CF-105 K. Owen P/STABILITY/101

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ANALYZED

AUTOMATIC STABILITY AND

CONTROL SYSTEM FOR CF-105

March 1956 J. Dobrzanski

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AUTOMATIC STABILITY AND CONTROL SYSTEM FOR CF-105

1. SUMMARY

This report reviews briefly the reasons which lead to the automatic stability and control equipment development in the form in which it is proposed for the CF-105 aircraft. Progress up to date is described in the analysis of the problems involved and solutions obtained and their translation into the hardware design.

Further planning of development program and time schedule is given. It is expected that the ground tests of damping system hardware will begin at Avro in September 1956 and first developmental systems for flight testing will be delivered to Avro in May 1957. Flight test will start in July 1957. Development program should be completed by May 1958.

2. INTRODUCTION

The general survey of the problems of aerodynamic stability of clean airframe was given in Avro report P/Stability/92. The necessity of automatic improvement of flight characteristics of CF-105 was clearly indicated. To recapitulate here briefly: early in the preliminary design phase of CF-105 it was concluded that it is not possible to provide adequate damping for an aircraft so that it would be a satisfactory armament launching platform at supersonic speeds and heights of 60,000 ft. and over and at the same time maintain reasonable performance. This view is generally recognized at present and considered to be fairly independent of any particular design configuration. In this situation it is evident that the aircraft will not be operational in case of failure of electronic damping system equipment. Exploiting this fact fully it was decided to increase the functional scope of the "black boxes", forming the damping system, to include also automatic stability augmentation thus permitting marginal aerodynamic stability design. Such increase in scope calls naturally for much more extensive analytical work and more sophisticated net-work design but has practically no effect on "black boxes" weight or reliability as the basic computing elements and sensors are already there to provide automatic damping. It does not involve any changes to the basic aircraft control system. However, on the other side of the balance sheet, that is dispensing with adequate directional stability by purely aerodynamic means, there are first order effects produced as far as weight and operational manoeuvrability is concerned. This argument is fully presented in the above quoted Avro report and so shall not be repeated here.

Such then is the historical background of CF-105 "black boxes" design philosophy . It has not resulted - as was suggested in some quarters - from wind tunnel test indication of inadequate static directional stability. In fact the fin effectiveness obtained during tests in Cornell Aeronautical Laboratories is very close to the theoretically expected.

2. INTRODUCTION (Continued)

In view of all that was said above it will be readily appreciated that the term "damping system" is something of a misnomer as the system also provides automatic stability augmentation and, as will be described later, incorporates several safety features and completely novel electrical feel system for the manual mode.

3. DYNAMIC STABILITY OF CLEAN AIRFRAME

Since the report P/Stability/92 was written considerable more work was done both analytically and on the analogue computer and therefore some statements concerning the dynamic stability must be brought up to date.

The region considered as manually unflyable without automatic stabilization and damping has decreased in level flight to an area shown on Fig. 1. A similar area expected previously at transonic speeds at low altitudes has disappeared all together. The unflyable area is now defined as one in which the amplitude of any disturbance is doubled in less than 3 seconds. The "Dutch Roll" mode damping characteristics are given on Fig. 2 for level flight and on Fig. 3 for 4 'g' flight. It is very interesting to note that at 4 'g' the aircraft is stable without automatic means throughout the flight envelope in spite of the fact that static directional stability decreases with 'g' and is definitely negative around Mach No. 2 and medium altitudes. Because of this decrease in static stability with 'g' it was previously expected that the unflyable area will enlarge with normal load factor, - as it happens it is non-existent at 4'g'. This fact illustrates very well how unreliable and useless it is to base speculations concerning dynamic stability of supersonic aircraft of modern layout on one or two static derivatives.

From Fig. 4 & 5 it can be seen that considerable region of flight envelope lies in satisfactory region for manual handling as defined by U.S.A.F. Spec. 1815B.

Realizing that the definition of unflyable area as given above is somewhat arbitrary it is intended to actually "fly" on the computer many points within the flight envelope to determine experimentally how realistic it is. So far a few points were "flown" with satisfactory results. The flying is performed through a stick which is provided with correct spring feel as per CF-105 design of the purely mechanical manual mode. The display is in the form of horizontal bar on the cathode ray tube representing aircraft wings and able to roll, sideslip and pitch. Record of such a flight in lateral degrees of freedom is shown on Figure 6. The flight condition is 20,000 ft. and Mach No. 1.5, - as can be seen from Fig. 1 it is very close to the unflyable boundary. It was quite easy after a few seconds of initial learning to keep the sideslip to less than $1/2^{\circ}$.

3. DYNAMIC STABILITY OF CLEAN AIRFRAME (Continued)

The flying on the simulator in pitch degrees of freedom, as was expected, is quite safe in all conditions of flight, there always being adequate dynamic stability. Damping is poor at altitudes of over 40,000 ft. and therefore a practical pilot technique is to move stick slowly and not to try and compensate for poorly damped oscillations as the period is too short for human response characteristics.

4. DAMPING SYSTEM FUNCTIONAL OBJECTIVES

As was already mentioned earlier under the currently used term "damping system" many other functions are included which are in no way related to damping. A very complete description of the functions to be provided by the damping system and its performance is given in the "Requirements for CF-105 Damping System and Associated Equipment" Issue 4 dated November 7th, 1955, See Appendix 1.
This document forms the basis of a contractual agreement between Avro and Hughes Aircraft Company for the design and development of the damping system.

5. SYSTEM STUDY

The study of the overall system stability and component requirements began in September 1953. In November 1953 first two units of Boeing Electronic Analogue Computer were installed and from that date the study was greatly accelerated by the extensive use of this facility.

