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EDWARDS AIR FORCE BASE, CALIF. 93523

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HONEYWELL AFCS FOR THE NO. 3 X-15 VEHICLE

. . . development
. . . flight testing

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February 1963

HONEYWELL AFCS FOR THE NO. 3 X-15 VEHICLE

By

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FOREWORD

This document presents a two-part history of Honeywell's development and flight testing program directed toward adaptive controls for the X-15.

Part I gives a brief history of the design and development of Honeywell's prototype X-15 adaptive flight control system in an F-101A. Adaptive principles of the X-15 system are also described.

Part II summarizes and describes the flights that have already been made by the No. 3 X-15 vehicle containing the MH-96 Adaptive Flight Control System.

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PART I

DEVELOPMENT AND FLIGHT TEST OF ADAPTIVE CONTROLS FOR THE X-15 IN AN F-101A

BY

A. J. BAILEY, JR.
Chief Engineering Pilot

DEVELOPMENT AND FLIGHT TEST OF ADAPTIVE CONTROLS FOR THE X-15

The X-15 with its extensive flight envelope and the accompanying large changes in performance characteristics presents a broad challange to the flight control system designer.

The conventional design approach starts with a detailed analysis of the theoretical aircraft characteristics to pre-determine a set of optimum control parameters for each flight condition. These pre-determined parameters are then scheduled into the control system by means of a complex air data sensing device which identifies the flight condition. However, predicted flight characteristics of a vehicle under development are not accurate enough for pinpoint determination of control system parameters. Even with extensive flight testing, certain compromises will exist in the final design. From a time and cost standpoint, it is no longer practical to optimize system gain scheduling by flying to all points of the flight envelope and sampling performance. This is true for the supersonic transport as well as the X-15 research vehicle.

Six years ago, ASD's Flight Control Laboratory awarded Honeywell a study contract to investigate advanced control techniques.

This study program was directed toward developing a system which could provide satisfactory control through extreme changes in flight environment without the need for:

- Accurate knowledge of the vehicle's characteristics
- Air data scheduling
- Extensive flight testing

Early in 1958, Honeywell's program produced encouraging results, and a decision was made to mechanize the fundamental idea and test it in an aircraft.

A. J. Bailey, Jr., Aero's chief engineering pilot, was designated to coordinate system design with pilot capability and characteristics in a way that would promote maximum efficiency in mission accomplishment. To assist in this pilot input to system design and to preclude the dangers of a one-pilot technique, a select group of pilots was scheduled at various development stages to evaluate and comment on the breadboard X-15 system in the F-101A test vehicle. These pilots included Joe Walker, Major Bob White, Major Bob Rushworth and Neil Armstrong. Pilot comments have played an important part in Honeywell's final control system configuration now being installed in the X-15.

Honeywell believes that future successes in space will be with manned craft as well as with missiles, and that man's selective judgement coupled with proper control techniques will prove to be a significant factor in accomplishing this success.

There are two major points covered in this paper:

1. Proving the feasibility of the adaptive concept.
2. Development of the X-15 adaptive system.

PROVING THE FEASIBILITY OF THE ADAPTIVE CONCEPT

Early in 1958, Honeywell installed the first adaptive hardware in a fully instrumented F-94C jet aircraft for feasibility testing. This was the culmination of Honeywell's research study effort to develop a stability augmentation system that would not be dependent on complex scheduling of system parameter gains. This project was administered under the direction of the Flight Control Laboratory, Wright-Patterson AFB.

The flight test program in the F-94C proved the feasibility of Honeywell's adaptive concept and corroborated results of earlier studies. When the Minneapolis phase of the flight test study was completed, the aircraft and equipment were sent to Wright-Patterson AFB for continued development testing and final evaluation. Captain Gus Grissom, USAF, was assigned as project pilot to conduct this flight test evaluation.

Based on the success of the F-94C adaptive feasibility study, Honeywell decided to fabricate and flight test a complete three-axis adaptive AFCS in a supersonic aircraft. Since there was little military funding available, company funds were obtained to support the design and flight test of the MH-90, a three-axis adaptive system. The Flight Control Laboratory made available a JF-101A (S/N 53-2422) at Minneapolis for a test vehicle. MH-90 program goals were:

- To demonstrate the three axis AFCS adaptive concept in a supersonic aircraft
- To develop satisfactory outer loop control (attitude hold, heading hold, altitude control, and control stick steering) without use of air data scheduling
- To match the adaptive system to optimum pilot control technique

These objectives were successfully met, and the adaptive concept was proven The program required approximately 12 months to complete and included 48 developmental flights. Major J. J. Knight, USAF, and Captain H. K. Wimberly, USAF, evaluated the system at the conclusion of the program.

DEVELOPMENT OF THE X-15 ADAPTIVE SYSTEM

In July 1959, Honeywell was awarded a contract for the development of an adaptive system to be flight tested in the X-15 space flight research vehicle by the Flight Control Laboratory, ASD, (AF33(616)-6610). The system proposed by Honeywell for the X-15 used the concept and many of the components developed for the MH-90 program.

The JF-101 test bed with the first complete adaptive hardware created considerable interest on the part of AF and NASA pilots and engineers. As a result, a series of flight demonstrations of the MH-90 were performed in Minneapolis by the following evaluation pilots:

Major W. F. Daniel, USAF AFFTC Edwards AFB
Major R. M. White, USAF X-15 Project Pilot
Capt. R. A. Rushworth, USAF X-15 Pilot
Capt. S. H. Nelson, USAF AFFTC Edwards AFB
Capt. N. C. Evans, USAF AFFTC Edwards AFB
Joe Walker, NASA X-15 Project Pilot
Neil Armstrong, NASA X-15 Pilot
LCDR Tex Birdwell, USN, NATC, Patuxent River
LCDR Taylor Brown, USN, NATC, Patuxent River

Comments of these pilots were studied and evaluated, and when appropriate, changes were made in the MH-90 system.

To more effectively evaluate the adaptive mechanization planned for the X-15, Honeywell modified the MH-90 to the electrical equivalent of the aerodynamic portion of the X-15 adaptive system. This modification made the system the electrical equivalent of the X-15 adaptive, although it did not include redundant networks with the associated comparators and monitors, reaction control circuitry, fixed gain fail safe circuits, or the facility for automatic pre-flight checkout.

No mechanical changes in the aircraft were necessary. This JF-101A control system includes an autopilot series servo integrated with the aileron and rudder power actuators. The stabilator is controlled by a parallel servo for gross motions or commands, and a series servo for fast damping motions. Therefore, only long term pitch commands are reflected back to the center stick.

Control is accomplished in the X-15 by utilizing right and left horizontal stabilizer series servos for control of the horizontal stabilizer power actuators and a yaw series servo for control of the upper and lower vertical stabilizer power actuator. These differences in control system mechanization did not affect the autopilot tests; results in aircraft response were the same.

The JF-101 test bed (Figure 1) was equipped with special flight test provisions to provide an efficient, easily maintained system for autopilot development testing. These provisions included a special gain adjust panel available to the pilot (Figure 2), an aircraft cabling junction box for making cabling modifications (Figure 3), and a test panel for checking all autopilot signals. To make the most effective use of flight time, an analog simulation of the aircraft was used for ground checking. This closed-loop test procedure provided an accurate measure of system operation prior to actual test flights.

ADAPTIVE PRINCIPLES (X-15 SYSTEM)

Consider the single axis control system shown in Figure 4. The MODEL defines the response of the system and is composed of a simple electrical network which shapes the command signal to represent the desired response of the aircraft.

The ADAPTIVE CONTROLLER slaves the aircraft to the output of the model. The ability of the controller to follow the model requires that the system gain, rate of control surface motion, be continuously held at or very near the maximum gain that the system will tolerate without being driven to instability. As the flight condition changes, the basic aircraft gain also changes. If the adaptive controller is to command the aircraft as prescribed by the model, it must vary its gain to maintain the loop gain either at or near the critical level. When the system is at the critical gain level, a small amplitude oscillation called a "limit cycle" will occur. The frequency of this "limit cycle" is determined by the compensation network in the adaptive controller.

