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INVESTIGATION OF LATERAL STABILITY AT STALL

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CORNELL AERONAUTICAL LABORATORY, INC.

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FOREWORD

This report was prepared by Gifford Bull of the Flight Research Department, Cornell Aeronautical Laboratory, Buffalo, New York, under Contract No. AF 33(038)-10480, RDO No. 472-106, "Supporting Research on Aircraft Guidance Systems." The work was administered under the direction of the Aeronautical Research Laboratory, Wright Air Development Center, with Mr. L. S. Wasserman and Mr. F. L. Daum acting as project engineers.

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Contrails

ABSTRACT

The investigation of the stability and control characteristics of a stalled airplane was continued to include the case of the lateral motion.

An automatic control was developed which successfully stabilized the stalled airplane, effectively preventing the rolling off and spinning which are characteristic of a normal, unstabilized, stalled airplane. The response of the stalled airplane to transient inputs from the rudder or ailerons was measured. The frequency response characteristics of the pertinent responses to aileron and rudder inputs in fully stalled flight were determined.

The data presented includes the recorded time histories of control inputs and airplane motions at various fully stalled angles of attack, and the experimentally determined frequency response characteristics at 28° angle of attack.

PUBLICATION REVIEW

This report has been reviewed and is approved.

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TABLE OF CONTENTS

LIST OF SYMBOLS x

SUMMARY 1

INTRODUCTION 2

DESCRIPTION OF THE TEST AIRPLANE 6

INSTRUMENTATION 8

- Automatic Control 8
- Signal Sources for the Automatic Control 11
- Recording Instrumentation 14
- Signal Sources for Recording Instrumentation 14
- Instrumentation Power Supply 16

FLIGHT TESTS 18

- Development of Automatic Control 18
- Data Flight Technique 25

TREATMENT OF THE DATA 28

- Method of Attack 28
- Oscillograph Records 29
- Analysis of Data 30
- Calculated Airplane Response 33
- Separation of the Effects of the Ailerons and Rudder 33
- Discussion of the Plots of the Response of the Airplane to a Single Control Motion 35
- Discussion of Validity of Data 37
- Autopilot Adjustment Using Experimental Derivatives 40

CONCLUSIONS 41

RECOMMENDATIONS 42

REFERENCES 43

Contrails

LIST OF TABLES

TABLE	TITLE	PAGE
I	Dimensions and Leading Particulars for PT-26 NX-69593..	44
II	Autopilot Sensitivities Used in Data Flights	46
III	Values of Airplane Parameters Used in Calculations.....	46
IV	Summary of Flights	47

Contrails

LIST OF FIGURES

FIGURE	TITLE	PAGE
1	Test Airplane	49
2	Three View of Test Airplane	50
3	Block Diagram of Automatic Control System	51
4	Frequency Response of Autopilot	52
5	Frequency Response of Roll and Yaw Rate Gyros	53
6	Static Calibration of Roll and Yaw Rate Gyros	54
7	Harmonic Content of Variables - Aileron Disturbance	55
8	Harmonic Content of Variables - Rudder Disturbance	56
9	ϕ/δ_a With Autopilot Operating Both Controls - Rudder Disturbance	57
10	ϕ/δ_a With Autopilot Operating Both Controls - Aileron Disturbance	58
11	r/δ_r With Autopilot Operating Both Controls - Rudder Disturbance	59
12	r/δ_r With Autopilot Operating Both Controls - Aileron Disturbance	60
13	Ratio p/ϕ - Flt. 105	61
14	ϕ/δ_a for Four Angles of Attack - Aileron Disturbance	62
15	ϕ/δ_a for Four Angles of Attack - Rudder Disturbance	63
16	r/δ_r for Four Angles of Attack - Aileron Disturbance	64
17	r/δ_r for Four Angles of Attack - Rudder Disturbance	65

Contrails

FIGURE	TITLE	PAGE
18	Calculated and Measured r/δ_r with Autopilot Operating – Amplitude Ratio	66
19	Calculated and Measured r/δ_r with Autopilot Operating – Phase	67
20	ϕ/δ_a with no Rudder Motion	68
21	r/δ_a with no Rudder Motion	69
22	β/δ_a with no Rudder Motion	70
23	ϕ/δ_r with no Aileron Motion	71
24	r/δ_r with no Aileron Motion	72
25	β/δ_r with no Aileron Motion	73
26	Lift Curve of Test Airplane, Determined in Flight	74
27	Elevator-to-Trim Curve, Determined in Flight	75

OSCILLOGRAPH RECORDS

RESPONSE OF AUTOMATICALLY CONTROLLED AIRPLANE TO A TRANSIENT DISTURBANCE. BOTH CONTROLS OPERATED BY AUTOPILOT. DISTURBANCE AMPLITUDE IN TERMS OF DISTURBANCE POTENTIOMETER SETTING.

28	α =28° Left Aileron Disturbance-Disturbance Amplitude .23	76
29	α =28° Left Aileron Disturbance-Disturbance Amplitude .30	77
30	α =28° Left Aileron Disturbance-Disturbance Amplitude .30	78
31	α =28° Left Aileron Disturbance-Disturbance Amplitude .30	79
32	α =28° Left Aileron Disturbance-Disturbance Amplitude .30	80
33	α =28° Left Aileron Disturbance-Disturbance Amplitude .30	81
34	α =28° Left Aileron Disturbance-Disturbance Amplitude .30	82

Contrails

FIGURE	TITLE	PAGE
35	α =28° Left Aileron Disturbance-Disturbance Amplitude .30.....	83
36	α =28° Left Aileron Disturbance-Disturbance Amplitude .40.....	84
37	α =28° Left Aileron Disturbance-Disturbance Amplitude .40.....	85
38	α =28° Left Aileron Disturbance-Disturbance Amplitude .40	86
39	α =28° Left Aileron Disturbance-Disturbance Amplitude .60.....	87
40	α =28° Left Rudder Disturbance-Disturbance Amplitude .40.....	88
41	α =28° Left Rudder Disturbance-Disturbance Amplitude .50.....	89
42	α =28° Left Rudder Disturbance-Disturbance Amplitude .53.....	90
43	α =28° Left Rudder Disturbance Disturbance Amplitude .60.....	91
44	α =28° Left Rudder Disturbance-Disturbance Amplitude .60.....	92
45	α =28° Left Rudder Disturbance-Disturbance Amplitude .70.....	93
46	α =28° Right Rudder Disturbance-Disturbance Amplitude .70.....	94
47	α =28° Left Rudder Disturbance-Disturbance Amplitude .70.....	95
48	α =28° Left Rudder Disturbance-Disturbance Amplitude .80.....	96
49	α =28° Left Rudder Disturbance-Disturbance Amplitude .80.....	97
50	α =28° Left Rudder Disturbance-Disturbance Amplitude .80.....	98
51	α =28° Left Rudder Disturbance-Disturbance Amplitude .90.....	99
52	α =24° Left Rudder Disturbance-Disturbance Amplitude .70.....	100
53	α =24° Right Rudder Disturbance-Disturbance Amplitude .70.....	101
54	α =24° Left Aileron Disturbance-Disturbance Amplitude .23.....	102
55	α =24° Right Aileron Disturbance-Disturbance Amplitude .23.....	103
56	α =20° Left Rudder Disturbance-Disturbance Amplitude .70.....	104

Contrails

FIGURE	TITLE	PAGE
57	α =20° Right Rudder Disturbance-Disturbance Amplitude .70....	105
58	α =20° Right Aileron Disturbance-Disturbance Amplitude .23...	106
59	α =18° Left Rudder Disturbance-Disturbance Amplitude .80.....	107
60	α =18° Right Rudder Disturbance-Disturbance Amplitude .80....	108
61	α =18° Right Aileron Disturbance-Disturbance Amplitude .23...	109
62	α =28° Manually Controlled Stall-Disturbance Applied by Pilot.....	110

Contrails

LIST OF SYMBOLS

α	angle of attack, degrees
β	sideslip angle, degrees
δ_e	elevator angle
δ_a	aileron angle (sum of both left and right aileron deflections)
δ_r	rudder angle
ϕ	roll angle, degrees or radians
p	roll rate, degrees/second or radians/second
r	yaw rate, degrees/second or radians/second
ω	frequency, radians/second
cps	cycles/second

Contrails

SUMMARY

The investigation of the stability and control characteristics of a stalled airplane was continued to include the case of the lateral motion.

An automatic control was developed which successfully stabilized the stalled airplane, effectively preventing the rolling off and spinning which are characteristic of a normal, unstabilized, stalled airplane. This automatic control may be found useful in tactical airplanes, allowing, for example, utilization of the maximum lift coefficient of the airplane in approaches and landings, and permitting rapid descents without picking up excessive speed.

The response of the stalled airplane to transient inputs from the rudder or ailerons was measured. These results were harmonically analyzed and the aileron and rudder effects were separated by a vector analysis. The frequency response characteristics of the pertinent responses to aileron and rudder inputs in fully stalled flight were determined.

The characteristics of the satisfactory autopilot system and the techniques used in arriving at it are presented. The data presented includes the recorded time histories of control inputs and airplane motions at various fully stalled angles of attack, and the experimentally determined frequency response characteristics at 28° angle of attack.

INTRODUCTION

Stalls have been important in aviation since the earliest days of the airplane. When an airplane stalls, its stability and control characteristics undergo a radical change and the airplane becomes extremely difficult or impossible to control. In addition, the lift coefficient becomes smaller and the drag coefficient much larger, necessitating either a rapid descent or a very large increase in thrust to maintain level flight. Finally, the turbulence of the flow over a stalled wing causes buffeting of the airplane which may be quite severe. An extensive bibliography on the literature of the stall was included in Reference 1.

For several years, the Cornell Aeronautical Laboratory has been engaged in an investigation of the first of these problems, the stability and control characteristics of a stalled airplane. This work has been done under contract to the Aircraft Laboratory and the Flight Research Laboratory of Wright Air Development Center, USAF. This work is the first serious investigation of the motion of a stalled airplane since the pioneering work in the field done by the British in the 1920's and early '30's, and is the first to be done on an airplane of reasonably modern configuration.

An exploratory program (Reference 1) indicated that the longitudinal and lateral motions of the stalled test airplane could be separated, although only with considerable practice on the part of the pilot. The longitudinal motion was selected for study first, because it was expected to be simpler to handle both experimentally and analytically. This program also showed that the longitudinal motion was repeatable, if careful attention was paid to the initial conditions. Recording instrumentation was installed in the airplane,

Contrails

and analysis of the flight test data confirmed the repeatability of the motion at the stall. It appeared that the airplane was behaving as a spring-mass-damper system and could be represented by linear equations of motion, although the numerical quantities involved were different for angles of attack below, at, and beyond the peak of the lift curve as would be expected. The test airplane, like most airplanes, had only sufficient longitudinal control to reach slightly beyond the peak of the lift curve. When the stall occurred the airplane oscillated between a stalled and a semi-stalled angle of attack.

The desirability of obtaining steady state data in the stall was clear. Accordingly, the stabilizer of the test airplane was made adjustable to an angle of incidence which would enable the airplane to reach steady, trimmed angles of attack as high as 12° past the peak of the lift curve. Improved instrumentation was installed. Flight tests were made in which the characteristics of the airplane were obtained from measurements made in steady stalled glides and from the response of the stalled airplane to step and impulse elevator inputs. Lift coefficient vs. angle of attack and elevator-to-trim curves obtained in that program are shown in Figures 26 and 27. A linear analysis fitted the flight test data reasonably well, except that a lag of approximately one second apparently existed between a change in angle of attack and the resulting change in lift. This work was described in Reference 2.

In all of the longitudinal tests, the pilot kept the lateral motion small by constant manipulation of the ailerons and rudder. This technique required considerable practice, and success could not be guaranteed for any one run. In general, the pilot was able to keep the airplane under control in about four stalled glides out of five.

When the longitudinal characteristics of the stalled airplane had been determined, the problem of the lateral motion was tackled. The decision was made to attempt to stabilize the stalled airplane by an automatic control. It was felt that if a human pilot could stabilize the stalled airplane, an autopilot could be made to do the job and would probably do it better. The lack of lateral stability derivative data in the stalled regime meant that the quantities to be sensed, the servo computer design and the autopilot settings would have to be based originally on certain assumptions and then adjusted by trial and error in flight. Accordingly, a rather extensive program to obtain a satisfactory autopilot installation, and later further refinements using measured derivatives were planned.

The plan was to disturb the stabilized airplane, measure its response, and thereby obtain the airplane's stability and control characteristics. Stability derivatives were to be determined if possible.

The response characteristics of the airplane were to be measured in steady stalled glides, and should apply as well to a case such as the entry to a spin, as long as the motion has not progressed too far. This information allows a quantitative discussion of the motion of the test airplane in the stall, and should give a qualitative idea of the behavior of most conventional airplanes in the stall. In some situations the test results may not be directly applicable. For example, the high rates of roll and yaw which occur in a spin undoubtedly affect the airplane's aerodynamic characteristics.

It was suspected that non-linearities in the airplane characteristics would cause more trouble than they had caused in the case of the longitudinal motion, primarily due to the generally more complicated nature of the lateral motion. However, a linear analysis was to be carried as far as possible to assist in showing where

non-linearities might exist, as it had in the analysis of the longitudinal motion . The intractability of a non-linear analysis, and the lack of knowledge of the form of the non-linearities made a linear analysis far more attractive.

In this report are presented the results of a linear analysis of the flight test data, in the form of frequency responses, and sufficient examples of the flight test oscillograph records to allow an analysis different from the one used here to be made if desired.

Contrails

DESCRIPTION OF THE TEST AIRPLANE

The test airplane was the same Fairchild PT-26 (Figures 1 and 2) which had been used in the previous Cornell Aeronautical Laboratory investigations of the longitudinal motion at the stall (References 1 and 2). Table I lists the pertinent aerodynamic characteristics of the airplane. The horizontal stabilizer had been made adjustable to provide sufficient nose up pitching moment to allow stabilized flight at angles of attack well above the stall. A 5½ foot mast just aft of the front cockpit carried a vane for measuring angle of attack, placing the vane 1.4 root chord lengths above the wing. A 4.7 foot boom was mounted 2 feet inboard and below the right wing tip to carry the vane which measured sideslip (Figure 1). This placed the sideslip vane 0.95 local chord lengths in front of the leading edge and .70 chord lengths below it. Another boom protruded 0.9 local chord lengths from the left wing tip to carry the total head and static head tubes. The booms and mast had no noticeable effect on the stall characteristics of the airplane. The rear cockpit was stripped of its normal equipment and a special floor installed to facilitate the installation of the test equipment.

The stall characteristics of the test airplane were average, i.e., neither outstandingly good nor bad. However, at the high angles of attack reached in the steady stalled glides the whole wing was stalled and the airplane was exceedingly unstable.

The airplane had Frise ailerons, and tuft studies showed that the aileron unstalled when it was deflected full up, opening the slot

Contrails

between the wing and the aileron. The extent to which this characteristic affected the controllability of the airplane is not known precisely, but it was believed to be favorable.

AUTOMATIC CONTROL

A modified Sperry A-12 autopilot controlled the rudder and ailerons in the stall. The elevator was operated manually while entering and recovering from a stall, but was locked during the stall.

A block diagram of the automatic control system is shown in Figure 3. The operation of the system will be explained first, and then the individual components will be described.

Signals proportional to roll angle, roll rate, yaw rate, sideslip and aileron position were fed into a mixer. The pilot set knobs on the mixer to select the proper amounts of the various signals to be fed to the aileron and to the rudder. The outputs of the aileron and rudder channels of the mixer were sent to the aileron and rudder channels of the autopilot amplifier, which in turn drove the servos to move the ailerons and rudder. A disturbance input, V_o , could be supplied to either the aileron or rudder channel of the mixer, allowing the pilot to apply a transient disturbance to the automatically controlled airplane.

AUTOPILOT

Most of the autopilot installation consisted of standard Sperry A-12 autopilot components. The autopilot amplifier was modified only to the extent of tapping one of the transformers to secure voltage of the proper phase for use in some of the rest of the cir-

cuits. The servo control, servos, and servo engaging mechansims were standard A-12 equipment. The travel of the push rods in the control system was relatively small. The servos were connected to the control system with solid links instead of cables, partly to reduce deformation in the servo control system connection and partly because the solid links simplified the installation problem for this particular airplane. The servo drum travel was therefore relatively small, instead of the complete revolution or so that it rotates in a conventional installation. The small travel magnified the effect of lost motion in the servo gears, since a given amount of play was a larger percentage of the hard-over to hard-over travel of the servo drum. The aileron control forces of the PT-26 were very light, especially at the low airspeeds associated with a steady stall, so the torque required from the aileron servo was well within its capabilities. The rudder forces were several times as large as the aileron forces. The forces at the rudder servo, because of the low mechanical advantage of the servo over the rudder, were high enough to affect the accuracy of the positioning of the rudder by the servo. The rudder control, therefore, felt somewhat spongy compared to the prompt, accurate aileron control.

The rudder servo was installed under the floor of the front cockpit and the aileron servo under the floor of the rear cockpit. Because of the relatively long control run between the servo and the control surfaces, backlash in the control system was present between the servo and the control surface. The entire control system in the PT-26 is made up of push-pull tubes and bell cranks, with ball bearings at every joint to reduce backlash. The play in the controls due to backlash was of the order of ±2° of total aileron, or approximately 2% hard-over to hard-over aileron, and about ±2.5° of rudder, or about 8% of hard-over to hard-over rudder.

Contrails

The play in the controls due to backlash and the apparent stiffness of the rudder servo could have been improved by redesign of the servo installation. However, the autopilot was able to stabilize the stalled airplane with the servo installation just described, so improvements in the installation were not considered justifiable.

The dynamics of the Sperry A-12 autopilot could be varied easily by means of adjustments provided in the amplifier assembly which attenuate the incoming signal, provide a variable amount of first derivative of the incoming signal, and attenuate the servo position repeatback signal. The airplane buffets severely in the stall, with a dominant frequency of about 10 cycles per second. To avoid producing high inertia loads in the control system, an effort was made to prevent high frequency signals from the signal pickups from being transmitted through the autopilot to the control surfaces. The autopilot frequency response was therefore adjusted to cut off at a frequency considerably less than 10 cps. Flight tests soon showed that the phase lag in the autopilot when so adjusted, added to phase lags in the rate gyros and in the airplane response, produced a continuous oscillation of the ailerons at a frequency of about 3cps. This point is discussed in more detail in the section describing the development of the automatic control. At this frequency, the airplane response was small and the airplane could actually be stabilized in the stall with the 3 cps. oscillation present, but the problem of producing usable data under those circumstances was formidable. The autopilot was then "stiffened", to provide a higher natural frequency and smaller phase lags, and the system made stable at the expense of transmitting a slightly larger amount of the buffeting signals to the control system. The frequency response of the autopilot, measured on the ground without simulated air loads, is shown in Figure 4.

Table II lists the autopilot sensitivities used in the data flights.

The automatic control system proved to be reliable and required very little maintenance.

ROLL ANGLE

Roll angle signals were provided by a Sperry A-12 vertical gyro. A switch was installed in the wire providing power to the gyro erection system. Shutting off the erection before starting any accelerated maneuvers prevented the gyro from erecting to a spurious vertical. The drift rate of the gyro with the erection turned off was so slow that no measurable error was introduced into the recordings.

ROLL AND YAW RATE

Roll and yaw rate were measured by Pioneer B-1 rate gyros. These instruments originally had an air dash pot connected to the indicator needle, to damp the motion of the needle. The dash pot was rearranged to damp the motion of the gimbal. In addition, the springs restraining the gimbal were stiffened to increase the natural frequency and decrease the phase lag over the operating frequency range. Considerable difficulty was encountered in obtaining consistent frequency responses of these instruments, although the static calibrations were repeatable and showed little scatter. Figure 5 shows the frequency response of the rate gyros, and Figure 6 their static calibrations.

The frequency response of the instruments varied with the amplitude and direction of the disturbance. Dry friction in the air dash pot possibly accounted for some of the discrepancies.

Because reliable quantitative knowledge of the motion of a stalled airplane was unobtainable (acquiring this knowledge being the object of the project), it was necessary to use instruments which covered a wide range of rates.

Contrails

These rate gyros were chosen because they had sufficient range for this exploratory program and for reasons of availability, size and weight, quality, power consumption and cost. However, the stabilization achieved eventually reached such a degree of refinement that the angles and rates to which the motions were confined were often so small that values near the threshold of these instruments became of importance and the limitations of these instruments became evident.

SIDESLIP

A vane was mounted on a boom projecting . 95 chord lengths ahead of the right wing tip. The boom was slanted downwards at an angle of 15° to the thrustline to make it line up approximately with the oncoming air when the airplane was stalled. A unit containing two autosyns driven by the vane was mounted at the end of the boom. One autosyn was connected directly to the shaft of the vane. It provided signals for the autopilot. The other autosyn was driven by a gear train from the vane, and operated a repeater autosyn on the pilot's instrument panel. The vane was mounted below the boom, partly to place it in less disturbed air at high angles of attack, and partly to simplify the problem of keeping dust from entering the mechanism around the shaft.

AILERON ANGLE

The rudder was moved proportional to aileron angle to counteract the yawing moment due to aileron deflection. The necessary aileron angle signal was obtained from an autosyn in the center section, connected to the aileron push-pull tube.

Contrails

RUDDER AND AILERON REPEATBACK SIGNALS

The A-12 autopilot is a "position" type of autopilot, i. e., the servo moves the control surface to a position corresponding to the input to the autopilot. The control surface position signal which is balanced against the autopilot input signal comes from an autosyn in the servo assembly. The system is arranged to make the servos follow the control motion when the airplane is being flown manually so that they are always "zeroed" and can be engaged at any time. The built-in surface position, or repeatback, signal takes as zero the position of the surface when the servo is engaged, and hence is useless as an absolute surface position indicator. Since surface position was needed for purposes other than repeatback to the autopilot, it was provided independently of the autopilot.

DISTURBANCE SIGNAL

To apply a transient input to the airplane in the stalled glide, a voltage was applied to the mixer input which added to the signals from the various signal pickups. The pilot was provided with a potentiometer to regulate the amplitude of the disturbance input, and switches to select the surface, aileron or rudder, to be disturbed, and the direction of the disturbance, e. g., left or right rudder. A momentary switch in the cockpit applied the disturbance input as long as the switch was held in the "on" position. The disturbance was thus a square wave superimposed on the signals entering the mixer. The disturbance signal passed through the mixer unchanged, and hence was also a square wave superimposed on the input to the autopilot amplifier.

RECORDING INSTRUMENTATION

A Consolidated Model 5-101A oscillograph, using 8 inch wide paper, was the primary recording instrument. Quasi-static quantities, and those of lesser importance in the tests were recorded on a photo observer. Switches to operate the recording instrumentation were installed on the pilot's stick grip, the elevator lock, and a switch panel on the throttle quadrant. The quantities recorded on the oscillograph and photo-observer were:

OSCILLOGRAPH	PHOTO-OBSERVER
Roll Angle	Altitude
Roll Rate	Airspeed
Yaw Rate	Time
Sideslip	Angle of Attack
Aileron Angle	Elevator Angle
Rudder Angle	Pitch Angle
Aileron Autopilot Input	Run Number
Rudder Autopilot Input	
Disturbance Input	

SIGNAL SOURCES FOR RECORDING INSTRUMENTATION

ROLL ANGLE, ROLL RATE, YAW RATE, SIDESLIP, AILERON ANGLE

Part of the signal output from the signal source that supplied the signals to the automatic control was used for recording. The signal was passed through a phase sensitive rectifier, filtered, and applied to the oscillograph.

RUDDER ANGLE

An autosyn, mounted on the fuselage and connected directly to the rudder by a linkage, provided the rudder position signal for recording.

Contrails

ALTITUDE AND AIRSPEED

A Kollsman sensitive altimeter and a helicopter type airspeed indicator were mounted on the photo-observer panel. The helicopter airspeed was used because it had small instrument errors and an expanded scale in the low airspeed range in which the stall tests were conducted.

A swivelling static head and shielded total head tube were mounted on a boom .9 local chord lengths ahead of the left wing tip. The axis of the total head tube was inclined downwards at an angle of 15°, to minimize the errors in total head reading at the high angles of attack of the stall.

No airspeed calibration was made in stalled flight due to the large rates of descent involved and the consequent impossibility of using pacing or tower pass methods. A trailing bomb would not have been accurate due to the large initial transients and short time available in stabilized stalled flight. It was assumed that the airspeed errors, using the system just described, would be small and could be neglected.

ELEVATOR ANGLE

An autosyn, mounted on the stabilizer and connected to the elevator by a linkage, drove a repeater autosyn on the photo observer panel. A second autosyn, similarly mounted, drove a repeater on the pilot's instrument panel. The stabilizer was set at maximum negative incidence for all the test runs.

ANGLE OF ATTACK

An autosyn driven by a vane, was mounted at the top of a 5-1/2 foot mast projecting up just behind the front cockpit. The vane was thus located 1.4 root chord lengths above the wing. The

Contrails

autosyn drove a repeater autosyn on the photo-observer panel. The position error of the angle of attack vane had been determined from flight tests, in both stalled and unstalled flight, in the previous test program involving the longitudinal motion of the stalled airplane. Elevator-to-trim curves for various c. g. positions (Figure 27) had also been determined from those flight tests, and a chosen angle of attack was obtained in flight by setting the elevator to the desired angle, using the elevator position indicator on the pilot's instrument panel.

PITCH ANGLE

Pitch angle was measured by a standard air driven artificial horizon, mounted on the photo observer panel.

INSTRUMENTATION POWER SUPPLY

The test airplane was equipped originally with a 12 volt power supply. Some of the instrumentation required 12 volts and some 24 volts. Two 12 volt 34 ampere hour aircraft batteries were therefore connected in series, providing a convenient source of both voltages. The 12 volt load was distributed as equally as possible between the two batteries to equalize the rate of discharge of the batteries. The 12 volt generator was replaced by a 24 volt generator, and a 24 volt voltage regulator was installed.

The current drawn when both the autopilot and recording equipment were turned on was more than the generator would provide, so the batteries were actually discharging during the test runs. Fortunately, due to the nature of the tests, each test run was relatively short (3 or 4 minutes), and was followed by a long

(10 - 15 minutes) climb back up to altitude. With batteries in good condition, the battery voltage would stay at a sufficiently high value for flights as long as three and a half hours.

A Holtzer - Cabot Model 153F, 1000 volt-ampere inverter supplied the 3 phase 400 cycle AC for the gyros, autopilot, and other instrumentation. One phase was regulated to supply a source of constant voltage to the instruments where the calibration depended on the voltage.

The pilot was provided with AC and DC voltmeters, which allowed him to monitor the operation of the power supply and discontinue the tests if the voltage strayed too far from the proper values.

Contrails

FLIGHT TESTS

Fifty one flights, totaling 107:20 hours were made in the course of the lateral phase of the project. Most of this effort was devoted to developing the automatic control to the point where it would stabilize the stalled airplane sufficiently to make it possible to obtain usable data. Table IV summarizes the test flights.

The flight tests were conducted at altitudes of 8000 to 4000 feet. The airplane stalled at 68 mph, calibrated airspeed. The airspeed was higher in the stalled glides due to the lower lift coefficients, and can be computed if desired from information available in this report.

DEVELOPMENT OF AUTOMATIC CONTROL

The difficulty in developing the automatic control to make it stabilize the stalled airplane was due primarily to the fact that the aerodynamic characteristics of the stalled airplane were not known well enough to allow the calculation of the proper sensitivities of the autopilot, or even of what quantities should be sensed. Determination of these airplane characteristics was one of the objects of the project.

Preliminary calculations had been made (Reference 3), using assumed values of the aerodynamic derivatives of the stalled airplane. According to these calculations, the stalled airplane could be stabilized in roll by driving the rudder by a combination of roll angle, roll rate, and roll acceleration signals. Use of the ailerons was not considered in these calculations, partly due to limitations

Contrails

of time and partly due to serious doubts as to the efficacy of the ailerons in the stall.

Subsequent experience in the investigation of the longitudinal motion of the stalled airplane indicated that the ailerons were useful in controlling the stalled airplane and produced rolling moments provided sufficient rudder was used to neutralize the adverse yawing moment due to aileron deflection. It was exceedingly difficult to stabilize the airplane at the high angles of attack attained in these tests (12° above the stalling angle of attack) using the rudder only, while after considerable practice stalled glides could be made by using both ailerons and rudder.

