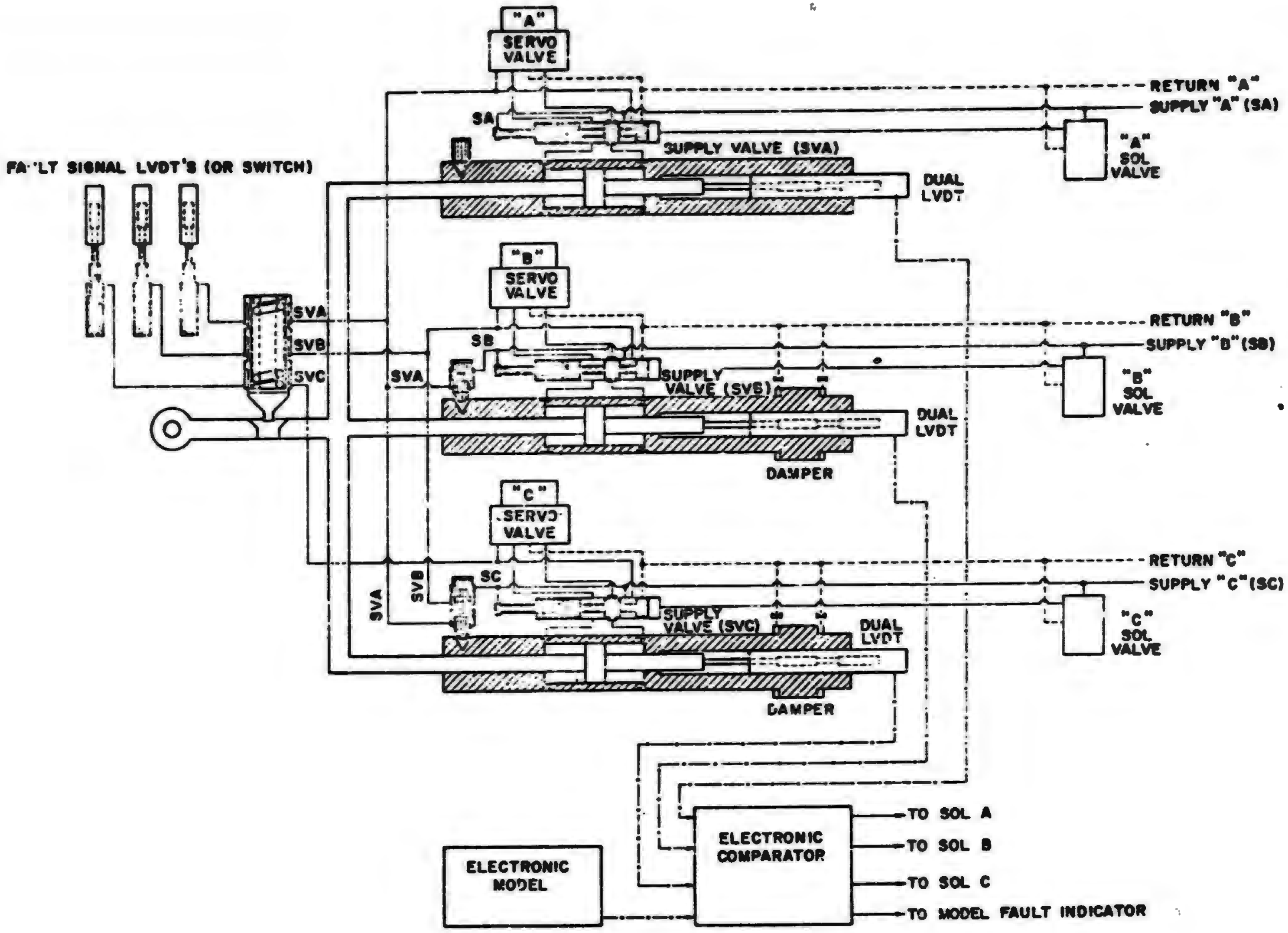


FIGURE 12 TRIPLEX ACTUATOR SCHEMATIC

ELECTRONIC LOGIC & SWITCHING

three pistons are connected together and form a common output. Their sleeves are mounted such that the channel "A"sleeve is always grounded or secured to the housing while the "B" and "C" sleeves are free to move with respect to the common output within a fixed mechanical limit. With the main output ram at its centered position the sleeve actuators are free to move ± 0.10 inch before reaching the mechanical limits. The sleeves can be locked to the housing depending upon the present failure conditions. Therefore, actuators "B"and"C"function as operating standby actuators.

