A Model-Following Variable Stability System for the NASA ARC X-14B

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A description is presented of the basic design concept, hardware design, ground and inflight acceptance testing of a Variable Stability System (VSS) installed on the NASA ARC X-14B, a twin-engine, single-seated VTOL airplane. The variable stability system was of the model-following type and provided variability in the three rotational degrees of freedom. A general-purpose airborne digital computer was used as the model residence and was an integral part of the hybrid model-following flight control system. The initial system design was achieved using optimal control techniques and the ASP computer program. Single-degree-of-freedom simulations were used to introduce nonlinear characteristics to the synthesis process, and these were extended to a six-degree-of-freedom simulation for final system synthesis and analysis. The digital computer software acceptance testing involved emulation of the system on the IBM 360 and extensive bench tests of the computer and data adaptor combination. Systems integration and acceptance testing was initiated in a ground test mode which was then extended to testing on a captive rig and culminated in flight acceptance testing in the hover flight mode. It is demonstrated that the system was capable of model-following to the NASA specifications. The major advantages of the digital computer were the reprogramming capability, reliability of operation, and the light weight and low volume of the installation which were critical in this application. The digital computer hardware and software performance was extremely satisfactory in the hybrid flight control operation, while the analog portion of the system took considerable time for alignment and adjustment.

Nomenclature

θ_M = model pitch attitude, deg
θ_{Model} = model pitch attitude, deg
θ = X-14B pitch attitude, deg
θ_{Airplane} = X-14B pitch attitude, deg
$\theta_{M_{ss}}$ = model steady state attitude command, deg
θ_c = model command
$\theta_{e\max}$ = maximum pitch attitude error
$\dot{q}_M$ = model pitch acceleration, deg/sec²
$\dot{q}_{\text{Model}}$ = model pitch acceleration, deg/sec²
$\dot{q}$ = X-14B pitch acceleration, deg/sec²
$\dot{q}_{\text{Airplane}}$ = X-14B pitch acceleration, deg/sec²
$\dot{q}_{e\max}$ = maximum pitch acceleration error
δ_Y = pitch nozzle deflection, deg
δ_X = roll nozzle deflection, deg
δ_Z = yaw nozzle deflection, deg
δ_p = pilot command, in.
δ_n = (nozzle thrust) × (wingspan)
δ_{py} = pilot pitch-stick command, in.
η = fuselage torque
F = transition matrix
G = feedback matrix
H = output matrix
J = input matrix
$K_{\theta m}$ = model attitude command gain, deg/deg
P = solution of Ricatti equation
Q = cost function output matrix
R = input weighting matrix
U = control vector
Y = output vector

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Index categories: VTOL Aircraft Design; VTOL Handling, Stability, and Control; and VTOL Testing.

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Introduction

THE systems synthesis, ground, rig, and flight test evaluation results of the model-following flight control system in the NASA ARC X-14B shown in Fig. 1 are presented in this paper. The vertical takeoff and landing NASA ARC X-14B was converted from the NASA ARC X-14A airplane, through the major modification items shown on the general arrangement, Fig. 1, and the control system block diagram of Fig. 2. A manually activated electrical motor is used to divert the thrust of both engines from the conventional position where the thrust is directed along the x-axis of the airplane, to the VTOL position where the thrust is directed along the z-axis of the airplane. Conventional surfaces such as elevator, ailerons, and rudder are used for control in the conventional flight

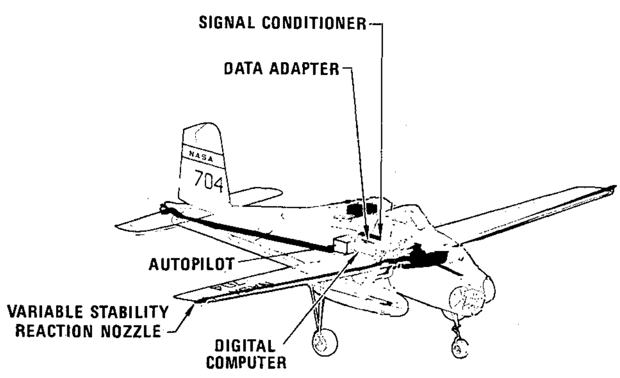

Fig. 1 X-14B VSS general arrangement.