As a result of this early phase of analytical and simulation work all essential characteristics of the electro-hydraulic and hydraulic components of the control system were specified to the Design Office before the actual design stated. These included maximum effort, maximum rate and frequency response. As design proceeded and various components were manufactured, they were tested in the Structural Test Dept., and tests results analyzed in the Aerodynamics Dept. Servo units tested were developed to the point of meeting frequency response requirements. Main elevator hydraulic jack phase lag is better than specified. Hydraulic jacks for ailerons and rudder were not tested so far but being smaller units than the elevator jack no difficulty in response is expected. From tests results actual analytical transfer functions for various components were obtained and are at present used on the computer. A check of the validity of results obtained from computer was performed recently. In this test one computation was performed using entirely computer simulation of aerodynamics, sensors, damping system and controls. Another computation was performed using exactly the same computer simulation with the exception of control system which was replaced by actual hardware from servos to control motion. Figure 7 shows a comparison of relevant variables. In both cases the same input of sharp edged gust was used. Figure 8 shows a similar comparison for pilot input. It can be seen that the agreement is excellent thus establishing confidence in computer results.

6. DAMPING SYSTEM DESIGN

The block diagram of the damping system is given by Figure 9. The essential feature of this design is the provision of two independent inputs into the hydraulic actuator: the parallel and the differential servo, the inputs from which are summed at the jack valve. Command outputs from the pilot or from the A.F.C.S., come into the parallel servo which moves at one end the control surface and at the other the stick. Independently differential servo is smoothing the response of the aircraft regardless of whether it originates from intentional command or random gust. The pilot at the stick is unaware of the differential servo movements (and associated control movements). As far as he is concerned he is flying an aircraft of excellent aerodynamic characteristics which do not deteriorate with either speed or altitude.

The signals actuating the differential servos are derived from aerodynamic feedbacks (such as angular rates, accelerations etc.) as sensed by the sensing elements (gyros etc.) and suitably shaped by the scheduling and compensating networks (the actual damping system "black boxes"). So far two distinct feedback systems were analyzed in fair detail:

(i) purely rate damping system using only gyros as sensing elements.

(ii) rate and acceleration system using gyros, accelerometers and sideslip weather vane as sensing elements.

The second system is by far superior as far as performance is concerned and is the one being now developed.

Very special feature of this design is the requirement (See App. 1. Para. 4.3.5) for ± 50% change in any loop gain (one at a time) and any time constant in compensating networks without causing a dangerous deterioration in the aircraft response. Such wide tolerance in damping system parameters has several important advantages:

(i) overall system performance will not be too sensitive to reasonable variations in control system due to manufacturing characteristics and airframe tolerances. Thus no individual adjustment of damping box from aircraft to aircraft should be necessary.

(ii) scheduling and compensating functions can be compromised without deterioration of overall performance. Simplifying these functions and limiting the number of scheduling parameters will result in simpler and therefore more rugged and reliable system.

(iii) finally the design will be less affected by any misestimates of the aerodynamic coefficients. This has obvious advantages of safer and shorter flight test development period.

6. DAMPING SYSTEM DESIGN (Continued)

Many doubts were voiced regarding the feasibility of this requirement. However, it has been shown recently that with a sideslip feedback it can be achieved not only without causing a dangerous condition but actually with very little deterioration in performance from the optimum. This is illustrated by Fig. 10 for Mach. No. 2.0 and altitude 30,000 ft. - a most critical condition of the whole flight envelope as can be seen by the extreme rate of divergence of clean airframe shown for comparison.

When on manual mode (with normal damping system in operation) pilot flies through control stick steering. This consists of electrical signal proportional to the pilot force being compared with normal acceleration in case of pitch axis and rate of roll in case of roll axis. The resultant signal is fed into the parallel servo. Thus by closing a feedback loop around the final aerodynamic results of pilot action (See Fig. 8) a feel characteristic is produced which gives constant stick force per "g" and constant stick force per unit rate of roll independent of flight condition. This is achieved by extremely simple means and the actual level of stick for per "g" can be altered by mere adjustment of the gain potentiometer. To approach such feel characteristics by mechanical means would lead to a very complicated design.

7. RELIABILITY

The damping system being an essential element of the aircraft operational capability every effort is made to achieve as high reliability as is possible at the present state of the art. Weight and expense considerations are to be subordinated to reliability.

Detail electrical circuitry arrangements to improve reliability are not yet know as the system just enters the hardware design stage at Hughes Aircraft Co. A general requirement dictated by reliability aspects is that only magnetic amplifiers shall be used.

8. SAFETY PROVISIONS

These are described in detail in Appendix 1, para 3.3, 3.5 and 3.6.
In general provisions are made to safeguard against the malfunction of equipment. There are no provisions for preventing the pilot from overloading the airplane in pitch save for high stick forces. In the roll axis, however, there is a command signal limiter even on the manual mode.

The yaw axis of the airplane being the most critical there is a complete duplication of the yaw axis of the damping system including sensors, computing and hydraulic servos. The duplicate yaw axis is called the emergency damping system. The switch over from normal to emergency in case of detected malfunction is automatic. The sensing element used is the sideslip vane. It is therefore necessary for a double failure to occur before the pilot is left without any artificial damping and stability.

8. SAFETY PROVISIONS (Continued)

Besides having a switch to switch off the entire damping system the pilot can, in emergency, override the parallel servo by exerting sufficiently high force.

9. MAINTENANCE EQUIPMENT

This equipment is developed concurrently with the damping system at H.A.C. and will consist of:

(i) closed loop overall tester which will by a fast simple test either pass or reject the whole system.

(ii) unit testers of simple "go", "no go" variety which will be used to test in detail every component.