Under normal operating conditions, the channel"A" actuator is driving the load. The actuators"B"and "C"are operating, but cannot drive the load since their sleeves are floating and provide no reaction. Normal tolerances between flow gains and driving signals are taken up by the floating sleeves. Since the flow gain may change in actuator"A"compared to "B" and "C" under dynamic loading conditions, an auxiliary damper is attached to the "B" and "C" sleeves to obtain some load sharing during maximum slew rate inputs.

When differences in actuator position exceed a reference threshold (channel"A"failure), the difference in voltage output of the Linear Variable Differential Transformers (LVDT) will be detected in the comparators. The electronic logic will determine the failed actuator and remove 28 volts d.c. from a three-way, two-position solenoid valve. This in turn ports one end of the bypass-shutoff valve to return. This action allows the valve spool to travel in the bypass direction due to the force of the unvalved supply pressure and spring acting on the opposite end. During its travel, the spool valve lands are made such that the pressure to the sleeve lock on the next actuator is shutoff and connected to return. This allows the next actuator sleeve to start being locked to the housing as the failed actuator pistons are bypassed. When the failed actuator is bypassed, the control ports from the servo valve are short circuited and oil is allowed to circulate so that no force is exerted on the output.

The operation of the actuator is set up such that if the channel"A"actuator fails first, its piston is bypassed and"B"actuator sleeve is

> locked to the housing and becomes the driving actuator. If channel B actuator fails after a channel A failure, then its piston is bypassed and the channel C actuator is secured to the housing. Now, if a third failure occurs in either the electronic model or actuator C, then the main ram is centerlocked to the housing in a position that corresponds to the normal neutral position of the actuator output.
>
> The outputs of the pressure switches shown in the schematic are sent to the electronic logic circuitry where they are used to reset the logic after a failure has occurred. The signals are also combined with the normal shutdown and failure indication logic to obtain an immediate fault indication if a pressure failure occurs.

Two-Fail/Operate Redundant Servo Valve Using Fail Passive Technique with Electronic Logic

The Sperry Phoenix Company has designed a two-fail/operate fail passive redundant servo actuator under their fly-by-wire contract with the Air Force Flight Dynamics Laboratory, and it is described in technical report AFFDL-TR-67-53. An excerpt from this report follows describing the operation of the fail passive actuator as shown in Figure 13.

> This technique employs a small redundant secondary actuator which mechanically drives the main control valve and power actuator with nearly unity feedback. The dual mechanical linkage can be sealed within the actuator body where it is protected and bathed in oil. Both the secondary and power actuators employ active redundancy. When dual hydraulic supplies are used, the secondary actuator is dual tandem with two single-stage jet-pipe valves driving each piston thus forming four inner servo loops. When triple hydraulic supplies are used, the secondary actuator is triple tandem with a single valve driving each piston thus forming three inner loops.
>
> The uniqueness of the configuration derives from the inner loops which are designed to have passive failure characteristics. A fail-passive channel fails in such a way that it has no output and it does not interfere with the normal operation of a parallel channel. In other words, active or hardover failures have been eliminated by design. Since a failed channel has no force output, the other good