Figure 1. F-101A Adaptive Test Vehicle

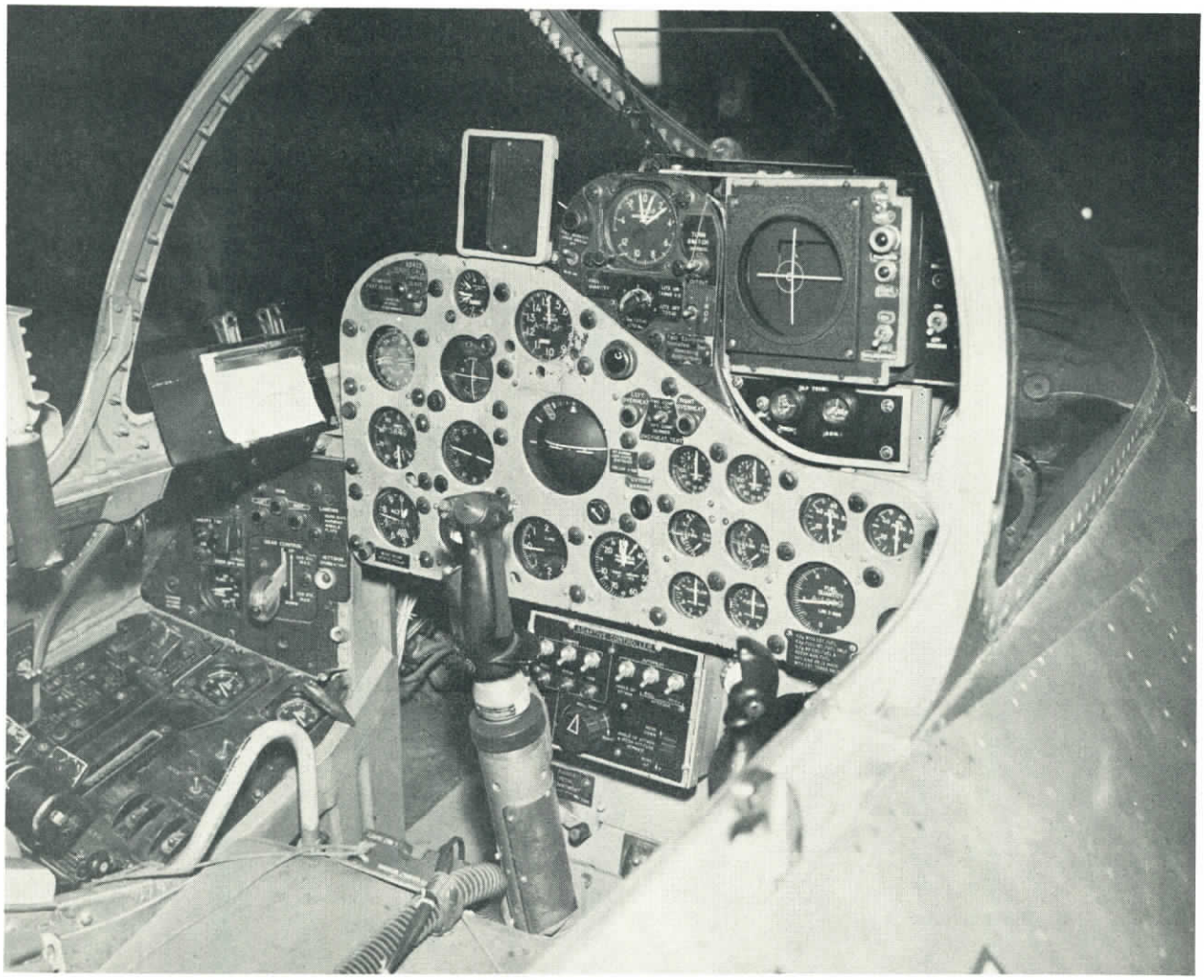


Figure 2. F-101A Cockpit

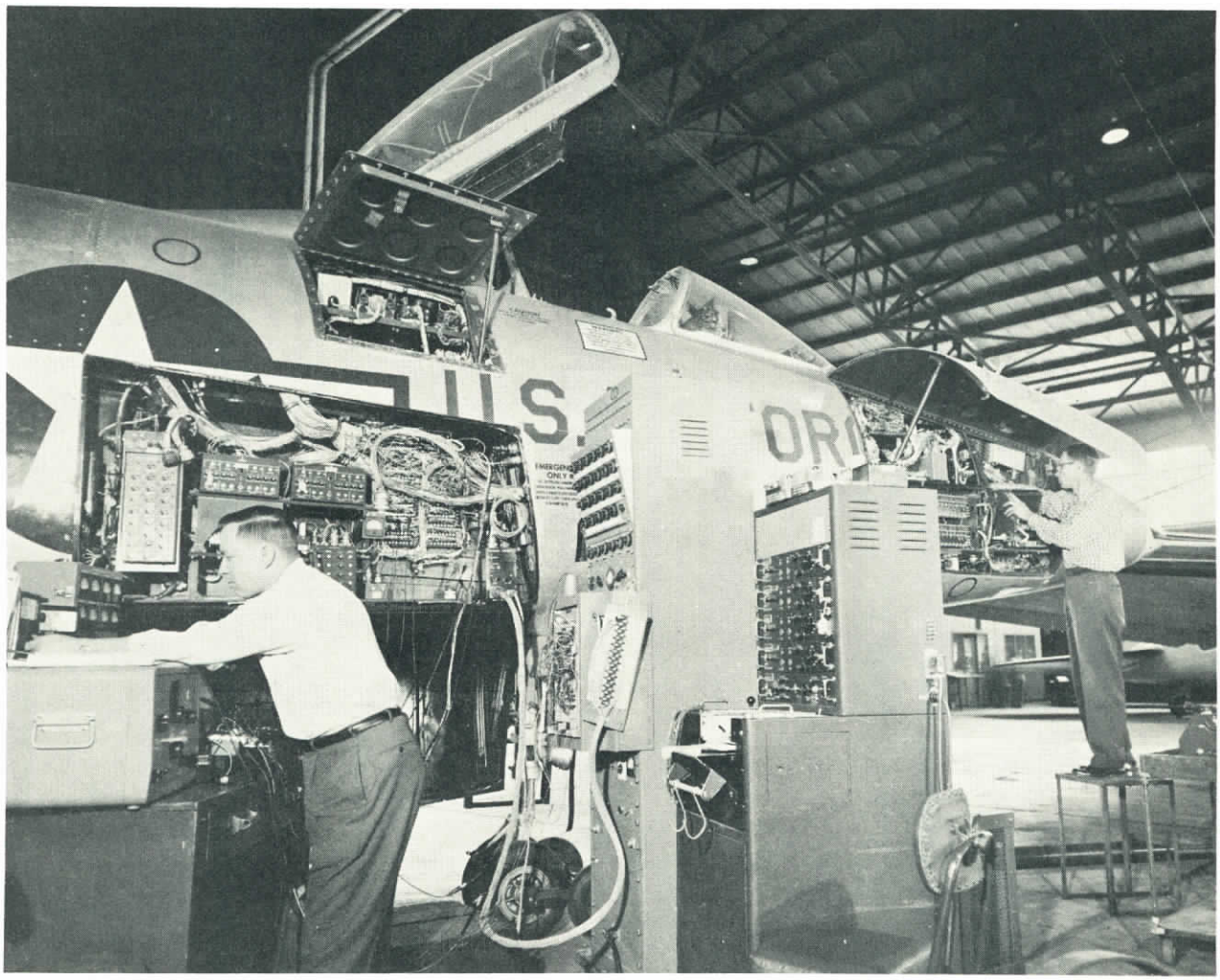


Figure 3. Analog Computer Next to F-101A

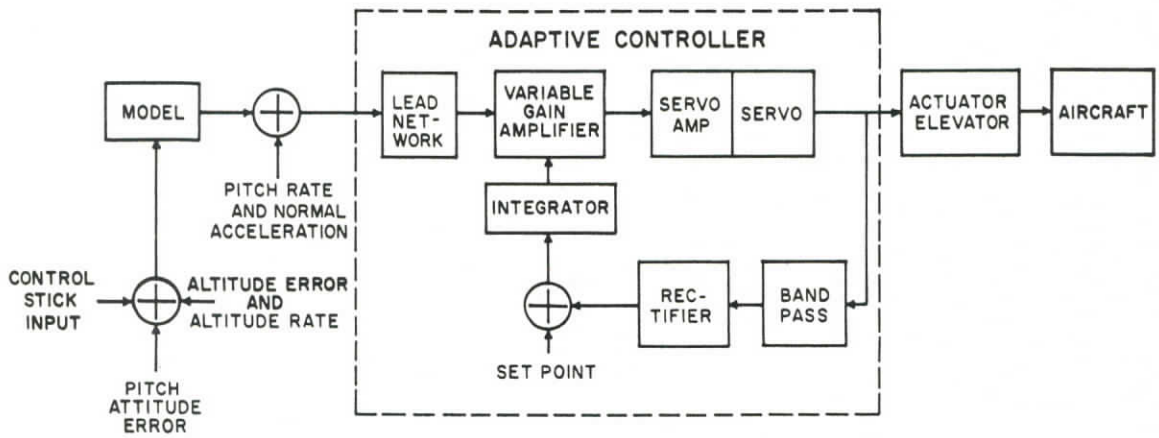


Figure 4. Adaptive Pitch Axis - Simplified Block Diagram

Gain changing is accomplished in the following manner: The residual motion is "sensed" by a limit-cycle amplitude detector, comprised of a bandpass network followed by a recitifer. The output of the rectifier is compared to a "set point" which establishes the level of the limit cycle to be sustained. This level is kept comparable in magnitude to the always present control system dead spot.

