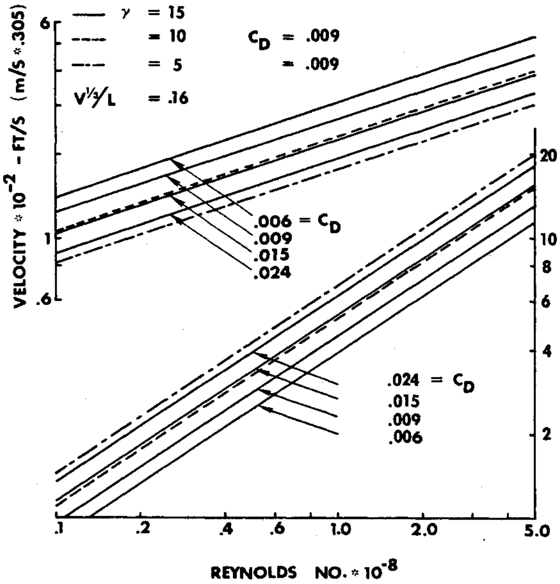

Fig. 1 Vehicle terminal velocities and lengths.

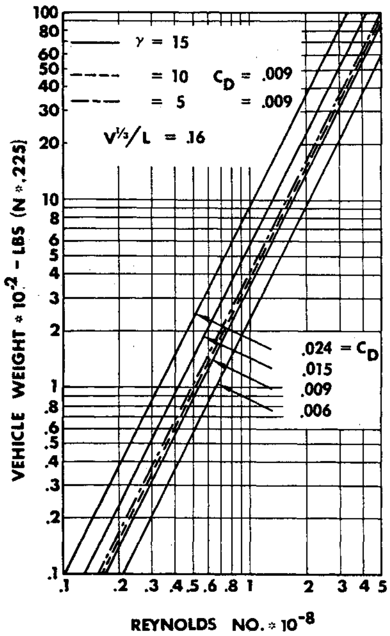

Fig. 3 Total vehicle weights.

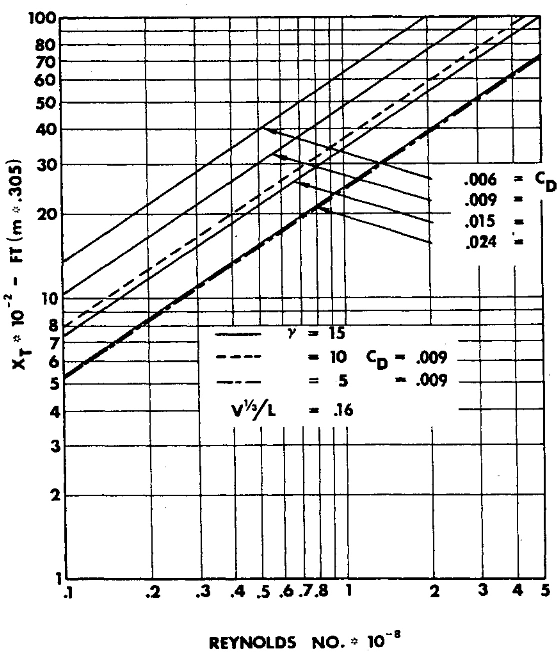

Fig. 2 Depth at which 99% of the terminal velocity is achieved.

oceans, but are not really needed since about 80% of the terminal speed is reached at dimensionless depth $s=1$ or about 275 m (900 ft). This is approximately within reach of the U.S. Navy DTNSRDC Lake Pend Oreille, Idaho facility, which is used for buoyant ascent tests from depths of 800 ft (244 m). When feasible, dropping such vehicles from heights of 30-40 m (100-130 ft) would reduce the depth requirements for 80% of the terminal speed by a factor of 2. Structural requirements are readily estimated from the relation for the collapse pressure W_c, cylinders of diameter D, thickness a, and elastic modulus E, which Ref. 1 gives as