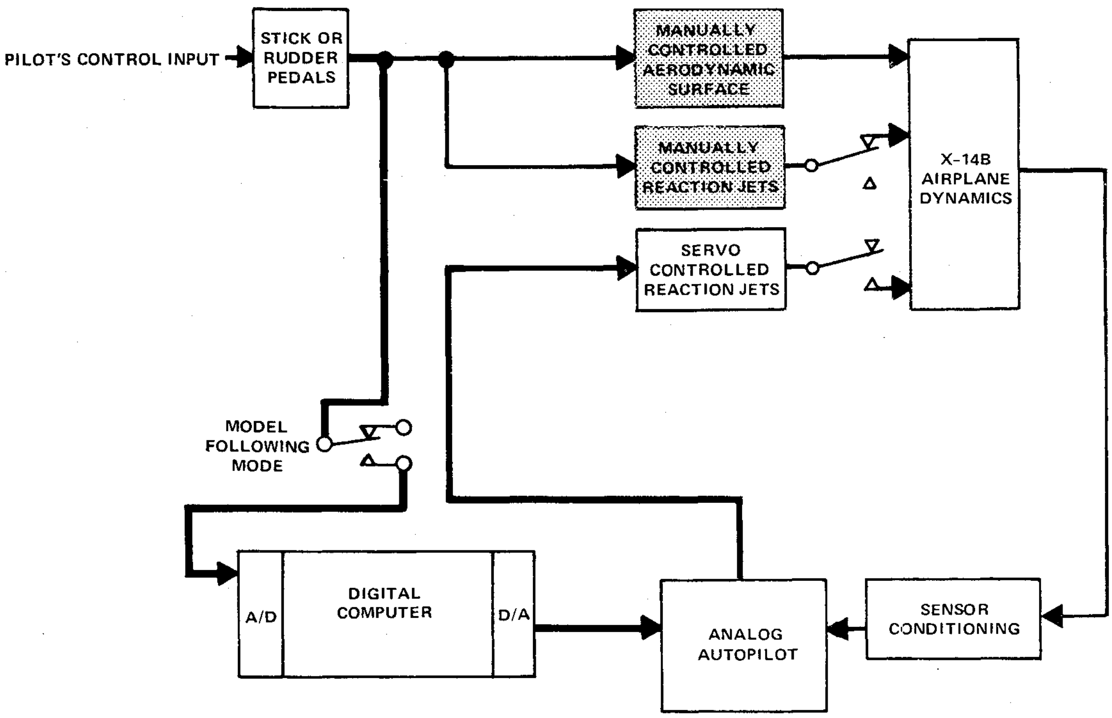

Fig. 2 X-14B VSS control block diagram.

mode. In the VTOL mode, the control of the airplane moments is achieved through the use of reaction nozzles, to which the bleed air from the engines is supplied by means of the installed system of hot air ducts. With this arrangement the X-14B airplane is capable of takeoff and landing as a conventional or as a VTOL airplane. However, the variable stability system which is discussed here is designed for application in VTOL flight only, specifically for research in hover.

The X-14B control system consists of the conventional aerodynamic surfaces, one set of mechanically controlled bleed air nozzles, one set of electrically controlled bleed air nozzles, a separate set of bleed air ducting for each of the two types of nozzles, and a "bleed air" switching valve to direct bleed air to either set of nozzles. A variable mechanical force feel system, the electronic units, the digital computer, the sensors, and the conventional stick and pedals for pilot control input complete the list of control system equipment. With this control system, the pilot has available four separate control modes: one conventional mode and three VTOL modes (manual, direct, and model). Each of the four modes uses the conventional control stick and rudder pedals as pilot command inputs. The control surfaces and manual nozzles are at all times mechanically connected to the control stick and pedals, as shown on Fig. 2, and reflect the pilot command motions in all four modes. The electrical nozzles move as a result of pilot command/sensor output, or as a result of computer command/sensor output. Only those nozzles supplied with bleed air have any impact on the motions of the X-14B in VTOL flight. In the conventional flight regime, if the variable stability system is engaged, the control surfaces simply overpower the effect of the bleed air nozzles. In VTOL flight, when the electrical nozzles are operating, the pilot is controlling the airplane through a "fly-by-wire" system.

The X-14B model-following system is unique in comparison to other fixed-wing, inflight simulator systems in that an airborne digital computer is a central part of the flight control system. A system of feed-forward and feed-back gains is employed to achieve good model following. In this way, the impact of servo-dynamics, sensor noise, and structural coupling can be minimized and the system made insensitive to parameter variations associated with flight conditions. The mathematical model of the airplane to be simulated is stored in the airborne digital computer. The pilot control motions are sensed by electrical displacement pickoffs which are of the analog type. The analog signals pass to the digital computer through the analog-to-digital converter unit as shown in Fig. 2. The digital computer is an IBM 4 Pi CP-2 computer which has stored differential equations for three airplane mathematical models. In the preacceptance configuration, Model 1 and Model 2 could be selected and flown in the ground simulation mode. Model 2 could be selected and flown in actual hover flight, and Model 3 was a model used for preflight checkout of the complete Variable Stability System. The major function of the NASA ARC X-14B is to provide inflight simulation capability in the vertical takeoff and landing flight regimes.

This paper begins with a discussion of the design of the model-following control system. A part of the specification to which the system was designed is presented. A brief discussion of the use of Kalman-type optimal control theory in the synthesis of the control system is presented. This is followed by a discussion in the system synthesis on the use of a nonlinear single-degree-of-freedom simulation and a six-degree-of-freedom simulation of the control system.

A man-in-the-loop moving base simulation is then discussed as it was employed to establish failure detection and mode switching times. The design and testing of the airborne digital computer hardware and software is then discussed. The ground test, captive rig test, and flight test results are shown in the form of time histories to demonstrate the performance achieved by the model-following control system.

The Design of the Model-Following System

The conceptual design of the flight control system had been accomplished by NASA ARC personnel (Ref. 1, 2, and 3). NASA ARC personnel had decided that the system would be of the model-following type. Provisions had also been made for the redesign of the pneumatic system, and a fail-safe concept established. The following discussion will cover the work from the completion of the conceptual design to the final acceptance testing.

The design of the model-following system was completed in three phases. In Phase I, linear analyses and optimization techniques were used in the synthesis of the control for each degree of freedom. Phase II included nonlinearity studies

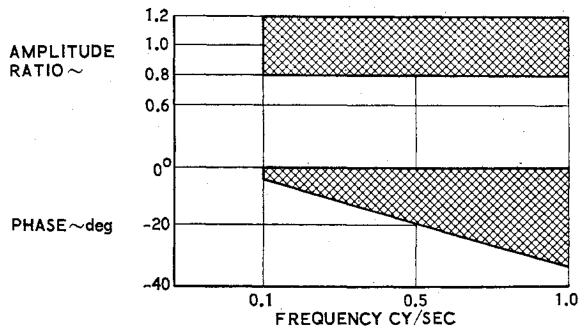

Fig. 3 Frequency response requirements for the VSS.

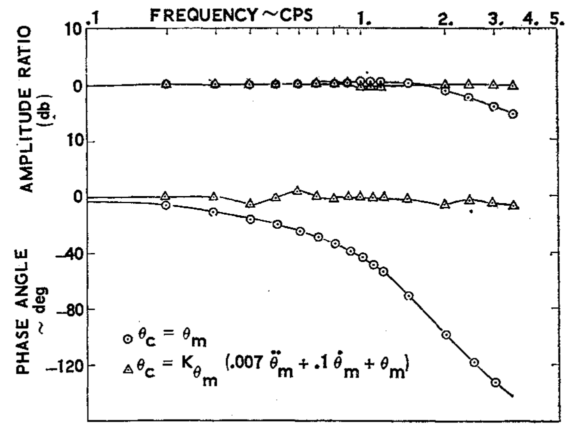

Fig. 4 Suboptimal system frequency response.

but was limited to each single degree of freedom. In Phase III, six-degree-of-freedom simulations were employed to complete the system design. A limited pilot-in-the-loop simulation was conducted in support of the design work.