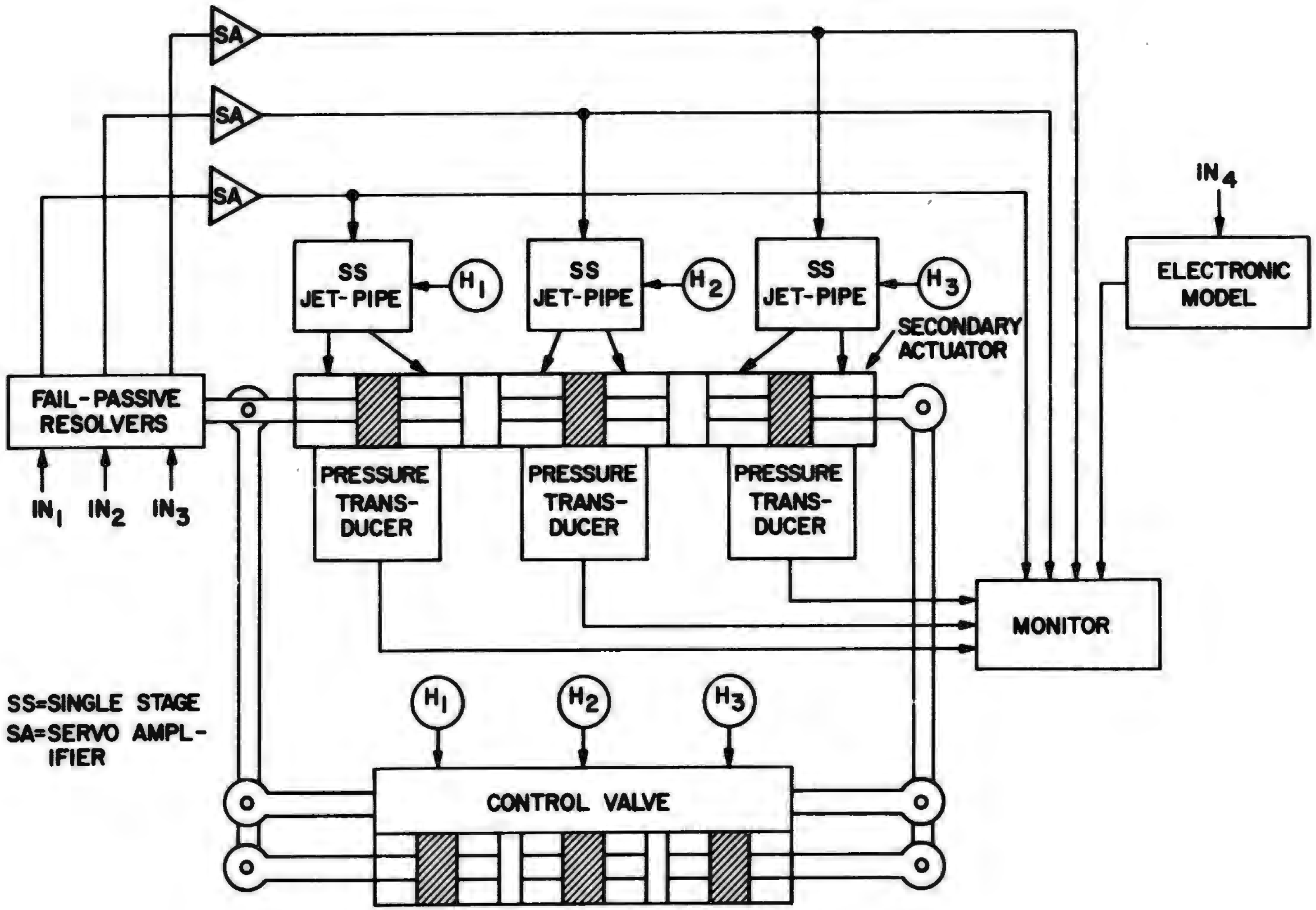


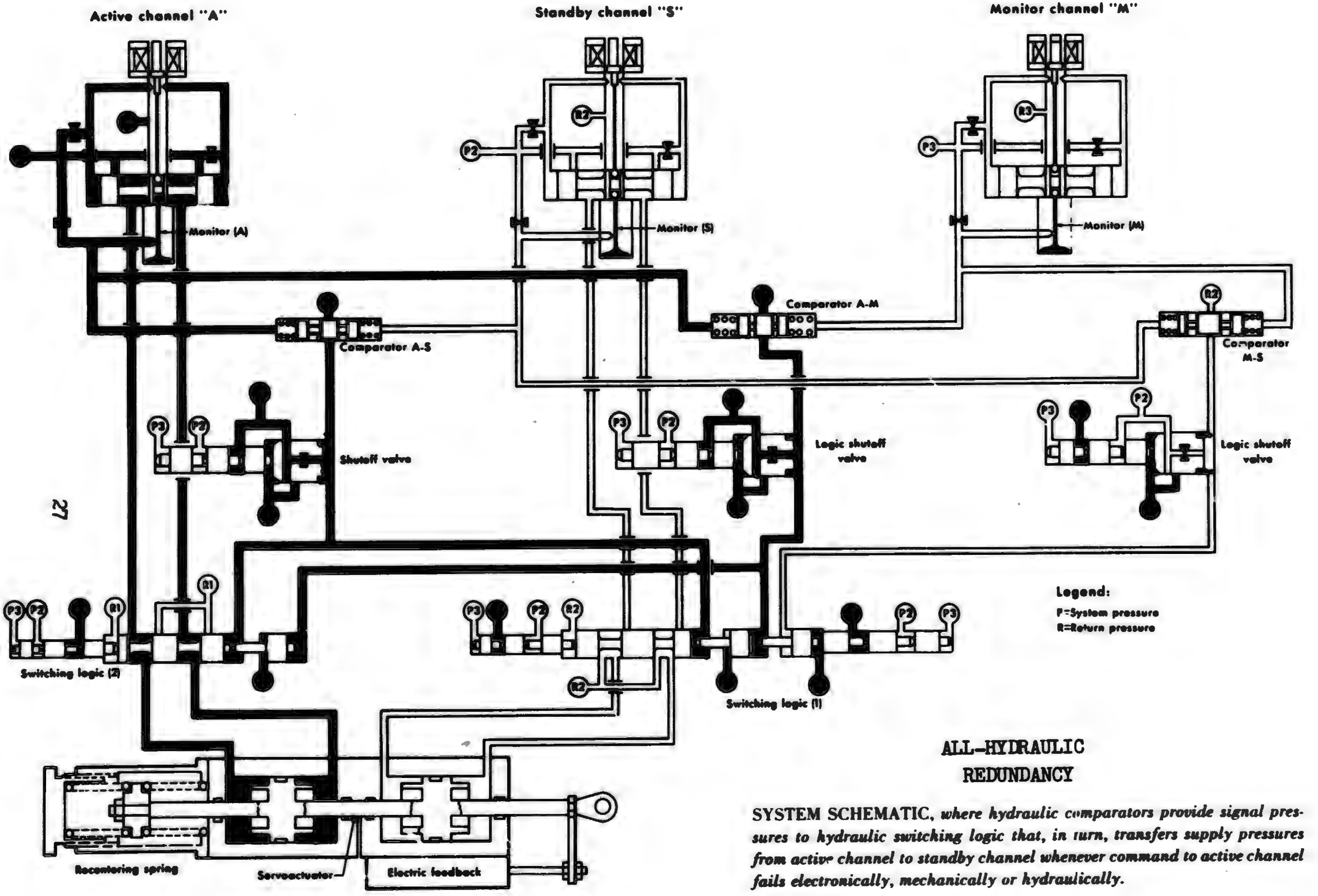
FIGURE 13 FAIL PASSIVE REDUNDANCY

channels can operate unimpeded. The single-stage jet-pipe valve not only has the proper failure characteristics, but also acts like a very open-centered valve so that fluid can be forced back through it with relative ease thus preventing hydraulic lock. The servo error signal is formed in the position feedback transducer, rather than in an amplifier as is normally done, such that a transducer failure blocks the command signal. This feature prevents the open loop condition that normally results from a loss of feedback. The electronics fail passively because a.c. signals are used. A hardover electronic failure causes a d.c. output to which the a.c. circuits are not sensitive.

If a hardover input should occur in a channel or as an input, the other channels collectively offset the output force of the failed channel at the force-summing actuators. The high loop gains reduce the resulting position offset to an insignificant level. Therefore, a triplex servo will operate after two failures. Further, for passive failures, no monitoring, switching, or engage valves are required. Monitoring is performed, however, primarily for failure reporting, and switching can be done where necessary such as for centering and locking.

Single-Fail/Operate Redundant Servo Valve Using Hydraulic Logic and Switching

Hydraulic Research and Manufacturing Company has designed and built a single-fail/operate redundant actuator using hydraulic logic and switching which will be installed and flight tested in an F-4C aircraft at Edwards Air Force Base this year . The actuator is an integral part of an Air Force Flight Dynamics Laboratory program in which the most promising advanced control equipment and technology are being combined to achieve improved tactical weapon delivery performance. A similar two-fail/operate redundant actuator has been designed and is presently being built by HRM for flight tests during Phase III of our in-house fly-by-wire program to be explained later in the paper. A description of the operation of the single-fail/operate hydraulic logic servo actuator shown in Figure 14 follows. Because of the need for fast failure detection and correction, the redundant hydraulic system uses continuous monitoring and automatic transfer to a standby system. The concept is designed to meet the needs of a supersonic transport or other high-performance craft, but the principles of the all-hydraulic redundant controls are applicable to other servo systems where nearly instantaneous command transfer to a standby channel without interrupted performance is needed. Failure detection and correction are achieved in 10 msec or less, bettering electronic systems.