Since a constant amplitude limit cycle identifies critical gain, any tendency of the system to operate at any other gain level is reflected by either too small or too large an amplitude of limit cycle. The system will immediately initiate corrective action to re-establish and sustain critical gain. The limit cycle frequency in this system is approximately four cps, and the set point amplitude is about 0.1 degree (double amplitude) of elevator position. This is not discernable to the pilot because it occurs at random intervals and seldom exceeds more than two full cycles at four cps before the gain changer reduces the gain. Simulator studies and actual flight tests show that this adaptive feature causes the gain control to vary as an inverse function of the control surface effectiveness.

Pitch Axis

The pitch control operates in any of four modes:

1. Damper with or without acceleration feedback
2. Pitch attitude-hold
3. Angle of attack hold
4. Altitude-hold

When operating on damper with acceleration feedback, the pilot commands a combination of pitch rate and normal acceleration, the sum of which is proportional to the force applied to the control stick. Combining normal acceleration and pitch rate feedbacks in the inner loop provides excellent pilot handling qualities in aircraft response. At the high dynamic pressure flight condition, the amount of normal acceleration relative to pitch rate

experienced during maneuvering flight is large. Consequently, the pilot is primarily commanding normal acceleration, and the resulting acceleration dynamic response is essentially identical to the model output.

At the low dynamic pressure flight conditions, the amount of pitch rate becomes large relative to the normal acceleration that the aircraft experiences. The pilot now primarily commands pitch rate, and the resulting pitch-rate dynamic response is essentially identical to the model output.

When operating on damper with the normal acceleration switched out, the inner loop operates on pitch rate only.

When on attitude hold or angle of attack hold, the inner loop is controlling pitch rate. This permits the use of a fixed pitch attitude or angle of attack feedback gain, since the relationship between pitch rate and pitch attitude or angle of attack is independent of flight condition.

The altitude control, although not a part of the final X-15 system, is discussed because it was mechanized and proven successful in the F-101A developmental system.

When the altitude hold mode is engaged, normal acceleration is the inner loop feedback configuration. The normal acceleration response of the basic inner-loop configuration is sufficiently invarient with flight condition to permit the use of fixed outer-loop altitude displacement and rate gains. Altitude displacement and rate signals are supplied by an altitude "blender" which has inputs of static pressure, rate of static pressure change, and normal acceleration. At low rates of change, altitude information is supplied by the conventional pressure sensors. At high rates, the altitude information is supplied by integration of normal acceleration. This scheme eliminates the undesirable effects of dynamic lags and transients associated with pressure sensing and transmission, and reduces the noise level of the altitude rate signal.

Figure 5 shows two actual flight recordings of altitude control performance involving engagements in ascents and descents (Note the smoothness of the recoveries). Altitude control performance has been outstanding through the entire program. This is attributed to the high inner loop gain provided by the adaptive system and addition of the acceleration signal to the "blender" Altitude retention during accelerations and decelerations at 10,000 feet from 200 to 550 KIAS caused deviations of only ±40 feet.

Roll Axis

The roll axis of the MH-90 operates on any one of three modes:

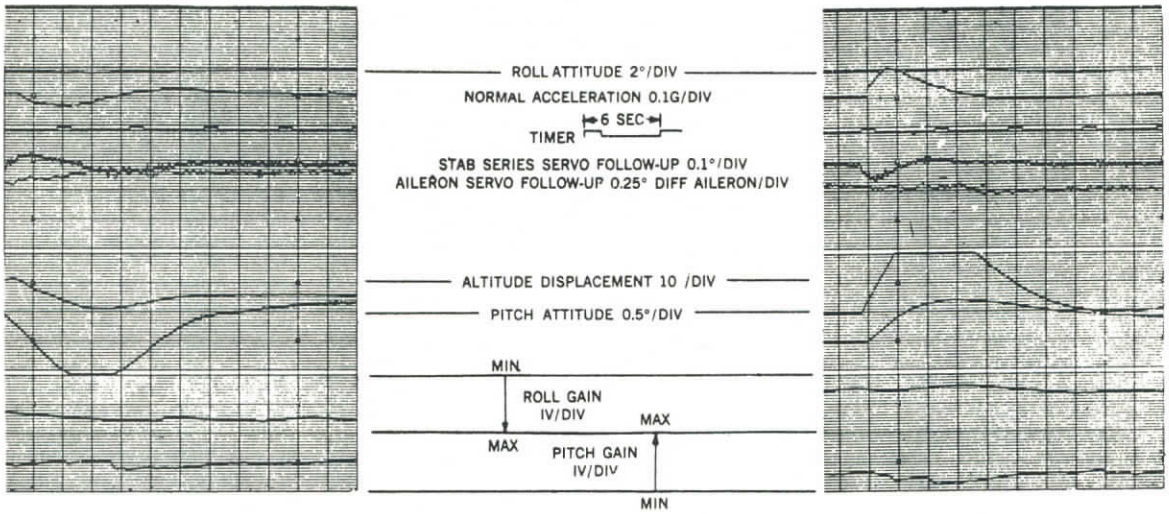
1. Damper
2. Roll-attitude-hold
3. Heading-hold

The adaptive gain controller is identical to that used in the pitch axis. When operating in the damper mode, the pilot commands roll rate which is proportional to the force applied to the control stick.

The use of a roll rate inner loop having a response that is uniform for all flight conditions enables the simple application of a fixed-roll-attitude-hold mode. For this mode, a fixed-gain, attitude error signal is applied to the model input. Since the roll-rate response is uniform for all conditions, the roll attitude response also is uniform.

The heading-hold mode permits the pilot to "lock-on" to any desired compass heading, after which the system will hold the selected heading. The existence of a heading error will command a proportional bank angle until corrected.

Figure 6 shows actual flight recordings of a series of roll attitude steps made of various flight conditions (Note the uniform aircraft response at all flight conditions).



Engage Altitude Control at 3000 Feet Per Minute Up — 35,000 Feet, Mach 0.9

Engage Altitude Control at 8000 Feet Per Minute Down 20,000 Feet, Mach 0.9

Figure 5. Altitude Control Flight Records

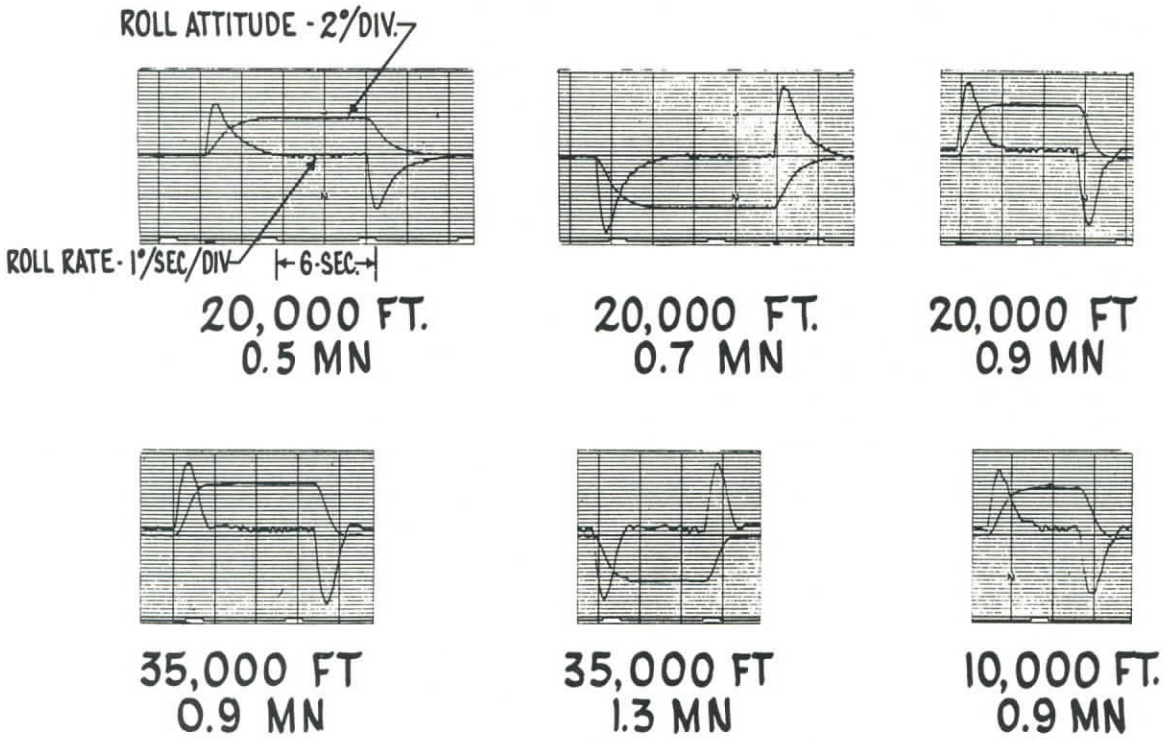


Figure 6. Roll Attitude Steps

Yaw Axis

The yaw axis has only one operating mode, using a combination of lateral acceleration and high passed yaw rate to provide turn coordination and yaw damping. The adaptive gain controller is the same as in the roll and pitch axis.