INITIAL ADJUSTMENT OF THE AUTOMATIC CONTROL

As a starting point, the ailerons were actuated by a combination of roll angle, roll rate, and roll acceleration, and the rudder by the same signals plus aileron angle, to neutralize the adverse yaw of the ailerons. A suitable angular accelerometer could not be obtained at the time, so one was designed and built specially for the purpose. The amount of the various signals to be used was estimated from the calculations mentioned above. Several flights were spent in exploring the effects of different amounts of the various signals. The autopilot could not stabilize the stalled airplane but it seemed to be close to success, for it could keep the airplane from rolling off for some time after the stall was reached. However, before long the airplane would always roll off to one side or the other and start to spin. It was also apparent that as soon as the roll rate or roll acceleration sensitivities were increased in an attempt to improve the ability of the autopilot to control the stalled airplane, the airplane auto-

Contrails

pilot system instability, mentioned in the section describing the autopilot, appeared in the form of a rolling oscillation driven by the ailerons.

The frequency was quite high (about 3 cycles/second) and the amplitude of the rolling motion was very small, but the aileron motion was large--nearly hard-over to hard-over. Closer examination of this phenomenon showed that phase lags in the roll rate gyro and angular accelerometer, the autopilot servos, and the airplane were sufficient to total 180° and produce the instability. The frequency of the oscillation was high enough to suggest that the aileron effectiveness and the airplane's moment of inertia were the only airplane parameters affecting the oscillation. An analysis using only these two parameters, with the measured dynamic characteristics of the signal sources and the autopilot, predicted an instability at a frequency of close to 3 cps.

Two ways of removing the instability presented themselves. The phase lag in the autopilot could be reduced, but only at the expense of making the autopilot capable of passing more high frequency signals. The signals that arose from the buffeting of the airplane in the stall had a frequency of about 10 cps. These signals, if transmitted by the autopilot to the control surfaces, would result in undesirably high inertia loads in the control system. The autopilot had purposely been adjusted so that it would not pass high frequency signals, to make it act as a sort of mechanical filter for signals due to buffeting. Reducing the phase lag in the autopilot did not, therefore, seem like a desirable course to follow if the trouble could be cured by some other means.

By altering the ratios of roll angle, roll rate, and roll acceleration signals, the phase lag in the net signal from the signal sources could be reduced at the frequency where the instability

Contrails

occurred, while still providing, at the lower frequencies used in stabilizing the airplane, a large enough signal with a phase leading the motion of the airplane sufficiently to be useful in stabilizing the airplane. This scheme was tried, and allowed larger amounts of the signals to be used without reaching the region of the system instability. The ability of the autopilot to stabilize the airplane was improved considerably, but it was still not good enough.

The signal gains were set still higher, and stalled glides were made with the 3 cps oscillation superimposed on the lower frequency motion of the controls, which were trying to stabilize the airplane. The 3 cps oscillation was of a high enough frequency to have very little effect on the airplane, and the increased signal gain was beneficial. Several stalled glides were made of over a minute's duration, in which about 3000 feet of altitude was lost, with good stabilization by the automatic control. However, the automatic stabilization still could not be depended upon. The initial conditions, i. e. the entry to the stall, had to be controlled extremely carefully, and the tests had to be conducted in completely smooth air or control would be lost and the airplane would roll off into a spin. Furthermore, the oscillograph records were extremely difficult to read because the 3 cps oscillation tended to mask other motions of the airplane and the controls.

The autopilot was made stiffer by increasing the servo position repeatback gain. The 3 cps aileron oscillation was lessened by this adjustment, and the amount of 10 cps buffeting signals sent to the controls was not materially increased. The sensitivities could be increased for the same amount of 3 cps control oscillation, and the stabilization of the airplane by the automatic control improved, but the effect was not great.

Contrails

At this point, the stiffness of the autopilot was increased considerably decreasing the phase lag in the system as a whole to the point where the 3 cps system instability vanished. Care was taken to choose an adjustment which provided the best balance between decreasing the phase lag at low frequencies (up to 3 cps) while still attenuating as much as possible of the 10 cps buffeting. A larger amount of 10 cps buffeting noise was noticed in the control motion, but it was not considered excessive.

The stabilization of the stalled airplane was not greatly improved by this adjustment, but the absence of the system instability made it much easier to follow the motion of the controls, both in flight and on the oscillograph records.

Sideslip sensing to the autopilot was tried, in an effort to find a signal, useful in stabilizing the airplane, which was not derived from roll angle and which therefore would not produce the troublesome system instability. Sideslip sensing did not prove to be satisfactory in stabilizing the airplane, probably for the following reason: the rolling moment due to sideslip is markedly higher at stalled angles of attack than at unstalled angles of attack. Therefore, it is desirable to keep the sideslip angle small in stabilized stalled flight to avoid producing large destabilizing rolling moments. However, if the lateral motion ever becomes large, the powerful rolling moment available from sideslip, applied in the proper direction, may be essential to overcome other destabilizing moments and return the airplane to a level attitude. The question then is, in what direction should sideslip signals be used? The sideslip should be kept small when all is going well, but should be allowed to become large when the lateral motion becomes large. After several flights devoted to experimenting with sideslip sensing to the ailerons and rudder, it was concluded that sideslip sensing

Contrails

could not be put to good use in stabilizing the stalled airplane.

While the flight tests devoted to perfecting the automatic control were in progress, attempts were made to find out what the human pilot did which made possible manual control of the stalled airplane. Whatever he was doing was obviously not too simple, for it had taken an experienced pilot a considerable amount of practice before he could do a good job of keeping the lateral motions of the stalled airplane small. Manually controlled stalls were done in which the pilot made a special effort to analyze what he was responding to when he moved the controls. The motion was so rapid and seemed so irregular that he could only say that aileron was moved proportional to roll angle, roll rate, and perhaps roll acceleration, and rudder largely proportional to aileron angle and perhaps yaw rate. One interesting point was that it was perfectly clear to the pilot when and how much the controls had to be moved, even though he was unable to say what facet of the airplane's motion made him aware of that necessity. To prove this, the controls were sometimes moved in a manner slightly different from what seemed to be called for; loss of control was the inevitable result. Manually controlled stalls were also made in which, in addition to using the controls in the required manner, the rudder was used to keep the sideslip zero, except that if the airplane seemed to be getting out of hand, sideslip was purposely used to "pick up a wing". This technique would work, but it seemed to be asking too much to expect the autopilot to switch sideslip sensing in the middle of a stall.

Oscillograph records of manually controlled stalls were studied, but the motions were too complex to allow extraction of much information as to what the pilot was doing.

Contrails

The elimination of the instability in the automatically controlled airplane suggested a different technique. One control was handled by the automatic control, and the other by the pilot.

The motion of each control could then be studied separately, while the other control was being moved in what was presumably the optimum manner by the pilot. It was also much easier to compare the motion of one automatically operated control to the corresponding motion when manually controlled than it was to compare the full automatic control to the full manual control. This technique proved to be successful. The roll acceleration signal was eliminated as unnecessary although it might be desirable in a more highly developed system. The aileron sensitivity to roll angle was increased considerably, making the ailerons operate proportional to roll angle and roll rate.

Inspection of oscillograph records taken in stalls controlled by the half automatic and half manual technique indicated the need for yaw rate sensing to the rudder. When yaw rate sensing was provided, the automatic control stabilized the airplane successfully, and in fact, was able to do a better job than the human pilot.

It was desired to study the response of the airplane to a disturbance, so a means was provided for applying a voltage to the autopilot input which was superimposed on the voltages fed to the autopilot from the various signal sources. The pilot could apply an impulse, of any desired magnitude, to the rudder or aileron. The duration of the impulse was determined by the length of time he held the impulse switch on. The automatic control would make the stalled airplane return to a wings-level attitude from rudder or aileron disturbances which produced roll angles as large as 20°. In practice, a 10° roll angle was selected as reasonable for the data flights.

Contrails

In its final form, the autopilot moved the ailerons proportional to roll angle and roll rate, and the rudder proportional to yaw rate and aileron position. It will be noted that the automatic control system was equivalent to a normal autopilot incorporating a yaw damper, with the addition of a correction for the adverse yaw of the ailerons. The autopilot sensitivities are listed in Table II.

At this point the airplane was ready for data flights.

DATA FLIGHT TECHNIQUE

The airplane was climbed to the test altitude in a region of smooth air. The instrumentation was turned on two minutes prior to reaching the test altitude, to allow the amplifiers to warm up and the gyros to come up to speed. The stabilizer was cranked to the full "nose-up" position, the gyro erection turned off, and power reduced to approximately zero thrust. The airplane was slowed up, and just before the stall the autopilot was engaged at a time when the wings were level and the roll rate, yaw rate, and sideslip were zero. The type of autopilot installed assumed as zero the conditions at the moment it was engaged, hence any bank angle or angular rate would be perpetuated as the norm throughout the ensuing stall. Zero sideslip was necessary to ensure that the controls were in the trim position when the autopilot was engaged. The autopilot was not engaged at higher speeds because the controls would not then be at their trim position for the stall. Actually, this point was not as critical as the zero motion of the airplane, for the autopilot would drive the controls to the necessary position and hold them there, but there seemed to be no reason for requiring more effort (and power consumption) on the part of the autopilot than was necessary.

Contrails

The airplane was then stalled fairly gently, although the rate of entry to the stall was not critical. Since the autopilot was trying to do the same thing as the human pilot (keep the airplane in a level attitude with no angular motion), the pilot did not have to exert much effort to keep the airplane level. The elevator was set at the angle necessary to produce the desired angle of attack, and locked in that position. Trim elevator angle vs angle of attack curves for several c. g. positions (Figure 27) had been determined in stalled and unstalled flight in the preceding test program, reported in Reference 2. Once the elevator was locked, the pilot removed his hands and feet from the controls, and the autopilot was given full charge of the airplane.

During the automatically controlled stall, small rolling, yawing and sideslipping motions were generally evident. The motion was irregular, rather than periodic. The maximum amplitude of the bank angle during these motions was about 2°.

The pilot selected a time when the airplane was as steady as possible, and applied a disturbance to the rudder or ailerons through the autopilot. The amplitude of the disturbance was usually set to produce nearly hard-over motion of the control, although a number of smaller amplitudes were also used. The duration of the disturbance was fixed by the desired amplitude of the resulting motion of the airplane. A pulse about one second long, when applied to the ailerons, produced a maximum bank angle of about 10°, which was a convenient amplitude from the standpoint of the recording instrumentation. Larger disturbances produced roll rates high enough to cause the roll rate gyro to hit its stops. The same duration pulse, when applied to the rudder, produced a suitably sized response of the airplane.

When the disturbance was removed, the autopilot promptly returned the airplane to a wings-level attitude, with a small overshoot. The motion was quickly damped to the level of the small, irregular motions that were characteristic of the automatically controlled stall. Four or five disturbances could be applied in a single stalled run, allowing sufficient time between them for the response to the disturbance to die out completely. Figures 28 to 62 illustrate the response of the airplane and the controls to these disturbances.

During the stalled glides the airplane lost altitude rapidly The stalled data runs had to be ended when the minimum safe altitude was reached, or when the airplane descended into rough air.

To recover from the stall, the autopilot was disengaged, the elevator unlocked, and the nose pushed down to return to an unstalled angle of attack. The instrumentation was turned off to conserve the batteries, the throttle opened part way to warm up the engine, and the stabilizer returned to the normal position. The airplane was then climbed back up for another run.

The autopilot could also control the airplane in full power stalled glides, in which the rate of descent was reduced to about 1500 ft/min, from over 3000 ft/min with power off. However, the complication of adding slipstream effects to the problem of analyzing the data appeared to outweigh the advantage of a longer data run, so all of the data flights were made with zero-thrust power.

METHOD OF ATTACK

Early in the project, a decision was made to handle the data as though the airplane was a linear system if the oscillograph records obtained in the flight tests looked at all pro nising. It was known that not all of the airplane's stability derivatives would be linear in the stall regime. However, in the study of the longitudinal motion of the stall (Reference 2), a linear analysis had proved to be reasonable, with a non-linear correction added late in the analysis. It was felt that a linear analysis would provide some indication of the degree of linearity of the data, and might point out where non-linear terms could be added as corrections. Furthermore, since the techniques of linear analysis are well known and relatively simple, a linear analysis seemed like a reasonable starting point.

The data was first harmonically analyzed and the appropriate calibrations applied to determine the harmonic content of the input and output traces. Figure 7 and 8 show samples of this analysis. Dividing the outputs by the inputs and comparing phases produced frequency response plots of which representative samples are given in Figures 9 - 19.

Since the autopilot operated both the rudder and the ailerons simultaneously to stabilize the stalled airplane, the harmonic analysis of the recorded data gave the response of the airplane to one control input for the case where the other control input was working to stabilize the airplane. Therefore, it was necessary to separate the effects of the two controls and obtain the response

Contrails

of the airplane to each control separately before the data could be analyzed further. This was done by the method described later in this section and the results at an angle of attack of 28° are given in Figure 20 - 25.

It had been planned to perform further analysis of the frequency responses to extract lateral stability derivatives or to obtain a check on the values assumed by comparing calculated and measured frequency responses. Due to the exploratory nature of this research in the stalled flight regime, more of the project funds were needed to perfect the automatic control and obtain the flight data than had been estimated. The analysis was therefore tailored to fit the available project funds, but the degree of agreement in Figures 18 and 19 indicates that the values used (Table III) are reasonable.

The frequency response is a useful form in which to have information on the airplane. If stability derivatives were presented, the first step in the solution of many problems would be to convert those derivatives to frequency responses of the airplane.

OSCILLOGRAPH RECORDS

Samples of the oscillograph records of the response of the stalled airplane to rudder or aileron disturbances are shown in Figures 28 - 61. The records present data for inputs of various magnitudes at an angle of attack of 28° and for one value of each input at angles of attack of 24°, 20° and 18°. Also presented for comparison in Figure 62 is a manually controlled stall at 28°. The oscillograph records show a general shape of the transient response which is quite reasonable with variables which promptly returned to nearly the initial conditions.

There is some irregular high frequency oscillation, or "hash" superimposed on the traces due to the heavy buffeting which occurs

in the stall. There is some residual oscillation in the motion of the automatically controlled stalled airplane, discussed in the section on the development of the automatic control, which makes it impossible to determine the exact initial or final values of the variables.

It will also be noted that the sideslip usually took much longer to return to its steady value than did the angular motion variables. This situation is quite plausible, of course, since sideslip was not controlled directly by the autopilot, but depended on the control of the bank angle and yaw rate. The rate of return of sideslip after the roll angle and yaw rate have settled down is dependent on the value of side force due to sideslip, or C_{Y_β}

ANALYSIS OF DATA

Several transients were harmonically analyzed on a Rolling Sphere Harmonic Analyzer. The disturbance input was applied through one of the controls, as indicated in the plot. Figures 7 to 19 show the response of the airplane to one control motion while both that control and the other were being operated by the autopilot to stabilize the stalled airplane, and Figures 20 - 25 show the response to one control motion with the effect of the other control motion removed. Inspection of the results (Figures 7 to 17) of the harmonic analysis showed that:

1. The harmonic content of the input disturbance to the airplane (Figures 7 and 8) had an appreciable magnitude over the frequency range of interest (0-3 cps). The amplitude of the disturbance for the very low frequencies was considerably smaller than for the higher frequencies, but it was expected that the response characteristics of the airplane would be such that a small input would produce a large output in terms of motion of the airplane at the low frequencies.

The magnitude of the input was also relatively low at frequencies of about 1 cps. The desired input would have a constant amplitude over the frequency range of interest. Although the actual input departed from the desired input, it was felt that the actual input was satisfactory. Ratios of the output to the input, e.g., ϕ / δ_a, would be expected to be less reliable for the frequencies where the input was small. In this case, the input (control motion) was small at very low frequencies and at about 1 cps, while the output (airplane motion) was small at frequencies in the neighborhood of 3 cps and higher.

The harmonic content of the input varied somewhat from transient to transient, due to different durations of the disturbance, but the examples shown are typical.

2. When plotted as ratios such as r / δ_r, ϕ / δ_a, etc. (Figures 9 - 12), there was some scatter in the amplitude ratio points and more scatter in the phase angles between different runs due to the presence of different amounts of effect from the other control, but the results were consistent enough to be rather encouraging.

3. When the ratio roll rate to roll angle, or p / ϕ, was plotted vs. frequency (Figure 13), the scatter was not excessive for frequencies below one cps but became worse at higher frequencies. Again, the scatter in the phase angles was greater than that in the amplitude ratio. The plot of p/ϕ vs. the frequency, ω, should be a straight line, since the amplitude of p should equal the amplitude of ϕ times the frequency and p should lead ϕ by 90°.

The deviation from a straight line should be due only to the fact that the indicated roll rate varies from the true roll rate by an amount indicated by the frequency response of the

instrument (See Figure 5). The correct value of ρ / ϕ multiplied by the measured frequency response of the rate gyro is included in Figure 12 for comparison. It should be kept in mind that considerable difficulty was experienced in obtaining a consistent frequency response curve for the rate gyros and part of the discrepancy on Figure 12 may be due to errors in ground measurements of the rate gyro characteristics. At the higher frequencies, the roll angle response of the airplane becomes small, making possible larger percentage errors in the measurement of ϕ. Part of the discrepancy in Figure 12 may be due to the lower accuracy of the ϕ measurements at the higher frequencies.

The ρ / ϕ plots give no information on the linearity of the data as a whole; they show only whether the raw data and the harmonic analysis are internally consistent. The rate gyros gave the least reliable indications of any of the instrumentation, so it is perhaps a little unfair to judge the rest of the data by their performance, but it was the best internal check on the data that was available.

4. Data for four stalled angles of attack are shown in Figures 14 to 17. Careful inspection of the data shows a variation of the responses with angle of attack, although this variation is obscured by scatter of the data, particularly for the phase plots.

5. It was found that the residual oscillation of the stabilized stalled airplane was large enough, compared to the airplane's response to the transient input, to affect the results of the harmonic analysis. If the variable had returned to zero, or some constant value, and stayed there, the length of the trace after the transient had died out would not have affected the results, but unfortunately, the residual oscillations existed.

The harmonic analysis was repeated for several of the transients, stopping the analysis as soon as the response to the transient had become small enough to be indistinguishable from the

residual oscillations. This altered the shape of the resulting curve somewhat. These results were used in the separation of the effect of the aileron and rudder on the motion, described below.

CALCULATED AIRPLANE RESPONSE

To obtain a better idea as to the shape of the curves, the frequency response of the airplane in the form of r/σ_r, was calculated using linear equations and the estimated stability derivatives used in Reference 3. Measured autopilot and rate gyro characteristics were used, and the equations included the effect of both the aileron and rudder being actuated to stabilize the airplane. The results are compared with the frequency response obtained by harmonically analyzing the flight test data in Figures 18 and 19. It can be seen that the agreement between calculated and measured amplitude ratios is good but that there is a consistent discrepancy in the phase angles.

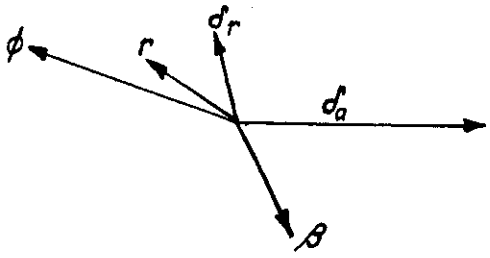
SEPARATION OF THE EFFECTS OF THE AILERONS AND RUDDER

When the stalled airplane was automatically controlled, both the rudder and the ailerons were working more or less continuously to stabilize the airplane. It was not possible to obtain in flight the response of the airplane to either control separately, since an aileron disturbance, for example, would cause yawing as well as rolling motion and would therefore start the rudder working. For many purposes, it is more desirable to have the response of the airplane to the rudder only or the aileron only.

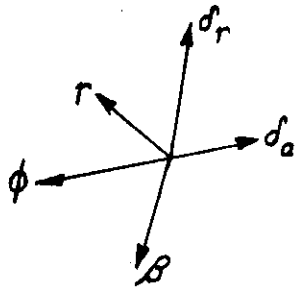
Contrails

The effects of the two controls were separated by the following method. Vector plots of the harmonic content of the variables were made for a given frequency. Vector plots from rudder and aileron disturbed transients, which gave quite different responses, were then manipulated and combined to eliminate the contributions of one of the controls. Ratios of the type r/δ_r for $\delta_a = 0$ were then obtained from the combined vector plot. By performing this operation for a number of frequencies, plots of the ratios of interest were obtained as a function of frequency. A more detailed description of the method follows.

Suppose that for one frequency the vector plots of the response of the stalled airplane to two different disturbances looked like this:



AILERON DISTURBANCE



RUDDER DISTUBANCE

If it was desired to eliminate the aileron motion, the δ_a vector of one of the plots, say the rudder plot, was multiplied by a factor which made its length equal to the δ_a vector of the other plot. All of the other vectors of the rudder plot were then multiplied by the same factor. The entire rudder plot was then rotated through an angle sufficient to make the δ_a vector of the rudder plot lie in a direction opposite to the δ_a vector of the aileron

Contrails

plot. The plots were then superimposed and the corresponding vectors added. When this operation was performed, the δ_a vectors dropped out, since the preceding steps made the two δ_a vectors equal in magnitude and opposite in direction. From the vector sums remaining the phase and amplitude of the ratios r/δ_r, β/δ_r etc., were obtained for a rudder disturbance with $\delta_a = 0$ for one frequency. The process was repeated for a number of frequencies in the frequency range of interest.

The response of the airplane to aileron motion only and rudder motion only was determined by the method described above, and the results presented in Figures 20 to 25. Each of the points at a given frequency was determined from a different transient. To reduce the chance of one of the transients used being an unrepresentative one, the aileron only results were obtained by combining three aileron transients with each of two rudder transients. The rudder only results were obtained by combining two rudder transients with one aileron transient and a third rudder transient with a different aileron transient.

DISCUSSION OF THE PLOTS OF THE RESPONSE OF THE AIRPLANE TO A SINGLE CONTROL MOTION

Examination of Figures 20 - 25 shows that the scatter previously observed in Figures 9 - 12 has been largely removed and curves showing the trends of the points appear to be much more reliable. There follows a discussion of the physical nature and reasonability of the responses given in Figures 20 - 25.

It will be noted that the roll angle, ϕ, was used to describe the rolling motion response, instead of the roll rate, p, while yaw rate, r, was used to describe yawing motion response. The reason for this apparent inconsistency was that the instrumentation was such as to provide reliable indications of roll angle, while the indicated roll rate was less reliable. Yaw rate had to be used, because it was the only measurement of yawing motion that was available.

Contrails

ϕ/δ_a

With the rudder fixed, the ailerons produce a rolling displacement which drops off rapidly as the frequency of motion increases. The roll angle lags the aileron angle by approximately 90° at low frequencies. This is reasonable, since the roll rate should be approximately in phase with the aileron motion, and the roll angle lags the roll rate by 90°. As the frequency increases, the phase angle between the roll angle and the aileron motion increases rapidly to values over 300°, or almost a full cycle.

β/δ_a

At low frequencies, the ailerons produce a large sideslip angle, which decreases rapidly with frequency since the roll angle produced by the ailerons decreases rapidly with frequency. At low frequencies, the phase angle shows the expected adverse yaw, i. e. left aileron produces a sideslip to the left. As the frequency increases, the phase lag between the aileron motion and the sideslip increases rapidly.

ϕ/δ_r, β/δ_r, r/δ_r

When the rudder is moved, it produces a yaw rate, and hence a sideslip. The sideslip and the yaw rate both produce a rolling motion. As the frequency of the rudder motion increases, the response rapidly becomes smaller for all three of the variables r, ϕ, and β. The phase angles at low frequencies for all three plots indicate that the motion is in the "proper" direction, i. e. left rudder produces left yaw rate, left roll angle, and sideslip to the right.

Contrails

r/δ_a

At the higher frequencies, r/δ_a is primarily a measure of the adverse yaw of the ailerons, while at low frequencies the ailerons, by producing a roll angle and thus some sideslip, produce a yaw rate in the "proper" direction, i.e. left aileron will produce yaw rate to the left. At some frequency in between, the two effects should cancel. The amplitude ratio plot should thus be high at low frequencies (though not as high as in unstalled flight, because of the low aileron effectiveness in the stall), and become small at an intermediate frequency. As the frequency becomes higher, so that the roll angle produced by the ailerons becomes small, the adverse yaw will be the primary effect and the r/δ_a should increase, with the phase changed 180° from the direction at low frequencies. The scatter in the phase plots at the frequencies where the amplitude ratio is small is not surprising, since errors become a large percentage of the value of the variable when the amplitudes are small. However, the scatter at all frequencies is large enough to make it impossible to be sure of the general trend. Inspection of the oscillograph records (Figures 28 to 62) indicates that the yaw rate response for the aileron disturbances is not large, leading to the conclusion that the reliability of the r/δ_a plots is not particularly good.

DISCUSSION OF VALIDITY OF DATA

It must be clearly understood that every step of the analysis described above assumed that the stalled airplane behaved as a linear system, and that therefore a linear analysis was applicable. This assumption is known to be somewhat in error. There are a

Contrails

number of nonlinearities which are well known as well as those which are more obscure. The reasons for using a linear analysis were pointed out at the beginning of this section. It is probably more remarkable that the results of a linear analysis appear as reasonable as they do than that there is as much scatter as there is.

The lift curve for the test airplane (Figure 26) determined in flight during the tests reported in Reference 2, is not linear above the stall. It is, however, much more nearly linear at the highest angles of attack, where most of the tests reported here were conducted, than at the peak of the lift curve.

In the previous longitudinal stall tests (Reference 2), a lag of close to one second was found between a change in angle of attack and the resulting change in lift. Presumably, there should be a similar lag in all of the lateral stability derivatives, such as C_{l_p}, which arise from local changes in angle of attack. This time delay is long enough to have a marked effect on the response of the stalled airplane.

There are probably nonlinearities in some of the other derivatives, such as those depending on sideslip and control deflections, but the extent of these is not known. On the other hand, the motion of the automatically stabilized stalled airplane never became very large, and the large secondary effects, such as the possibility of the roll rate derivatives being also dependent on sideslip, were not as important as they would be in a maneuver such as a spin.

The effect of various nonlinearities is rather difficult to assess, but it is felt that they account for a large part of the scatter in the data.

Rough air could also cause scatter in the data, by applying unknown forces and moments to the airplane. It was assumed that

Contrails

the only disturbances to the airplane came from the measured control displacements. In practice, particular care was taken to conduct the flight tests in smooth air, and data flights were discontinued whenever even very light turbulence was encountered. It is felt that it is safe to assume that there were no effects from rough air in the data presented in this report.

The recording instrumentation was such that the difference between the indicated and true value for any of the variables can be considered negligible for the frequency range of interest in these tests, except for the roll and yaw rates, and to some extent the sideslip angle. Considerable difficulty was experienced in obtaining consistent frequency response calibrations of the rate gyros and the indicated value may be in error by an amount that is hard to determine. It is for this reason that any results of this analysis pertaining to roll are presented in terms of roll angle rather than roll rate.

Incremental values of sideslip were used, so the static position error of the sideslip vane installation was unimportant. It was of the order of 5°, and did not affect the linearity of the sideslip recording circuit. The installation of the sideslip boom was designed to minimize errors due to aerodynamic interference with the vane, but no way was found to determine the variation, if any, of local sideslip angle at the vane with the motion of the airplane.

The small residual motion of the automatically controlled stalled airplane was never fully explained. It may have been due to resolution difficulties in the autopilot, caused by the play between the servo and the control surface. Another possibility is that the motion may have been caused by the irregular character of the separation of the airflow from the stalled wing. If a stalled wing is held steady, the flow over it probably fluctuates sufficiently

Contrails

so that at any given instant the moment on the wing may be somewhat unsymmetrical, even though the average moment over some length of time is symmetrical. The fin, rudder and part of the fuselage lay in the wake from the wing, so a momentarily unsymmetrical flow could produce a yawing moment and a side force. It is not known, however whether fluctuations in the flow over the stalled wing can occur at frequencies which are low enough to affect the motion of the airplane. Moments due to a fluctuating flow caused by the separation characteristic of the stall would affect the repeatability of the data since they would apply an input to the airplane that was not accounted for; it was assumed in the analysis that disturbing inputs were due only to the control surface motion.