In the Phase I studies it was assumed that the high gain model-follower would cause the motion in the three axes to be decoupled. Single-degree-of-freedom analyses were then conducted on each axis. No attempt was made to model the sensor dynamics. The basic idea in the analysis was to devise a feed-forward and feed-back gain set that would cause the follower response to be the same as the model response to within some specified accuracy.

The specifications imposed by NASA ARC involved frequency response requirements and step response requirements, and included specifications on acceptable long-term drift. The specification involved tolerances on both acceleration and attitude response. The frequency response requirements are shown on Fig. 3 and were established to guarantee adequate model following with an acceptable lag while limiting the use of resonance-producing lead terms in the command loop. The step response requirements, shown on Table 1, were established to limit the pure lags associated with model-following and ensured that the steady-state performance of the system was acceptable.

In each degree of freedom, all signals, attitude, rate, and angular acceleration were used for the model matching. Each aircraft motion variable was measured by an independent sensor, such as an attitude gyro, rate gyro, or angular accelerometer. The design activity was concentrated on finding the optimum gain for each loop. There were six loops: the three feed-forward loops from the model and the three feed-back loops from the following sensors. As a result, a set of six gains had to be determined for each axis.

Optimization techniques were applied to obtain the best set of gains. Consider the equations of motion as represented by

$$\begin{aligned}\dot{X} &= FX + JU \\ Y &= HX\end{aligned} \tag{1}$$

where

$$X = \text{State vector, e.g. } [\theta_m,\ \dot{\theta}_m,\ \delta_y,\ \dot{\delta}_y,\ \theta,\ \dot{\theta}]^T \tag{2}$$

and using the quadratic performance index

$$2V = \lim_{\tau\to\infty} \int_0^\tau (X^T H^T Q H X + U^T R U) dt \tag{3}$$

the gain sets can be obtained, partially, by solution of the Ricatti equation

$$\dot{P} = PF + F^T P - PGR^{-1}G^T P + H^T Q H \tag{4}$$

An automated method (ASP) of solving this equation (Ref. 4) based on the work of Ref. 5 was available at Northrop and used in the synthesis process as reported in Ref. 6. Typical results are shown on Fig. 4. It can be seen that the use of a reduced number of feed-forward loops results in a follower response that is inconsistent with the response requirements. The use of the full set of feed-forward loops, referred to on the figure as the combined input, results in a follower response that is well within the specified requirements. Depending on the weighting assigned to the output error matrix Q and the input control power matrix R as a ratio, the follower dynamic response will be affected. Designer judgment, involving realizability and reasonableness of the gains required, is needed to select the best Q/R ratio. In the initial synthesis work on the X-14 follower, the $Q/R = 100$ gain set was used as the basic set.

Results obtained from the use of the optimum linear technique are only as good as the representation given to the X-14B airplane dynamics and control system. A rigid body representation was valid for the pitch and yaw axis, but not for the roll. The force created by the roll reaction control nozzles opening and closing excited the wing bending modes. The frequency of the fundamental bending mode was in the vicinity of the natural frequency of the roll variable stability system and therefore could not be neglected.

The effect of the bending mode on the roll stability was analyzed. An analytical expression for the fundamental bending mode was established and expressed in the transfer function form:

$$\frac{\eta}{\delta_N} = \frac{3600(S/600 + 1)}{[S^2 + 2(0.05)60S + (60)^2]} \tag{5}$$

Table 1 Step response criteria as a percentage of model maximus

	Roll			Pitch			Yaw		
Time after step (sec.)	$\leq .2$	$>.2$ $\leq .4$	$>.4$	$\leq .2$	$>.2$ $\leq .4$	$>.4$	$\leq .2$	$>.2$ $\leq .4$	$>.4$
Angular acceleration error	±60%	±25%	±20%	±30%	±20%	±20%	±30%	±20%	±20%
angular Displacement error	±10% or .5 deg	±10% or .5 deg	±10% or .5 deg	±10% or .5 deg	±10% or .5 deg	±10% or .5 deg	±10% or .5 deg	±10% or .5 deg	±10% or .5 deg

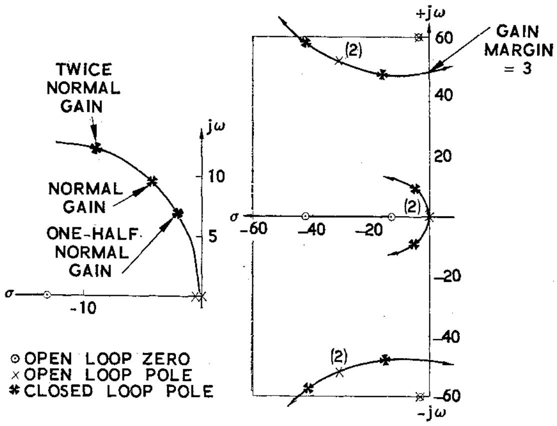

Fig. 5 Case 1 root locus.

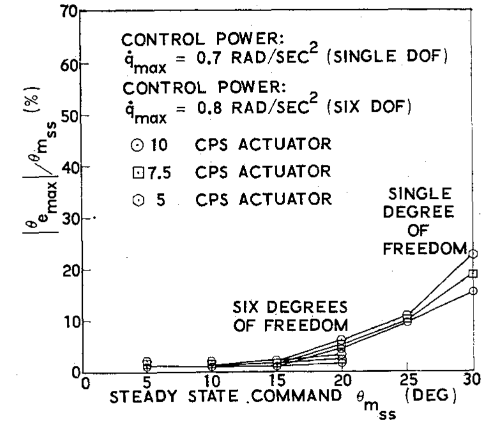

Fig. 6 X-14B error from a step displacement command.