10. DEVELOPMENT AND HARDWARE EVALUATION PROGRAM

The development and evaluation will be accomplished in several distinct phases:

(i) analytical and computer simulation leading to detail specifications for the design. At present some 50% completed.

(ii) aerodynamics, sensors, computing networks simulated on the computer with actual hydraulics of the control system rig tied in to the computer. Control surface deflections, rates and accelerations fed back to the computer. At present only a few preliminary tests of this type were run with satisfactory results. This phase expected to start in July 1956.

(iii) all elements simulated on the computer with the exception of damping system computing and scheduling unit. The actual unit as obtained from Hughes will be used in this phase. Provisions will be made to power this unit from the mock-up of the aircraft electrical power supply situated in the Production Division. Effect of transients in the power supply (e.g. icing loads) will be investigated. This phase is expected to start in September, 1956.

(iv) in this phase fully representative cockpit (near the rig) cabling and hydraulic equipment will be tested together with the analogue computer. The only elements left on the computer will be aerodynamics and sensors. Pilot display will be transmitted to the rig and the pilot will be able to "fly" the airplane using all actual control components. The simulation of both the manual and emergency made will be possible. A standard panel of instruments is now developed to respond to aircraft variables generated on the computer. New ideas in cockpit display such as combining 3 degrees of freedom in one instrument will also be evaluated.

10. DEVELOPMENT AND HARDWARE EVALUATION PROGRAM (Continued)

In this phase following conditions will be evaluated:

(a) Simultaneous demands to all controls (any necessity of policing will be evident).

(b) Aerodynamic loads simulated on all controls.

(c) Engine r.p.m. for:-

- full power
- half power
- idle

(d) One engine out case.

(e) One hydraulic system out case.

(f) Ambient temperature variation.

11. TIME SCHEDULE FOR DEVELOPMENT OF THE DAMPING SYSTEM

1 January, 1956	-	Contractual coverage given to Hughes Aircraft Company - Design work started at full effort.
1 June, 1956	-	Preliminary installation data.
1 July, 1956	-	Agreed upon equipment specification for development systems.
	-	Agreed upon characteristics of special development test equipment.
1 September, 1956	-	Delivery of simulation hardware and operational characteristics.
1 December, 1956	-	Supplementary installation data.
1 May, 1957	-	Delivery of first flight test system.
1 June, 1957	-	Delivery of special development test equipment.
1 July, 1957	-	Delivery of second flight test system.
	-	Start of flight test by Avro with Hughes Aircraft engineers participating in the analysis of test results conducted at Avro.
1 August, 1957	-	Delivery of flight test back-up system plus additional spares.
1 January, 1958	-	Agreed upon equipment specification and design freeze for pre-production systems.
1 April, 1958	-	Delivery of pre-production drawings.
1 May, 1958	-	Conclusion of damping system developmental flight test program.
1 December, 1958	-	Completion of Final Engineering Report.