$$W_c = KE(a/D)^3 \tag{6}$$

The coefficient K is a function of the length-to-radius ratio and is presented by Ref. 1 in graphical form. For an aluminum cylinder $E=700{,}000$ atm, $K=3$ for length-to-radius ratios greater than 16. Taking $a/D=0.05$, the collapse pressure is calculated to be 260 atm. With some internal stiffening and meridional curvature even thinner shells could be used safely. The vehicles could be abandoned, or recovered, by several means. For example, at a preset depth, the internal ballast could be ejected or a ballast section could be separated by explosive bolts. The part of the vehicle containing instruments, or even onboard recording equipment, would then be recovered at the surface.

The vehicle weight and size requirements appear to be quite reasonable. Test facilities, particularly with an air drop, can be easily found, and recovery of parts of the vehicle does not present any major problems. The concept could be used for hydrodynamic research, or as a very high speed delivery system.

Reference

[1] Baumeister, T., Avallone, E. A., and Baumeister III, T., eds., *Marks' Standard Handbook for Mechanical Engineers,* 8th ed., McGraw-Hill Book Company, New York, pp. 5-50.

AIAA 81-1620R

Portable Servoactuator Test System

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Nomenclature

A	= piston area
B	= load viscous damping
F_L	= load force
I	= servovalve current
K	= load, spring (if any)

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*First Lieutenant, USAF.

Kq = servovalve flow gain
Ksv = gain of servovalve first stage
M = load mass
P_L = load pressure
Rv = linearized slope of pressure vs flow
S = servovalve damping
V = actuator volume
Wn = servovalve frequency response
Xv = displacement of second stage spool
β = bulk modulus of the oil

Introduction

THE portable servoactuator test system (PSTS) is an Air Force project aimed at reducing support costs associated with electrohydraulic servovalves. The project is divided into two phases. Phase I will design and test a breadboard PSTS. Phase II will produce and field test six PSTS suitcase testers. This paper will discuss the background, description, and concept of the portable servoactuator test system.

Background

At present the Air Force has no way of testing electrohydraulic servoactuators onboard aircraft. As a result, many manhours are spent troubleshooting hydraulic problems. This leads to a considerable expense in terms of time and money.

The portable servoactuator test system (PSTS) will eliminate this expense. By repackaging existing reprogrammable test equipment into a suitcase, the flightline mechanic will be able to get a go/no-go status while the actuator is still on the aircraft. Not only will this reduce the maintenance expenses, but it will also expedite the pinpointing of hydraulic problems.

A total of 24 different USAF aircraft have been identified as having electrohydraulic servoactuators. The PSTS will be useable on all these aircraft plus any new aircraft that will use this type of actuator.

Description

The portable servoactuator test system is to be an electronic device, packaged in a portable case (see Fig. 1), that will test an actuator on the aircraft. Using 28-V dc power and 400-cycle 120-V single-phase ac power, the PSTS will drive the actuator under test via an electrical current input into an electrohydraulic servovalve. Hydraulic power, supplied by an onboard aircraft APU or an external source (mule), will also be required. In addition, outputs from position and/or pressure transducers will be needed. The tester will be configured for a triplex or quad redundant level and will have five major functional elements. The system is divided functionally as follows:

1) servoactuator control circuits
   a) LVDT excitation
   b) synchronous demodulator
   c) servoamplifier
2) servoactuator model and comparator circuits
   a) electronic actuator model
   b) comparator (dynamic)
   c) comparator (static)
3) function generator circuits
4) test/evaluation logic circuits
   a) program input circuitry
   b) system configuration logic
   c) evaluation logic
5) servoactuator interface control
   a) servoactuator I/O cable and adapters
   b) servoactuator program card

The functional circuits of 1 and 2 will be generic in configuration with certain variables capable of being programmed for the required values of a specific actuator to be tested. The function generator circuits will also be programmable for the necessary command signal characteristics required in a given test of a particular actuator. The logic circuits of 4 will receive the program inputs from the specific servoactuator program card, input the correct program commands to configure the test system (circuits 1, 2, and 3), and evaluate the test results received back from the comparators. This evaluation will result in a go/no-go status given on the actuator (see Fig. 2).