ALL-HYDRAULIC
REDUNDANCY

SYSTEM SCHEMATIC, *where hydraulic comparators provide signal pressures to hydraulic switching logic that, in turn, transfers supply pressures from active channel to standby channel whenever command to active channel fails electronically, mechanically or hydraulically.*

FIGURE 14

To grasp the idea, consider the single-fail/operate case shown in Figure 14 using an active channel, a standby channel, and a monitor channel. For detection, each channel condition must be monitored. For single-fail/operate redundancy, at least two command channels are needed, an active channel and a standby channel. Both the active and standby channel are coupled by a switching logic valve to the tandem actuator.

Each channel, active, standby, and monitor, consists of a conventional flapper-nozzle, two-stage servo valve. The second-stage power spools drive the tandem actuator. Hydraulic position monitors coupled to the servo valve power spools transmit a signal which is proportional to power-spool displacement to hydraulic comparators. Each of three comparators, placed between the three channels, receives two signals and detects any phase and/or amplitude difference between them. As long as the signals agree in phase and amplitude, the comparator output remains constant. A disagreement between signals causes the output of the comparator to change. This output change is used to trigger shutoff valves which control the position of the switching logic valves. Once the shutoff valve has been triggered, it locks, preventing the system from cycling between a failed channel and an operational channel should the failed channel momentarily correct itself. For any first failure, the system either transfers to the standby channel or in the situation where the first failure is the monitor or the standby channel, activates itself so that the standby channel cannot be engaged upon the occurrence of a second failure. For a second failure, both halves of the actuator are bypassed and the actuator is spring (or hydraulically) centered. Since monitoring is carried out at the servo valve spool, just before the actuator, hardover failures of the servo valve, input, or servo amplifier are detected without requiring a deviation of the actuator from commanded position.

Current AFFDL Fly-by-Wire Programs

The fly-by-wire effort of the Control Elements Branch, Flight Control Division, Air Force Flight Dynamics Laboratory is divided into two parts: an in-house effort being conducted in the Control Techniques Laboratory in Building 195 with the assistance of Hydraulic Research and Manufacturing Company personnel working under contract; and two contracted efforts (facetiously referred to as our out-house efforts) with Douglas-Long Beach and Sperry Phoenix Company.

In-House Programs

The in-house program consists of the design, manufacture, assembly, and flight test of a single axis (pitch) fly-by-wire flight control system for a B-47 aircraft. The specific objectives of this program are: (1) to demonstrate that a fly-by-wire system capable of controlling a B-47 or similar type aircraft can be built within the present state-of-the-art; (2) to demonstrate that building such a fly-by-wire system is not only feasible but also practical; (3) to establish an assurance level or level of confidence in fly-by-wire control systems among military operators and among the aircraft manufacturers and designers; (4) to demonstrate that a fly-by-wire system overcomes control problems that are inherent in mechanical control systems such as stiction, friction, and hysteresis; (5) to demonstrate that fly-by-wire systems will minimize the adverse effects of structural changes due to bending, thermal expansion, etc., on the performance of aircraft or aerospace vehicles; (6) to demonstrate that a fly-by-wire control system can effect a significant weight and volume savings over the conventional mechanical flight control system; (7) to fly a pure fly-by-wire system in military aircraft and thereby demonstrate to industry a strong military interest in fly-by-wire control systems for use on future military aircraft and aerospace vehicles; (8) to provide a flying test bed for future tests of contractor owned fly-by-wire control systems; and (9) to raise the level of fly-by-wire technology in the U. S. A. This program will be accomplished in the following four phases:

Phase I - The existing B-47 control stick and feel system will be used in conjunction with a simple nonredundant fly-by-wire system. Linear displacement transducers (LVDT's) connected to the pilot's control stick will operate a servo actuator (modified B-47 actuator) in parallel with the existing aircraft pitch actuator. During tests of the fly-by-wire system, the normal aircraft pitch actuator will be bypassed.

Phase II - A side-stick controller will be installed in the pilot's cockpit and a C* feedback system will be installed to provide the necessary feel/response. The same nonredundant servo actuator as was used in Phase I will be used here.