Difficulties due to play between the servos and the control surfaces should not seriously affect the data used in this report, since the measured control surface motions were taken to be the input.

AUTOPILOT ADJUSTMENT USING EXPERIMENTAL DERIVATIVES

It was originally intended to use the derivatives obtained experimentally to calculate how to improve the autopilot behavior. The extent of the investigations made in adjusting the autopilot prior to the start of the data flight phase and the agreement between calculated and measured frequency responses (Figures 18 and 19) indicated that achievable improvements in further adjustment of the installed equipment would not produce any large improvement in the actions of the autopilot, and no further adjustments were made. As discussed elsewhere in this report, further improvements might be obtained by changes in the rate gyros and servo installations.

Contrails

CONCLUSIONS

1. Automatically controlled stalled glides were possible up to the highest angle of attack (12° above the peak of the lift curve), that the test airplane was able to reach. The violent roll off at the stall, characteristic of the uncontrolled airplane, was eliminated. The stabilization was sufficiently good to make the airplane recover from disturbances in which the bank angle reached 20°. The airplane would recover from gentle turns made while stalled, but not steep turns. Stabilized stalled glides could be made in rough air, and with any amount of power from idling to full throttle.

2. The action of the stabilizing equipment was sufficiently positive and repeatable that automatic control in the stall may be found applicable to tactical airplanes. Stabilized stalled flight may be useful in any situation in which it is desired to:
   (a) Prevent loss of control in an inadvertent stall
   (b) Utilize the maximum lifting potentialities of an airplane
   (c) Perform rapid descents without picking up excessive speed
   (d) Obtain high drag at low speeds, as in minimizing the landing, roll of airplanes

3. The agreement between calculated and measured responses and the character of the frequency response plots obtained when rudder and aileron action were separated are sufficiently good that further efforts at analysis should result in obtaining worthwhile information on the lateral stability derivatives in stalled flight. In any such studies, non-linearities due to time lag in circulation buildup and the like should be considered.

4. These results, especially the better insight into the physical nature of some of the responses, may be useful in studying spin entries.

Contrails

RECOMMENDATIONS

1. The potentialities of automatic stabilization of an airplane at and beyond the stall should be investigated thoroughly. It is entirely conceivable that some practical use could be made of this hitherto neglected regime of flight.

2. The data obtained in these flight tests should be analyzed further, particularly with a view toward obtaining the variation with angle of attack of the various airplane characteristics. This analysis should include the best possible estimates of the various non-linearities in the airplane characteristics in an attempt to duplicate the type of airplane responses found in the flight tests. This implies more theoretical work on the aerodynamics of separated flows.

3. In any further data gathering flight programs, the difficulties encountered in this program with the rudder servo and rate gyros should be eliminated.

4. In any efforts to utilize the stalled regime tactically, the problem of tail buffeting should be carefully investigated.

5. A study should be made of the application of these results and of the results of any further analysis of this data to the problem of spins and spin entries.

REFERENCES

1. Campbell, G. F. and Whitcomb, D. W. "Theoretical and Experimental Investigations of the Longitudinal Stall Dynamics of a Conventional Airplane" Cornell Aeronautical Laboratory TB-498-F-1 28 May 1948

2. Campbell, G. F. and Bull, G. "Determination of the Longitudinal Stall Dynamics of a PT-26 Airplane" Cornell Aeronautical Laboratory Report TB-498-F-2 24 February 1950

3. Campbell, G. F. and Bull, G. "Analytical Investigation of Lateral Motions of an Airplane at the Stall With and Without Autopilot" Cornell Aeronautical Laboratory Report TB-556-F-1 7 April 1949

Contrails

TABLE I

DIMENSIONS AND LEADING PARTICULARS FOR

PT-26 NX69593

I. PRINCIPAL DIMENSIONS AND AREAS

A. Airplane - General

Span	36 ft. - 0
Length (over-all)	27 ft. 11-27/64 in.

B. Wings

Airfoil Section	
Root	NACA 2416
Tip	NACA 4409
Incidence (average)	+1°
Dihedral - Trailing edge in wing ref. plane	center section 4° Outer panel 6°
Sweepback (Leading edge)	3° 36′
Rear beam (straight line) at	69% root chord
Aspect Ratio	6.48
Taper Ratio (λ)	.600
Mean Aerodynamic Chord (MAC)	69.7 in.
Total Wing Area	200 sq. ft.
Wing Flap Area (total)	15.86 sq. ft.
Total aileron area aft of hinge line (including tabs)	17.64 sq. ft.

C. Empennage

Horizontal Stabilizer	
Tail length (distance of C/4 of wing MAC to elevator hinge line)	187 6 in.
Tail length (distance from C/4 of wing MAC to C/4 of tail MAC	168.16 in.
Span	145-5/8 in.
Incidence to thrust line	
Max. leading edge up	-3.7°
Max. leading edge down	-9.1°
Aspect Ratio (AR)	3.70
Taper Ratio (λ)	.464
Elevator - ratio of elevator area to area of horizontal stabilizer (including elevator)	.336

Contrails

TABLE I (contd.)

Total horizontal tail area	39.9 sq. ft.
Elevator area aft of hinge line (including tabs)	13.4 sq. ft.
Elevator trim tabs	.49 sq. ft.
Total vertical tail area	20.56 sq. ft.
Rudder	14.41 sq. ft.

D. Fuselage

Plan area	*54.9 sq. ft.
Side area	*96.7 sq. ft.
Distance from nose to 25% MAC	106.82 in.

II. CONTROL SURFACE TRAVEL

	UP	DOWN
Ailerons	36°	16°

Elevator -

STABILIZER SETTING	STAB. INCIDENCE	ELEVATOR TRAVEL UP	DOWN
Original configuration	+ ¼°	35°	25°
Modified, max. leading edge-up	-3.7°	35°	17°
Modified, max. leading edge-down	-9.1°	32°	20°

Elevator trim tabs	11°	11°
Wing flaps		50°
Rudder (measured at bottom of rudder)	32° Right	32° Left

III. CENTER OF GRAVITY (c.g.) RANGE:
(C.A.A. Spec. for NC Airplane)

Forward limit	19.7% MAC
Aft limit	32% MAC

IV. GROSS WEIGHT - FLIGHT TEST CONFIGURATION 2911 lb.

*approximate.

Contrails

TABLE II

AUTOPILOT SENSITIVITIES USED IN DATA FLIGHTS

Aileron/Roll Angle	δ_a / ϕ	10 deg/deg
Aileron/Roll Rate	δ_a / p	1.4 deg/deg/sec
Rudder/Yaw Rate	δ_r / r	3.8 deg/deg/sec
Rudder/Aileron Angle	δ_r / δ_a	0.5 deg/deg

AUTOPILOT AMPLIFIER SETTINGS

	RUDDER	AILERON
Displacement Tap	11	11
Rate Tap	0	0.6
Repeatback Tap	2.4	6.2

AUTOPILOT SERVO GEAR RATIOS

	RUDDER	AILERON
Power Gear	223:1	148:1
Repeatback Gear	105:1	750:1

TABLE III

VALUES OF AIRPLANE PARAMETERS USED IN CALCULATIONS

(Taken From Ref. 3)

C_L	= 1.025	I_x	= 2155 slug ft.^{2}
α	= 20°	I_z	= 4040 " "
γ	= 8.36°	I_{xz}	= -580 " "
U_o	= 107.6 ft/sec	L_p	= .754 1/sec
C_{l_p}	= .117	L_r	= 2.000 1/sec
C_{l_r}	= .310	L_v	= -.0675 1/ft sec
C_{l_β}	= -.175	L_{δ_a}	= -.050 1/deg sec^2
$C_{l_{\delta_a}}$	= -.0013	N_p	= .392 1/sec
C_{n_p}	= .114	N_r	= -.348 1/sec
C_{n_r}	= -.101	N_v	= -.0084 1/ft sec
C_{n_β}	= .044	N_{δ_a}	= .0185 1/deg sec^2
$C_{n_{\delta_a}}$	= .00090	N_{δ_r}	= -.0195 1/deg sec^2
$C_{n_{\delta_r}}$	= -.00095	Y_v	= -.246 1/sec

Contrails

TABLE IV

SUMMARY OF FLIGHTS

DATE	FLT.	DURATION	REMARKS
5-8-50	57	0:40	Check new sideslip boom for vibration
5-8-50	58	0:50	Same as Flt. 57, boom shortened one foot
5-23-50	59	1:10	Determine filter requirements for sideslip and rate gyros.
6-5-50	60	1:05	Determine filter requirements for sideslip and rate gyros.
6-6-50	61	1:05	Determine filter requirements for sideslip and rate gyros.
6-22-50	62	1:30	Determine filter requirements for sideslip and rate gyros.
9-18-50	63	1:50	Movies of stalled airplane
10-2-50	64	2:30	Movies of stalled airplane
12-21-50	65	3:00	Shakedown autopilot installation. Unstalled.
1-9-51	66	2:20	Shakedown autopilot installation. Unstalled and stalled.
1-10-51	67	0:55	Exploration of automatically controlled stalls. Oscillograph door opened.
2-9-51	68&68a	1:45	Exploration of automatically controlled stalls Stabilizer stuck down on Flight 68.
3-5-51	69	2:25	Exploration of automatically controlled stalls. Also shakedown new engine.
3-6-51	70	2:30	Exploration of automatically controlled stalls
3-9-51	71	1:15	Investigation of autopilot system instability. Unstalled
3-22-51	72	2:40	Check effect of varying sensitivities on autopilot system instability
3-26-51	73	1:10	Check operation of disturbance input circuit.
3-27-51	74	1:40	Check effect of autopilot dynamics on autopilot system instability.
4-5-51	75&75a	4:00	Stalls with autopilot. Effect of varying sensitivity on stabilization of airplane.
4-18-51	76	1:40	Stalls with autopilot. Effect of varying sensitivity on stabilization of airplane
4-26-51	77	2:00	Stalls with autopilot. Autopilot stabilized stalled airplane for 20 - 70 sec.
5-4-51	78	3:10	Stalls with autopilot. Check repeatability of good performance of Flight 77.
5-10-51	79	2:55	Stalls with autopilot. Autopilot did not stabilize airplane very well
5-15-51	80	2:00	Stalls with autopilot. Autopilot did not stabilize airplane very well.
5-31-51	81	2:10	Better control than Flights 79 and 80

Contrails

TABLE IV (contd.)

6-5-51	82	2:45	As 81. Unstalled runs to check autopilot performance.
6-12-51	83	2:25	Autopilot made stiffer. Less autopilot system instability. Stabilization still not good.
6-15-51	84	1:55	Calibration of autopilot. No stalls.
6-18-51	85	2:00	System instability worse (higher gains were used). Stabilization mediocre.
6-26-51	86	1:30	Movies of uncontrolled, and manually and automatically controlled stalls.
7-19-51	87	2:00	Automatic stabilization fair. Check on β sensing.
7-25-51	88	2:45	Automatic stabilization fair. Check on β sensing. Unstalled runs for trouble-shooting.
8-10-51	89	2:10	Automatic stabilization fair. Check on β sensing.
2-15-52	90	1:00	Exercise and shakedown of airplane and instrumentation.
9-4-52	91	2:20	Autopilot system instability eliminated. Stabilization still only fair.
9-11-52	92	3:05	Human pilot flew one control, autopilot the other, to isolate troubles.
9-29-52	93	2:40	Yaw rate sensing provided to rudder. Stabilization excellent.
10-1-52	94	2:50	Check repeatability of stabilization, and effect of power. Stabilization excellent.
10-16-52	95	1:30	Select amplitude of disturbance input.
10-24-52	96	2:45	Select amplitude of disturbance input. Movies of stabilized stall.
12-1-52	97	1:10	Improve disturbance input. Servo clutch slipped.
12-4-52	98	0:55	Improve disturbance input. Servo release cable broke.
12-19-52	99	1:50	Improve disturbance input. Check effect of autopilot adjustments. Bad weather.
1-6-53	100	2:55	Improve disturbance input. Check effect of autopilot adjustments. Voltage regulator failed.
1-13-53	101	3:05	Improve disturbance input. Check effect of autopilot adjustments.
2-10-53	102	2:15	Data flight. Transients in automatically controlled stall. Stabilization excellent.
2-19-53	103	1:35	Batteries gave out. No data
3-11-53	104	1:00	Short dircuit in photo-observer. No data.
3-12-53	105	3:20	Data flight. Transients in automatically controlled stall. Stabilization excellent.
3-18-53	106	3:00	Data flight. Transients in automatically controlled stall. Stabilization excellent.
3-24-53	107	3:00	Data flight. Transients in automatically controlled stall. Stabilization excellent.
4-2-53	108	1:20	Data flight. Transients in automatically controlled stall. Stabilization excellent.

Contrails

Figure 1. TEST AIRPLANE

Contrails

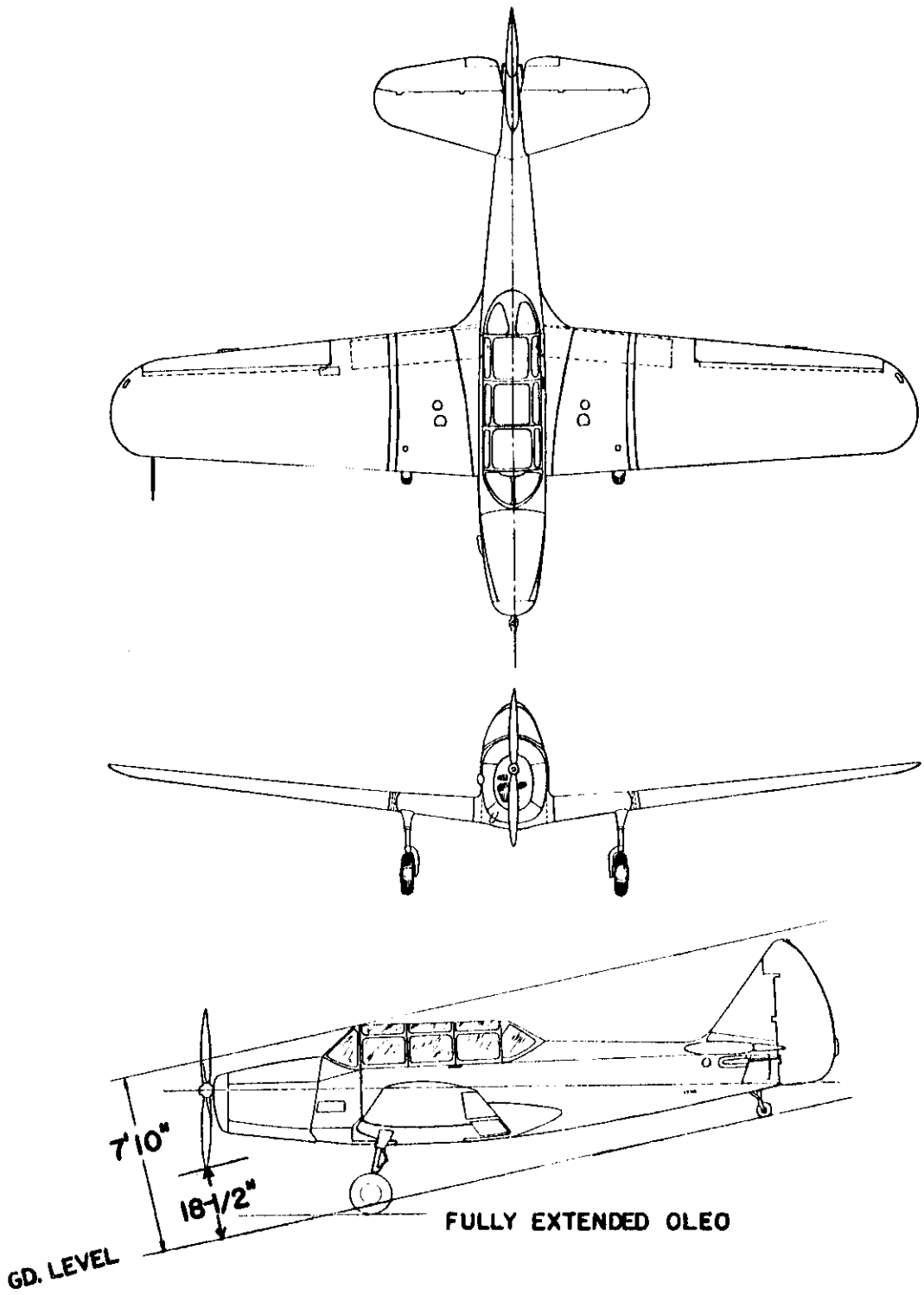
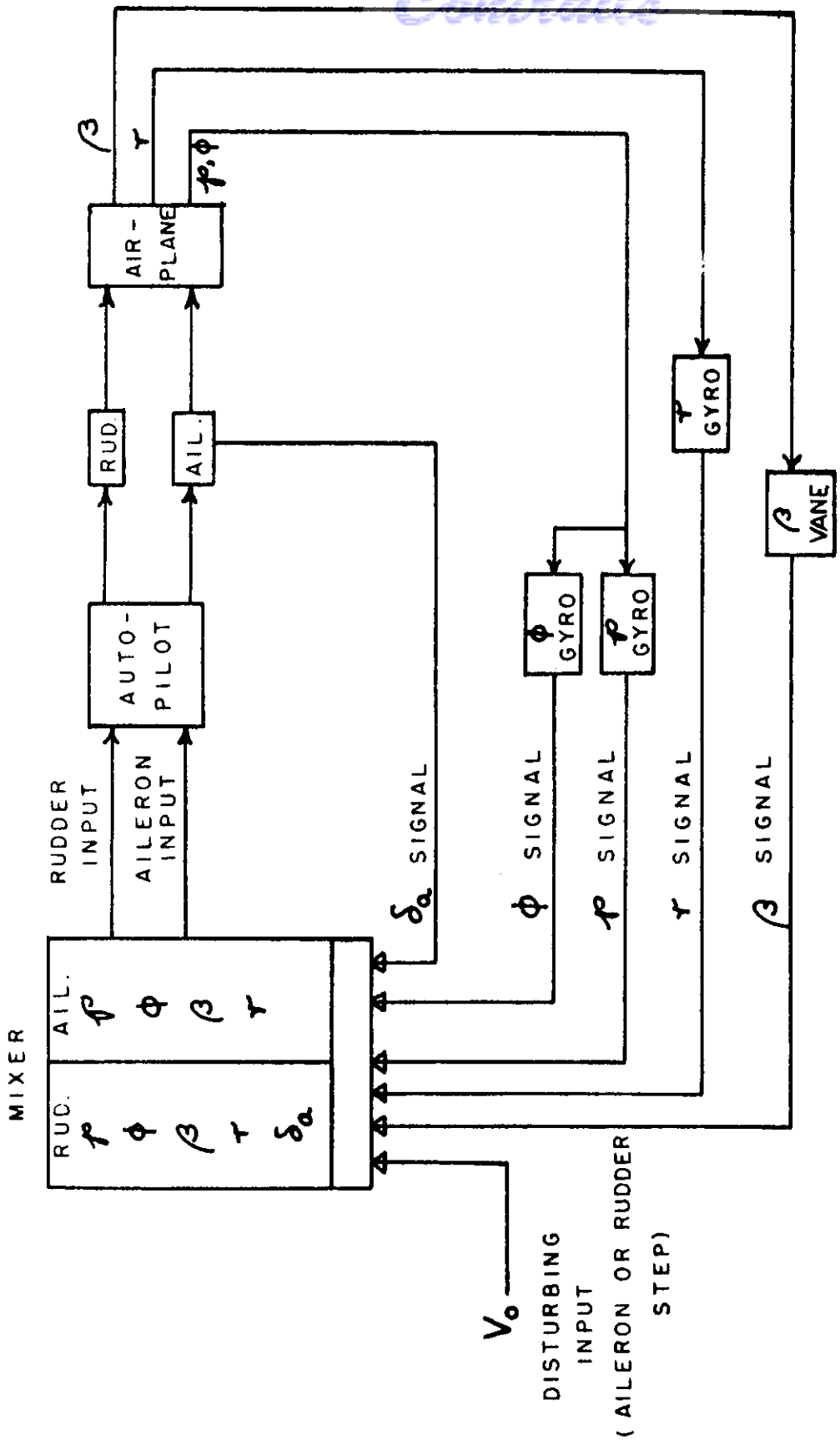