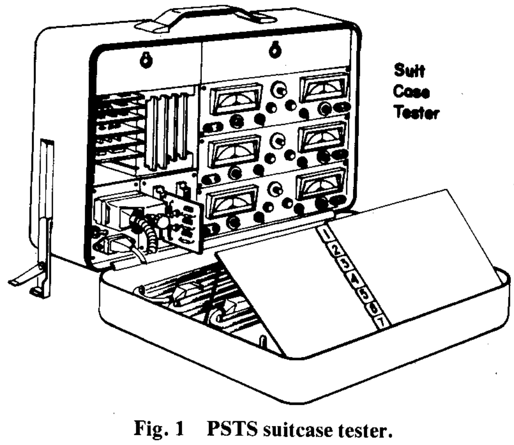

Fig. 1 PSTS suitcase tester.

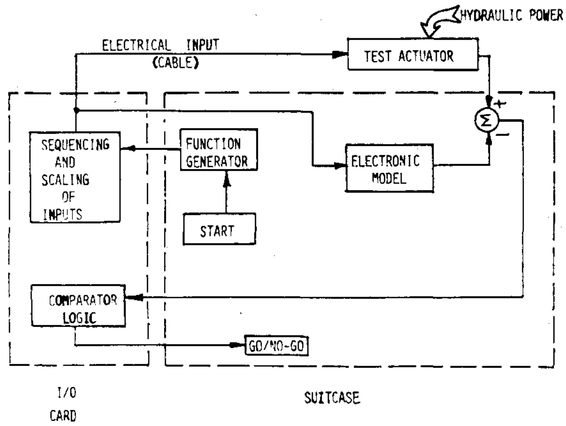

Fig. 2 PSTS functional diagram.

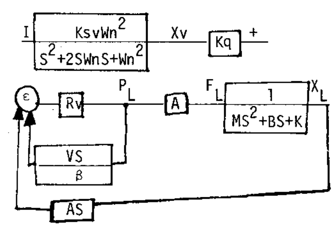

Fig. 3 Electronic model of a given actuator.

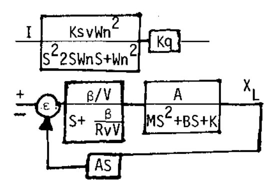

Fig. 4 Block diagram reduction.

Concept

An electrohydraulic servoactuator is a fairly complex system. By reducing it to a mathematical expression and simplifying, the primary variables can be isolated. This expression can be mechanized in modern electronic circuitry and the variables can be made programmable. This results in a fairly accurate electronic model of a given actuator (see Fig. 3). The block diagram reduces to the expression shown in Fig. 4.

The overall transfer function is

$$\frac{X_L}{I}=\frac{KsvWn^2}{S^2+2SWnS+W^2}\,\frac{Kq\beta A/V}{MS^3+(\beta M/RvV+B)S^2(K+\beta B/RvV+\beta A^2/V)S+K\beta/RvV} \quad (1)$$

Simplifying Assumptions

1) B is very small compared to the terms it adds to (generally true).

2) The servovalve is reasonably well damped at the frequencies of interest so that it can be represented as a single time constant, Tsv.