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Jan 1956

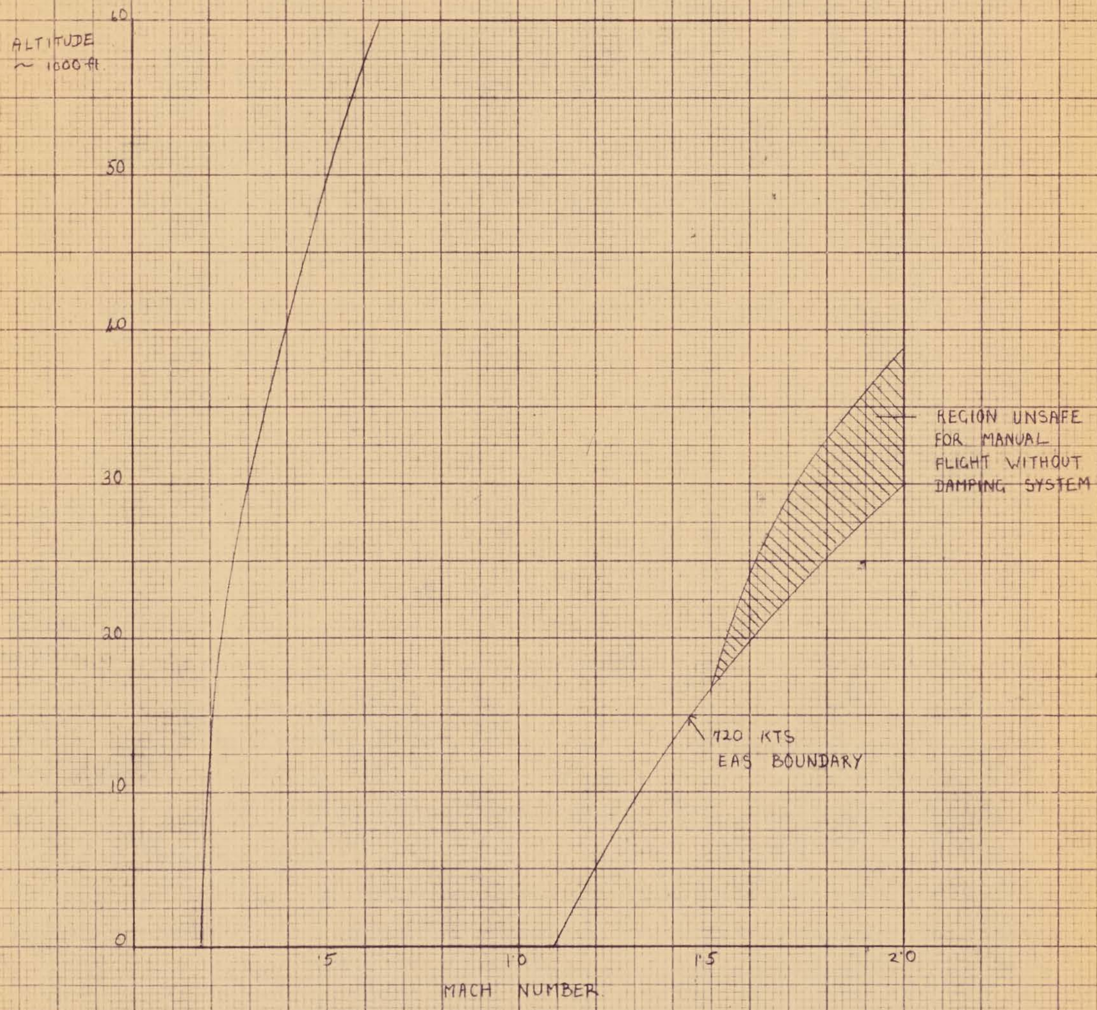
J. Rose

1.6

C-105

SAFETY BOUNDARY FOR DUTCH ROLL OSCILLATION

WING - ELASTIC, NOTCHED, EXTENDED + DROOPED
C.G. = ·31 c̄
WEIGHT - 47,000 lb



C_{Y_n}

$V_e \frac{\theta}{\sqrt{\beta}}$

FIG. 1

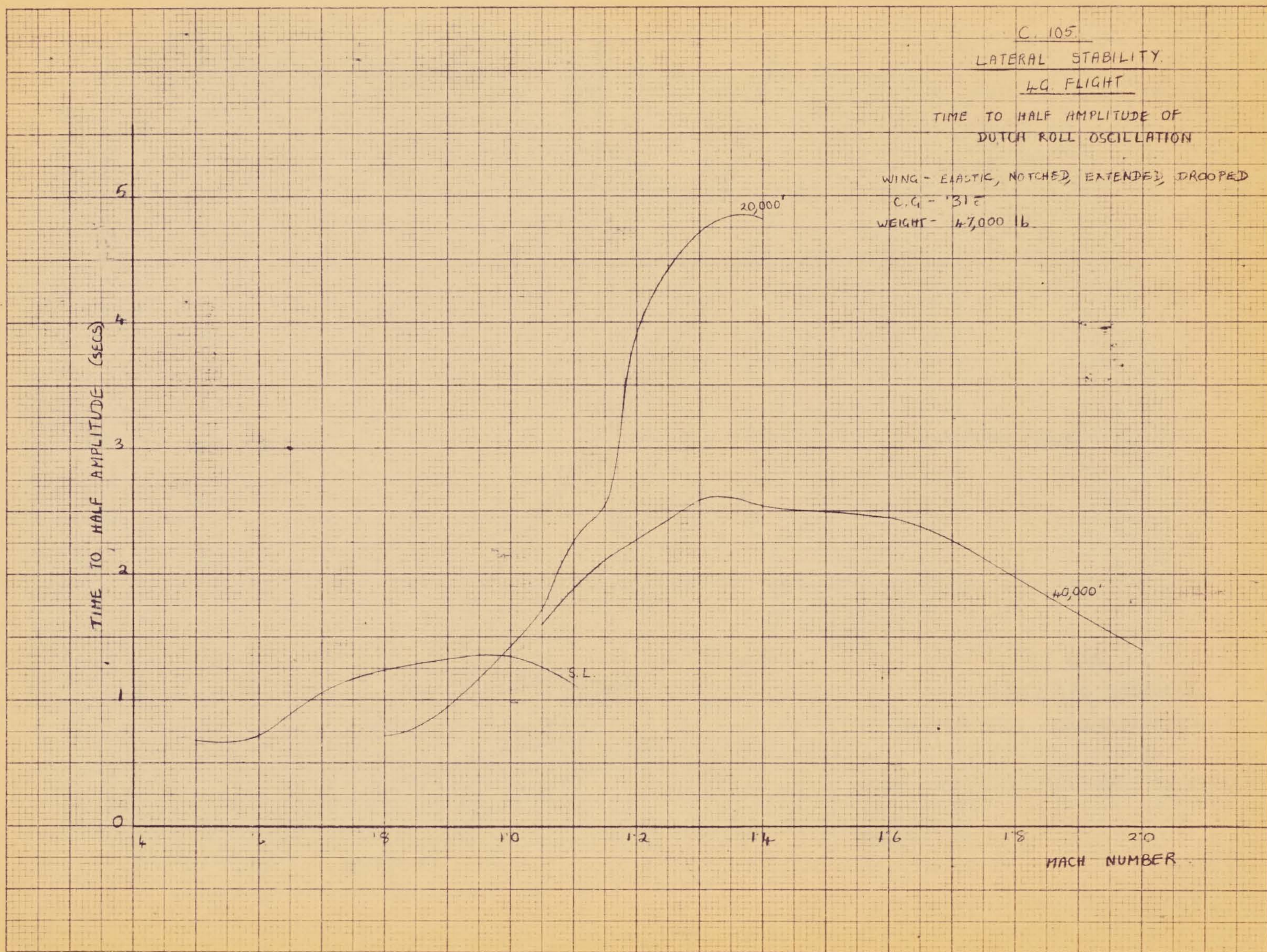


FIG. 3

P/Stability/100
March '56

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K+E 10 X 10 TO THE ½ INCH 359-12 KEUFFEL & ESSER CO. MADE IN U.S.A.

P/Stability/96.