An attempt to use the ASP optimal control technique of Ref. 4 with this functional representation of the structural response failed because of computer core size requirements for the ASP program. The digital program did not produce an optimum solution, so analyses were performed using the more conventional root locus technique.

Figure 5 shows a typical root locus plot for the total system including the structural mode. A second-order notch filter was used to suppress wing bending mode effects. In order to achieve an effective cancellation of the first bending mode, a notch filter of the same frequency as the measured bending mode frequency was selected. As a result, the lightly damped bending mode was replaced by a heavier damped notch filter mode. The lag effectively adds to the actuator lag and is shown in the root locus as a double pole. The addition of this lag to the system created enough phase shift in the feedback loop so that it was impossible to achieve the over-all damping ratio that had been achieved in the pitch and yaw axes.

Estimates of the bending mode frequency had established 60 rad/sec as the lowest bending mode frequency. The notch filter was manufactured at 52 rad/sec to ensure phase stable structural mode cancellation should the mode frequency turn out to be lower than 60 rad/sec.

The Phase II studies involved analog simulation of each single degree of freedom. Nonlinearities, the wing bending mode, and the notch filter were simulated where appropriate. The simulation provided confirmation of the optimal analysis work and allowed the impact of nonlinearities, such as available control power, to be established.

The Phase III studies extended the simulation to six degrees of freedom, and the analog simulation was superseded by a digital simulation using the DSL-90 language.

Results typical of the outcome from the single- and six-degree-of-freedom studies are shown on Figs. 6 and 7. On Fig. 6 is shown the attitude error to a step input, as a percentage of steady-state attitude command, as it is affected by the magnitude of the attitude command, the magnitude of the available control power, and the natural frequency of the control actuator. Figure 7 shows the acceleration error as a percentage of the maximum acceleration command.

The single-degree-of-freedom and multi-degree-of-freedom results are similar, as might have been expected. The point to be taken from these figures is the effect that reducing control power available has on model-following. In the single-degree-of-freedom studies the control power was limited to 0.7 rad/sec^2 with the result that model-following deteriorated drastically as attitude commands exceeded 15 deg. In the six-degree-of-freedom studies, the control power was limited to 0.8 rad/sec^2, and no deterioration in model-following was noted at attitude commands as high as 20 deg. The effect of reducing actuator natural frequency is clearly seen to be an inability to keep angular acceleration errors to within specified tolerances as the natural frequency of the actuator falls below 7.5 cps. The attitude errors are not appreciably affected by actuator natural frequency. It should be emphasized that the acceleration errors shown on Fig. 7 were of very short duration, 50–100 milli-seconds, and not steady-state errors. However, they were of some concern, from the point of view of the impact they would have on simulation fidelity. To get some indication of the significance of the impact of the errors, a man-in-the-loop six-degree-of-freedom moving-base simulation was conducted on the Northrop Three-Axis Visual Display Simulator.

The X-14B project pilot evaluated the two mechanizations shown on Fig. 8. Mechanization 1 was the model and prefilter, assuming perfect model following. Mechanization 2 was the model and prefilter and the X-14B with feed-forward and feed-back loops. The model-following errors of Figs. 6 and 7 thus were simulated in Mechanization 2. The pilot was unable to distinguish between Mechanization 1 and Mechanization 2. A word of caution in interpreting this information. The acceleration cues in ground-based simulators are obtained mainly from the motion drive system. The model-following capability, so to speak, of ground-based simulator motion system is much less than the inflight simulator in question. The errors associated with the poor model-following capability of the ground-based simulator could have swamped the model-following errors of Mechanization 2. While the simulation was not entirely successful in helping to quantify model-follower error tolerances, it was useful in establishing minimum elapsed times for failure detection in the control system.

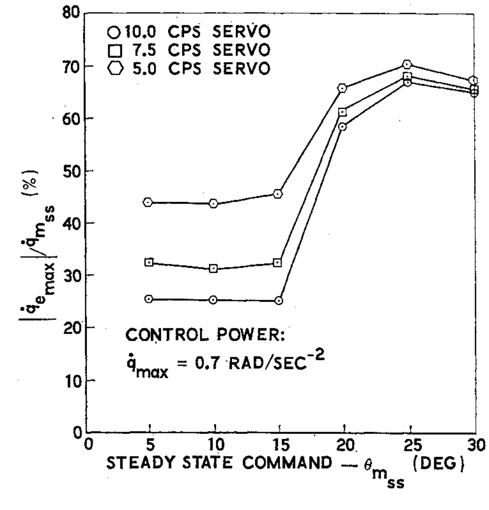

Fig. 7 X-14B error from a step displacement command.

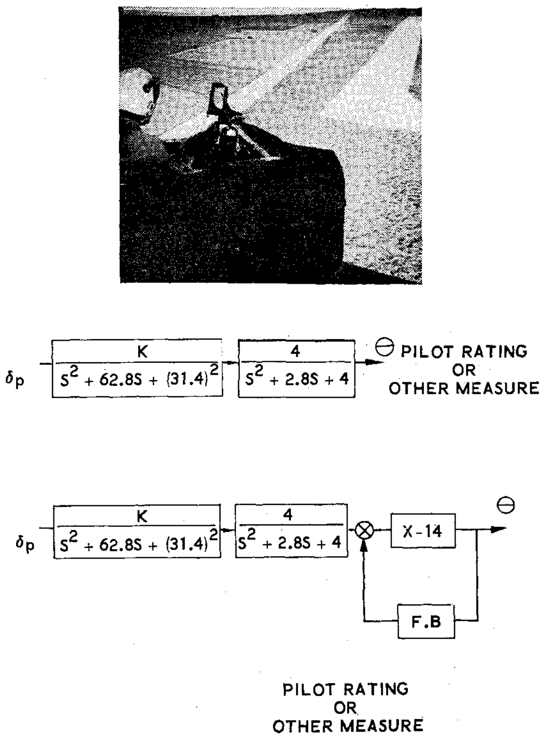

Fig. 8 Better criteria.