SUMMARY

Flight test programs conducted in the F-101 on the X-15 adaptive system were successfully completed:

- The adaptive concept feasibility was proven in a three-axis AFCS in a supersonic aircraft. Adequate flight data was taken showing invariant response of the system in all three axes, at all flight conditions.
- Excellent outer-loop control was achieved without use of air data scheduling. Altitude control, according to evaluation pilots, is the finest ever tested.
- Pilot acceptability is attested to by 78 evaluation flights performed by 18 pilots representing NASA, Navy, AF, The Boeing Company and McDonnell Aircraft.

The developmental breadboard adaptive system in the F-101 has totaled up the following operating record:

- 1300 hours of system ground time
- 300 hours of system flight time
- 213 total flights*
- 78 evaluation and demonstration flights
- 135 development and research flights

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*Number of flights and system operating hours includes those made on electric side stick development program.

Reliability was excellent during this operating period. No malfunctions in the adaptive feature occured which were not caused by human error. The adaptive mechanization has not shown any unusual characteristics detrimental to the aircraft control system.

The adaptive system developed and evaluated in the F-101 was limited to the electrical equivalent of the X-15 adaptive. It was not practical to test or evaluate features such as reaction controls circuitry, fixed gain, etc. However, the entire X-15 adaptive system was extensively evaluated on the North American Aviation X-15 simulator. These are briefly discussed in the following paragraphs:

Fixed Gain -- Dual fixed-gain channels parallel the dual adaptive channels in each axis. These provide an emergency, or backup system which will permit control of the aircraft to be maintained if the adaptive channels fail. The fixed gains are set at approximately the lowest critical gains in the flight envelope. For increased reliability, the fixed-gain channels are operative at all times, eliminating switching if a failure occurs.

Control in the aerodynamic and reaction control regions is provided by the fixed gain system.

Reaction Controls -- Automatic transition from aerodynamic to reaction controls allows the pilot to maneuver the aircraft with a single control stick throughout the entire flight envelope. As the altitude increases, the system gains increase to maintain the limit cycle. When the gains reach about 80 per cent of their maximum values (at about 100,000 feet), the reaction controls are energized. Due to a large deadspot, the reaction controls will not necessarily be used immediately after being energized. Instead, the aerodynamic surfaces will continue to provide a large part of the control with the reaction controls operating only when the system errors exceed the deadspot values.

As dynamic pressure increases during re-entry, the limit cycle will be resumed at about 100,000 feet and the reaction controls will be de-energized when the gain decreases to 60 per cent of maximum. Therefore, the system uses a minimum of reaction control fuel since maximum use is made of the aerodynamic controls.

Redundancy and Failsafety -- Each system axis is mechanized in a parallel redundant manner to provide fail safety. The redundant mechanization includes several comparators and limitors which allow the system to remain operative on the adaptive loop for 90 - 95 per cent of all failures. A failure in one axis will not disengage any other axis. Furthermore, a hardover in the outer-loops will revert the system to the damper mode; a hardover in the adaptive loop will, at worst, only revert that axis to the back up fixed gain circuitry.

Rate dampers in each axis are dual channel, incorporating dual rate gyros and servo pickoffs in addition to dual electronic channels. The stick and rudder pedal position transmitters are also dual. Dual fixed-gain channels parallel to the adaptive channels provide adequate damping to maintain control of the aircraft if the adaptive channels are switched out.

Dual lateral accelerometers are utilized since lateral acceleration is a part of the yaw axis damper system. Dual normal accelerometers are used in the pitch axis to provide protection on autopilot modes. The autopilot inputs themselves (roll and pitch attitude, angle of attack, and heading) are not dual, but are summed into both channels of their respective axis. Servo monitors will disengage the servos if they fail.

Figure 7 shows location of the various flight control devices in the X-15. Figure 8 shows cockpit location of the pilot function select panels. Figure 9 shows the pilot's center control panel:

- DAMPER switches -- Pitch, roll, and yaw dampers are engaged by moving the appropriate spring-loaded switch to the DAMPER position. Indicator lights go out when the dampers are engaged.

1. Q-Ball*
2. Pitch and Yaw Reaction Valves
3. Pitch Trim Actuator* and Pitch Stick Transmitter
4. Right Side Stick CSS and Trim Controls*
5. Center Stick CSS and Trim Controls*
6. Center Pedestal Function Selector
7. Roll Stick Transmitter
8. Left Console Function Selector
9. Rudder Pedal Transmitter
10. Stable Platform*
11. Adaptive Controller and Accelerometers
12. Roll Reaction Valves
13. Rate Gyros
14. Pitch-Roll Servocylinders*
15. Yaw Servocylinder*