Jan 1956

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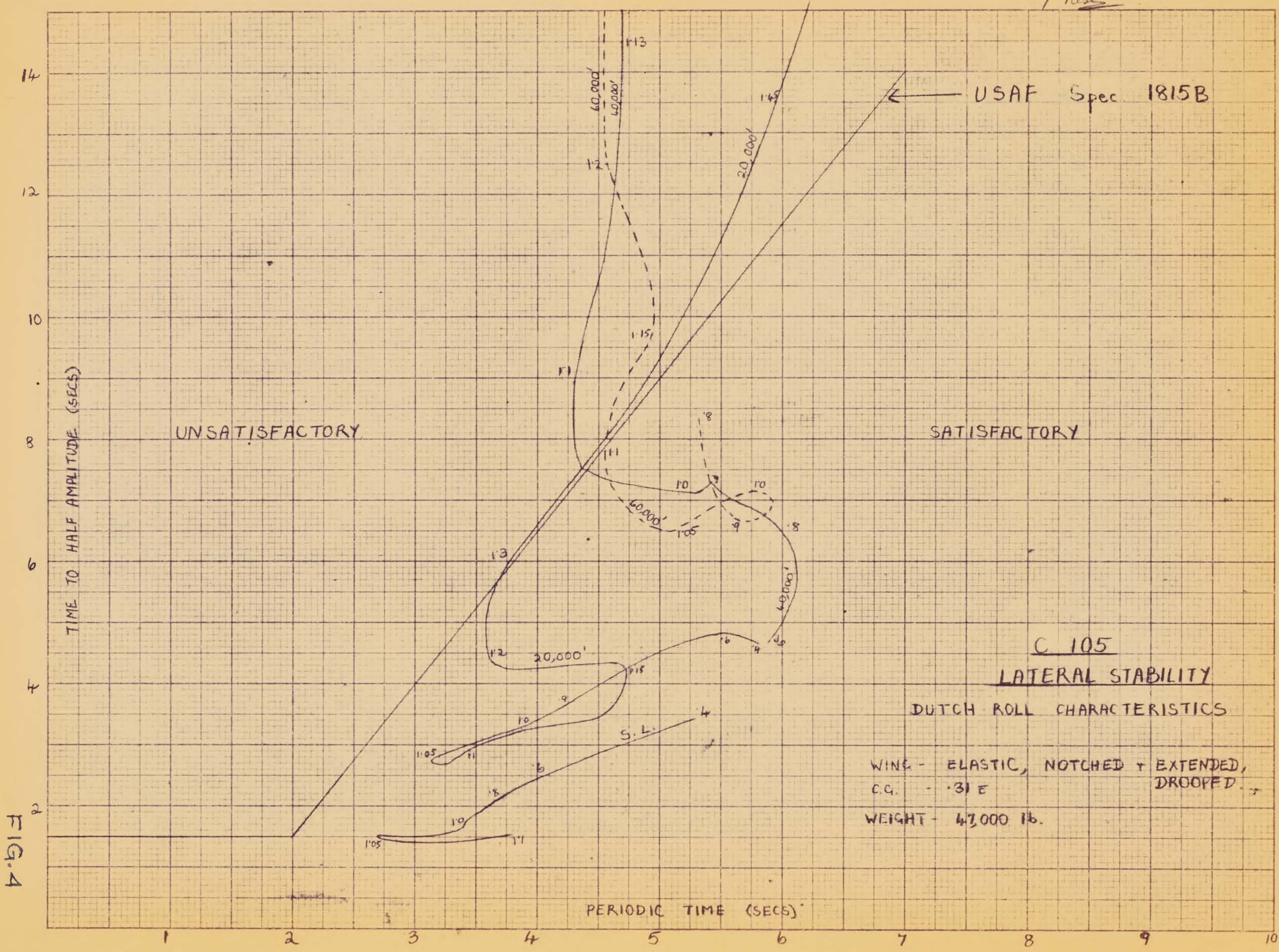
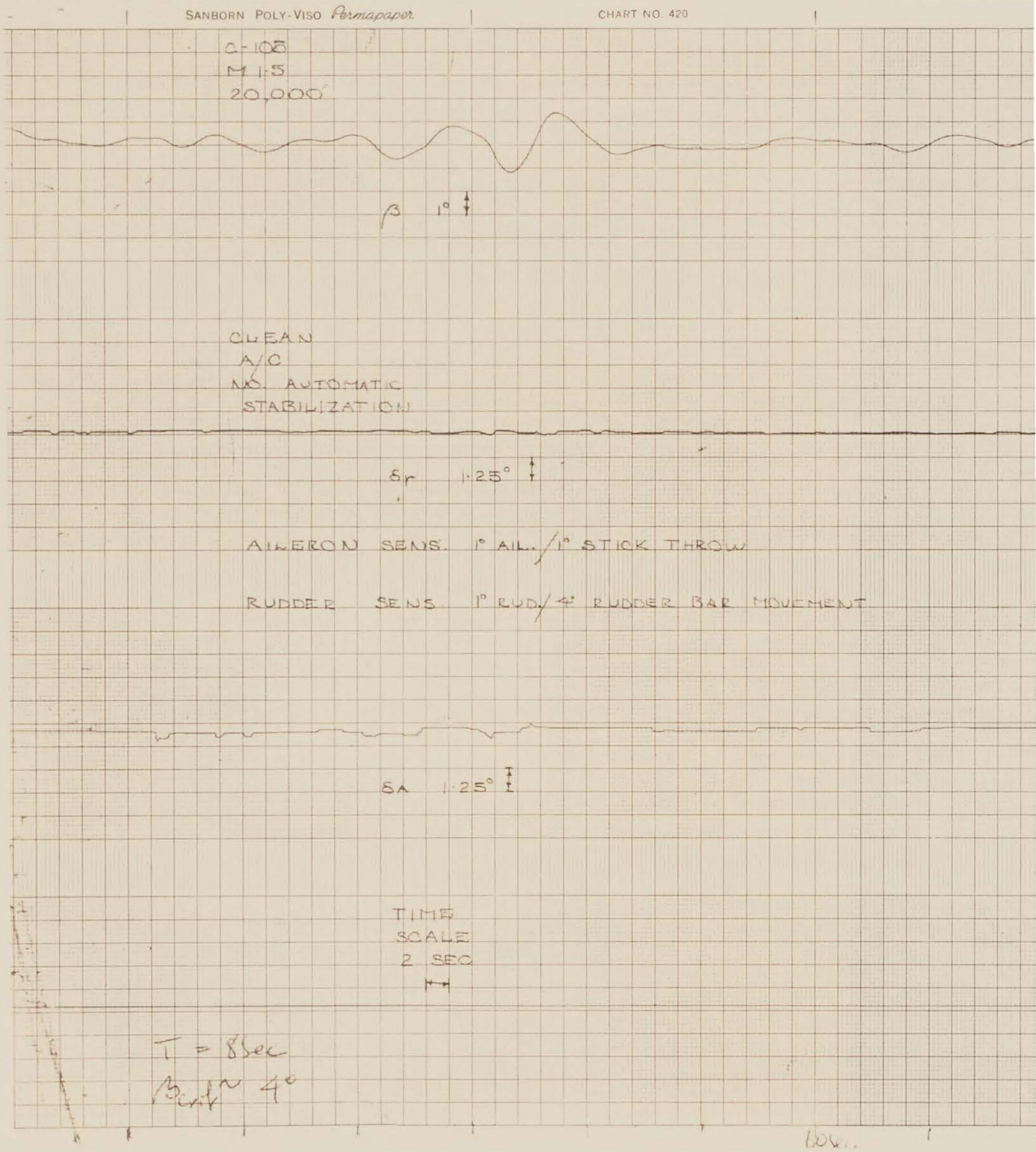