Under these conditions the transfer function is

$$\frac{X_L}{I}=\frac{Ksv}{TsvS+1}\times\frac{Kq\beta A/V}{(MS^3+(\beta M/RvV)S^2+(K+\beta A^2/V)S+K\beta/RvV} \quad (2)$$

Further Simplification

Assumption—the servovalve time constant is small enough that it has no significant effect in the frequency range of interest and can be neglected. Combining Ksv and Kq into a single gain term, Kq', the transfer function becomes

$$\frac{X_L}{I}=\frac{Kq'\beta A/V}{MS^3+(\beta M/RvV)S^2+(K+\beta A^2/V)S+K\beta/RvV} \quad (3)$$

Having an electronic actuator model, its output can be compared to the output of an actuator under test when both are subjected to like commands. Limits are set, within a comparator, to established tolerance levels and a logical pass/fail signal results. Through a variety of test configurations and the resultant data an assessment of the actuator's performance is achieved and determination of the nature of indicated problems is possible.

Conclusions

Since a variation of this concept has already been successfully utilized by the contractor, there should not be any major problems with the PSTS project. Except for a few variations, the PSTS will be a repackaging of existing electronic equipment. Once the PSTS is operational, the Air Force should be able to save large sums of money.

AIAA 82-4086

Wing/Control Surface Flutter Analysis Using Experimentally Corrected Aerodynamics

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Introduction

EXPERIENCE has shown that in most cases where aircraft have encountered flutter problems, control surfaces were involved. For this reason it is important that the wing/control surface/tab be accurately modeled when doing flutter analysis on an aircraft. Problems associated with aerodynamic modeling of control surfaces were discussed by Wassweman et al.[1] as early as 1944 when they suggested reducing the tab aerodynamic coefficients by 30% to account for the poor airflow found over control surface tabs. Therefore it is standard practice when using strip theory aerodynamics to include the variations of control surface aerodynamic coefficients as parameters in the flutter analysis. Considering the improvements that have been made in the area of theoretical unsteady aerodynamics over the last 15 years, should the variations in control surface aerodynamics remain a parameter in the flutter analysis, and if so, how should it be done? Experimental pressure data have been obtained by Hertrich[2,3] for both steady and oscillatory motion for several wing/flap configurations. In using this data to compare experimental and theoretical lifting pressure distribution on an airfoil with oscillating flap in two-dimensional flow, Albano and Rodden[4] indicated that the theory would overpredict the pressure. A similar comparison was made by Tijdeman and Zwaan[5] which also indicated that the pressure over the control was overestimated by theory, but they indicated that the differences are of the same order as for the pressure distributions over wings without control surfaces. Therefore corrections for one mode could be used on another mode under similar conditions: Mach number, frequency, etc. Forsching et al.[6] obtained experimental oscillatory pressure distribution data on a wind-tunnel model similar to one used by Hertrich. A detailed comparison of this data with theoretical pressure distribution was made by LaBarge.[7] He indicated that at the trailing edge the theoretical results tend to overpredict the pressure values, thus modifications to the theoretical pressure values will be necessary when using the predicted values to make hinge moment estimates. The above experimental and theoretical comparisons were reviewed by Ashley and Rodden[8] with similar comments being made on the significant differences that appear between the experimental and theoretical pressure distribution at the trailing edge of the control surface. Rowe et al.[9] indicated that considerable variation in results may be obtained when using the doublet-lattice program to model wing/control surfaces depending upon the method used in defining the control point distribution, but results would approach asymptotic values provided that either a sufficiently large number of control points were used or by using a smaller number of carefully spaced control points. They also indicated that theoretical and experimental data were in good agreement except for a small area around the wing/control surface hingeline. In a paper by Rowe et al.[10] it was found that the theoretical unsteady hinge moment could vary by as much as 20% from experimental results on a wing-aileron-tab configuration.

In each of the above comparisons, the indications are that theoretical unsteady aerodynamics overpredicts the pressure distribution on the wing/control surface interface and the control surface itself. What has not been discussed in these studies is the effect this will have on the predicted flutter speed. If these effects tend to make the flutter analysis unconservative or change the flutter mechanism, then these effects must be included in the flutter analysis in some manner.

Flutter Study

To determine if the above effects are critical to the flutter analysis, an analytical flutter model of the experimental model used by Forsching[6] et al. was developed.

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