*Components Not Supplied by Honeywell

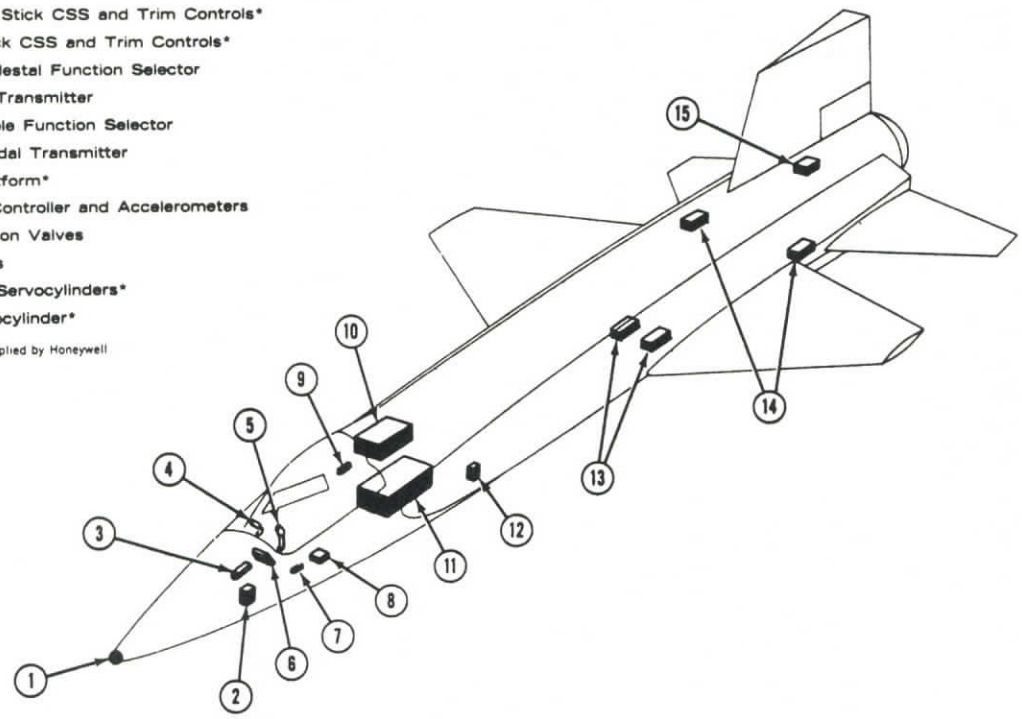

Figure 7. Location of X-15 Adaptive Flight Control System Devices

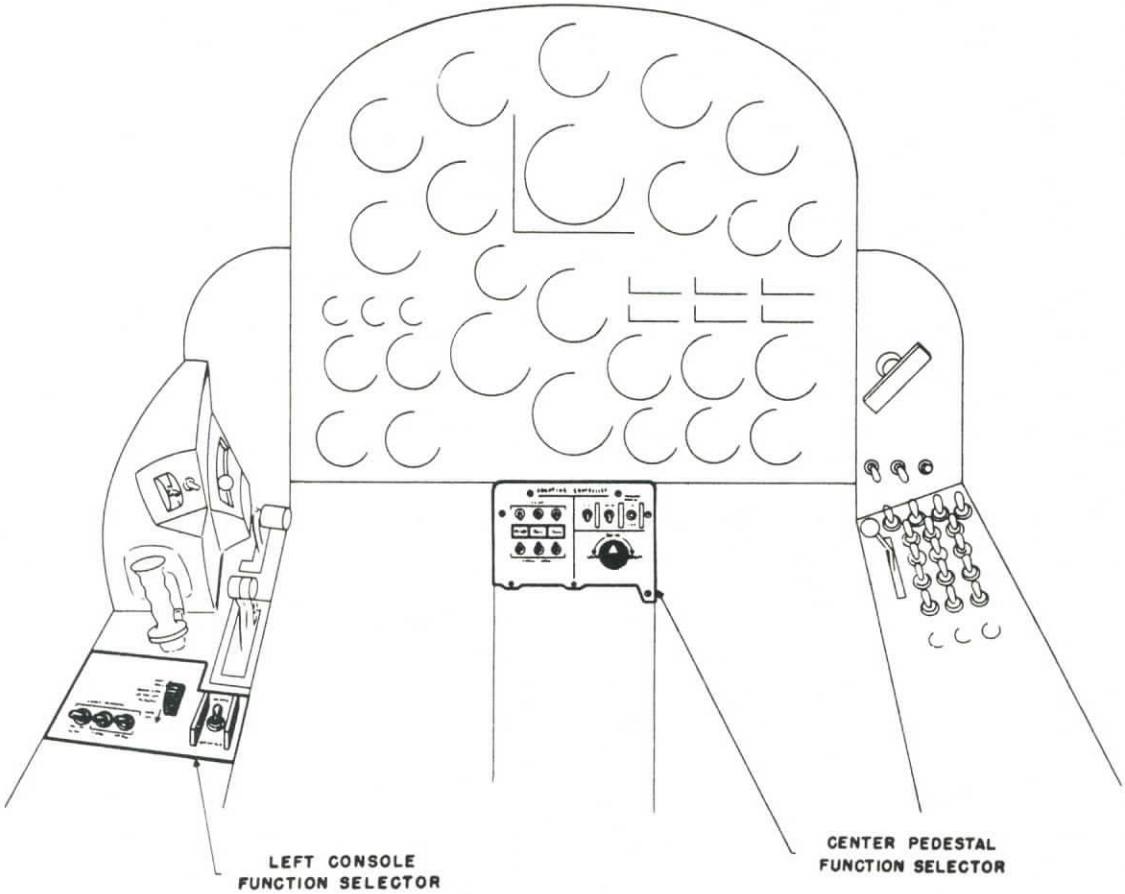

Figure 8. Function Selectors in Cockpit Layout

Figure 9. Function Selector Center Console

- FIXED GAIN Switches -- Fixed gain operation may be selected with the momentary-contact, spring return fixed gain switches. The indicator light will come on and the damper switch remains in the damper position indicating the system is on fixed gain.
- CONTROL STICK STEERING (CSS) Switch -- This switch controls whether the normal acceleration feedback is switched in or out. CSS switch on provides acceleration feedback.
- SPARE Switch -- No function at present.
- REACTION CONTROLS Switch -- Provides pilot option for using automatic activation of reaction controls, reaction controls always activated and controls off.
- ROLL TRIM Knob -- Pilot adjust of small electrical or mechanical mistrims in roll.

Figure 10 shows pilot's left console control panel:

- ROLL-HEAD Switch -- This switch may be engaged whenever both yaw and roll dampers are engaged. If this switch is engaged when bank angle is less than 7.5 degrees, the aircraft will roll level and lock on to heading. If bank angle is greater than 7.5 degrees, the roll attitude will be retained. The CSS switch on the stick will drop out the hold reference as long as depressed.
- PITCH AND ANGLE OF ATTACK HOLD REFERENCE Switches -- When the pitch damper is engaged, either angle of attack or pitch attitude hold may be selected. Momentarily depressing the CSS button on the control stick drops the reference -- release of button locks on the new reference.
- VERNIER WHEEL -- This wheel provides precise control or adjustment of pitch attitude or angle of attack.

Figure 11 shows the complete automatic flight control system. The box marked Airborne Analyzer is mounted in the B-52 and is used for pre-flight checkout of the flight control system.