Figure 2 – THREE VIEW OF TEST AIRPLANE

Contrails



BLOCK DIAGRAM

PT-26 AUTOMATIC CONTROL SYSTEM

Figure 3

Contrails

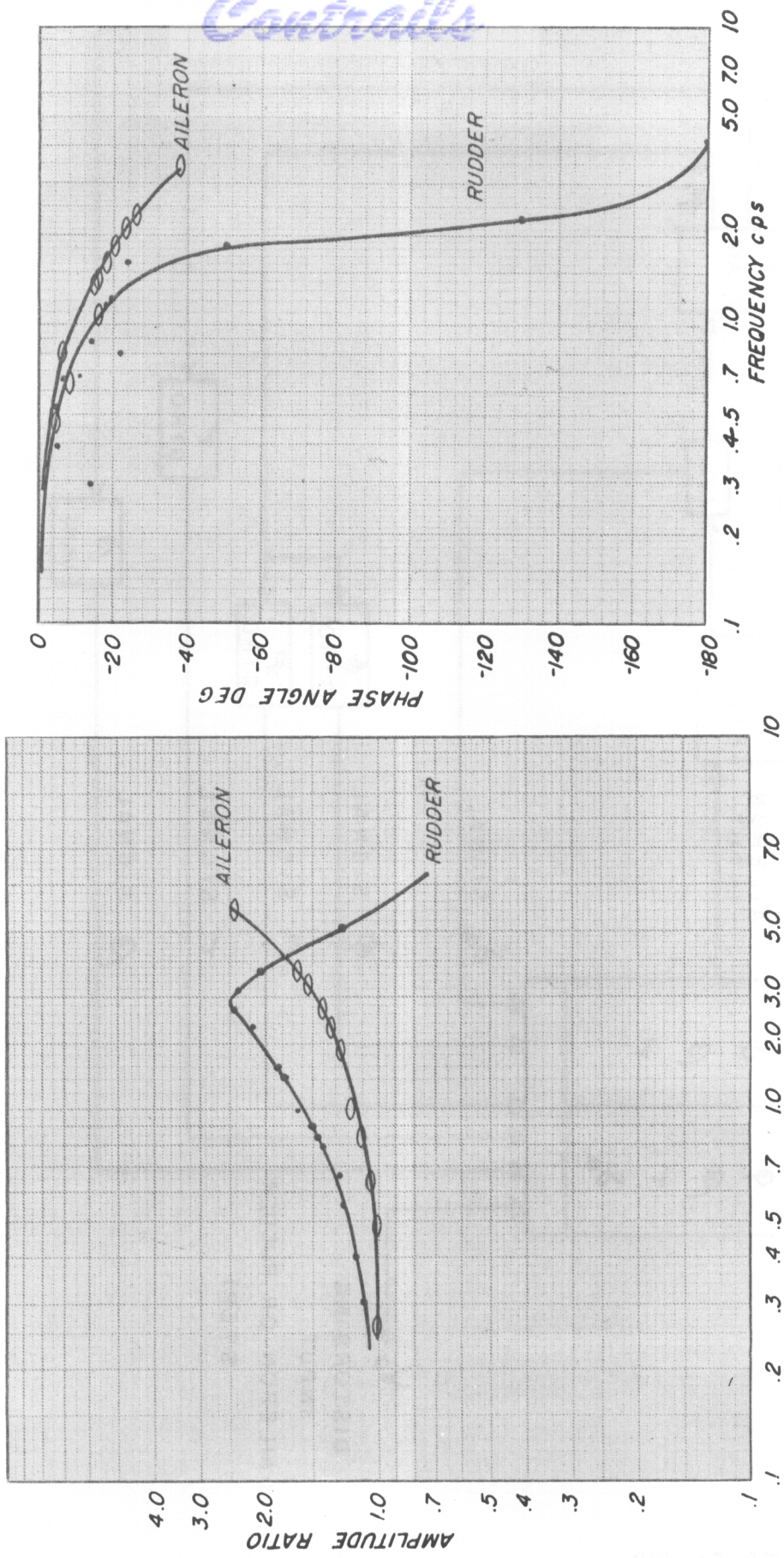


Fig. 4

Contrails

PT-26

ROLL & YAW RATE GYROS

FREQUENCY RESPONSE

8-7-52

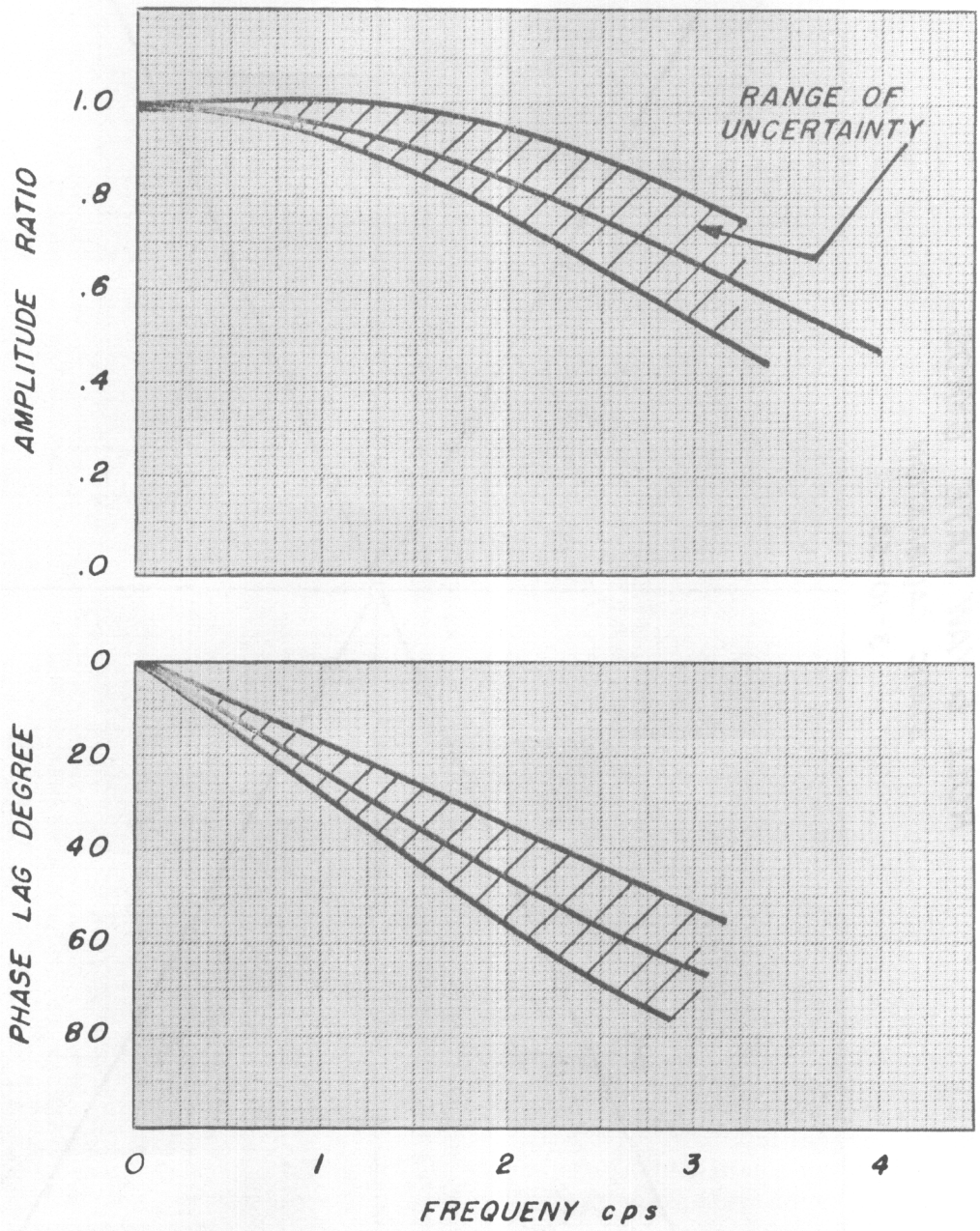


Fig. 5

PT-26

ROLL & YAW RATE GYROS

STATIC CALIBRATION

3-10-53

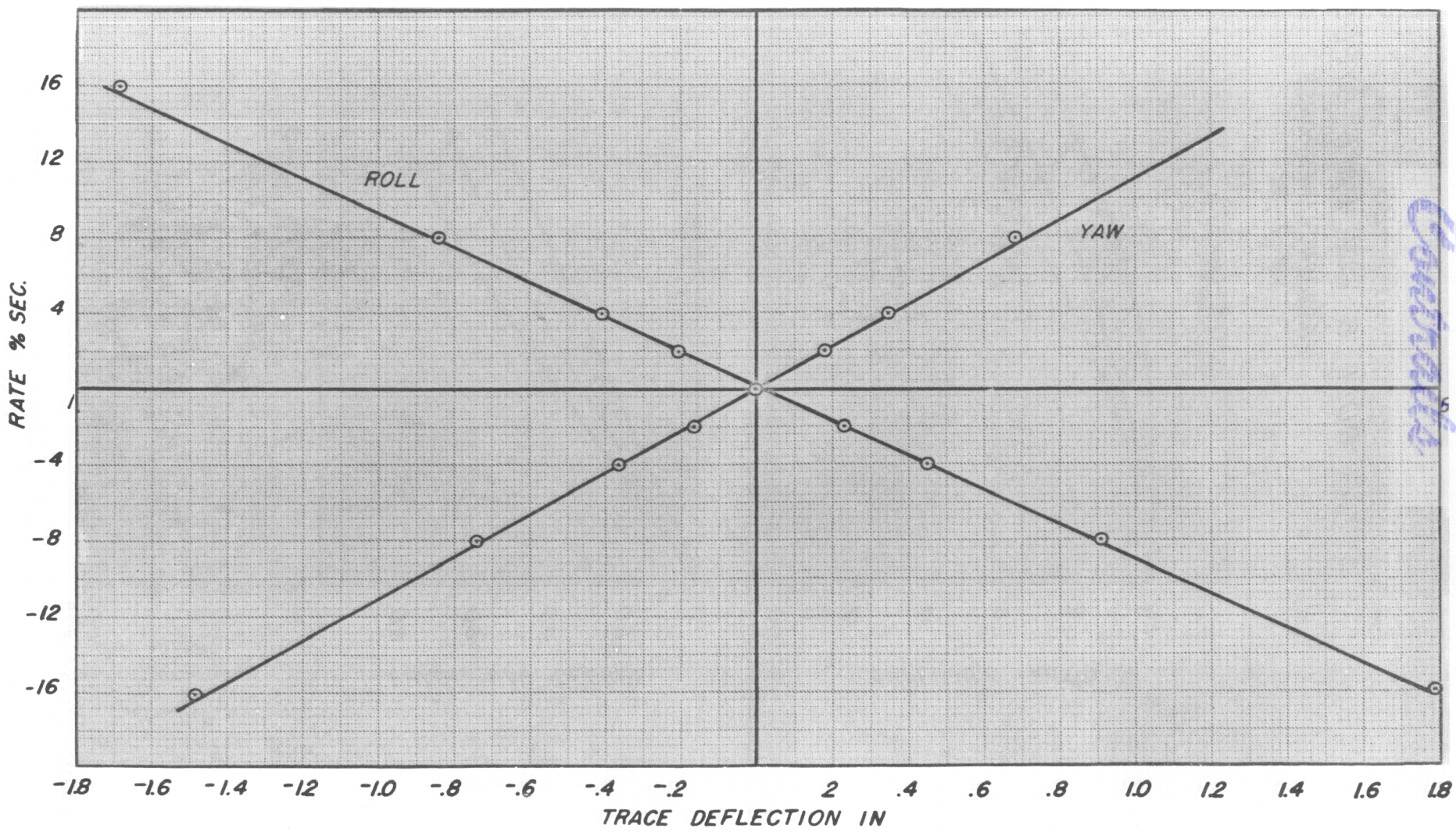


Fig. 6

PT-26 $\alpha = 28°$ HARMONIC CONTENT OF VARIABLES
AILERON DISTURBANCE RUN 105-10b
BOTH CONTROLS OPERATED BY AUTOPILOT

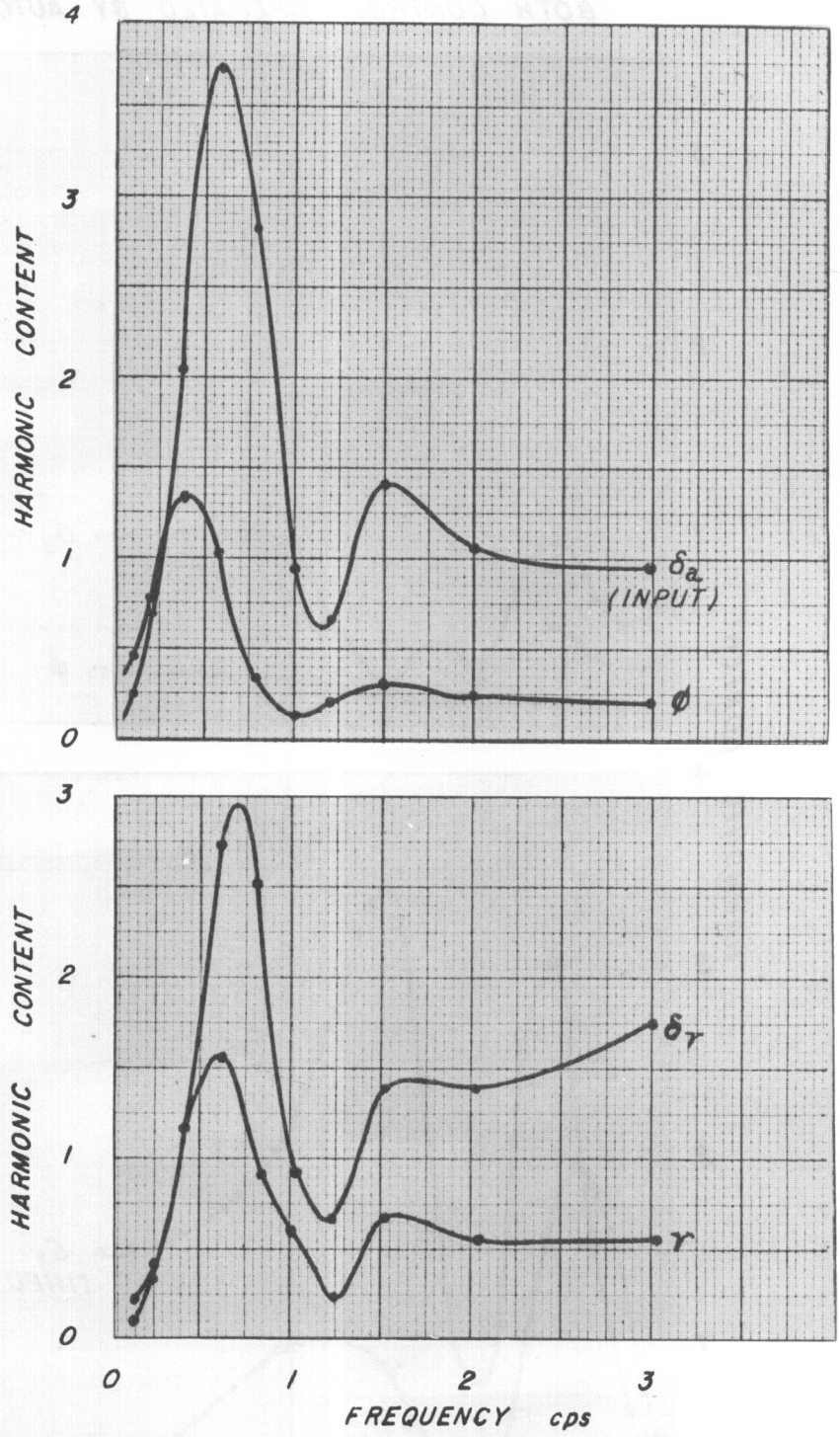


Fig. 7

PT-26 $\alpha = 28°$

HARMONIC CONTENT OF VARIABLES
RUDDER DISTURBANCE
RUN 105-6d
BOTH CONTROLS OPERATED BY AUTOPILOT

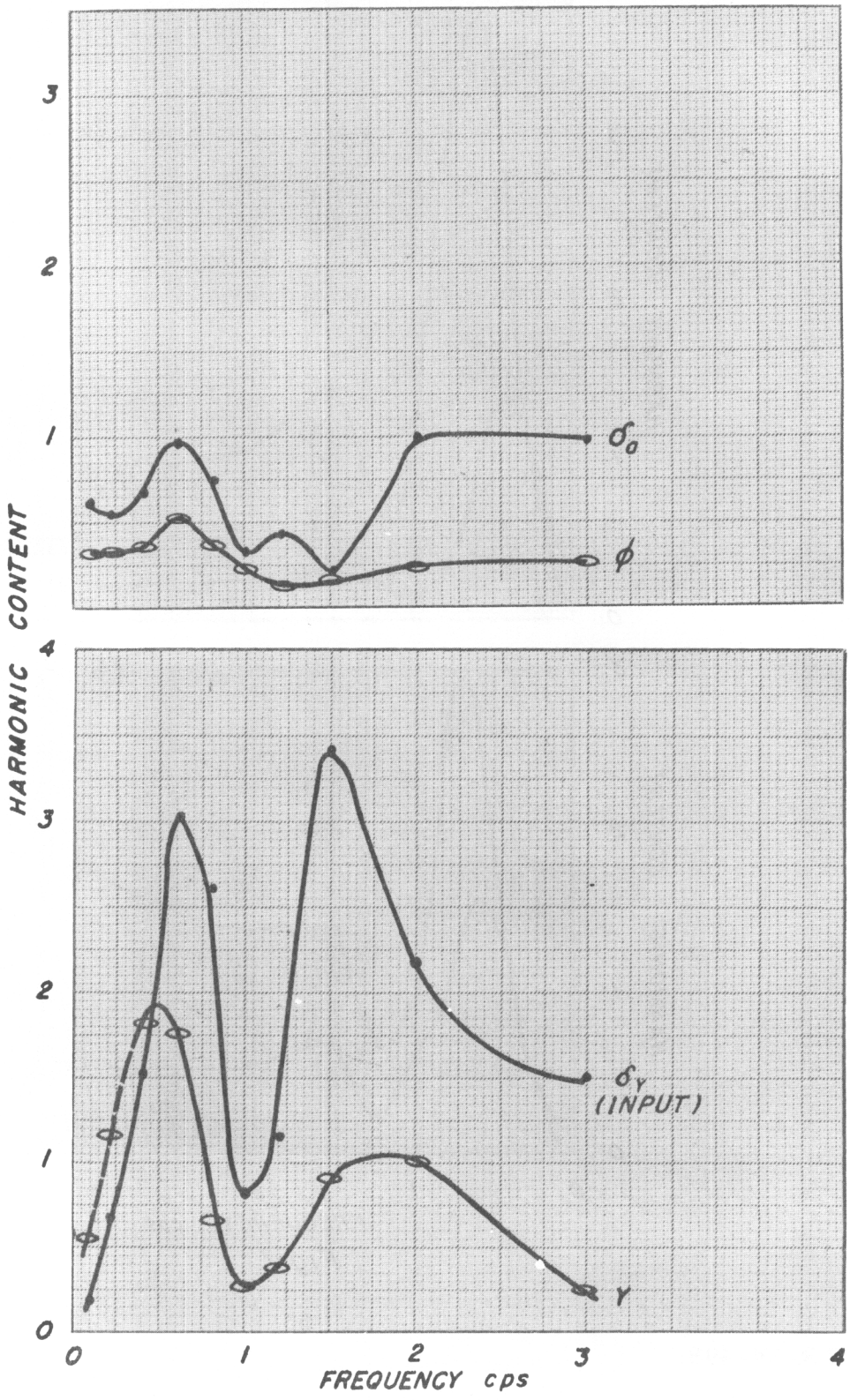


Fig. 8

PT-26 $\alpha = 28°$

ϕ/δ_a WITH δ_r OPERATING

CONTROLS OPERATED BY AUTOPILOT

RUDDER DISTURBANCE

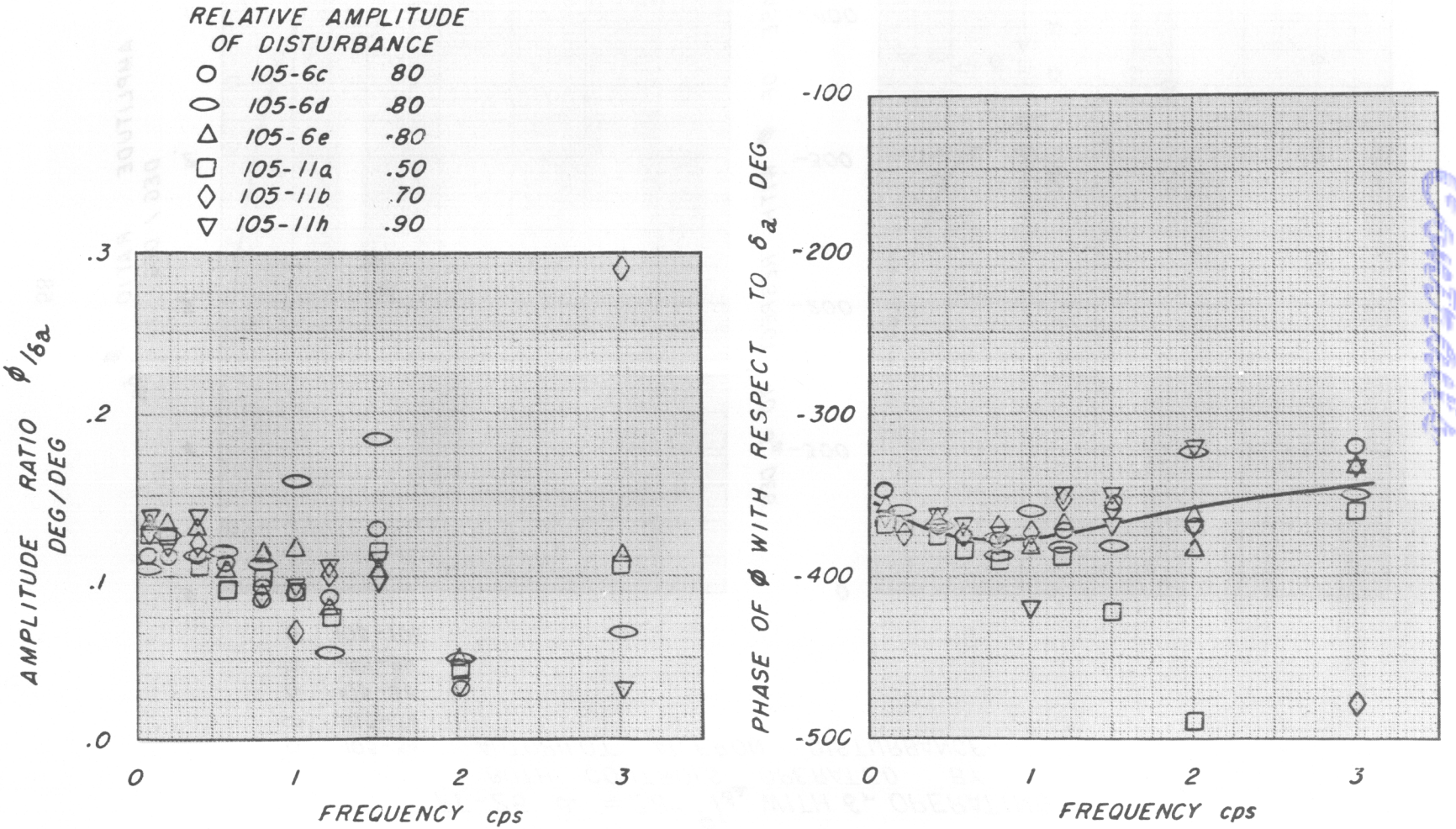


Fig. 9

Contrails