FIG. 4

4.1


SANBORN POLY-VISO Permapaper
CHART NO. 420
C-105
M 1.5
20,000'
β 1°
CLEAN
A/C
NO AUTOMATIC
STABILIZATION
δr 1.25°
AILERON SENS. 1° AIL./1° STICK THROW
RUDDER SENS. 1° RUD/4° RUDDER BAR MOVEMENT
δa 1.25°
TIME
SCALE
2 SEC
T = 8sec
βcrit ~ 4°

Dow.

A. V. ROE CANADA LIMITED
MALTON ONTARIO
TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT		REPORT NO.	
		SHEET NO.	
		PREPARED BY J. H. Jo...ski	DATE March 20, '56
		CHECKED BY	DATE

BLOCK DIAGRAM OF THE DAMPING SYSTEM.

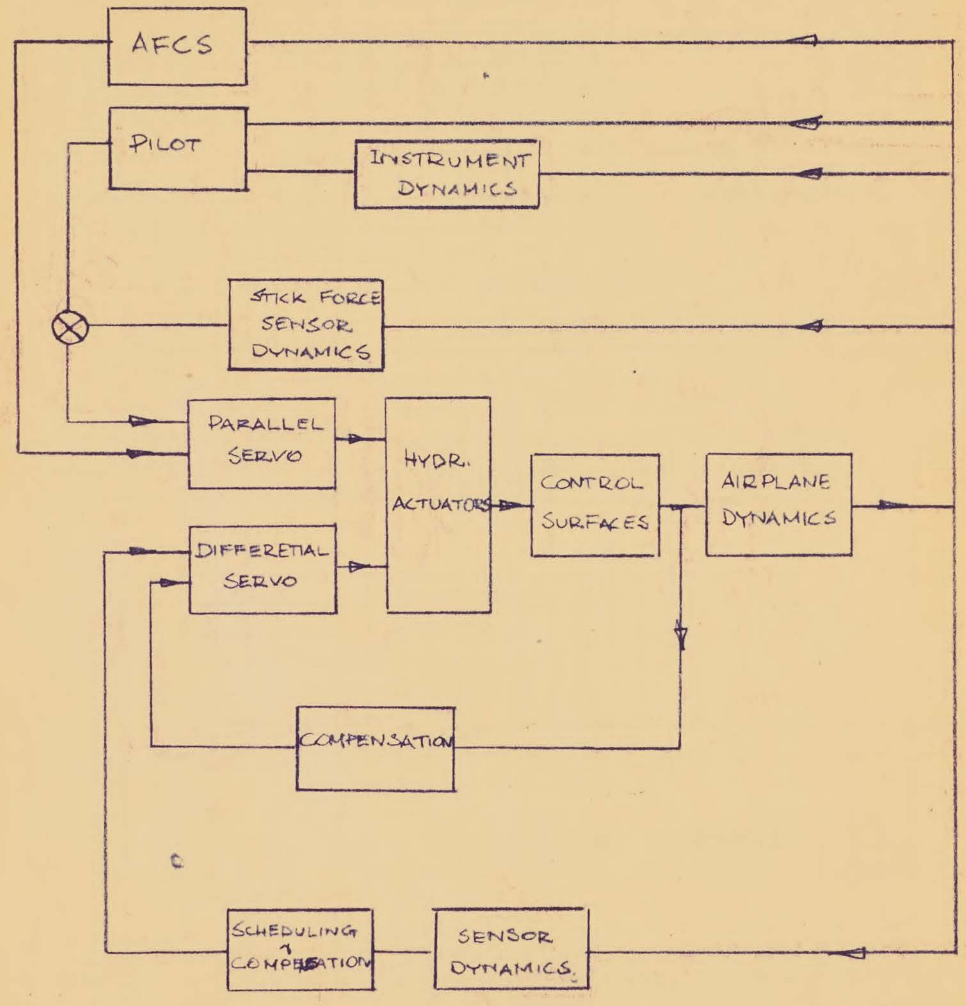


FIG. 9

AVRO AIRCRAFT LIMITED

REQUIREMENTS FOR CF-105 DAMPING SYSTEM AND ASSOCIATED EQUIPMENT

November 7th, 1955.