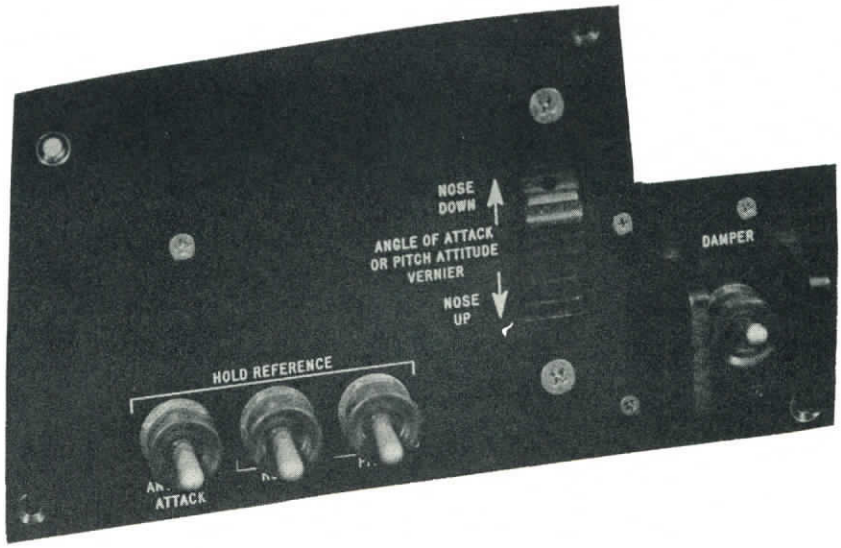

Figure 10. Function Selector Left Console

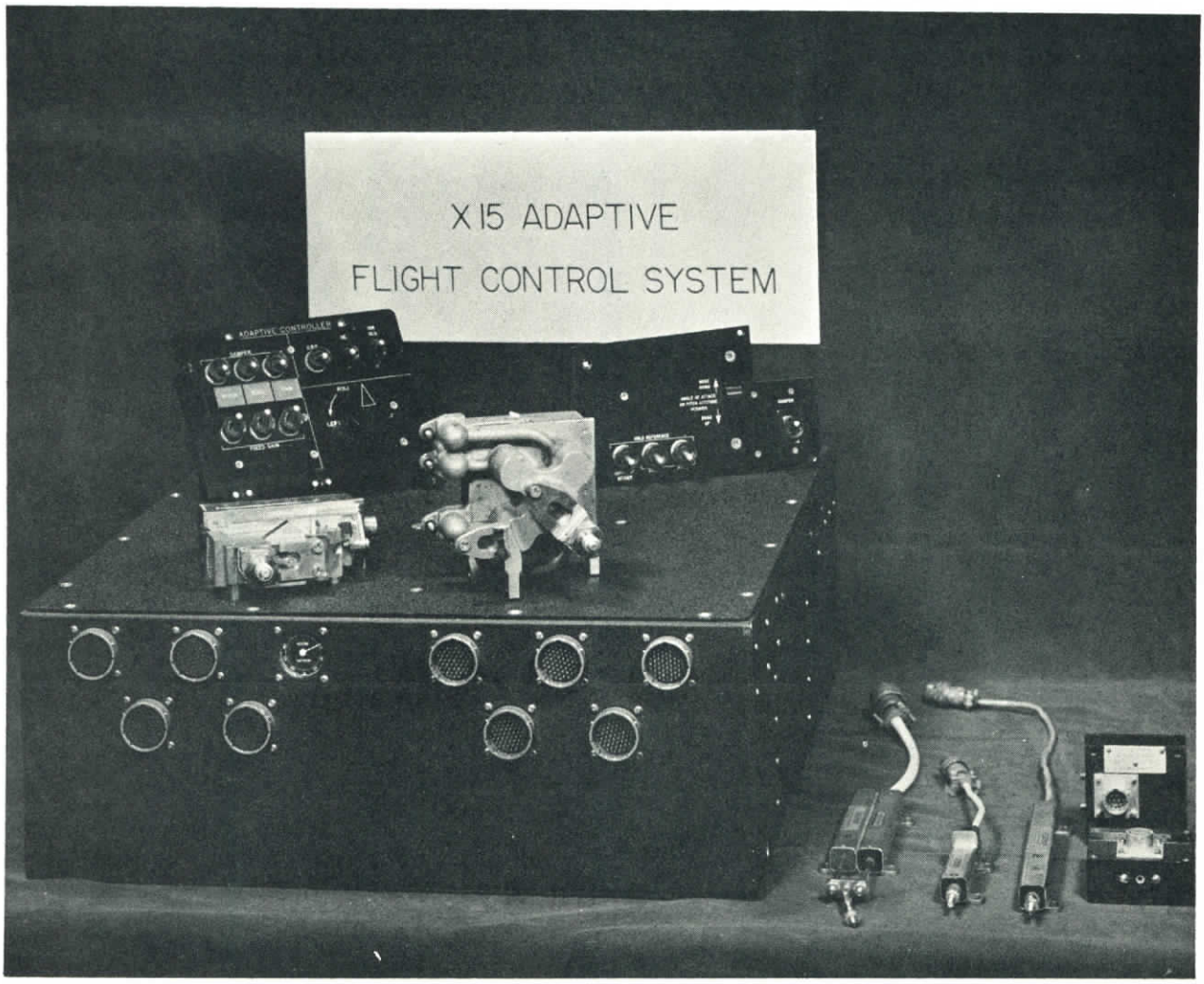

Figure 11. Display of All AFCS Components

CONCLUSION

Honeywell has developed a system for the X-15 capable of optimum performance over an extensive range of flight conditions with a high degree of reliability and failsafety. There are a number of vehicles in various stages of design and development where these characteristics will be primary considerations in the selection of a flight control system. These include the supersonic transport, STOL, VTOL, and various spacecraft.

PART II

X-15 FLIGHT TEST SUMMARY

BY

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X-15 FLIGHT TEST SUMMARY

SUMMARY

Fourteen flights of the No. 3 X-15 rocket aircraft containing the MH-96 Adaptive Flight Control System have already been made. Flight data is summarized in Table 1.

The first four flights totaling 44 minutes were developmental, and resulted in complete system checkout. Flight No. 5 familiarized a second pilot with the MH-96 system. Flights No. 6 and 7 were high-performance contractual demonstration flights and resulted in USAF acceptance of the MH-96 system. A record altitude for winged aircraft of 314,750 feet was also established in flight No. 7. Objectives of flight No. 8 were to familiarize a third pilot with the system and to evaluate minor system changes. A new re-entry technique was tried in flight No. 9. Flight Nos. 10-14 were directed toward evaluating system and aircraft performance with the lower ventral fin removed. In addition, a fourth pilot was familiarized with the system during the tenth flight.

All 14 flights can be considered successful with all system modes and functions operating as planned. No system failures occurred. A component failed during one flight, but because of the redundant system mechanization, this failure was not discovered until the post-flight system check. In addition, the effect of the failure was barely discernible during flight and was not recognized as being caused by a failure.

This flight test program has shown that the Honeywell adaptive concept is capable of providing the required gain in each system axis, independently, and over a wide range of flight conditions. The flight envelope in which adaptive operation occurs covers a speed range from less than 200 mph at landing to over 4000 mph, and an altitude range from sea level to about 100,000 feet.

Table 1. No. 3 X-15 Flight Test Summary

Flight Number		Date	Pilot and Organization	Purpose	Maximum Altitude (feet)	Maximum Mach Number	Maximum Velocity (fps)	Minimum q (psf)	Maximum q		Flight Duration (min)
MH	NASA								Exit (psf)	Re-entry (psf)	
1	3-1-1	12/19/61	Aborted								
	3-1-2	12/20/61	Neil Armstrong, NASA	System Checkout	81,000	3.75	3660	100	550*		10
2	3-2-3	1/16/62	Neil Armstrong	System Checkout	134,000	5.5	5520	75	730	420	11
3	3-3-4	3/29/62	Aborted								
	3-3-5	3/30/62	Aborted								
	3-3-6	3/31/62	Aborted								
	3-3-7	4/5/62	Neil Armstrong	System Checkout	180,000	4.12	4180	10	1404	920	11
4	3-4-8	4/20/62	Neil Armstrong	System Checkout	207,000	5.31	5557	6	1082	720	12
5	3-5-9	6/12/62	Major R. White, USAF	Pilot Familiarization	184,600	5.02	5158	10	1070	670	10
6	3-6-10	6/21/62	Major R. White	Demonstration	246,700	5.07	5270	2	780	1086	10
7	3-7-11	7/10/62	Aborted								
	3-7-12	7/11/62	Aborted								
	3-7-13	7/16/62	Aborted								
	3-7-14	7/17/62	Major R. White	Demonstration	314,750	5.40	5520	0	845	1186	10
8	3-8-15	8/1/62	Aborted								
	3-8-16	8/2/62	Joseph Walker, NASA	Pilot Familiarization	144,500	5.07	5140	61	1284	730	9
9	3-9-17	8/10/62	Aborted								
	3-9-18	8/14/62	Joseph Walker	Constant Attitude Re-entry	193,600	5.25	5520	10	648	986	9
10	3-10-19	10/4/62	Major Rushworth, USAF	Pilot Familiarization	112,000	5.17	5150	82	828	572	10
11	3-11-20	10/23/62	Major Rushworth	Ventral-Off Stability	134,000	5.47	5600	88	624	740	10
12	3-12-21	12/13/62	Aborted								
	3-12-22	12/14/62	Major White	Ventral-Off Stability	141,400	5.65	5560	22	959	696	10
13	3-13-23	12/20/62	Joseph Walker	Ventral-Off High Altitude	160,400	5.73	5620	22	1774	1000	9
14	3-14-24	1/17/63	Joseph Walker	Ventral-Off High Altitude	271,700	5.47	5660	0	498	1889	10

*Maximum for entire flight

At higher altitudes, the control surface effectiveness decreases rapidly, so reaction jet controls are required for stability and control. Since the moment produced by these jets changes insignificantly with flight condition, fixed gain operation is completely satisfactory. The system makes maximum utilization of the aerodynamic controls, however, with the result that most flights use only a small percentage of the available reaction jet fuel. The amount of fuel used depends greatly on the inputs made by the pilot in evaluating the system. Since this was an objective of most of the earlier flights, fuel consumption was correspondingly high.

A detailed description of the first seven flights is contained in Honeywell Aero Report 2373-TR9, "Advanced Flight Vehicle Self-Adaptive Flight Control Syster Acceptance Flight Tests", currently being published by the Air Force as WADD TR 60-651, Part V. Flight Nos. 8-11 will be described in Honeywell Aero Report 2373-TR11, "Advanced Flight Vehicle Self-Adaptive Flight Control System, Final Flight Test Report". The following paragraphs briefly summarize all flights made to date. Information on the first 11 flights is excerpted from preliminary copies of the previously described reports.

Pre-acceptance Flight Test Phase

Flight No. 1 -- The first flight of the MH-96 system was also the first flight of the No. 3 X-15. Therefore, the objectives were to check out the aircraft and to evaluate the control system performance. Neil Armstong was the pilot for this and the three following flights. The adaptive and fixed-gain dampers, pilot trim inputs, and automatic trim operation were evaluated in this flight. Operation with adaptive dampers was smooth and well damped The pilot reported that it was easy to hold a desired angle of attack as well as nominal zero rates in the pitch and roll axes. As expected, the fixed-gain damping was looser, but still provided acceptable operation. Pitch fixed-gain damping was increased prior to the third flight.

Operation of the pitch axis trim system in flight was much easier than operation on the simulator. Control stick motion due to automatic trim activity was noticeable but not objectionable. Adaptive limit cycle motions were not evident to the pilot, although the flight recordings verify that the gains were cycling as expected. No switching transients were noticed at any time during the flight.

The hold modes were inoperative for this flight because the platform and Q-ball were excited by one of the two 400-cps sources in the X-15, while the amplifier-demodulators in the MH-96 system were excited by the other source. This was caused by an error in the aircraft cabling. Proper operation of the hold modes requires accurate synchronization of the attitude signals and amplifier-demodulators. This synchronization is not obtained when two independent sources are used.

Other problem areas during this flight were the simultaneous disengagement of all axes at launch, a 12 cps rudder oscillation at low dynamic pressure with yaw adaptive operation, inability to re-engage pitch adaptive operation after evaluation of the fixed-gain system, and drift of the roll trim knob during powered flight.

The disengagement of the dampers at launch was probably caused by a momentary drop in the aircraft 28-volt d-c supply. The pilot was able to re-engage all axes on the first try, which followed engine shut-down. Capacitors were added to the switch solenoid circuits to enable them to withstand a 0.040 second open transient in the d-c supply without dropping out. There has been no recurrence of this problem.

Post-flight analysis showed that the rudder oscillation was caused by coupling between the rudder bending mode and the control system through the lateral acceleration feedback loop. Simple lag network filters have eliminated this problem.