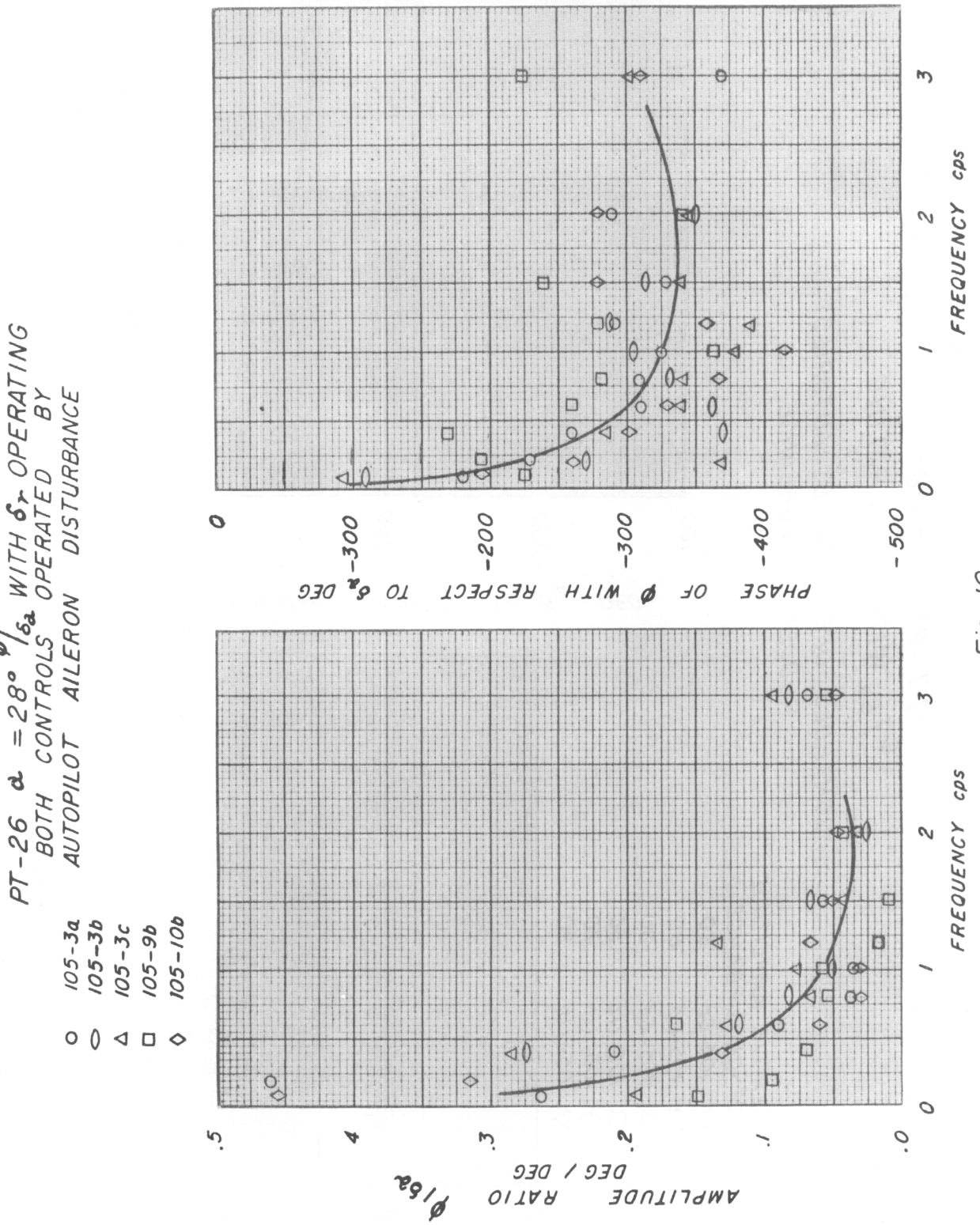


Fig. 10

PT-26 $\alpha = 28°$ r/δ_r WITH δ_a OPERATING
BOTH CONTROLS OPERATED BY AUTOPILOT. RUDDER DISTURBANCE

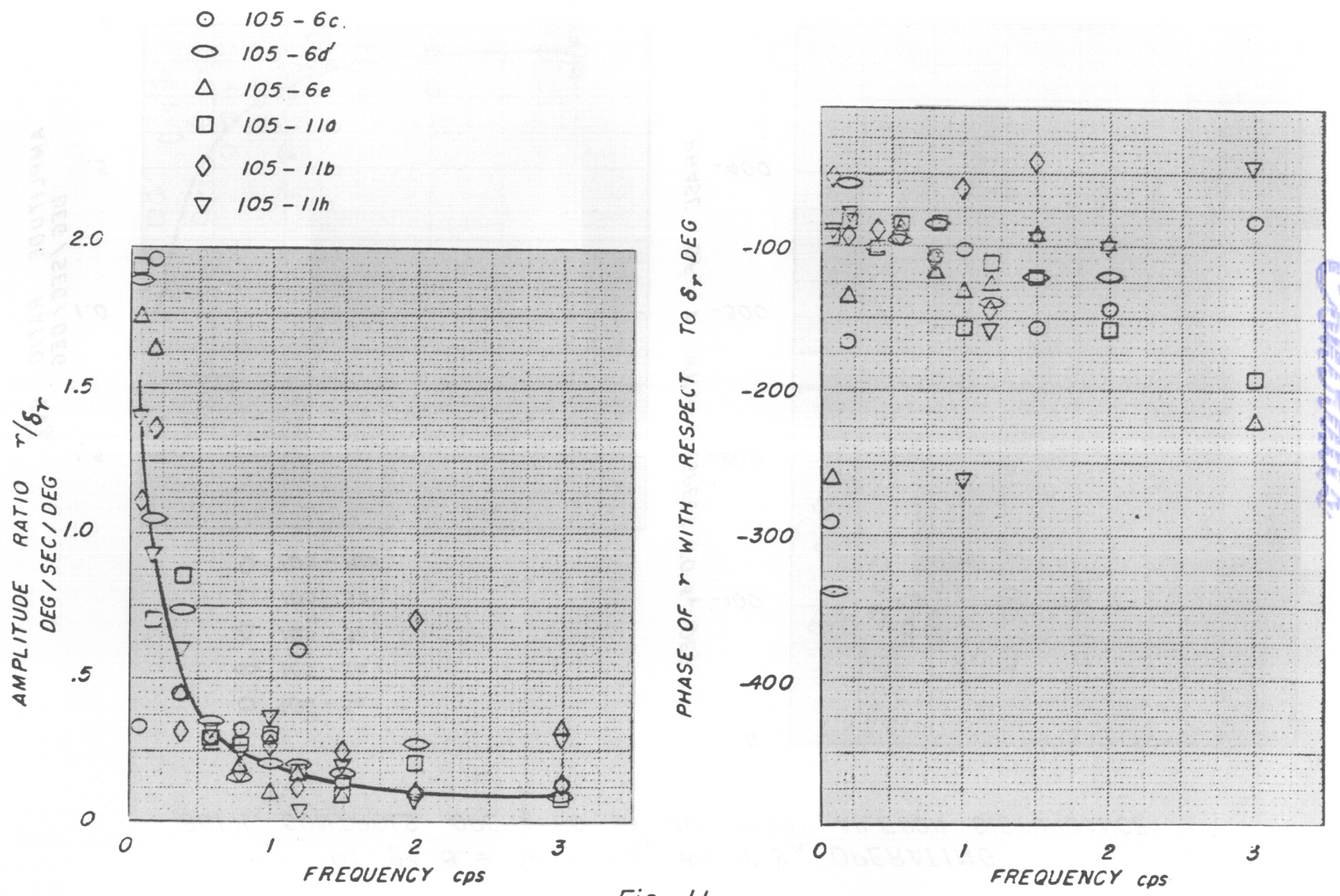


Fig. 11

PT-26 $\alpha = 28°$ r/δ_r WITH δ_a OPERATING
BOTH CONTROLS OPERATED BY AUTOPILOT AILERON DISTURBANCE

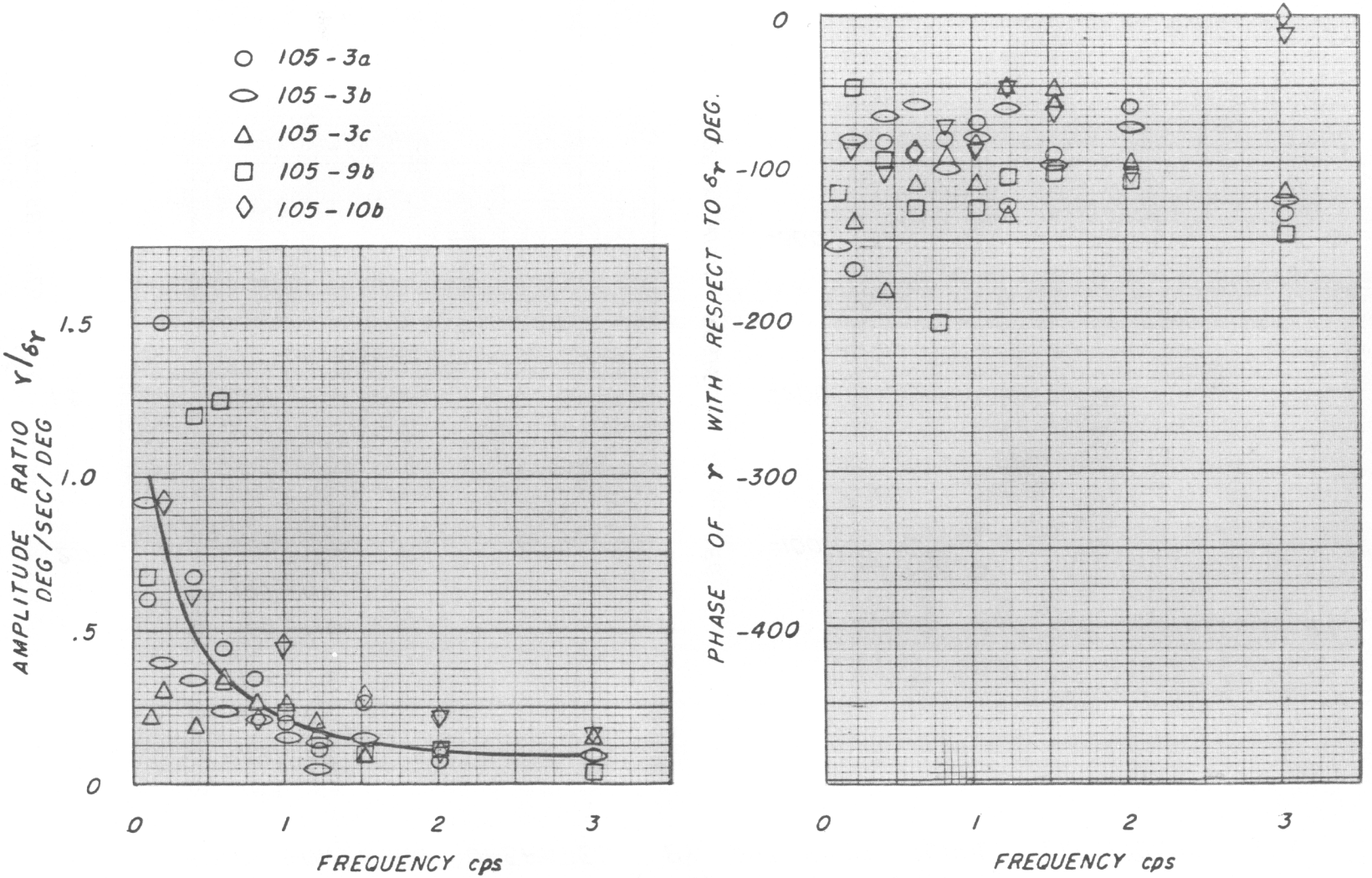


Fig. 12

Contrails

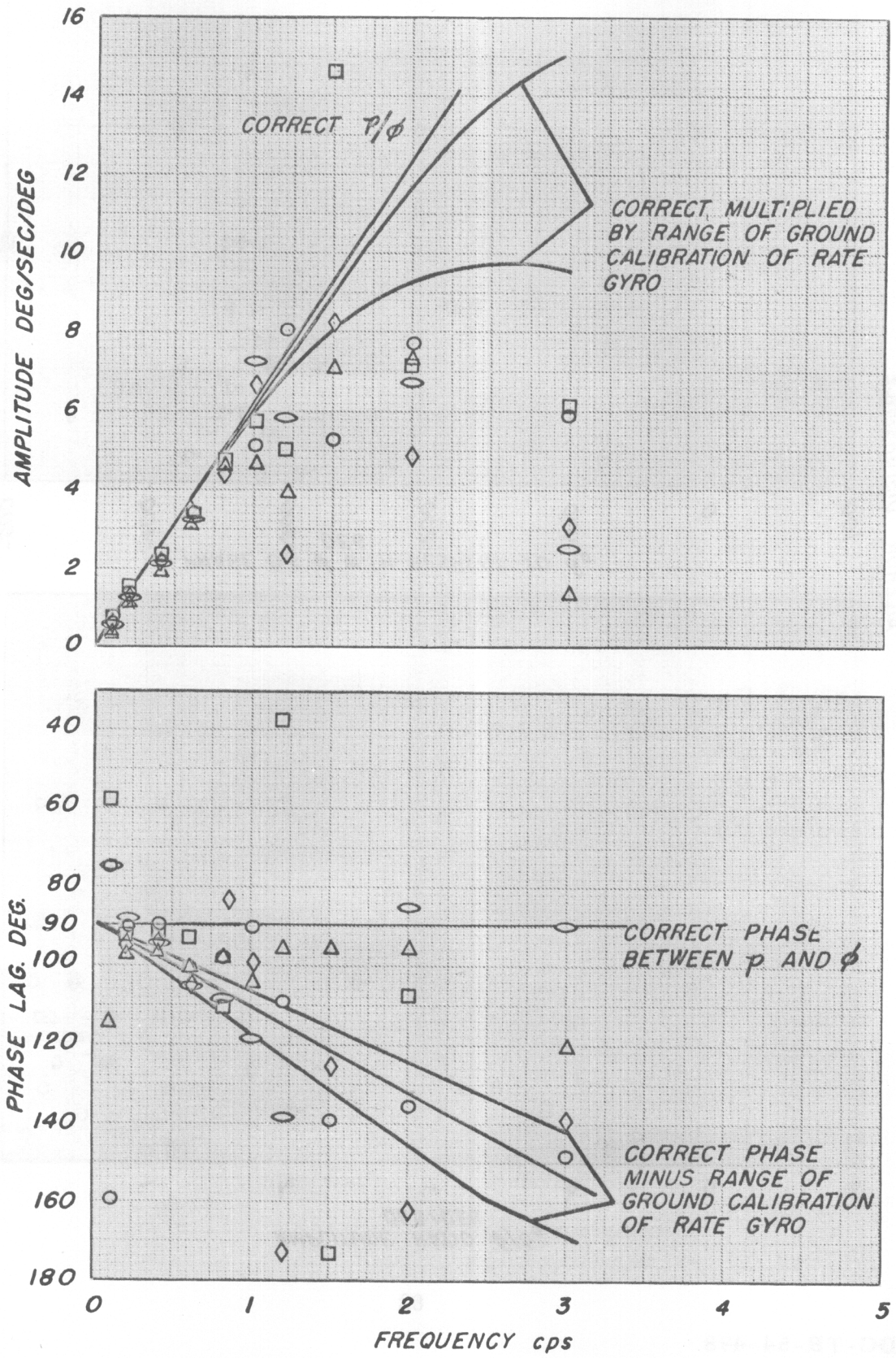


Fig. 13

PT-26 α AS NOTED

ϕ/δ_o WITH δ_r OPERATING

BOTH CONTROLS OPERATED BY AUTOPILOT

AILERON DISTURBANCE

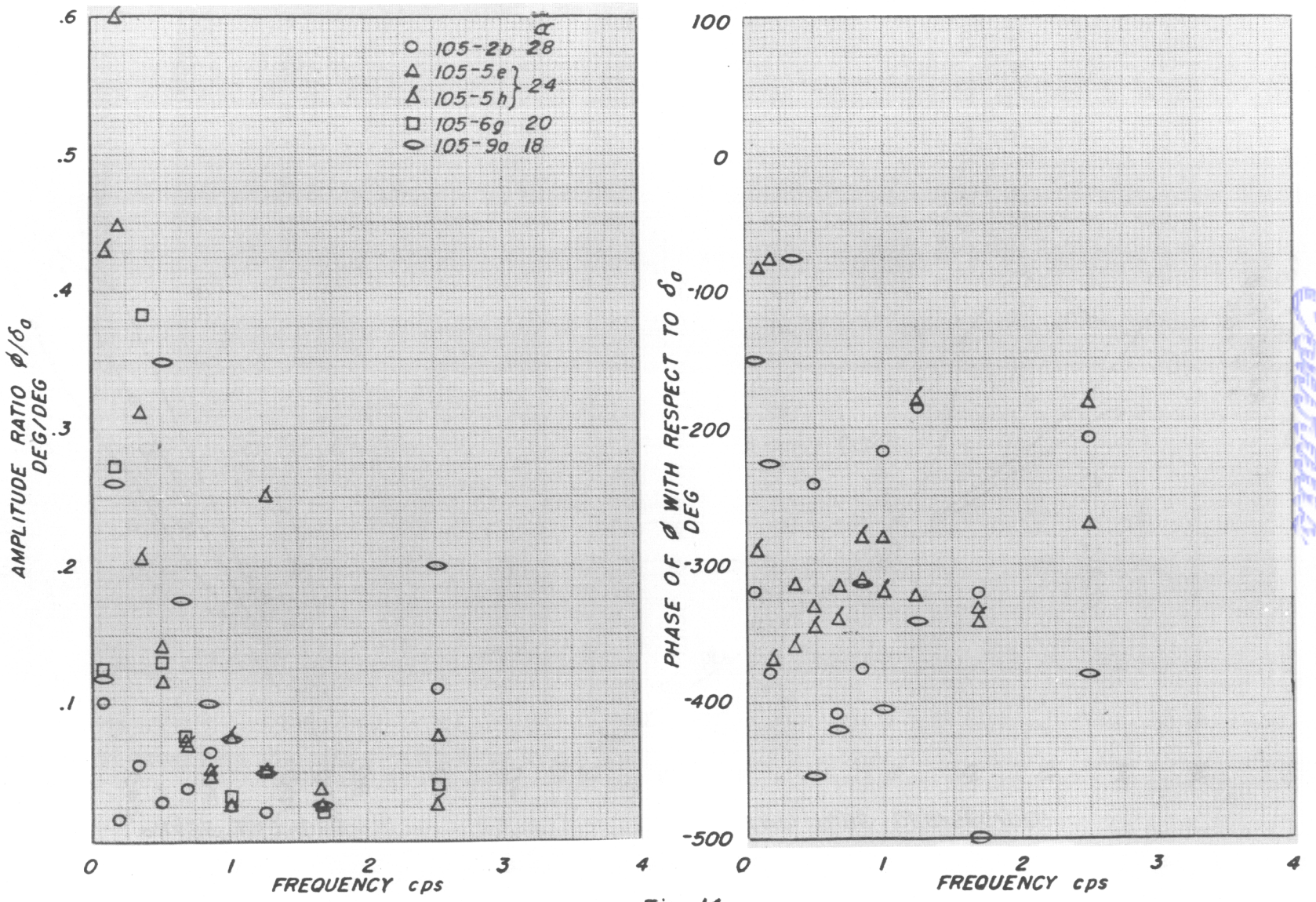


Fig. 14

Contrails

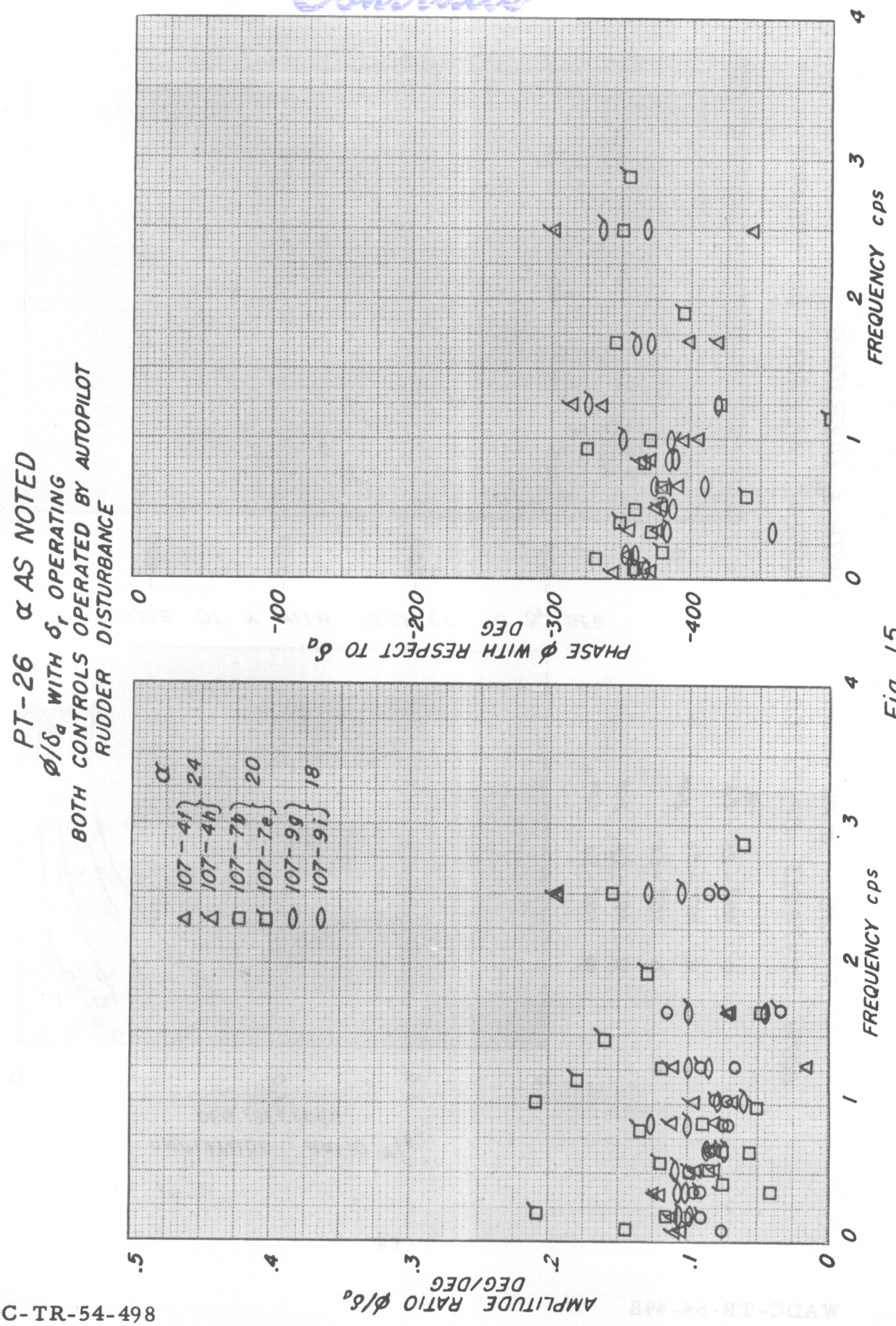


Fig. 15

Contrails

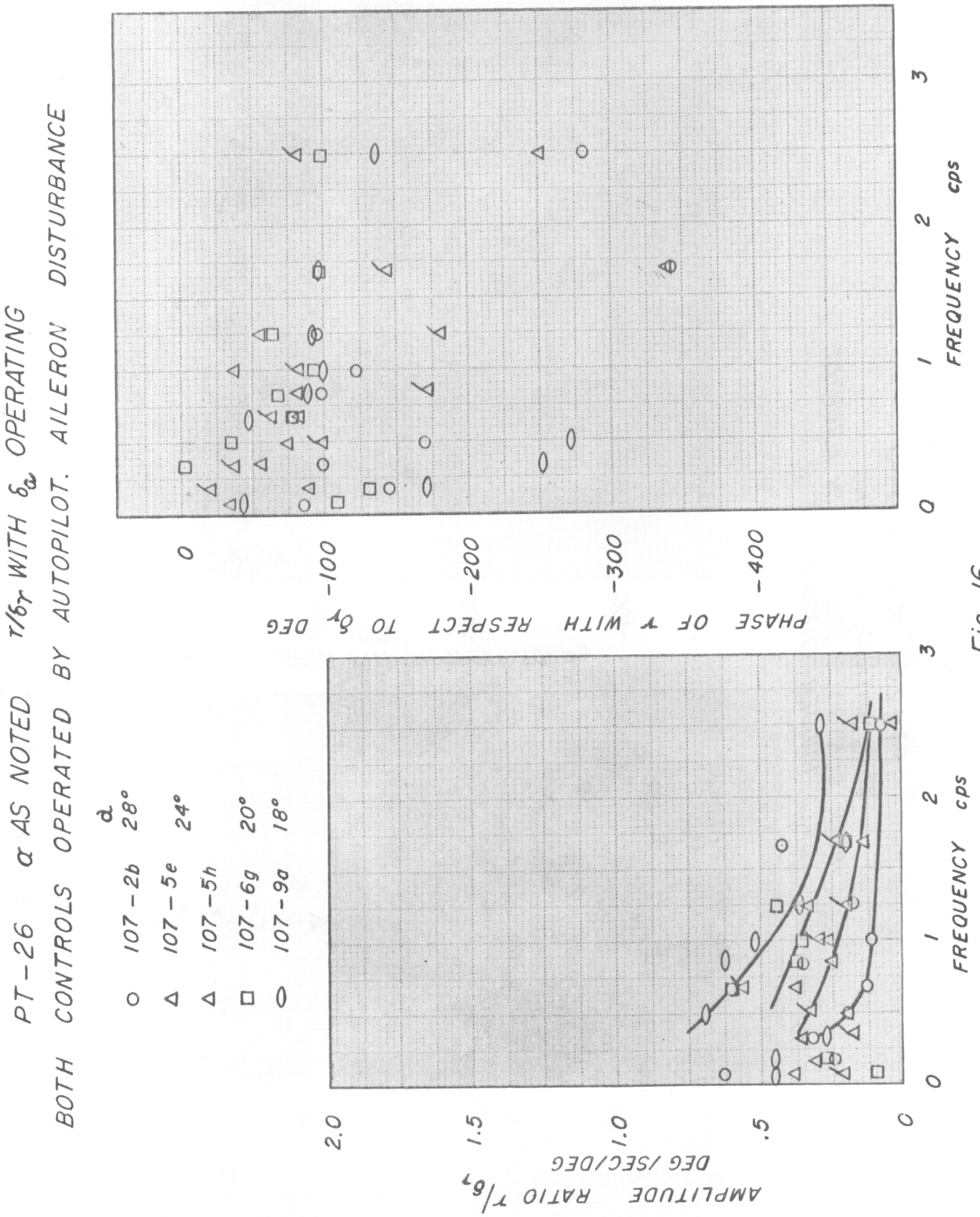


Fig. 16

Contrails

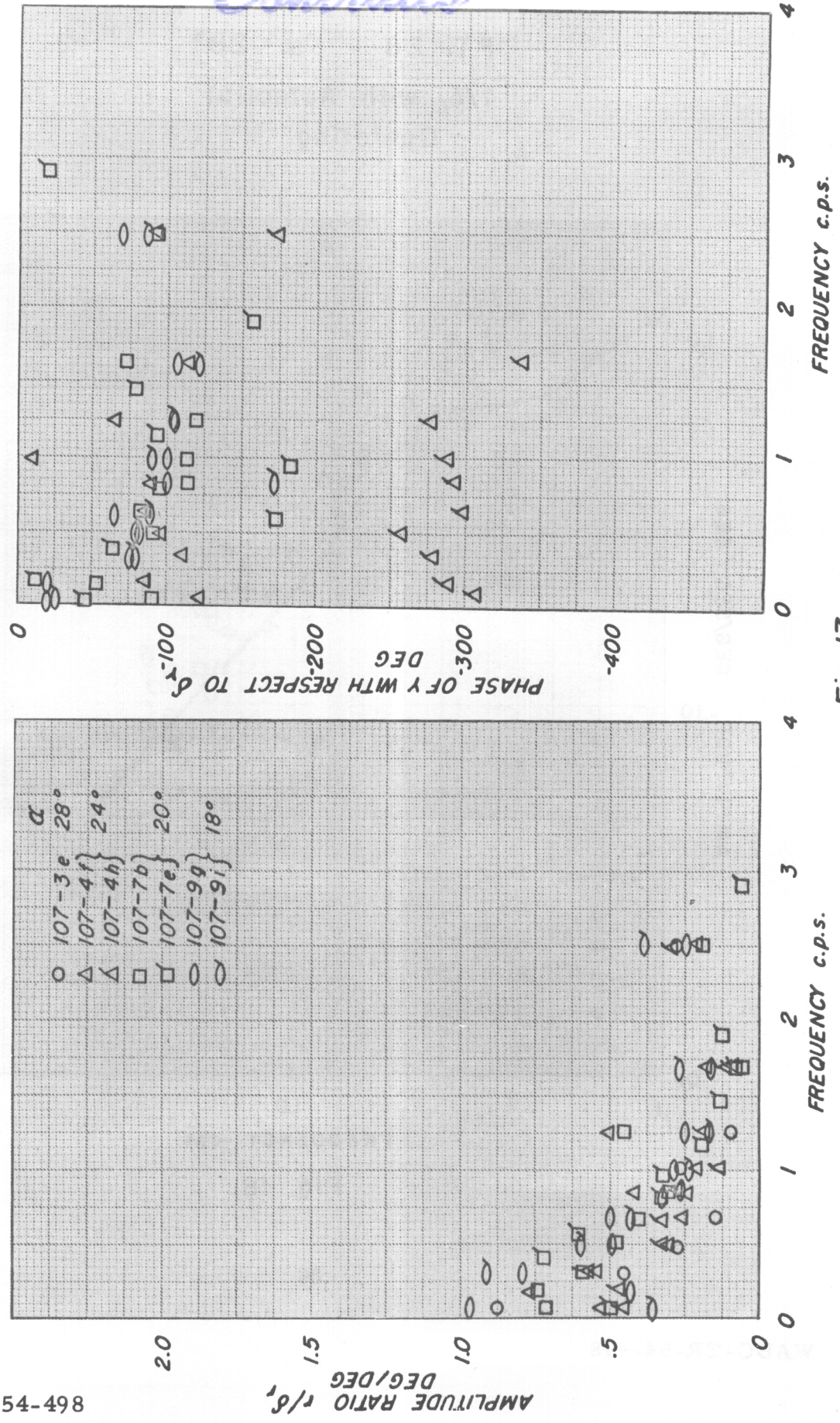


Fig. 17

Contrails

PT-26 $\alpha = 28°$

r/δ_r with Autopilot Operating

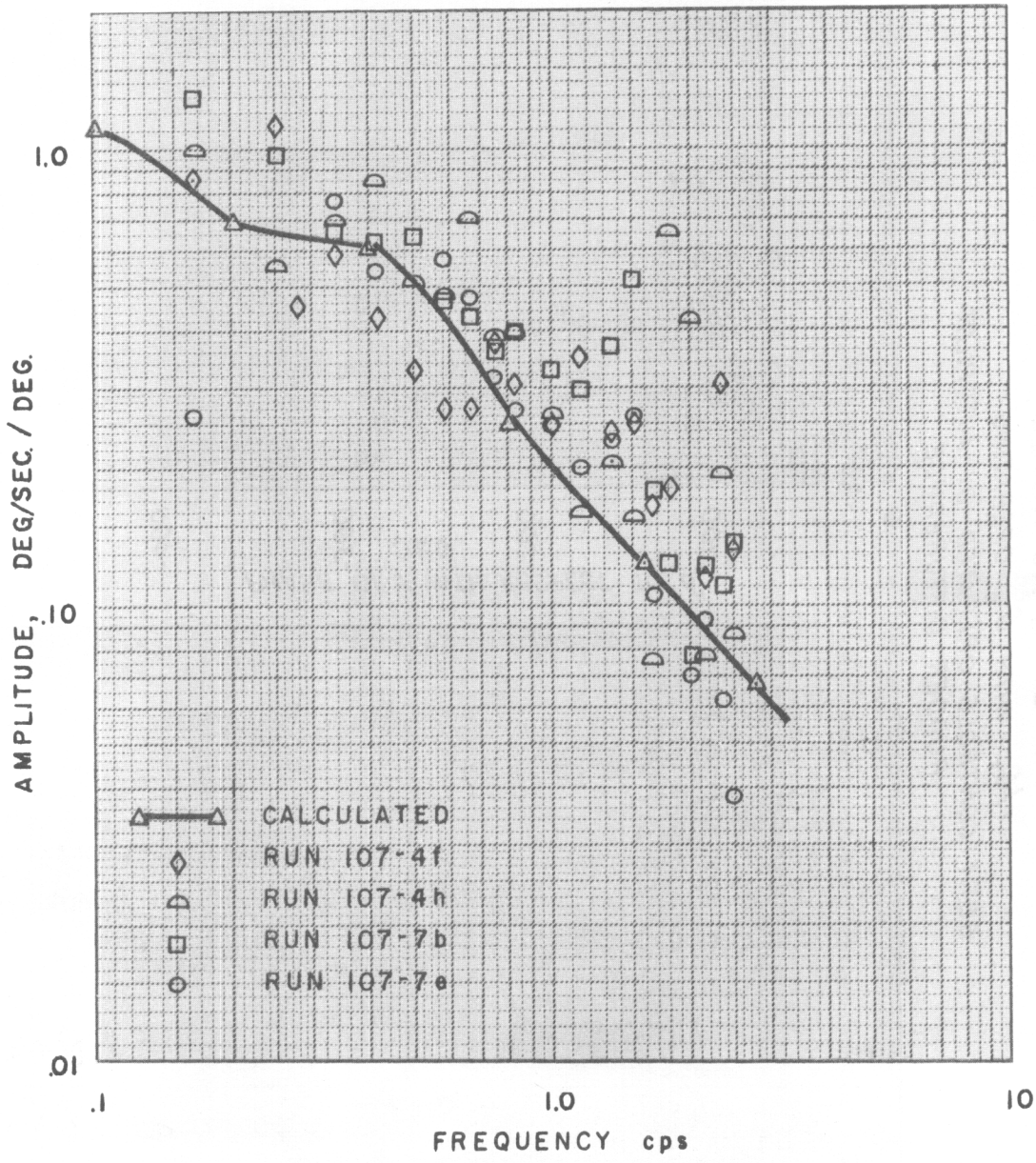


Fig. 18

Contrails

PT - 26 $\alpha = 28°$