1. INTRODUCTION

The damping system, as defined in this statement of requirements, when acting in conjunction with certain airframe components, is intended to improve the flying qualities of the aircraft to a point where the aircraft is easily controllable by the pilot under all conditions of flight, and is to provide the means by which the pilot controls the aircraft. The system must be so designed as to operate in conjunction with an automatic flight control system which is integrated with and essentially part of a Hughes MX-1179 or MA-1 type of integrated interceptor electronic system. Sections 1, 2 and 3 describe functionally the damping system and associated equipment covering the whole system, parts of which is AVRO responsibility and parts of which is HAC responsibility. Detail responsibilities for the various items that integrate into the damping system are defined in Section 4 for the HAC and Section 5 for AVRO Aircraft.

2. DAMPING SYSTEM CONFIGURATION

The damping system shall be comprised of:

2.1 Normal damping system

2.2 Emergency damping system

2.3 Switching system

2.4 Electrical control stick steering

2.5 Safety cut-out devices

2.6 Roll rate command signal limiter

3. FUNCTIONAL REQUIREMENTS OF DAMPING SYSTEM

3.1 Normal Damping System

3.1.1 To damp the short period oscillations about all three axes. The damping ratios and periods shall meet USAF Spec. 1815-b.

3.1.2 To damp the longitudinal long period oscillations, provided this can be accomplished without excessive complication. This shall be further studied and final determination shall be subject to mutual agreement.

3.1.3 To provide spiral stability such that the spiral mode of the augmented airframe shall not diverge at a rate greater than double amplitude in twenty seconds.

3.1 Normal Damping System (Continued)

3.1.4 To provide turn coordination and sideslip minimization in any operational manoeuvres up to + 6g in pull-outs and + 4g in turns and negatively to - 2g, such that:

3.1.4.1 50% of structural integrity limit on sideslip shall not be exceeded (this is a design objective).

3.1.4.2 The steady state sideslip shall be less than or equal to 2.5°.

3.1.4.3 Fire Control System requirements shall be met wherever more restrictive than 3.1.4.1 and 3.1.4.2.

3.1.5 Further study may indicate that for reasons of compatibility with the requirements of 3.5.1 it will be necessary to modify the above limits of 3.1.4. To the extent that this is necessary the requirements of 3.5.1 as effected by a simple mechanical "g" cut-out (i.e. without anticipation) shall prevail.

3.1.6 The artificial damping features shall not materially deteriorate the roll response characteristics of the basic aircraft.

3.1.7 To provide a means of manual control which is also suitable for accepting flight commands from the electrical control stick steering and the automatic flight control system.

3.1.8 To provide for uncoordinated manoeuvres at the option of the pilot. This shall be implemented by a coordination cut-out switch on the rudder bar.

3.2 Emergency Damping System

3.2.1 The critical dynamic characteristics of the yaw axis of the basic airframe make it mandatory that there be two independent damping systems on the yaw axis only; one being the normal system, and the other the emergency system. These systems shall be independent of one another; i.e., both systems will be provided with separate transducers, equalization, hydraulic power supply, and electrical power supply.

3.2.2 To insure maximum reliability, the emergency damping system shall be operated from the aircraft power supply.

3.2.3 The emergency damping system shall provide stability and damping of the Dutch Roll mode.

3.2.4 50% of structural integrity limit on sideslip shall not be exceeded in any pull-out or turn 2g manoeuvre.

3.2.5 To provide for uncoordinated manoeuvres at the option of the pilot. This shall be implemented by a coordination cut-out switch on the rudder bar.

3.3 Switching System

3.3.1 When the normal damping system fails to meet the requirement of 3.1.4.1, the emergency damping system shall be engaged and the normal damping system shall be disengaged. This shall be implemented by automatic switching to the emergency mode. See 3.4.1.2 and 3.2.

3.3.2 The command signal to actuate the switch described in 3.3.1 shall be a combination of signals originating from a sideslip sensor and other sources as required to provide adequate anticipation.

3.3.3 The automatic switch shall be a fail safe switch. The definition of "fail safe" shall be mutually agreed upon by AVRO and HAC at a later date.

3.3.4 The pilot shall have the option of re-engaging the normal damping system through the emergency and manual mode selector switch, see 3.4.3.

3.4 Electrical Control Stick Steering

3.4.1 There shall be two modes of flying through the stick:

3.4.1.1 Manual mode in which the inputs into the parallel servos shall be signals proportional to stick forces and normal acceleration and rate of roll. Normal damping system is engaged.

3.4.1.2 Emergency mode in which stick motion is mechanically linked to the control valves of the main surface actuators. The differential servos of the ailerons and elevators shall be centred, and the corresponding parallel servos shall be disengaged. In this mode, only emergency yaw axis damping is retained.

3.4.2 The stick force transducers shall be of either strain gauge or linear differential transformer type. The damping system shall accept signals from a pitch axis stick force transducer which shall be located aft of the bobweight in one of the control links. The damping system shall also accept signals from a roll axis stick force transducer.

3.4.3 There shall be a switch which shall be used to select either the emergency mode or the manual mode of control.

3.5 Safety Cut-Out Devices

3.5.1 There shall be a mechanical "g" limiting device which will disengage the pitch axis of the normal damping system so that the structural integrity limit of + 7.3g and - 3.4g on the normal load factor shall not be exceeded. This shall be accomplished in a fail safe fashion. The definition of "fail safe" shall be mutually agreed upon by AVRO and HAC at a later date.