No cause was found for the failure of the pitch axis adaptive mode to engage after the fixed-gain modes had been evaluated. Satisfactory adaptive operation had been obtained earlier in the flight. The post-flight checkout revealed nothing wrong with the system, and it was not possible to duplicate the failure on the ground. The problem has not recurred.

Drift of the roll trim knob in later flights was prevented by changing the trim potentiometer shaft locking nut.

Despite problem areas, this first flight was considered successful and received favorable pilot ratings.

Flight No. 2 -- The second flight included evaluation of the modes checked during the first flight, plus use of pitch and roll attitude hold, control stick steering (CSS), and reaction controls. Again, all modes provided satisfactory operation and resulted in favorable pilot comments. The entire flight, including landing, was flown using the right side stick. Landing was made with adaptive dampers engaged in all three axes. Owing to added visual and sensory cues and better instrumentation, the aircraft was reported to be much easier to fly than the simulator.

No transients were noticed during any of the switching, although a 15 cps oscillation in pitch was noticed when CSS was used. This oscillation, caused by surface resonance feedback through the normal acceleration loop, was eliminated by a second-order lag filter. Limit cycle oscillations at 3 to 4 cps were occasionally apparent in the yaw axis. This was eliminated by adding a small amount of filtering to the yaw adaptive channels and by reducing the yaw gain computer set points and bandpass limits.

Since the maximum altitude attained in this flight was only 134,000 feet, reaction controls had little effect on the aircraft control. However, through visual observation and examination of the flight recordings, it was verified

that these controls functioned normally with both fixed-gain and adaptive operation. Some changes in the reaction control engage logic and fixed-gain switching were made following the flight to prevent use of reaction controls unless needed. This reduces reaction fuel consumption.

Flight No. 3 -- For the third flight, the profile was extended to include both high (1400 psf) and low (10 psf) dynamic pressures. Angle-of-attack hold was evaluated, as well as all of the modes checked in the previous flights. The landing was made using CSS, which the pilot felt required higher stick forces than the basic rate damper mode. At the highest dynamic pressures encountered shortly after launch, a roll oscillation of five degrees per second amplitude occurred. This had not been experienced on the simulator; since the pilot did not consider it objectionable, little effort was devoted to its elimination.

The high altitude and low dynamic pressure encountered during this flight enabled a more thorough evaluation of the reaction controls. The responses to control pulses were well damped in all axes; hold mode reference retention was also satisfactory. Following the flight, the rudder-pedal - heading switching circuitry was changed to permit the heading reference to synchronize to the new aircraft heading after rudder pedal inputs. This was necessary to prevent opposition between the aerodynamic and reaction yaw control systems. Autopilot hold mode indicator lights were added to the left console, within the pilot's field of vision, since it became evident in this flight that the pilot is not always able to see the left console function selector switches.

Flight No. 4 -- In the fourth flight, the flight profile was extended to still higher altitudes and lower dynamic pressures. All MH-96 system modes and functions were again evaluated and provided satisfactory operation throughout the flight. The pilot considered the damping and controllability in ballistic flight to be comparable to aerodynamic operation. Some mismatch between the aerodynamic and reaction control system was evident in all axes, resulting in excessive reaction fuel consumption. This was corrected and the reaction control switching dead zones were widened before the fifth flight.

The pilot intended to check the normal acceleration limiter during re-entry on this flight. This limiter was set to add a large normal acceleration feedback and drive the angle-of-attack reference down at a normal acceleration of about 5.5 g's absolute. However, acceleration did not exceed 4.5 g's. Since this value was maintained past level flight, the aircraft climbed from 83,000 feet back up to 97,000 feet. Angle-of-attack hold was engaged throughout this period, with a reference of 20 degrees. The flight recordings indicate that pilot control stick inputs were opposed by the autopilot, causing the control servos to bottom out in the pitch-up direction. If the hold mode had been disengaged, or if CSS had been used, the pilot should have been able to prevent the aircraft from skipping as high.

Flight No. 5 -- The fifth flight was of lower performance than the fourth since the primary objective was to familiarize Major White with the MH-96 system preparatory to the high performance demonstration flights. All modes except CSS were evaulated and performed satisfactorily. Again the side stick was used throughout the flight; landing was made with three-axis adaptive dampers. Also, as in the other flights, no transients due to switching or operation of the flaps, speed brakes, or landing gear were observed. Control stick motion due to automatic trim follow-up was observed but was not objectionable; the pilot considered the trim changes to be normal and similar to those he would otherwise have made manually. No gain cycling or other oscillations were noticed at any time except for a brief, unobjectionable roll oscillation shortly after launch. Reaction control fuel consumption in this flight was about 20 per cent of the consumption in the third flight, during which a similar altitude was attained.

The pilot was favorably impressed with all functions of the MH-96 system. He considered the pilot task to be much less than with the system in the Nos. 1 and 2 X-15 aircraft. Even during re-entry, the pilot felt that, if required, he could have diverted his attention to other tasks.

Flight No. 6 -- Flight No. 6 was the first of two contractual demonstration flights. The aerodynamic and ballistic adaptive dampers and all hold modes were used. The pilot's over-all opinion of the system was again highly favorable, although the hold modes were somewhat looser at low dynamic pressures than he had expected. However, he considered their operation to be satisfactory, and it appeared that some of the looseness may have been due to his operation of the hold mode vernier control.

An angle of attack of 20 degrees was used during re-entry. The normal acceleration increased smoothly to about 5 g's where the limiting circuitry cut in and prevented further increase. Sideslip and lateral acceleration oscillations of up to one degree and 0.2 g respectively were detected briefly during the re-entry.

As in other flights, a slight roll oscillation occurred during rotation after launch, and automatic trim activity was detected at various times. The pilot considered neither of these to be objectionable. The system was well damped and handled very easily during the approach and landing.

Flight No. 7 -- This flight was the second contractual demonstration flight, and similar to the flight No. 6, except for the peak altitude. The same modes and functions were evaluated, with essentially the same results. With the additional flight and simulator experience, operation of the hold modes was considered completely satisfactory, even in the ballistic region. In flight Nos. 6 and 7, angle-of-attack hold was not used at the peak altitudes since the Q-ball becomes unreliable at altitudes above about 220,000 feet.

Re-entry was made with an angle of attack of 23 degrees. Again, the normal acceleration built up smoothly, with the limiter taking effect at about 5 g's. The maximum sideslip oscillation was about 1.5 degrees, and the maximum lateral acceleration was 0.3 g's. The flight recordings show that immediately prior to normal acceleration buildup, a small-amplitude 12 to 14 cps oscillation occurred in all three axes. This was not detected by the pilot. The flight data indicates that this oscillation was due to the control surface resonances, which

were excited by the angle-of-attack feedback. The Q-ball has a resonant frequency between 12 and 14 cps, which is the same range as that of the control surface bending frequency. No other problems were evident in this or the previous two flights; no changes had been made since prior to the fifth flight. Therefore, Honeywell and NASA considered the pre-acceptance flight test program to be completed.

Flight Nos. 8 and 9 -- Flight Nos. 8 and 9 were piloted by J. Walker. The main purpose of the first of these was to familiarize him with the MH-96 system operation. A change in the yaw axis fixed gain had been made since the seventh flight; operation in flight No. 8 revealed somewhat improved damping. Operation of the system was considered satisfactory, although the pilot felt that he had to force the aircraft to the ground during landing. This apparently was due to the fact that with the rate command system the aircraft lacks speed stability. None of the other pilots have commented on this.

The primary purpose of flight No. 9 was to evaluate a re-entry technique in which a constant pitch attitude rather than angle of attack is maintained. This technique would enable safe re-entries to be made in case of Q-ball failure, and appears to offer some advantages over constant angle of attack for re-entries from high altitudes.

This flight followed the profile essentially as planned through re-entry, which was quite satisfactory with the attitude technique. Owing to an inadvertent disengagement of the roll damper, a severe Dutch roll oscillation occurred for several seconds, but was brought under control when the roll damper was re-engaged. The pilot also reported having to hold in a small amount of lateral stick and rudder pedal throughout the flight. Following flight No. 9, guards were added between the damper engage switches on the center pedestal function selector and a light which will flash whenever any axis is disengaged or operating in the fixed-gain mode was added to the instrument panel. Post-flight checkout of the MH-96 system showed that a resistor had opened in one yaw lead network amplifier, causing a constant bias to be applied to the rudder servo.