r/δ_r with Autopilot Operating

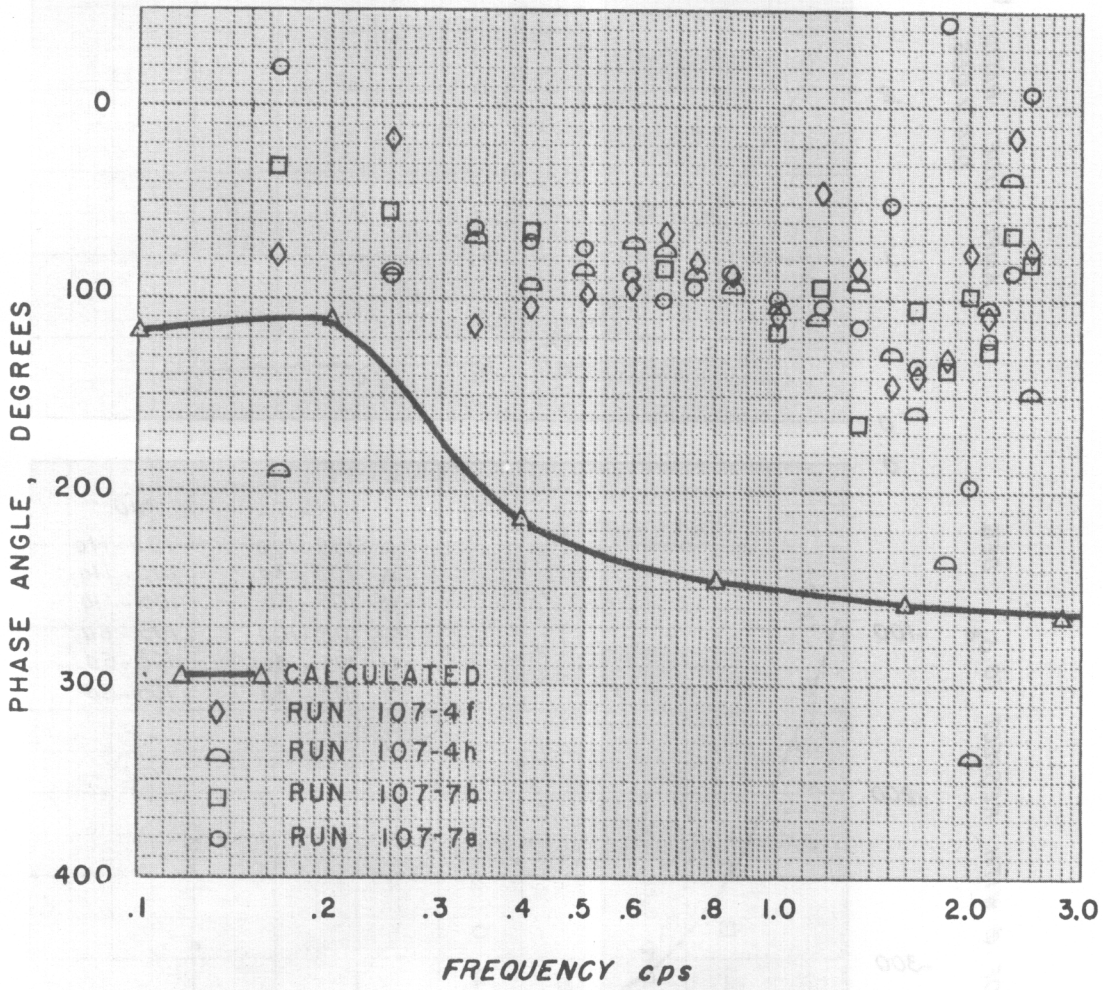


Fig. 19

Contrails

PT-26 $\alpha = 28°$ ϕ/δ_a WITH $\delta_r = 0$

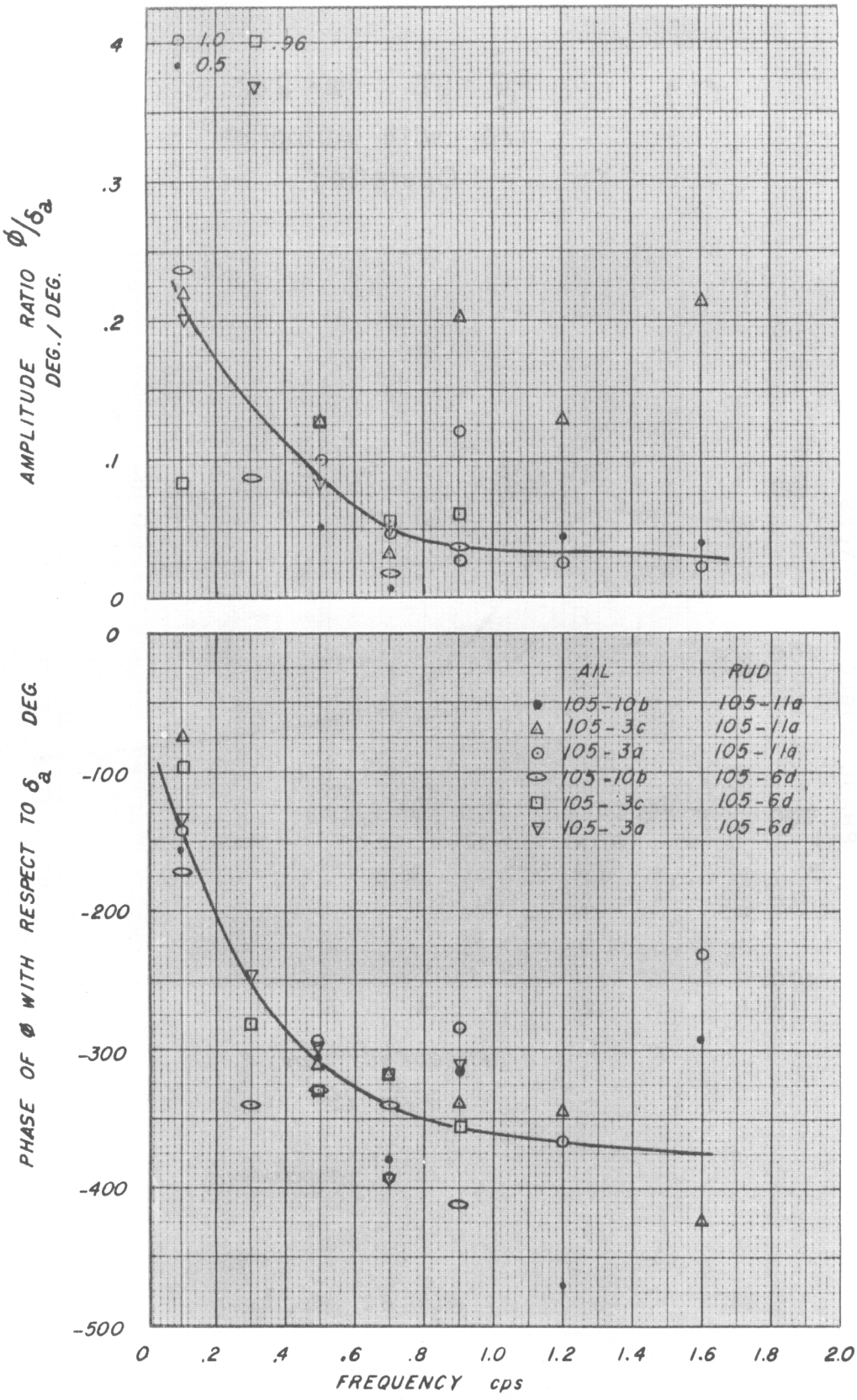


Fig. 20

PT-26 Controls $\alpha = 28°$ r/δ_a WITH $\delta_r = 0$

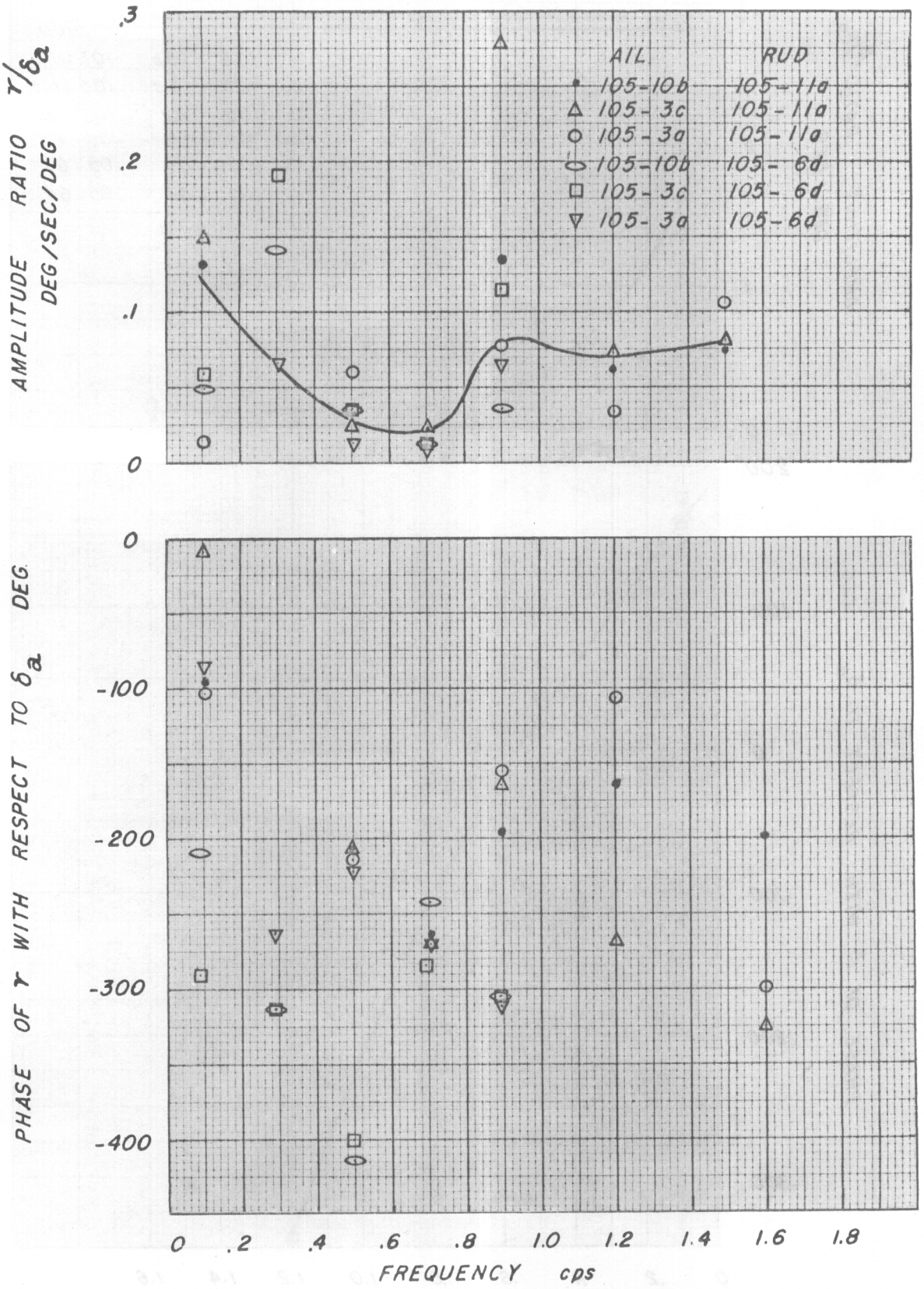


Fig. 21

PT-26 $\alpha = 28°$ β/δ_a WITH $\delta_r = 0$

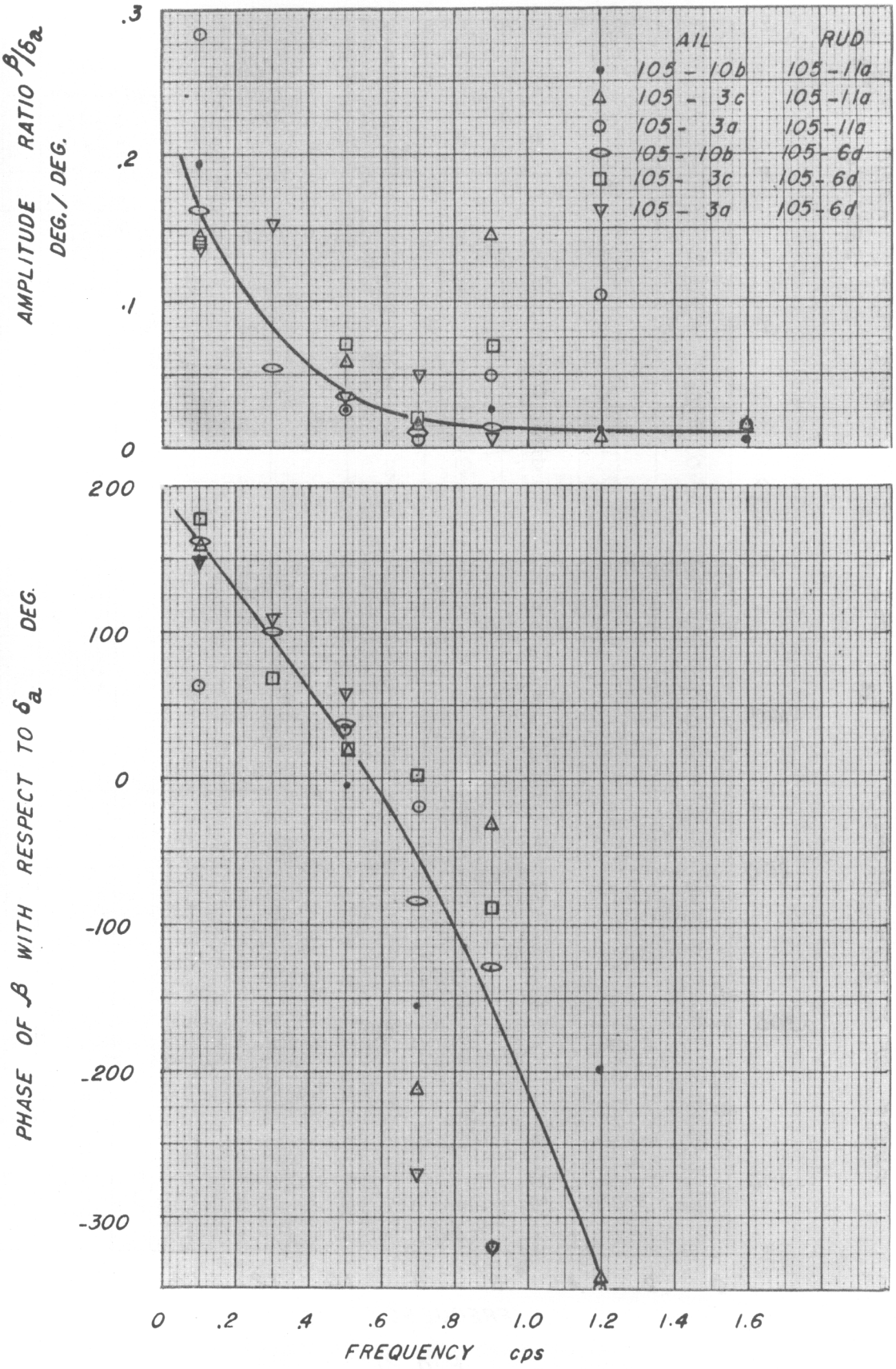


Fig.22

Controls

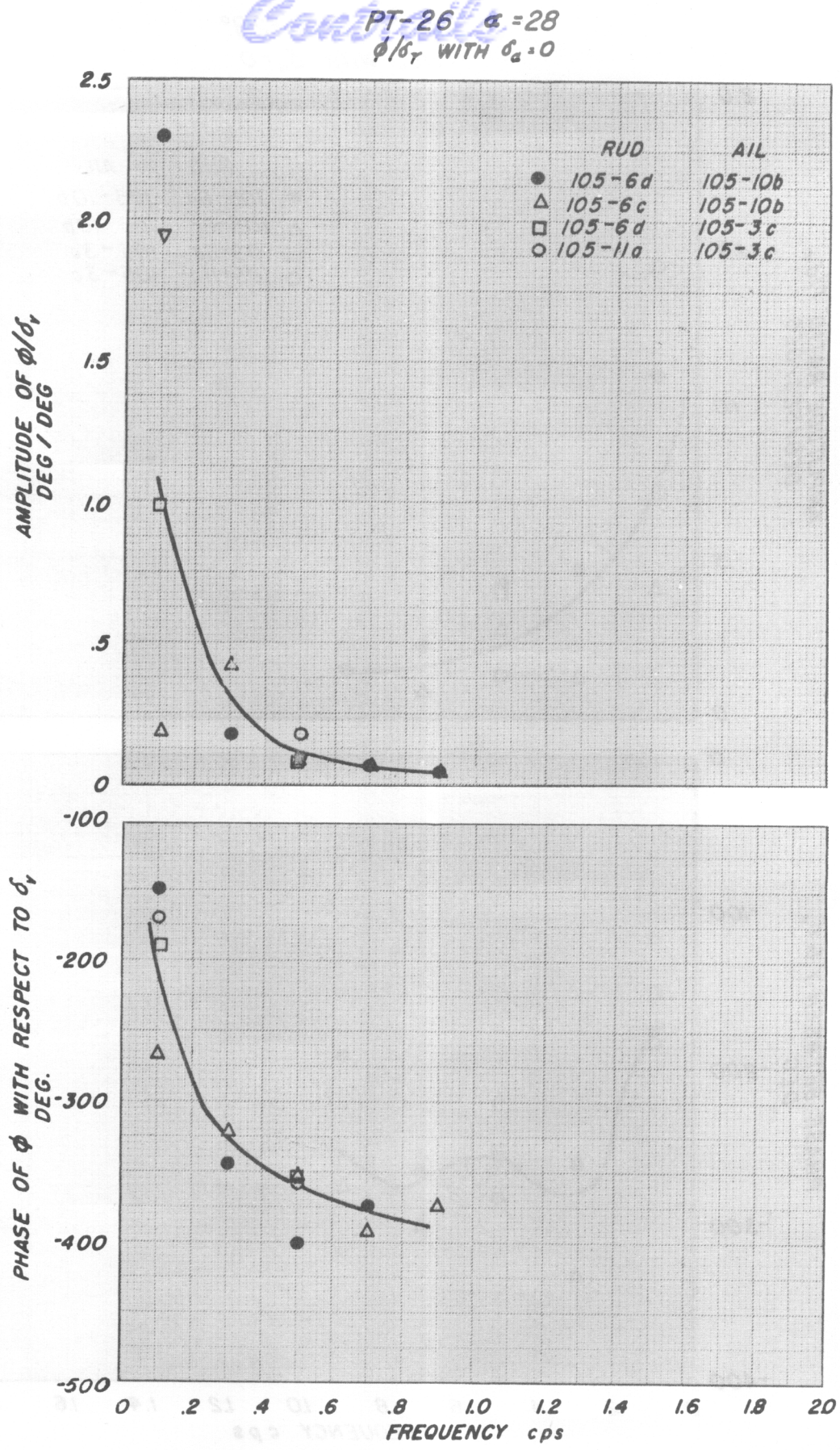


Fig. 23

Contrails

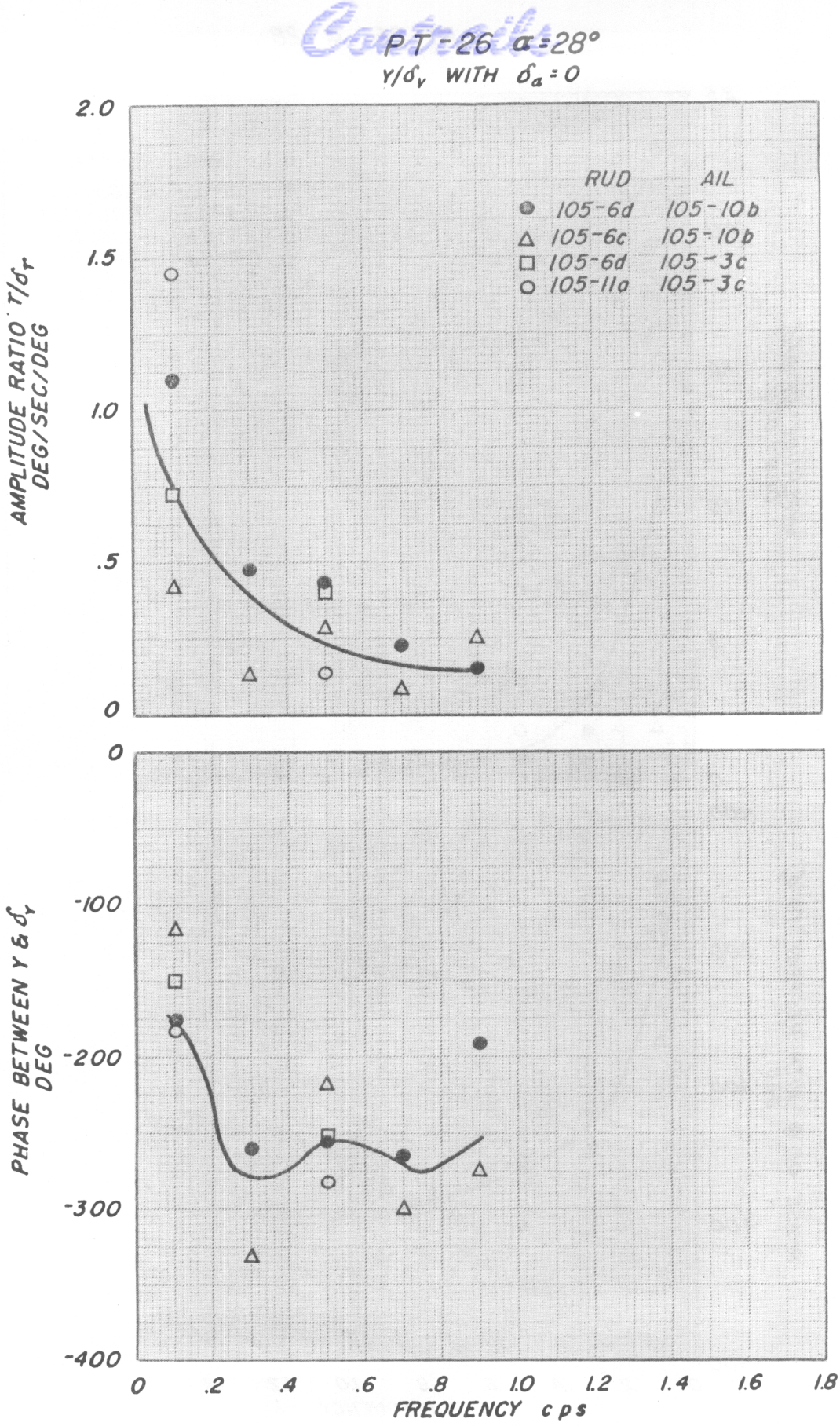


Fig. 24

Contrails

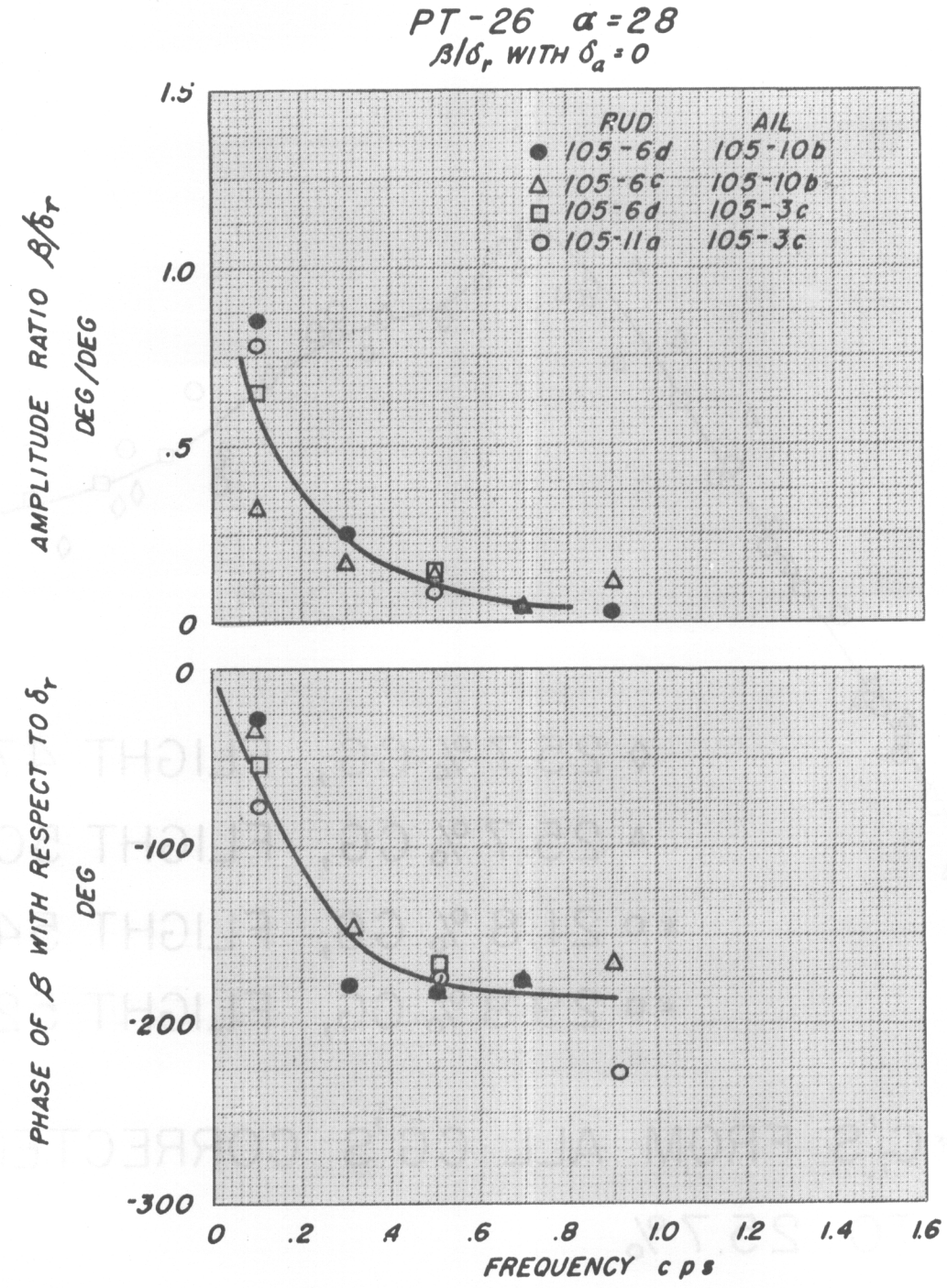
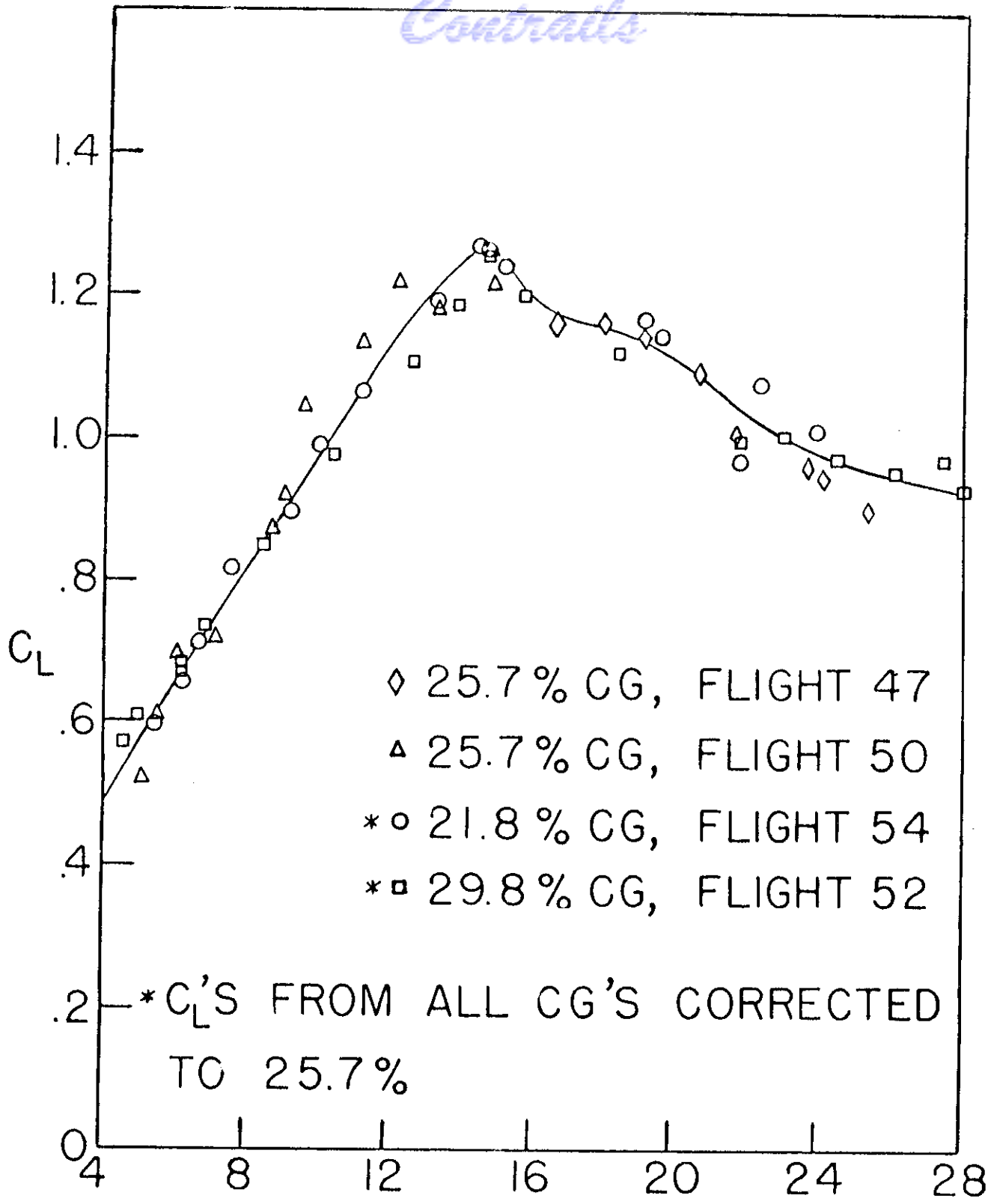


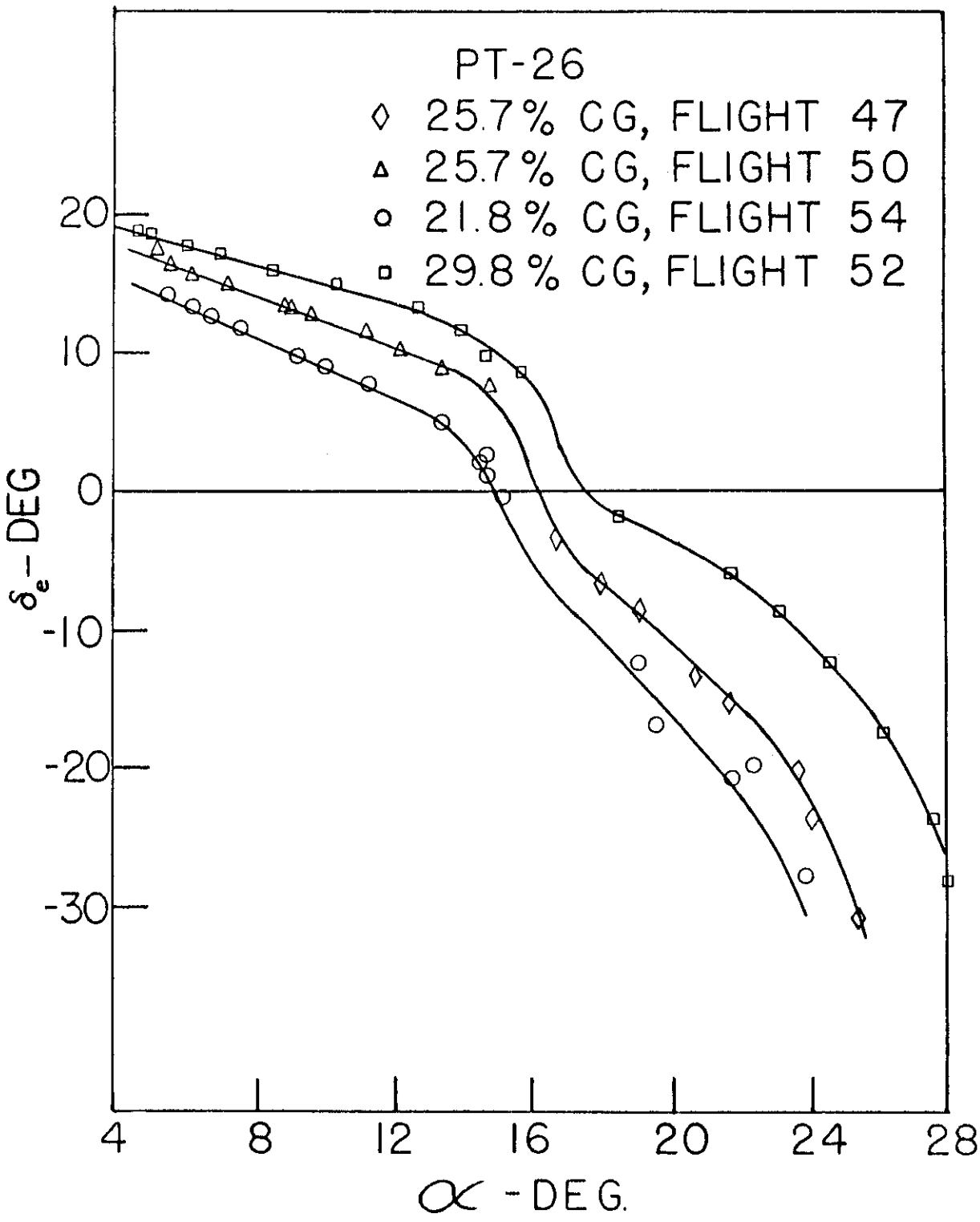
Fig. 25



LIFT CURVE DETERMINED FROM FLIGHT

Fig. 26

Contrails



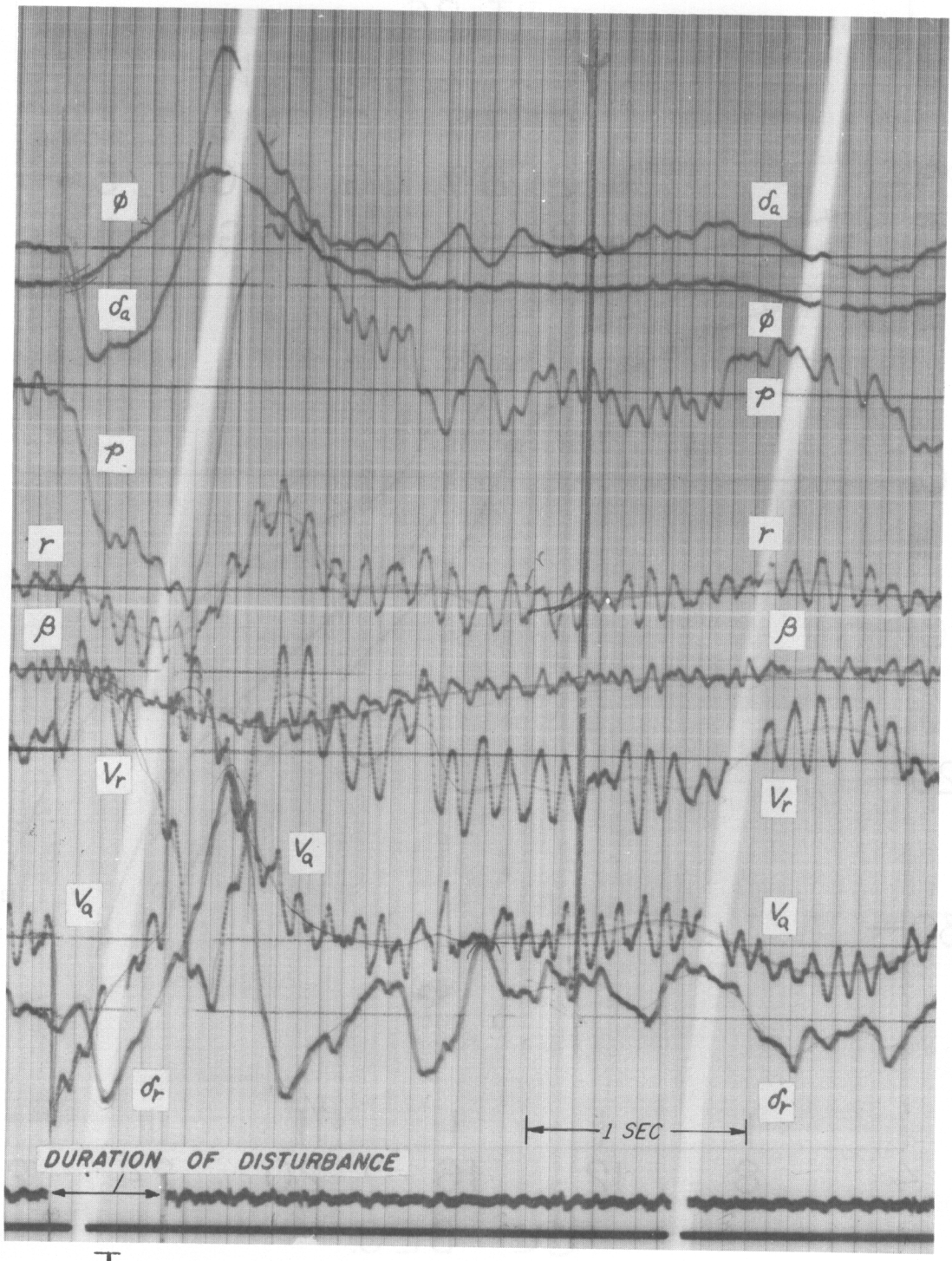
ELEVATOR TO TRIM DETERMINED FROM FLIGHT

FIG. 27

PT – 26 RUN 106 – 4g

$\alpha = 28°$

AILERON DISTURBANCE .23



THIS LENGTH EQUALS :