Systematic failure testing indicated that the allowable elapsed time between the occurrence of an electrical failure and the instant that manual control was available was 250 msec.

The outcome of the model-follower loop design was that the actuator frequency should be at least 10 cps and pitch control power should be 0.8 rad/sec² to ensure model-following of commands up to 20 deg. A notch filter was specified that would minimize wing bending effects on model following. While optimal control techniques had been useful in initial design, it was necessary to use more conventional techniques such as root locus and simulation to finalize the design.

The Design and Testing of the Model Residence

The X-14B computing system, or model residence, consists of airborne hardware, ground support hardware, and airborne and ground-based software. The testing of the computer system started with the software, then proceeded to hardware testing of individual units, and, finally, the whole computing system was checked for operation and for physical compatibility. The computer system was a combination of an off-the-shelf computer, a custom-built data adapter, and IBM-generated software.

The basic philosophy for the computer system testing was to demonstrate that the digital computer behaved in a manner similar to an analog computer. There were two reasons for this: 1) the customer was accustomed to airborne analog computers, so in this manner the operation of the digital computer could be easily evaluated; 2) the model-following autopilot was analog, and therefore an analog-type input from the computer was required.

The software was designed by IBM to Northrop specifications, and testing started many months before the computer hardware was delivered. The tests demonstrated that: 1) The computing cycle time was not more than 10 milliseconds; 2) The computing cycle involved all sensor correction calculations, one of three models, and part of the self-test routine; 3) The complete self-test routine execution could be performed in not more than three basic computing cycles; 4) All specified analog-type time histories were within a specified error band. The accuracy specification was such that: 1) one-percent error was allowed for a steady-state condition; two-percent error was specified for all dynamic responses; b) DSL-90 time histories were provided for comparison with the computer output, for each specified test input and for each model.

In the initial steps of the testing, the computer output was an output of the airborne computer simulator, the CP-2 dynamic simulator, with Fortran statements for all analog inputs and outputs. The simulator was loaded into the IBM 360/65 computer system. A dummy load was provided for the self-test routine. The simulator used the same sequencing and basic statements as the IBM 4 Pi CP-2 computer. Each CP-2 detailed operation which may take only a few μsec was listed in the computer printout for each computational cycle time and for the duration of the specified time of the input. From these listings it was possible to determine and confirm that the CP-2 computer actually performed all the specified calculations in a much shorter time than specified. The CP-2 actually idled in the "wait" state until the 10-msec time expired.

DSL-90 was a scientific program, for use on IBM 7090 computers. The program used a special type of shorthand instruction and Fortran statements which enabled the programmer to program the equations of the X-14B models for the airborne computer in more precise form than on the airborne computer. The DSL-90 calculations were performed using 36 bits, a better integration routine, and a shorter time interval for computations than the CP-2 computer. Thus, the DSL-90 results could be compared with the CP-2 simulator and be used for error evaluation of the CP-2 computer.

A typical time history comparison between the CP-2 dynamic simulator output and the DSL-90 results is shown in Fig. 9. Shown are the responses in roll rate and roll acceleration of a model, referred to as Model 1, for pilot step input in all three axes: pitch stick, roll stick, and rudder pedals. Model 1 simulated a typical experimental VTOL with many nonlinear characteristics, and therefore took a long computing time as compared to Model 2 or Model 3. Model 2 was a linear second-order type system to be used in flight test evaluations, and Model 3 was a ground test model useful in preflight checkout. Note the excellent agreement for the roll responses between the DSL-90 and CP-2 simulator output.

The individual tests of the hardware units were performed at the factory. The CP-2 computer and the FOU (a ground support unit for CP-2) were tested in Owego at IBM facilities. The DAU (an analog-to-digital and digital-to-analog conversion unit) was tested in Tempe, Ariz. at DSE facilities, and the tests performed were typical hardware unit acceptance tests.

The over-all computer system acceptance testing was performed at Northrop facilities. Here, the airborne computer was loaded with the demonstration programs and tested in

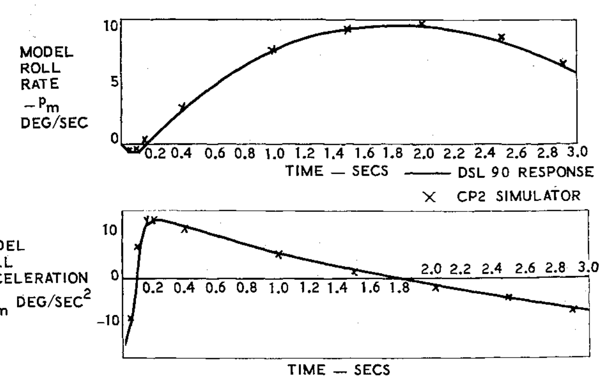

Fig. 9 Comparison of DSL-90 and CP-2 simulator dynamic responses.

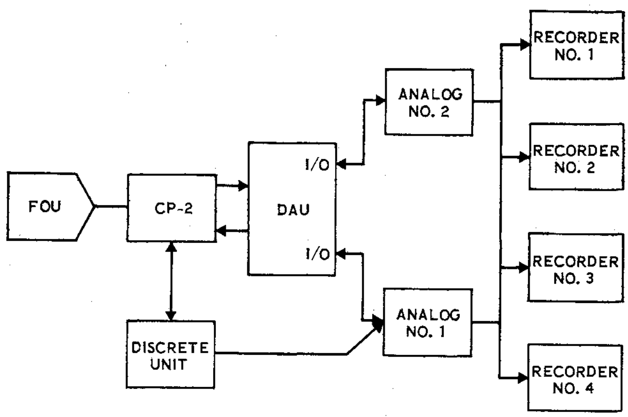

Fig. 10 Demonstration and acceptance test setup at Northrop.

the test setup shown in Fig. 10. Two Comcor CI-175 analog computers were used to provide inputs and accepted output from the DAU. Four 8-channel strip-chart recorders were used for the X-14B computer system evaluation. The CP-2 computer was provided with 400 Hz 3-phase 115 v electrical power. A discrete function generator unit provided discrete inputs for the CP-2 computer. The FOU unit was used to to load punched tapes of the demonstration programs and to insert special "test patches." The computer was tested with the FOU connected and disconnected from the CP-2.