Flight Nos. 10 and 11 -- Flight Nos. 10 and 11 were piloted by Major Rushworth. These flights evaluated aircraft performance with the roll and yaw dampers disengaged and the lower ventral fin removed. Simulator operation had indicated that the free aircraft controllability might be better with the ventral off at high angles of attack. Flight tests at these conditions were considered necessary to verify simulator operation. The system configurations checked were roll and yaw fixed gain, roll fixed gain and yaw disengaged, and roll and yaw disengaged. The adaptive pitch axis was engaged at all times. With the lower ventral removed, much of the adverse roll due to yaw effect was absent. Therefore, it was necessary to remove the yaw rate to roll axis feedback. The lateral acceleration feedback was also removed at the request of NASA so that its effect on adaptive operation could be observed. This does not change system operation in the fixed gain configuration, since lateral acceleration is only fed back to the yaw adaptive channels.

Operation of the MH-96 system in both flights was normal and satisfactory at all times that it was engaged. During flight No. 10, the No. 1 auxiliary power unit (APU) shaft sheared, removed hydraulic power to the rudder servo and thus eliminated yaw damping during the approach and landing phase of that flight. The pilot did not experience any major control difficulties, however.

Flight No. 12 -- The purpose of this flight, piloted by Major White, was to evaluate stability and control at angles of attack of 20 to 25 degrees, again with the lower ventral removed. This time, however, the adaptive dampers were engaged throughout the flight. In the two previous flights, evaluations were made at attack angles up to 15 degrees, with the roll and yaw axes operating in the fixed gain mode or disengaged.

The flight was completely successful with no apparent problem areas. The MH-96 system performed satisfactorily during the entire flight. Roll stick pulses were made at angles of attack of 20 and 23 degrees; the pilot reported the responses to be very well damped.

Prior to this flight, a switch was installed in the landing gear to disengage the MH-96 system upon landing skid touchdown. This reduces the load on the skids. If the system remains engaged, the pitch rate that develops when the nose comes down after skid touchdown causes a servo deflection of about seven degrees in the pitch-up direction. The resulting air loads tend to overstress the skids at landing. To prevent inadvertent disengagement of the MH-96 system during flight, the circuit which disengages the system is not energized until after re-entry. With the system disengaged at touchdown and the pilot pushing forward on the control stick, the landing loads on the skids in this flight were the lowest ever experienced.

Flight Nos. 13 and 14 -- These two flights extended the evaluation of ventral-off performance to higher altitudes with J. Walker as pilot. These flights demonstrated that with the adaptive dampers engaged, aircraft performance and stability with the lower ventral removed is essentially the same as when the ventral is present. The only exception is that at high altitudes directional stability is slightly less with the ventral off. However, reaction controls are normally used at these altitudes and provide the necessary additional directional control.

Adaptive dampers were used through both flights, including landing. Operation of the MH-96 system was completely satisfactory, although the pilot again reported having to force the aircraft to the ground at landing. Both flights essentially followed the planned profiles. There were no malfunctions of the MH-96 system during either flight. During flight No. 14, the alternator driven by the No. 1 APU failed near peak altitude. This failure had no significant effect on aircraft operation since dual alternators are used. If either alternator fails, all systems using that unit are switched to the other. After re-entry, the hydraulic pump driven by the No. 1 APU failed. This was more serious since the rudder servo uses the No. 1 hydraulic system. Consequently, yaw damping was not available for the remainder of the flight. The flight, including landing, was completed without further incident.

Aborted Flight Attempts

Table 1 shows that in addition to the 14 successful flights, there were 10 flight attempts where the B-52 carrier aircraft became airborne but the X-15 was not launched. Eight of these flight attempts were aborted for reasons completely unrelated to the MH-96 system. An attempt, prior to the successful third flight, was cancelled when a temporary servo monitor failure indication occurred during the prelaunch checkout with the airborne analyzer. During ground checkouts prior to this attempt, a considerable amount of noise pickup was evident in the wiring between the analyzer in the B-52 and the MH-96 system in the X-15. It was the opinion of the Honeywell flight test personnel that the failure indication, since only temporary, was due to such noise pickup. However, NASA flight personnel ruled that it was grounds for no launch. While returning to the base, the test procedure was repeated twice with no recurrence of the failure indications. By this time, however, the X-15 fuel had been jettisoned so no flight was possible.

A flight attempt prior to the successful seventh flight was aborted when separation of the umbilical connector, which joins the B-52 and X-15 wiring until launch, separated prematurely. This prevented running of the prelaunch checkout so verification of system operation immediately prior to launch was not possible.

Gain Changed Operation

The approximate correlation between pitch rate gain and the elevator effectiveness, M_{δ_e}, is shown in Figure 12 for flight No. 5. For these curves, it was assumed that at the limit cycle frequency, ω, the aircraft response is given by

$$\left| \frac{\dot{\theta}}{\delta_e} \right| = \left| \frac{M_{\delta_e}}{j\omega} \right|$$

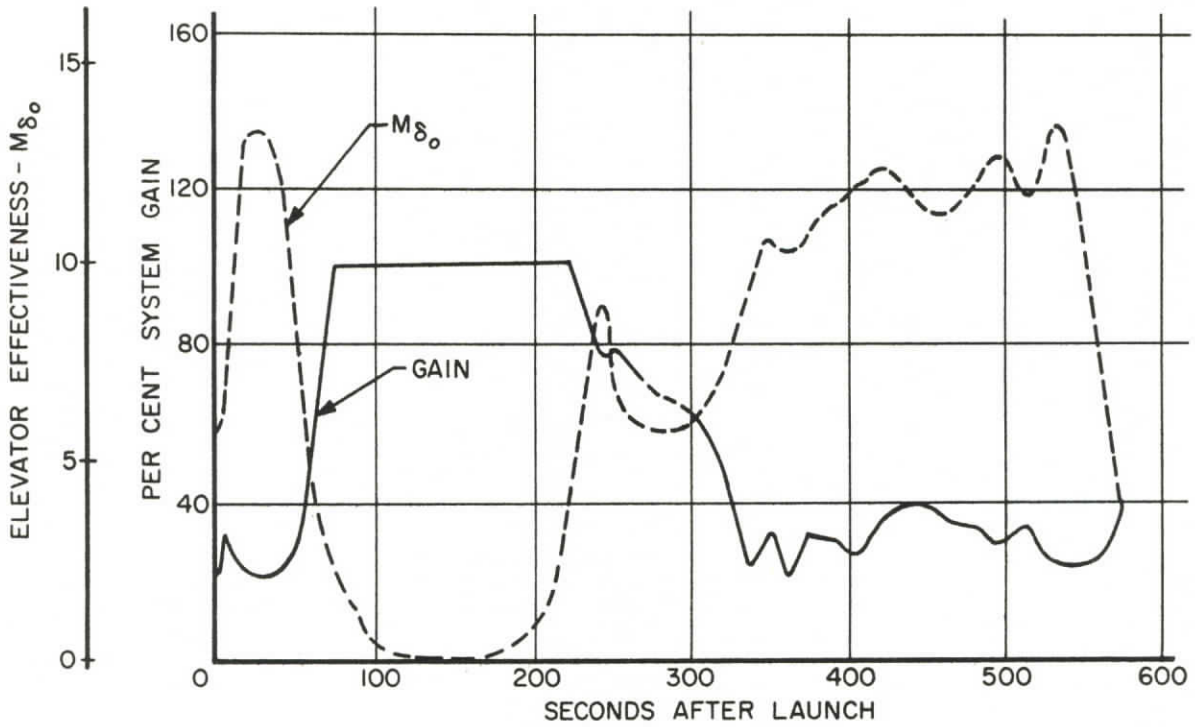

Figure 12. Surface Effectiveness and Gain Variations During Typical Flight

This is almost exact for the X-15 at nearly all flight conditions for ω greater than 15 radians per second (limit cycle frequency of 2.5 cps) and becomes more exact for higher frequencies. The quantity M_{δ_e} was computed from X-15 wind tunnel data using Mach number, dynamic pressure, and angle of attack from the flight recordings. The gain curve shown is the sum of the adaptive and fixed gains at the limit cycle frequency, as determined from the flight recordings. A vector sum was required since the frequency response characteristics of the adaptive and fixed-gain channels are different.

These curves show a good inverse correlation between the system gain and changes in M_{δ_e}. At the beginning of the curve, the gain increases even though M_{δ_e} also is increasing, since at launch the gain is clamped at its minimum value. Since this is less than critical for the launch condition, the gain rises until it attains the value needed for the limit cycle. The flat segment of the gain curve from approximately 80 to 220 seconds represents the period when the adaptive gain is at its maximum value and reaction controls are engaged. The dashed portion of the gain curve from about 275 to 300 seconds represents a period when the pilot switched to fixed gain. Therefore, there is no correlation between the two curves.