Phase III - The side-stick controller and C* feedback system from Phase II will be used but with a quadruply redundant actuator installed in place of the nonredundant servo actuator from Phase I. This actuator will use hydraulic logic and will be powered by four 3,000 psi hydraulic power supplies installed in the tail section of the aircraft.

Phase IV - The fly-by-wire system from Phase II will be used in conjunction with a liquid metal actuator package which will be installed in lieu of the quadruply redundant actuator from Phase III. These flight tests will be done as part of the liquid metal actuation program being conducted by the Air Force Flight Dynamics Laboratory. Phase I of these flight tests is scheduled to begin in the late fall of 1967 with Phase III flight trials terminating in December 1968. Phase IV flight trials of the liquid metal actuator package will be done during the spring and summer of 1969.

The Douglas Effort

This consists of the completion of an extended program in which much effort was expended in an attempt to design and build a pure electrical (nonelectronic) a.c. primary flight control system. This task, as originally directed by AFFDL, proved impractical and the program has recently been redirected to permit the use of electronics and d.c. in an effort to obtain more positive results from the remaining resources. A breadboard model of a redundant fly-by-wire system will be designed and built.

The Sperry Phoenix Effort

This consists of: (1) a fly-by-wire study and research contract (just completed), the results of which are included in Technical Report AFFDL-TR-67-53, "Fly-by-Wire Techniques", prepared by Mr. F. L. Miller and Mr. J. E. Emfinger of the Sperry Phoenix Company under the direction of Mr. V. R. Schmitt and F/L J. P. Sutherland, Project Engineers, FDCL, AFFDL: and (2) a recently-started contract to design and build a three-axes, quadruply redundant experimental laboratory model of a fly-by-wire system for a B-47 aircraft.

Proposed Future Efforts

AFFDL's future programs in fly-by-wire are anticipated to include a fly-by-wire flight test definition and trade-off study to determine the most economical and fruitful approach to demonstrating fly-by-wire technology through flight test. Dependent on the results of this study, a program is proposed for the flight testing of a complete three-axes, two-fail/operate (quadruply redundant) fly-by-wire system. This program would most likely be a joint aircraft prime contractor/flight control contractor effort where the flight control contractor would be responsible for the design and manufacture of the fly-by-wire system and the aircraft prime contractor would be responsible for the interfacing and flight testing of the fly-by-wire system.

Summary and Conclusion

Simple direct mechanical linkages, cables, and feel springs for manual control can no longer cope with many of the control system problems associated with modern high performance aircraft and aerospace vehicles. In an effort to meet the greater demands of these advanced aircraft control system requirements, the flight control designer has been forced to increase the complexity of the mechanical system with a resulting increase in weight, volume, and cost, and a decrease in flexibility and reliability. Invariably he is forced to compromise between the desired performance and design requirements and a practical mechanization. Fly-by-wire offers not only to meet the demands of these advanced control system design requirements, but also promises to do so with a decrease in complexity, weight, volume, and cost and an increase in flexibility and reliability. Why then is fly-by-wire not in common use today? The answer to this question was given in the first part of this paper; i.e., a lack of confidence in th concept of fly-by-wire and a feeling of false security in mechanical flight control systems. These are the principal factors which are now retarding the growth and general acceptance of fly-by-wire. The Air Force Flight Dynamics Laboratory fly-by-wire programs are aimed at establishing the assurance level or level of confidence in fly-by-wire control systems among military operators and aircraft manufacturers and designers which is necessary to overcome this stigma. We recognize the inevitable existance of many engineering problems which must be solved in going from a drawing board design to flight worthy hardware. Although our programs will not necessarily establish the best solutions to these problems, they should demonstrate conclusively that the solutions are feasible and practicable. We firmly believe that fly-by-wire is not only inevitable for use in advanced military and commercial aircraft and aerospace vehicles but is, in fact, on the immediate horizon. If this paper has helped to convince you of this fact, or even encouraged you to re-evaluate your previously held opinions, then it has served its purpose well.

Verily I say unto you ,

as it has been said of old,

The era of Fly-By-Wire has come!

Paul 8.10

SECURITY IS A FLY-BY-WIRE FLIGHT CONTROL SYSTEM
HUMMM
BUZZZZ
HUMM

FIGURE 15

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