3.5 Safety Cut-Out Devices (Continued)

3.5.1.1 The disengagement of the pitch axis shall be implemented by the mechanical recentering of the elevator differential servos and disengagement of the elevator parallel servo. The signal for disengagement shall be an electrical signal.

3.5.1.2 Simultaneously with the disengagement of the parallel servo the trim feel system shall be engaged. The signal for engagement of the trim feel system shall be an electrical signal.

3.5.1.3 The pilot shall have the option of re-engaging the pitch axis of the normal damping system.

3.5.2 Roll Rate

There shall be a mechanical roll rate limiting device which will disengage the aileron axis of the normal damping system so that the structural integrity limit on the rate of roll of 200 deg. per second shall not be exceeded. This shall be accomplished in a fail safe fashion. The definition of "fail safe" shall be mutually agreed upon by AVRO and HAC at a later date.

3.5.2.1 The disengagement of the aileron axis shall be implemented by the mechanical recentering of the aileron differential servos and disengagement of the aileron parallel servo. The signal for disengagement and recentering shall be an electrical signal.

3.5.2.2 Simultaneously, with the disengagement of the parallel servo the trim feel system shall be engaged. The signal for engagement of the trim feel system shall be an electrical signal.

3.5.2.3 The pilot shall have the option of re-engaging the roll axis of the normal damping system.

3.6 Roll Rate Command Signal Limiter

3.6.1 The stick force command signal to the aileron parallel servos shall be limited in such a fashion that the pilot may not request roll rates in excess of 180°/sec.

4. DAMPING SYSTEM ITEMS WHICH SHALL BE PROVIDED BY HUGHES AIRCRAFT COMPANY

4.1 Amplifiers

4.1.1 Amplifiers for both differential and parallel servos.

4.1.2 Amplifiers for the α, β sensor signal. Pick-offs may be required for other sub-systems, which shall be mutually agreed to by AVRO and HAC.

4.1 Amplifiers (Continued)

4.1.3 Amplifiers for air data pressure transducer signals. Pick-offs may be required for other sub-systems, which shall be mutually agreed to by AVRO and HAC.

4.2 Sensors

4.2.1 Gyros and accelerometers as required for the damping system. Pick-offs may be required for other sub-systems, which shall be mutually agreed to by AVRO and HAC.

4.2.2 Pressure transducers for air data source.

4.3 Computing and Scheduling Circuitry

4.3.1 The normal damping system should have separate and self-contained computing and scheduling circuitry; however, a possibility and advantages, if any, of using facilities of the Integrated Electronic and Control System shall be studied.

4.3.1.1 If this scheduling relies on the facilities of the Integrated Electronic & Control System then in the interim period during which the Integrated Electronic & Control System is not available substitute scheduling facilities shall be provided. This shall functionally approximate the scheduling of the normal damping system.

4.3.2 Provisions shall be made for accepting flight commands from the electric control stick and the automatic flight control system.

4.3.3 The emergency damping system shall have separate and self-contained computing and scheduling circuitry.

4.3.4 The normal and emergency damping systems shall be so designed that when all the deviations due to manufacturing tolerances of the components (as defined in Mil E-5400) are considered in the most unfavorable combination, the performance requirements of Section 3 shall be met.

4.3.5 The normal and emergency damping system shall be so designed that a 50% change in any damping system functional gain (and time constant as a design objective) from its design value, with other parameters at their design value, shall not result in an unstable condition (excluding phugoid) of the augmented airframe.

4.4 Power Supplies

4.4.1 It shall be a design objective that the normal damping system shall operate from either the aircraft power supply, or from the integrated electronic power supply, except that when operating from the aircraft power supply it shall not be required to accept control signals from the automatic flight control system.

4.4 Power Supplies (Continued)

4.4.1.1 In, the interim period during which the integrated electronic power supply is not available substitute units approximating the final unit shall be provided.

4.4.2 The emergency damping system shall be operated from the aircraft power supply.

4.5 Pilot Operated Switches

4.5.1 Main standby, operate and "off" switch which energizes the entire damping system.

4.5.2 Emergency and Manual Mode selector switch (including the automatic manual to emergency switch provision, See 3.3.1).

4.5.3 Re-engage switch for pitch axis safety cut-out.

4.5.4 Re-engage switch for roll axis safety cut-out.

4.6 Safety Cut-out Devices

4.6.1 Normal acceleration cut-out device.

4.6.2 Rate of roll cut-out device.

4.6.3 The electrical signals to implement hydraulically cut-out action as described in 3.5.1.1, 3.5.1.2, 3.5.2.1 and 3.5.2.2.

4.7 Command Signal Limiter

4.7.1 Rate of roll command signal limiter.

4.8 Switching from Normal to Emergency Damping

4.8.1 The electrical signals to implement this switching action as described in 3.4.1.2, 3.3.1 and 4.5.2.

5. DAMPING SYSTEM AND ASSOCIATED ITEMS SUPPLIED BY THE AIRFRAME MANUFACTURER WHICH SHALL NOT BE COVERED BY THE DAMPING SYSTEM CONTRACT

The detailed specifications of those items affecting damping system performance shall be arrived at by a joint AVRO-HAC effort.

5.1 Hydraulic system comprising pumps, reservoirs, actuators, and control valves.

5.2 Electro-hydraulic servos actuating the main hydraulic valves (both differential and parallel servos), including the centering and disengaging.

5.3 Control Stick Feel System.

5.4 Control Stick Trim System.

5.5 Air Data Pressure Sources

5.6 α , β sensor with variable reluctance type linear differential transformer output.

5.7 Stick force transducers which shall be of either strain gauge type or differential transformer type.

5.8 Rudder bar coordination cut-out switch.

5.9 Control surface transducers for the damping system and ground testing equipment.