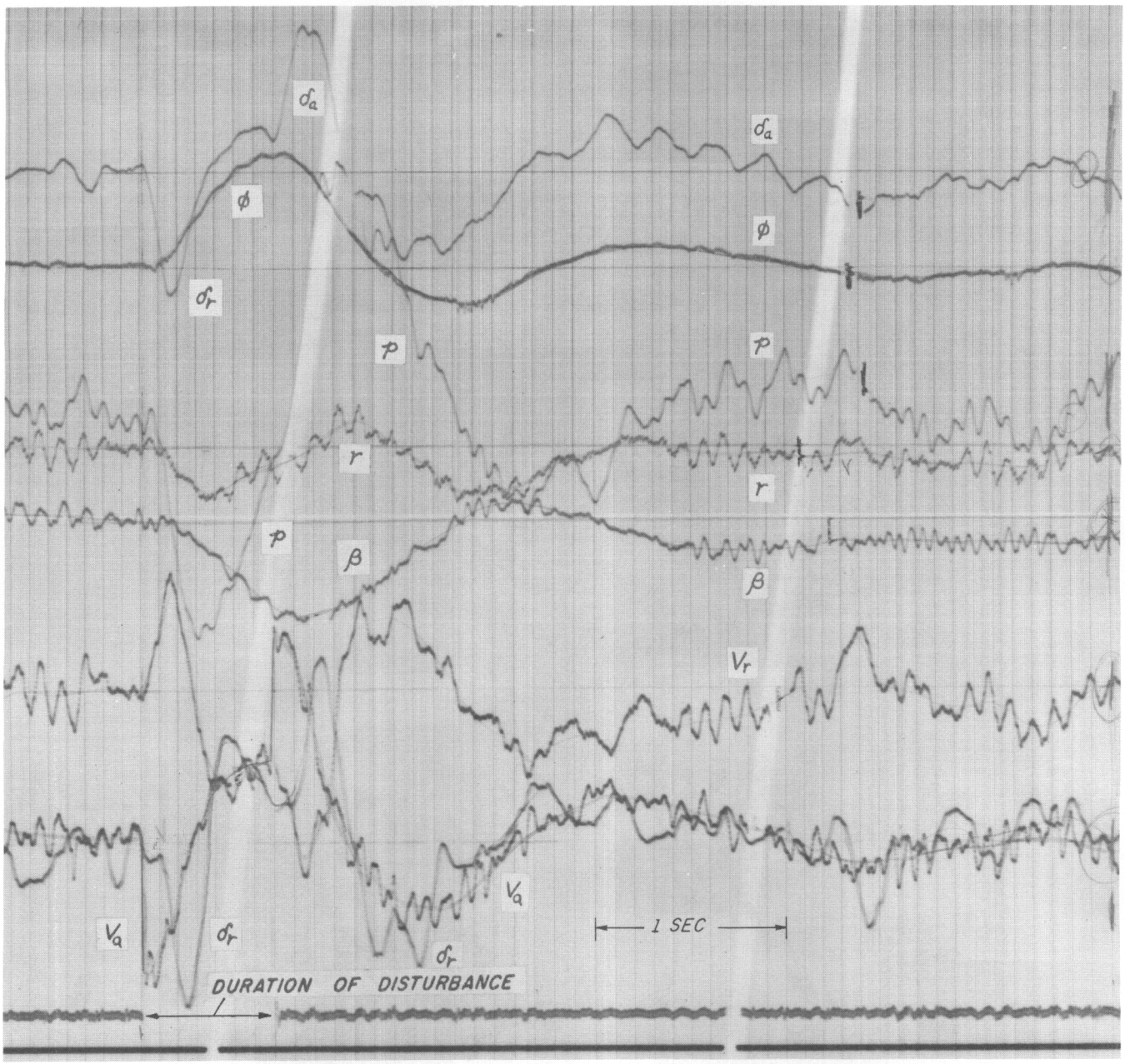
p + 11.2°/SEC	r + 8.75°/ SEC
ϕ – 9.2°	δ_a – 50.5°
β + 4.62°	δ_r – 15.3°

Figure 28

PT – 26 RUN 105 – 3b

$\alpha = 28°$

AILERON DISTURBANCE .30



THIS LENGTH EQUALS :

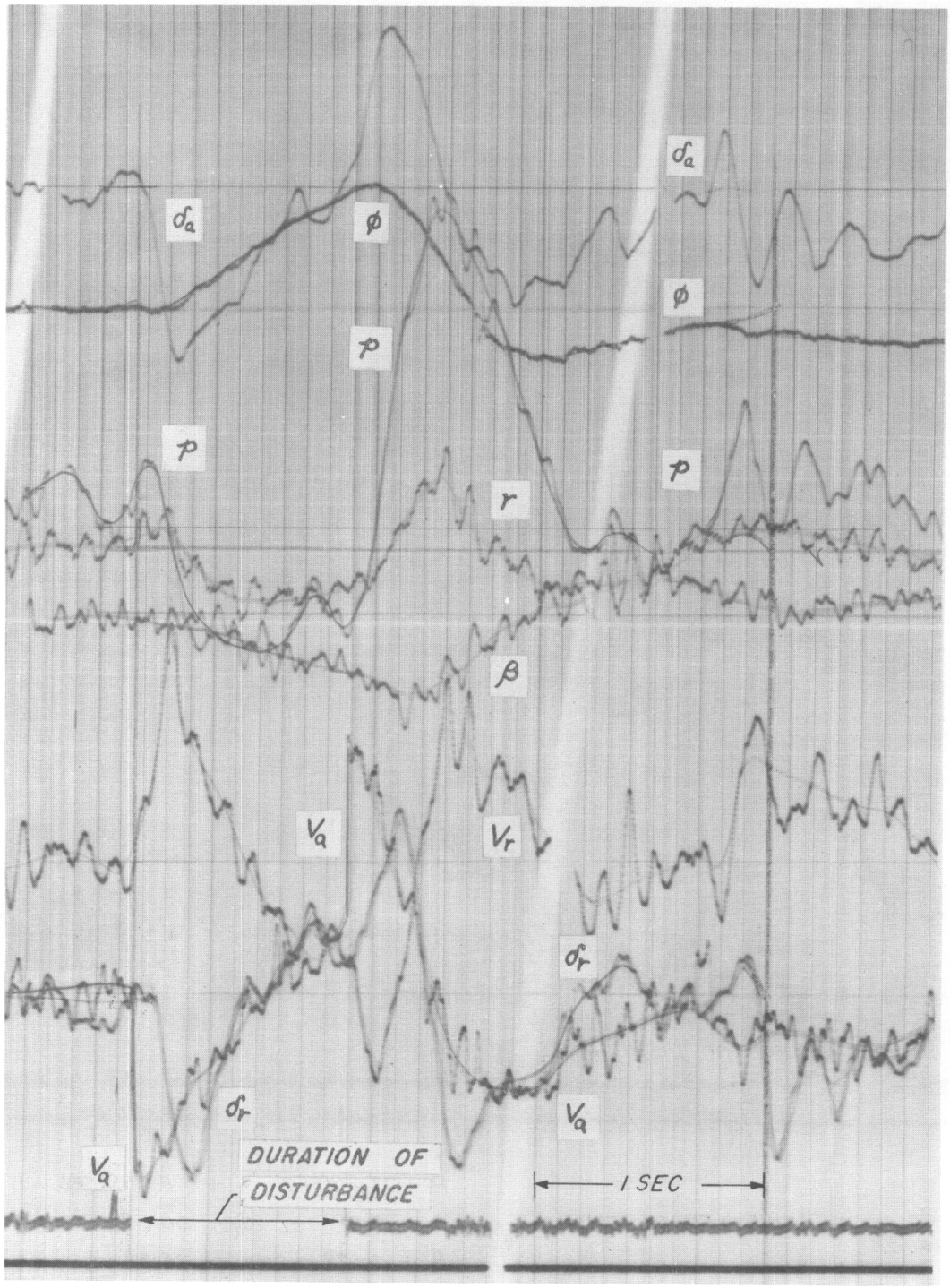
p + 11.2°/SEC	r + 8.75°/SEC
ϕ – 9.2°	δ_a – 50.5°
β + 4.62°	δ_r – 15.3°

Figure 29

PT – 26 RUN 105 – 3c

$\alpha = 28°$

AILERON DISTURBANCE .30



THIS LENGTH EQUALS :

p + 11.2°/SEC	r + 8.75°/SEC
ϕ – 9.2°	δ_a – 50.5°.
β + 4.62°	δ_r – 15.3°

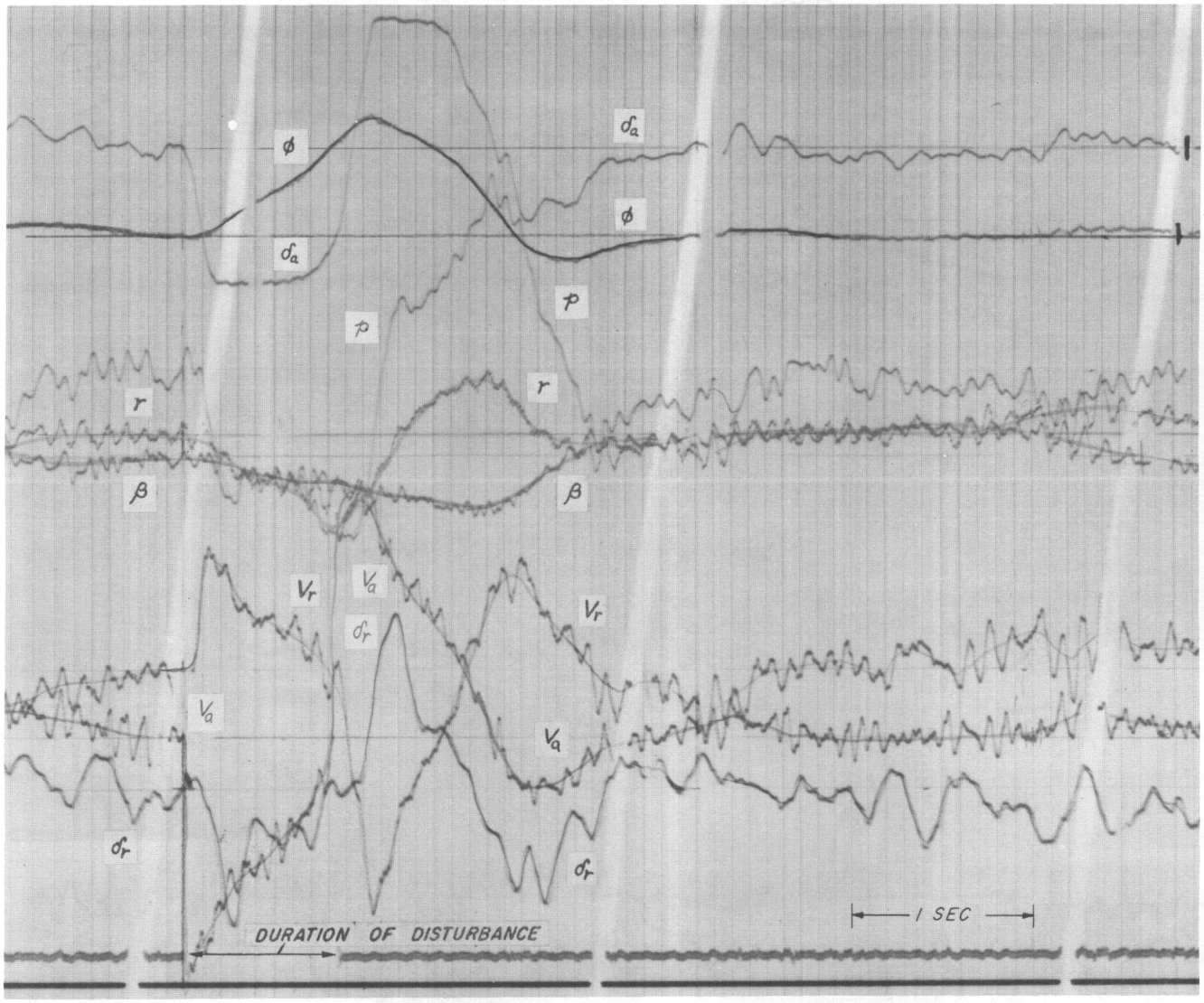
Figure 30

Controlled

PT – 26 RUN 105 – 10b

α = 28°

AILERON DISTURBANCE .30



THIS LENGTH EQUALS :

p + 11.2°/SEC	r + 8.75°/SEC
ϕ – 9.2°	δ_a – 50.5°
β + 4.62°	δ_r – 15.3°

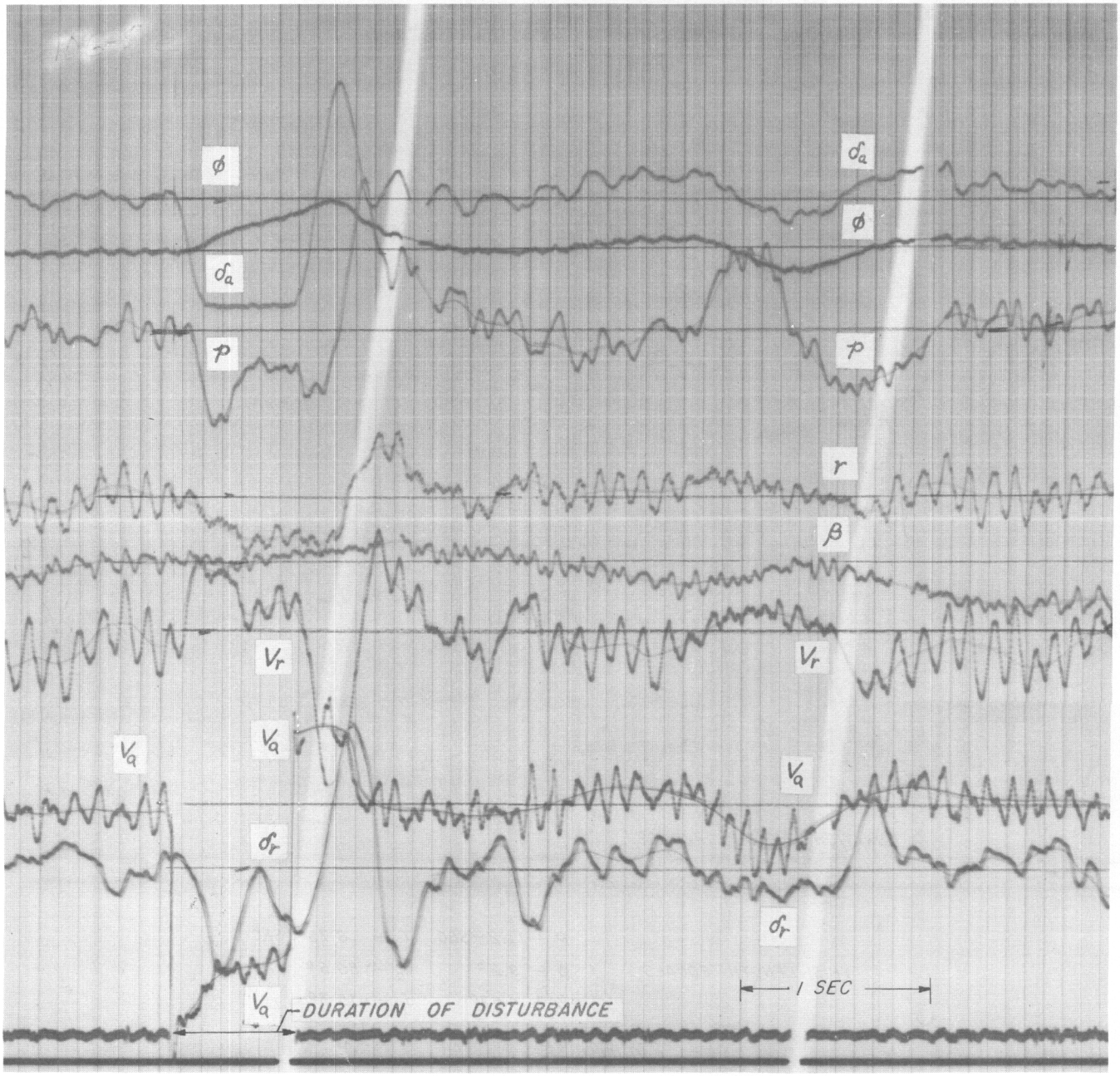
Figure 31

Contrails

PT – 26 RUN 106 – 5e

α = 28°

AILERON DISTURBANCE .30



THIS LENGTH EQUALS :

p + 11.2°/SEC	r + 8.75°/SEC
φ – 9.2°	δa – 50.5°
β + 4.62°	δr – 15.3°

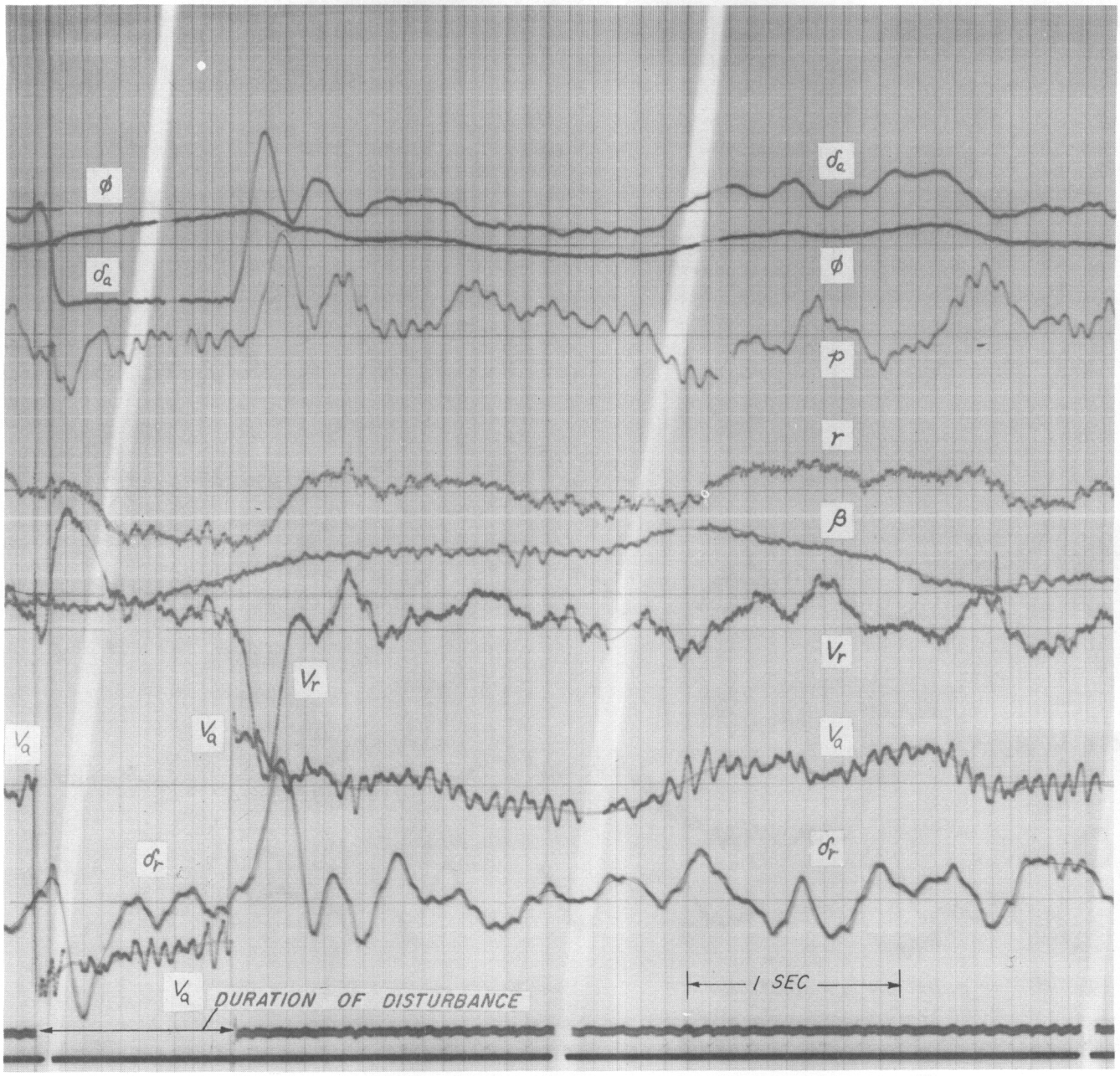
Figure 32

Contrails

PT – 26 RUN 106 – 6h

α = 28°

AILERON DISTURBANCE .30



THIS LENGTH EQUALS :

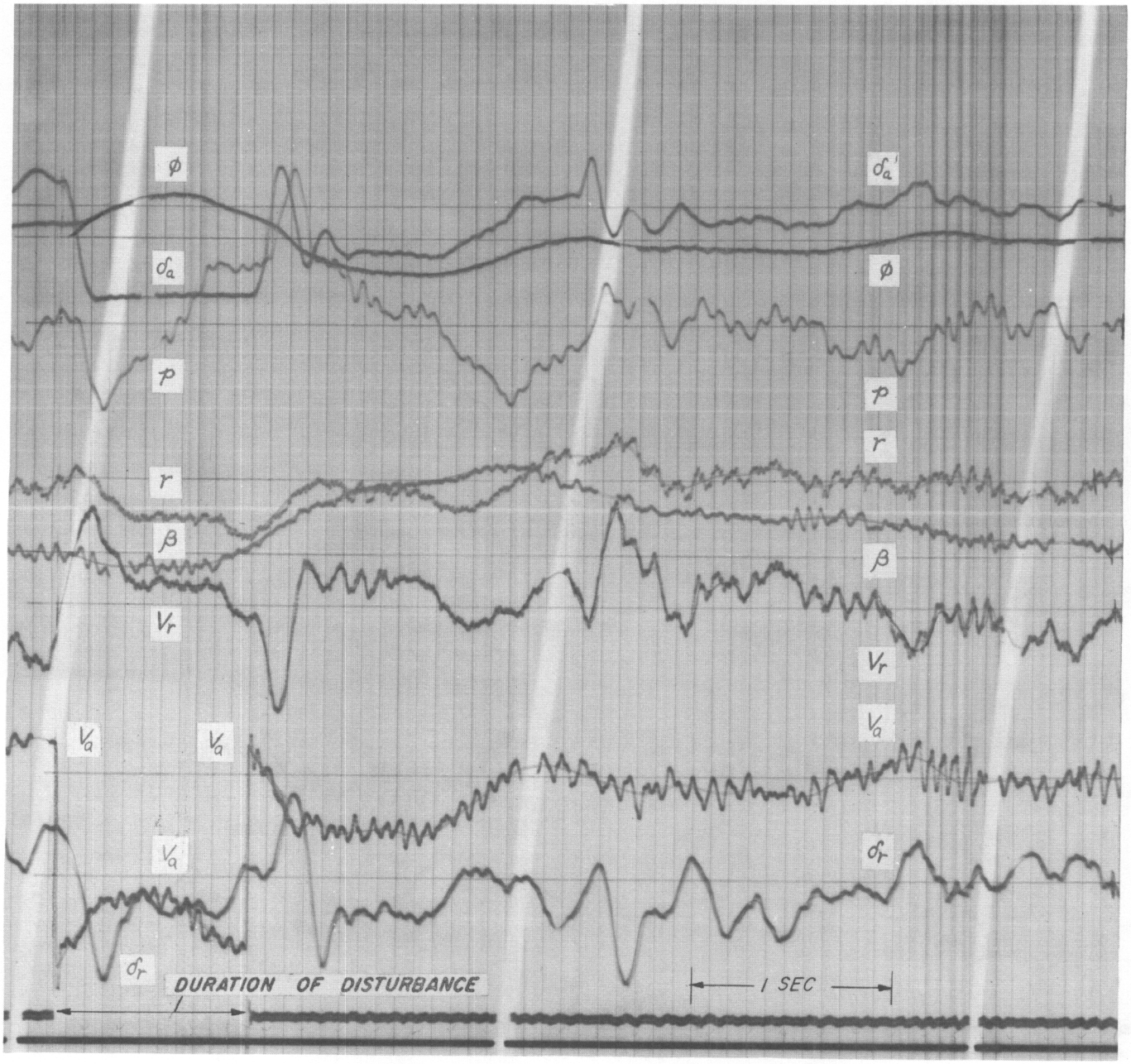
p + 11.2°/SEC	r + 8.75°/SEC
φ – 9.2°	δa – 50.5°
β + 4.62°	δr – 15.3°