The basic test requirements were: 1) record cycle time; 2) for a specified step test input record time history on analog computer and compare with DSL-90 traces; 3) run frequency response of Model 2 for the pitch, roll, and yaw axes; 4) perform logic switching.

The computer cycle time was checked from the recorded time histories. The recorder was set at top speed during the computer operation. The computer cycle time was distinctly shown in each varying analog signal. Since during a computing cycle the output is in a "hold" stage, a series of steps, each 10 msec long, was created.

Figure 11 shows a plot of the CP-2/DAU hardware output in comparison with the DSL-90 data. Note that the time histories of rate and acceleration are very close to the DSL-90 time histories. The discrepancies in the attitude traces, during the 1-2 sec time period, are due to improper DSL-90 representation of the CP-2 scale factors. The DSL-90 attitude was limited to 32 deg while the CP-2 output was limited to 30 deg

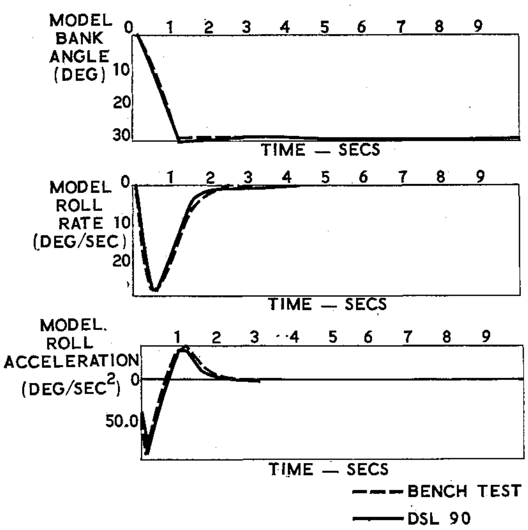

Fig. 11 Comparison of DSL-90 and CP-2/DAU hardware responses.

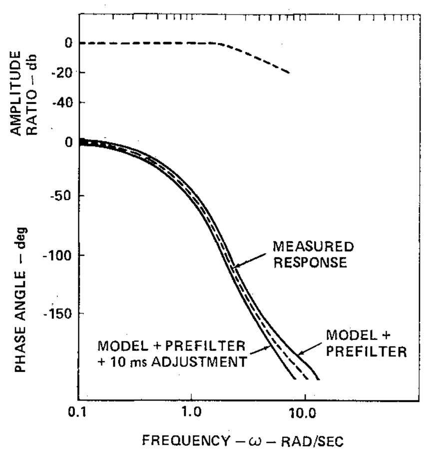

Fig. 12 Frequency response phase/gain.

Frequency response testing on each axis of the model was performed using a frequency sweep from $f = 0.1$ Hz to $f = 2.0$ Hz. Figure 12 shows the criteria used for roll frequency-response acceptance. The test data were within specified tolerances as shown on Fig. 12.

The outcome of the airborne model residence design and testing was assurance that the combination of hardware and software procured from IBM would perform its prescribed tasks. In fact, the only subsequent difficulty with the model residence was in the digital computer power supply, a difficulty unassociated with this specific application.

Ground Testing of the Variable Stability System

A method of testing the NASA ARC-14B VSS on the ground was devised. Later, this test was used as a comprehensive preflight check which was always performed with the VSS systems installed in the aircraft, with power provided from a ground unit. The test was utilized for detail system checkout operation and for the rig test "dry run".

A realistic closed-loop operating environment was devised. Except for the engine and bleed air system, the whole control system was energized and functionally set into operation. The free airframe dynamics of the X-14B were simulated on the Comcor CI 175 analog computer. Three test cables were connected between the analog computer located next to the airplane and the Lear Siegler Astronics VSS electronics units (signal conditioner unit and model-following autopilot unit). These cables carried electrical signals to and from the analog computer, and in this capacity were used as a link for the feedback loops. Positions from the four aircraft nozzles were fed to the analog computer. The analog computer used these nozzle positions first to compute the control moment inputs, then to calculate and output each angular degree-of-freedom state variable to the summation points of the corresponding sensors located in the aircraft. At the summation points the analog output variables were of the same polarity and scale as the corresponding sensors and, therefore, repositioned the nozzles in the same manner and through the same hardware channels as during rig and flight test. All the sensors were energized, causing some output noise at the summing points, and therefore produced an environment similar to the actual rig or flight test environment. To make sure that the sensors were working properly before each test, the aircraft was set into sinusoidal motion in each axis by hand. Because the airplane was light, it was easy to get a constant-amplitude sinusoidal motion of $f = 0.5$ Hz. For each axis, a strip chart recorder was used to record simultaneously the motion from all three angular sensors. Outputs from the angular accelerometer and rate gyro were compared with output of the attitude

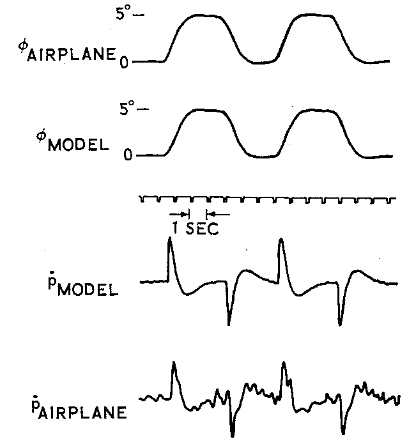

Fig. 13 Model vs airplane response in ground simulation mode.

gyro to insure the system was functioning properly and that the fail-safe system was properly adjusted to catch failures and eliminate nuisance disconnects.

A typical time history taken during the ground test, and as recorded on a strip chart recorder, is shown on Fig. 13. For this run, the VSS was switched into model mode (model-following mode). Through a discrete switch located in the cockpit, the pilot injected a 5 deg roll attitude command into the digital computer. After a few seconds, he removed the test input. The computer produced output, which was the roll model attitude and roll model acceleration, is shown in Fig. 13 as ϕ_{model} and as $\dot{p}_{model}$. The simulated aircraft response is shown on the same figure as $\phi_{airplane}$ and as $\dot{p}_{airplane}$. Note the good model matching. The ground-based airplane simulation did not include a wing bending mode, but the VSS hardware had installed a notch filter in both roll channels. As a result, excessive noise was generated which is shown in the airplane acceleration trace, $\dot{p}_{airplane}$.

The ground simulation confirmed anticipated problems associated with the use of the high-loop gains. Both the pitch and yaw axes had lower maximum control power available than the roll axis. The 2 Hz loop frequencies desired on all axes were achieved by means of setting a high v/v gain between the sensors and control nozzle position on the pitch and yaw axes. The high v/v gain amplified the system noise and in this case created an excessive nozzle chatter. To reduce the noise effects in both axes, the gains were reduced by 50% from the original estimates. The reduced gains were used in the rig and flight tests.

Rig Testing of the Variable Stability System

The test rig employed in the rig testing of the VSS system was designed, built, and operated by NASA ARC personnel. Figure 14 shows the NASA ARC X-14B airplane mounted on the test rig.

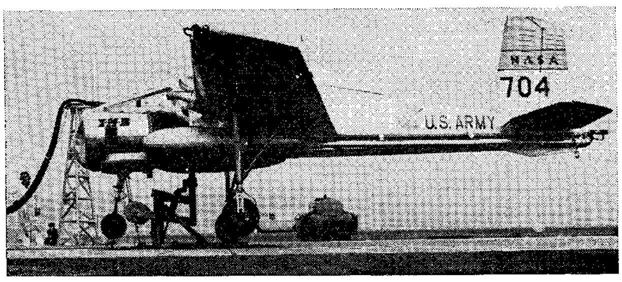

Fig. 14 X-14B mounted on rig test facility.

Rig testing of the ARC X-14B VSS produced some important measurements which helped to establish and redefine some of the more critical control system parameters. Among these parameters were the control power available in the pitch, roll, and yaw axes. In addition, the VSS performance and safety criteria were evaluated during these tests.

The rig test stand was specifically designed to test engine performance at hover flight conditions. In order to be able to minimize the ground effects during certain tests, the X-14B airplane was placed high above the ground. A movable grill-like platform was built over the testing pit to hold the airplane. The height of the platform was remotely controlled. When fully extended the platform was capable of placing the aircraft at a height of about five feet above level ground, which meant that the total distance to the bottom of the pit was about 10 ft. The X-14B was attached to the platform through a device that held the aircraft captive in the horizontal plane so that no translational motion could be executed either in the forward or in the side degrees of freedom. However, the motions in the other degrees of freedom were permitted by the rig. The aircraft was free to move in pitch, roll, yaw, and in altitude. All angular travels of the aircraft were restricted to 12 deg; the height displacement was limited to one ft.

The X-14B VSS was evaluated in the hover mode with both engines running and with the sensors measuring and producing not simulated, but real outputs. Within the provided space, the test pilot could fly, i.e.: 1) use throttle for control of engine thrust to liftoff from the platform to a one-ft hover altitude, and then maintain hover altitude; 2) use stick and pedals to command and maintain a desired attitude.

Three 75-ft cables were used to transmit data from the aircraft to four strip chart recorders located inside a trailer. To prevent excessive loading to the sample and hold stages of the digital-to-analog conversion unit, each cable was specially made with a low capacitance shielded wire. At the aircraft, these cables were attached to the same connectors as the three cables used during ground testing.

First preliminary rig test results showed that the X-14B VSS was performing properly. All the polarities in each loop were correct, and the aircraft appeared to be following the model. However, a closer check revealed that the VSS was not quite satisfying the specified error criteria (Fig. 3 and Table 1). For example, to a pitch step command, the attitude error was about 12% instead of the permissible 10%. The yaw frequency response at 1 Hz showed over 56 deg phase shift, which was about 20 deg over the acceptable limit. Roll frequency response showed a low-amplitude ratio on the acceleration signal.

A quick ground test was conducted to find the sources of error. Since the ground testing simulated a hover condition where the aircraft dynamics were represented mainly by inertia effects, and since all loop gains were known and fixed in terms of v/v ratios (nozzle position-to-sensor signal level or nozzle position-to-computer command), the only parameters in question were control power and aircraft inertias. To measure control power or aircraft inertia directly is very difficult. However, the ratio of these two parameters is angular acceleration, which could be measured directly by an accelerometer. The maximum nozzle opening which represented maximum commanded acceleration also could be measured and recorded. Therefore, by comparing the rig test results with the results from the ground tests, it was possible to get an estimate of the maximum control power. It was determined that the original estimate of the roll control power was about 30% high. The actual control power on the roll axis, for example, appeared to be 1.1 rad/sec^2 and not the 1.6 rad/sec^2 used in the analysis and simulation.

Taking into consideration the revised values for the control power, all the loop gains were adjusted. When the final rig testing was completed with the adjusted gains, it was found that VSS was performing within the specified error band. The pitch response to test step input now was showing, for example,

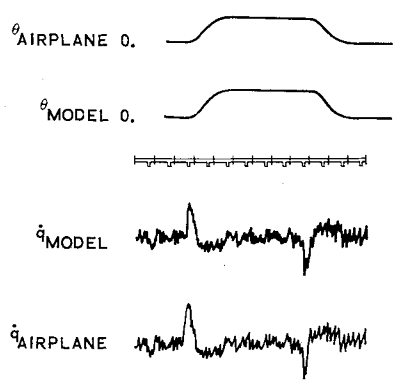

Fig. 15 Model vs airplane on NASA Ames rig.

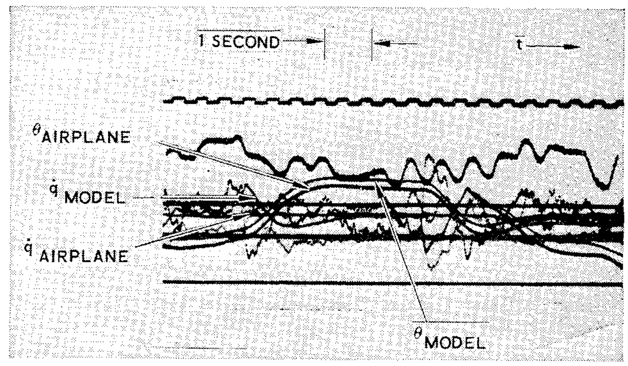

Fig. 17 Comparison of the time histories for the model and the airplane in hover.

only about a 2% error. A time history of the recorded rig test results is shown in Fig. 15. The same pitch test input was used here as during ground testing. The test pilot used a switch (a discrete input) to command a precise step input into the digital computer. Good model-following is indicated by the data on Fig. 15. The aircraft attitude trace has the same form as the model attitude, thus demonstrating good model matching was achieved. Note a high frequency noise in the acceleration traces. The noise was generated by engine vibration.

Flight Test of the Variable Stability System

The final phase of the ARC X-14B VSS acceptance activity was the flight testing. Unlike the rig tests, no system adjustments were required here, and therefore the flight test program went smoothly and lasted a comparatively short time.

The flight tests were performed to satisfy the final performance requirements of the X-14B VSS program and were conducted as specified by the NASA contract. The flight test records showed that the X-14B airplane at low speeds was following the computer commands in yaw, in roll, and in pitch. The test results were certified to indicate that the contract specifications were met.

A helicopter, with open side doors for good visibility, was used as a chase vehicle. The NASA ARC X-14B performance at hover speeds was recorded on film from the helicopter The helicopter was also used to confirm the speed and the altitude of the test vehicle. The direct electrical mode was engaged on the ground, and the pilot performed a conventional takeoff. The airplane climbed at an airspeed of 80 knots to 2500 ft altitude. Winds of 17 knots normally were present at this altitude. Transition to hover speed started after the airplane was pointed into the wind. At an airspeed of about 60 knots, the test pilot turned the bleed air to the VTOL control system "on". The thrust diverters were lowered and model mode was engaged at level attitude, at speeds below 20 knots, and at an altitude of about 2500 ft. Test inputs were introduced after hover speed was confirmed. The pilot first engaged the yaw discrete, then the roll discrete, and then the pitch discrete. Each discrete was engaged for about 3 to 4 sec and then removed. One discrete at a time was engaged. A few seconds after the discrete input was removed, the pilot switched back to direct mode. Failsafe operation was checked by using the emergency button located on the "B-8 grip" to switch from direct into manual mode. In manual mode the pilot made a conversion to conventional flight and then landed conventionally.

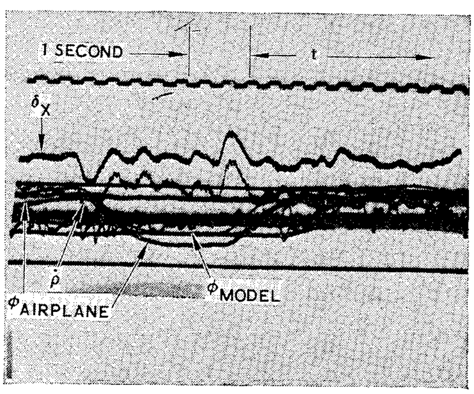

Fig. 16 Comparison of the time histories for the model and the airplane in hover.

The test results were recorded on galvanometer-type flight test recorders. Each trace in the recorder was color coded, so the signals could be identified. Figure 16 shows the recorded response to the roll discrete input. The discrete input command was engaged for about 3 sec and then removed. The roll discrete commanded 5 deg of right-wing-down roll attitude. Note that almost no errors exist between the model attitude, ϕ_{model}, and the airplane attitude, $\phi_{airplane}$. Figure 17 shows the response to a pitch input discrete. The discrete input command was engaged for about 4 sec and then removed. The discrete commanded 5 deg of nose-down pitch attitude. A noticeable increase in the forward velocity was reported by the pilot during this test. The pitch acceleration traces are easily identifiable by the triangular wave shape. The acceleration trace of the X-14B, $\dot{q}_{airplane}$, is right inside and below the model acceleration ($\dot{q}_{model}$) trace. The model pitch attitude, θ_{model}, and pitch attitude of the X-14B, $\theta_{airplane}$, have identical responses during the transition, and both have the same steady-state incremental values. The flight test data shown in Figs. 16 and 17 indicate good model-following.

Summary

A practical design of the NASA ARC X-14B VSS with a digital computer as an integral part of the flight control system was successfully performed and flight tested. Use of the airborne general purpose digital computer in the system not only made the system feasible in the light of weight restrictions, but also showed several beneficial features. The most important feature of the digital computer was the ability to repeat test results. For a switch-activated discrete input, the computer produced the same time history of the model output: on the bench during the ground test, during the rig test, and in the flight test. Thus, with this repeatability of the model command, it was possible to correlate all of the X-14B VSS engineering data.

The digital computer hardware and software performance was extremely satisfactory in the hybrid flight control operations. The analog portion of the flight control VSS took considerable time for alignment and adjustments.

As was shown, optimal control techniques were used only initially because of limitations discussed in this paper. A A nonlinear single degree-of-freedom representation of the X-14B dynamics and the VSS was very useful in basic design analysis and later in the ground testing. However, it was necessary to employ a six degree-of-freedom simulation to complete the design of the system.

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