Figure 33

PT – 26 RUN 106 – 6i

α = 28°

AILERON DISTURBANCE .30



THIS LENGTH EQUALS :

p + 11.2°/SEC	r + 8.75°/SEC
ϕ – 9.2°	δ_a – 50.5°
β + 4.62°	δ_r – 15.3°

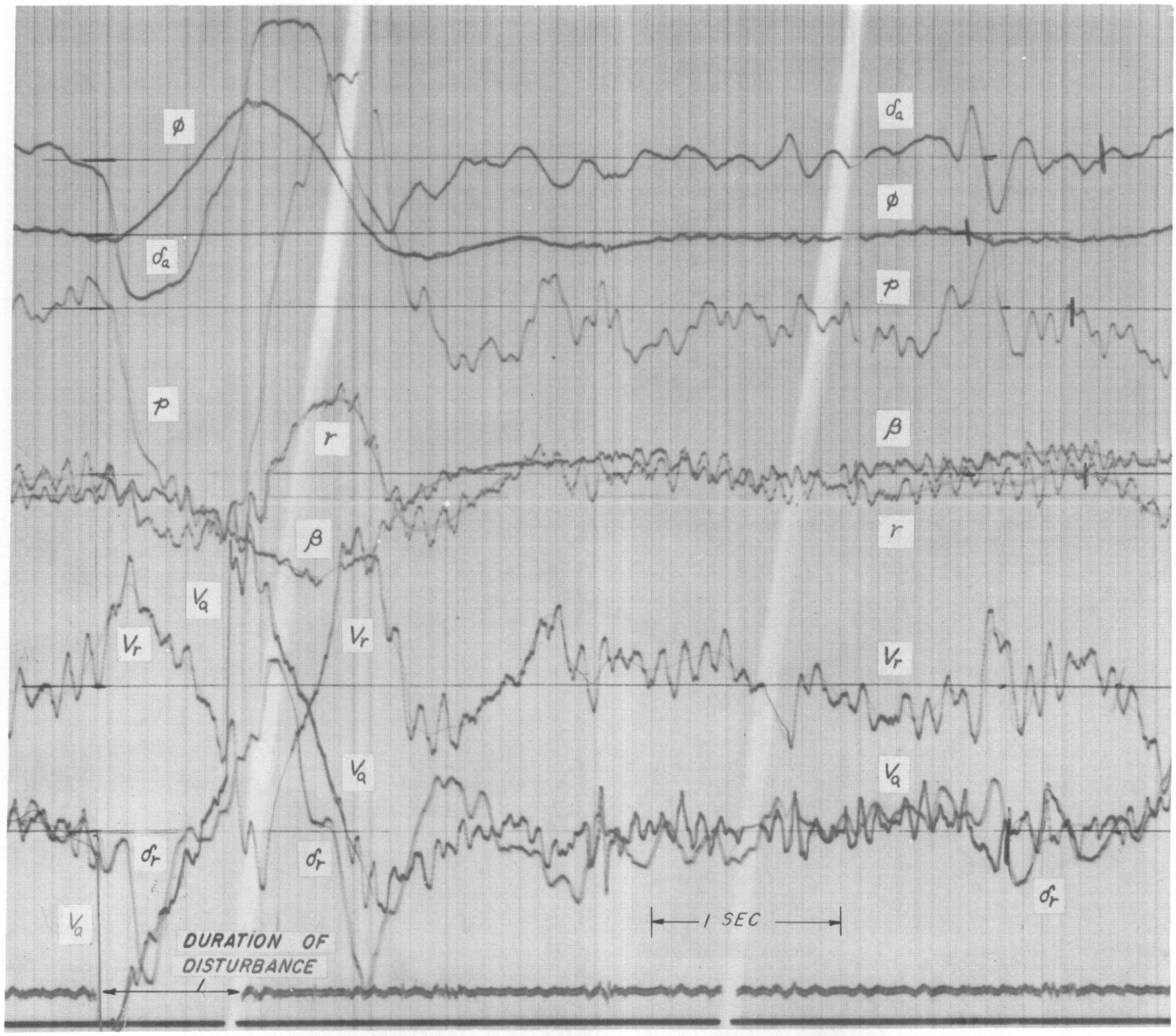
Figure 34

Controls

PT – 26 RUN 107 – 2b

α = 28°

AILERON DISTURBANCE .30



THIS LENGTH EQUALS :

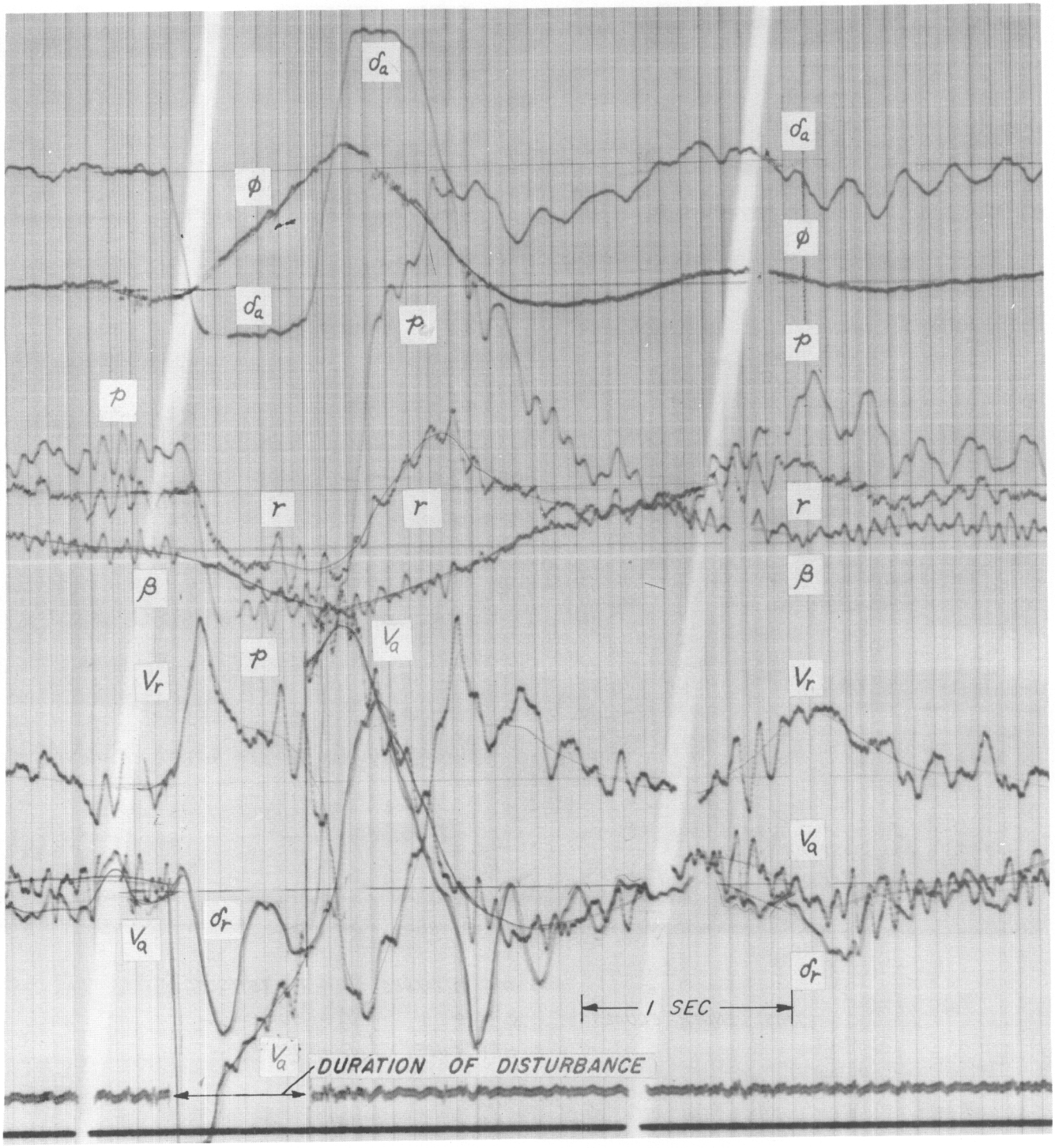
p + 11.2°/SEC	r + 8.75°/SEC
ϕ – 9.2°	δ_a – 50.5°
β + 4.62°	δ_r – 15.3°

Figure 35

PT – 26 RUN 105 – 3a

α = 28°

AILERON DISTURBANCE .40



THIS LENGTH EQUALS :

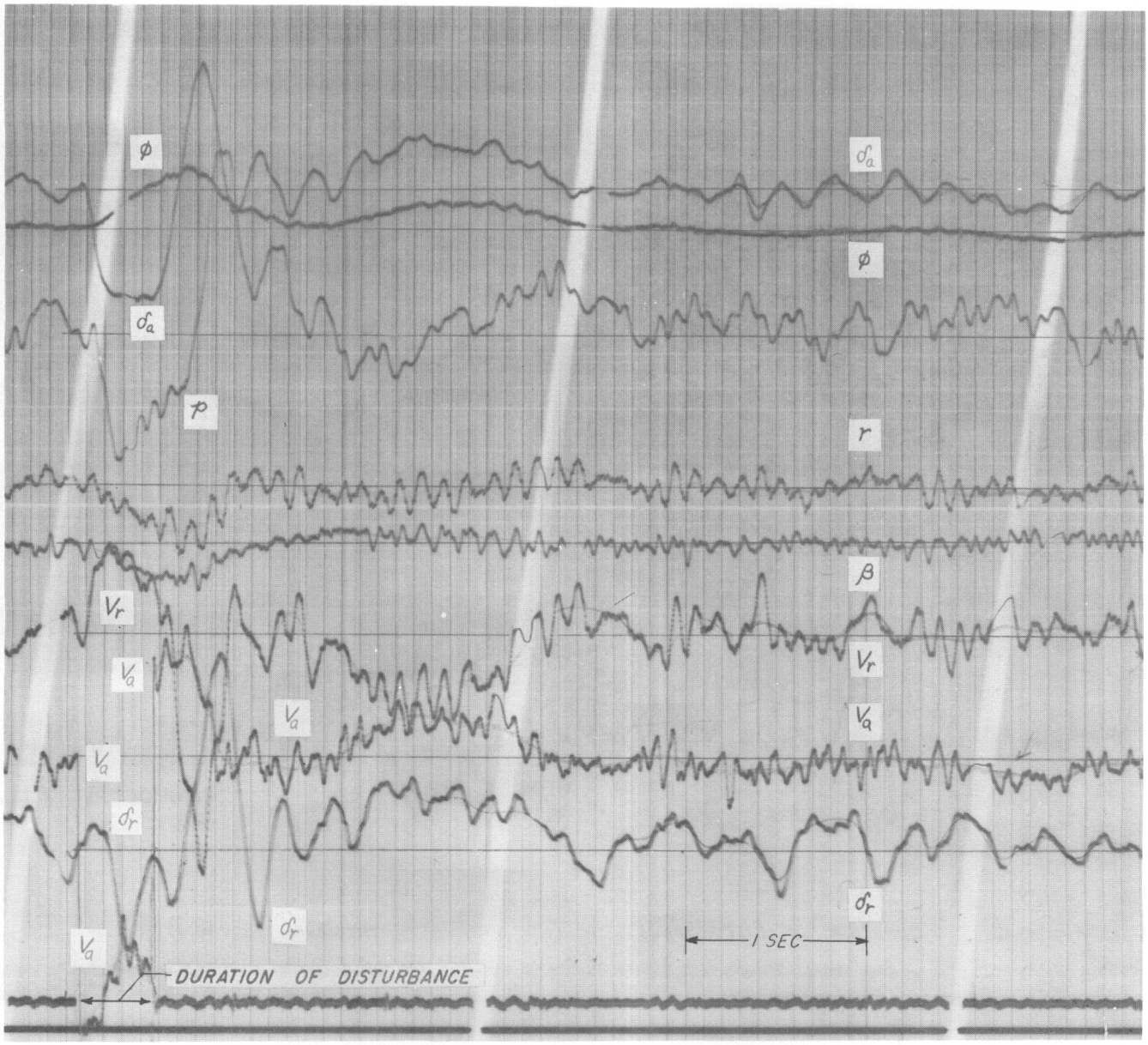
p + 11.2°/SEC	r + 8.75°/SEC
φ – 9.2°	δ_a – 50.5°
β + 4.62°	δ_r – 15.3°

Figure 36

PT-26 RUN 106-4b

α = 28°

AILERON DISTURBANCE .40



THIS LENGTH EQUALS :

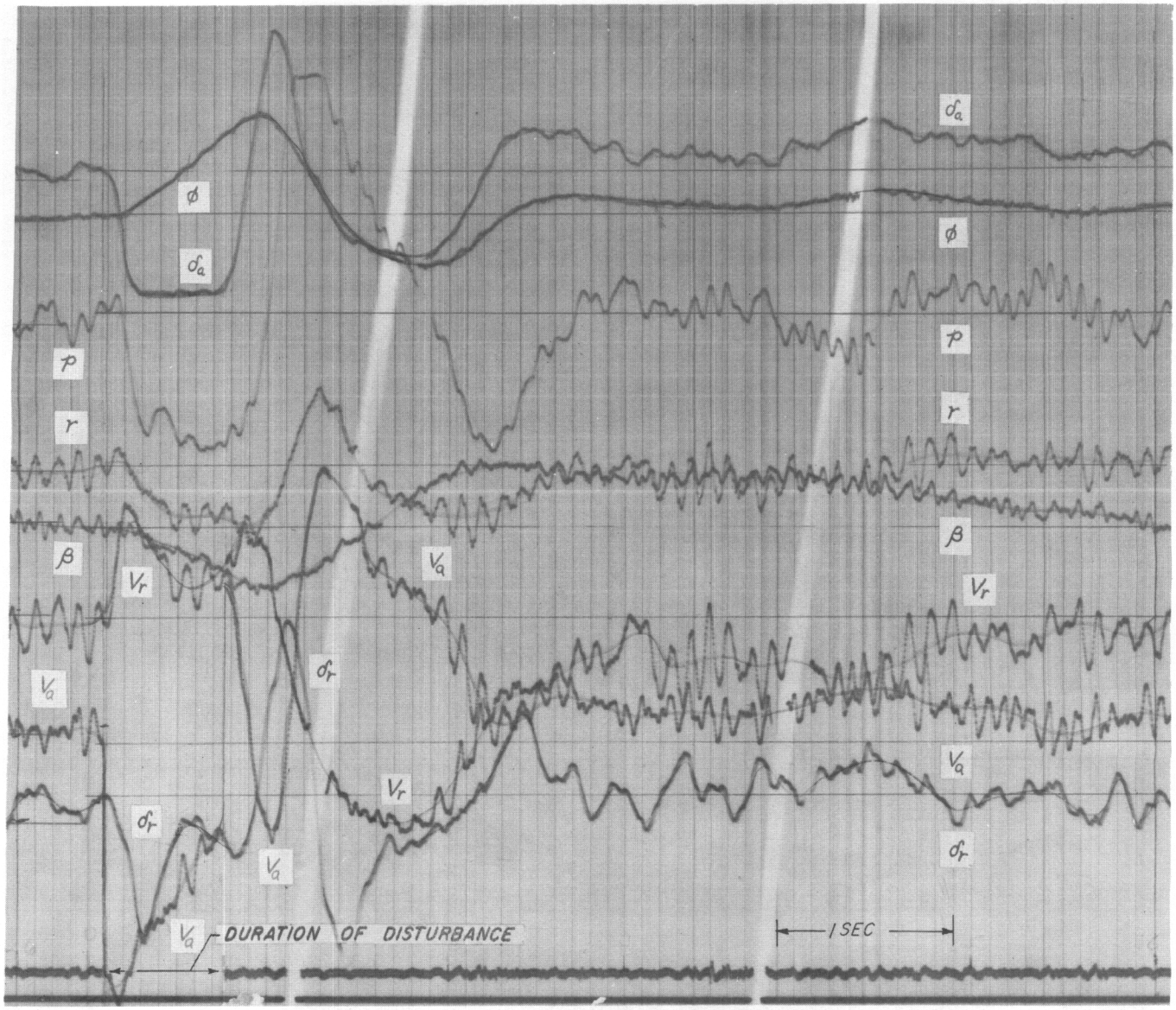
p + 11.2°/SEC	r + 8.75°/SEC
φ − 9.2°	δa − 50.5°
β + 4.62°	δr − 15.3°

Figure 37

PT – 26 RUN 106 – 4c

α = 28°

AILERON DISTURBANCE .40



THIS LENGTH EQUALS :

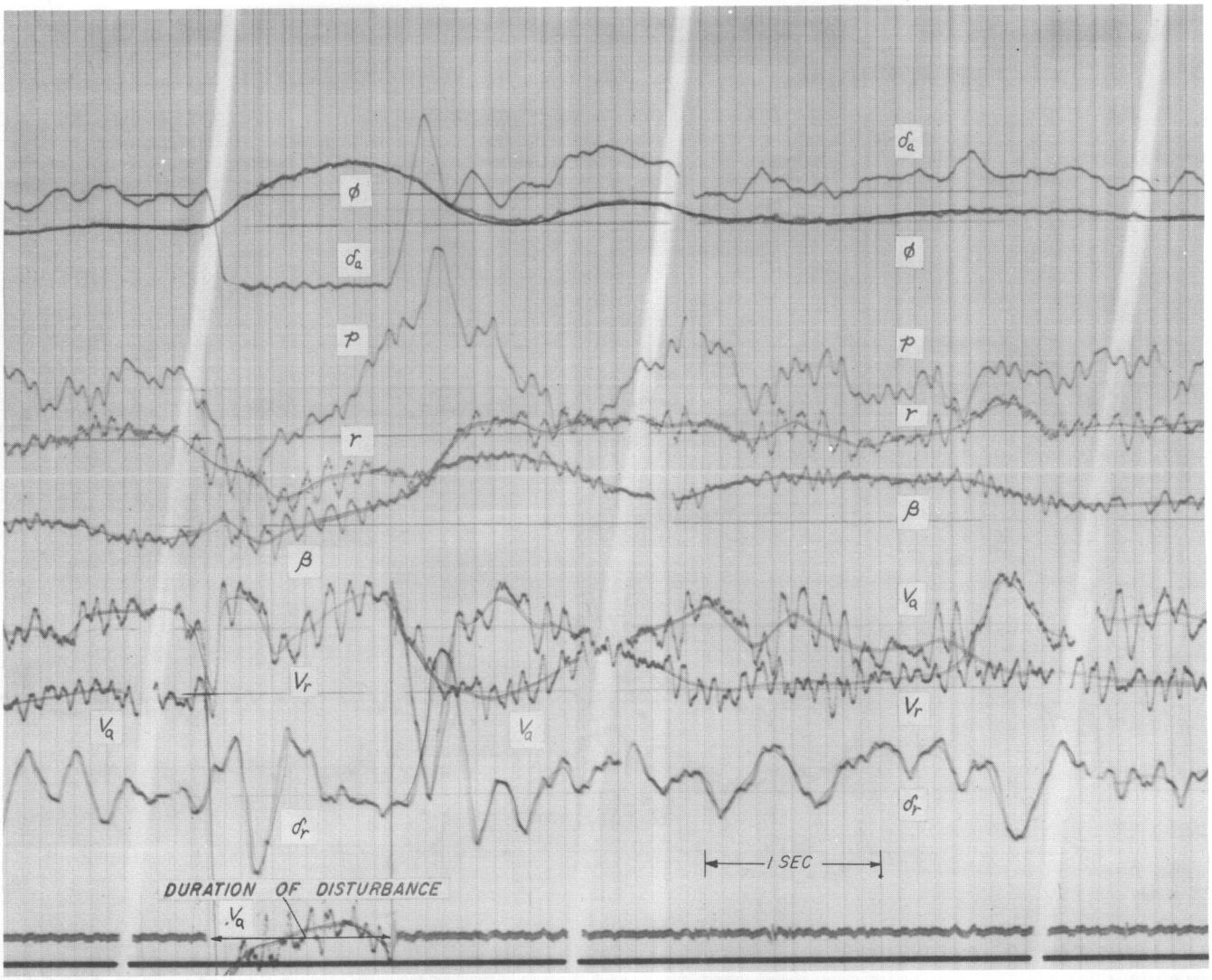
p + 11.2°/SEC	r + 8.75°/SEC
ϕ – 9.2°	δ_a – 50.5°
β + 4.62°	δ_r – 15.3°

Figure 38

PT – 26 RUN 105 – 9b

$\alpha = 28°$

AILERON DISTURBANCE .60



THIS LENGTH EQUALS :

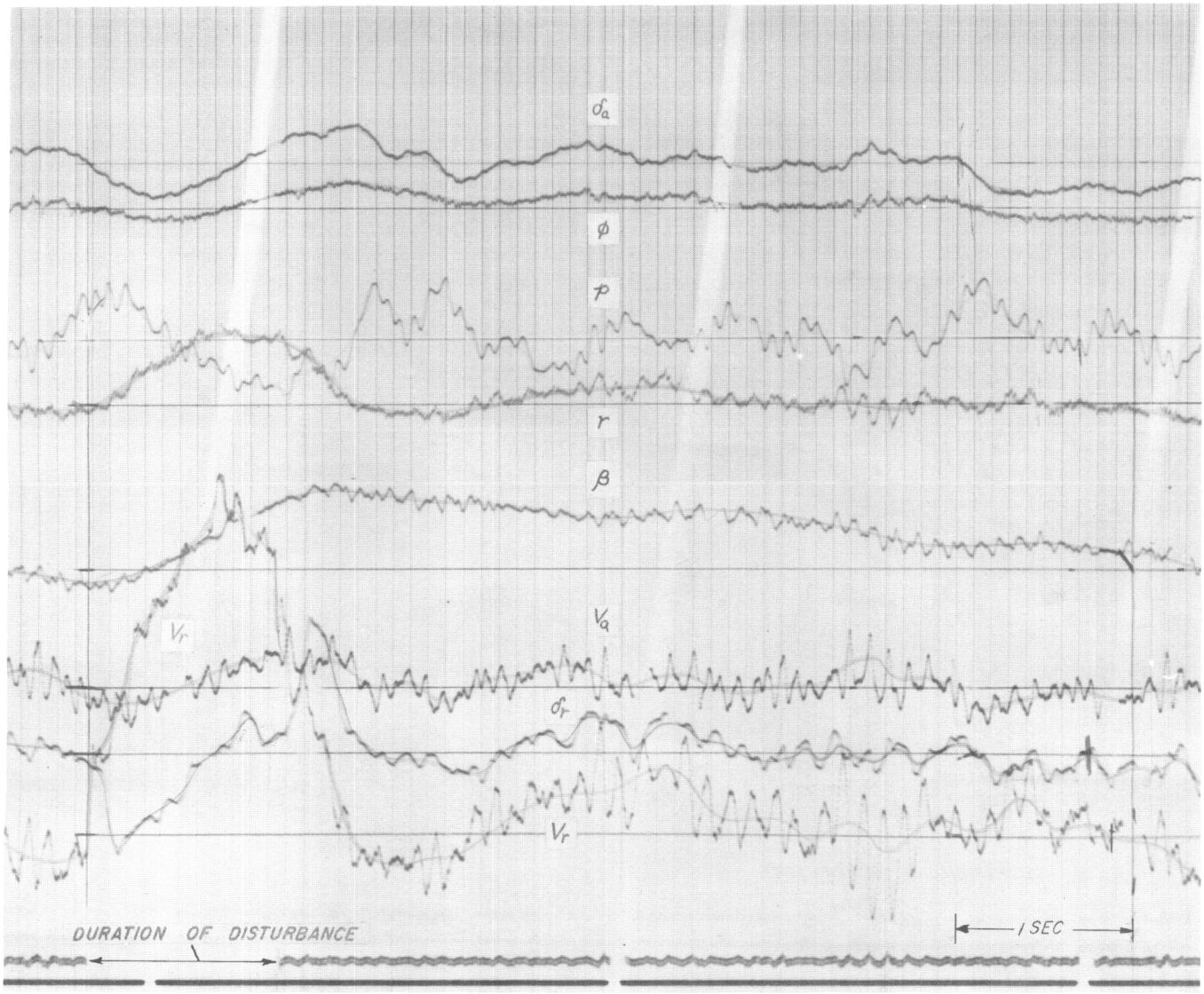
p + 11.2°/SEC	r + 8.75°/SEC
ϕ – 9.2°	δ_a – 50.5°
β + 4.62°	δ_r – 15.3°

Figure 39

PT-26 RUN 102-2b

$\alpha = 28°$

RUDDER DISTURBANCE .40



THIS LENGTH EQUALS :

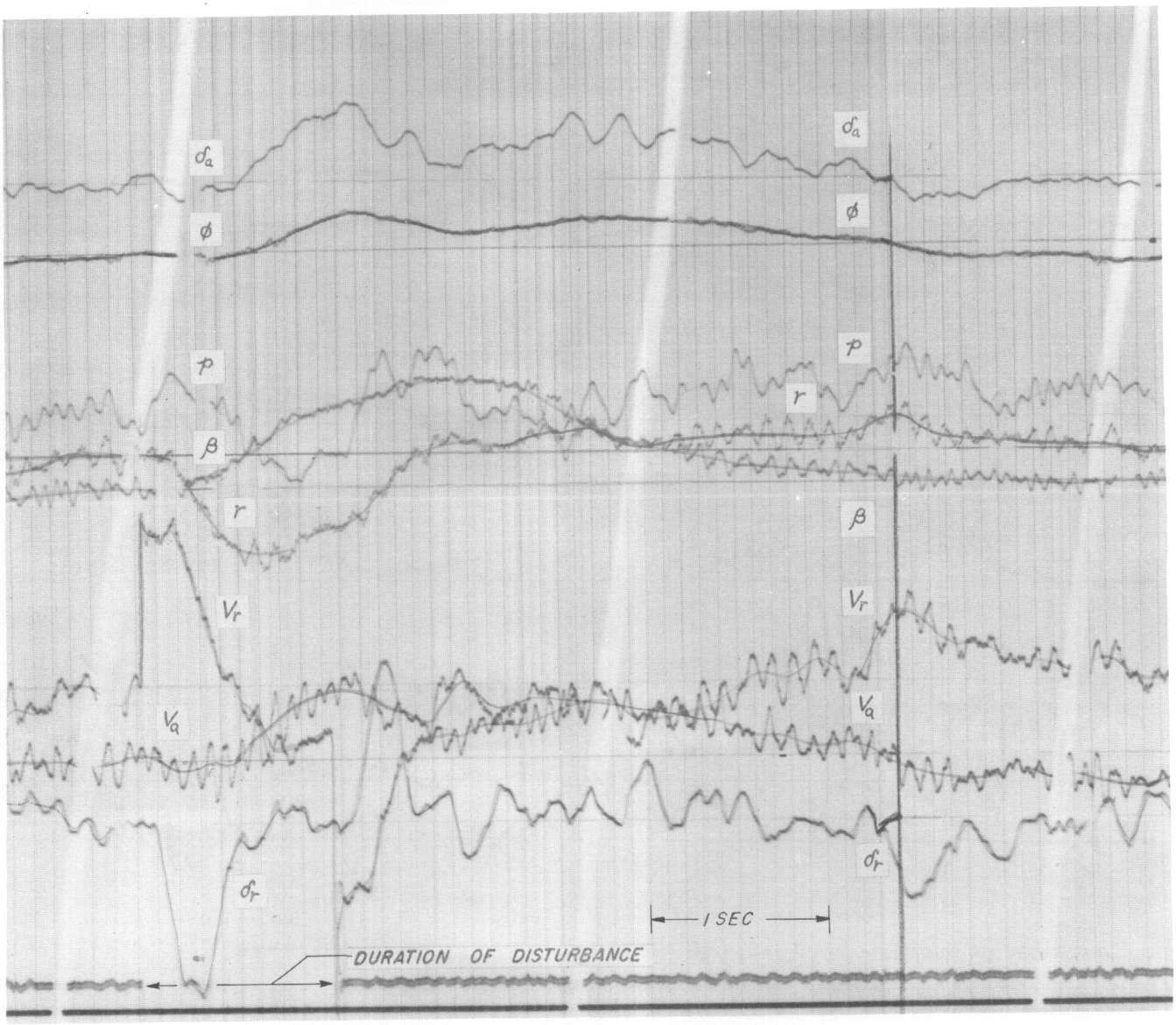
p + 9.24°/SEC	r — 9.92°/SEC
ϕ −10°	δ_a — 47.2°
β + 4.72°	δ_r — 14.6°

Figure 40

PT-26 RUN 105-11a

$\alpha = 28°$

RUDDER DISTURBANCE .50



THIS LENGTH EQUALS :

p + 11.2°/SEC	r + 8.75°/SEC
ϕ − 9.2°	δ_a − 50.5°
β + 4.62°	δ_r − 15.3°

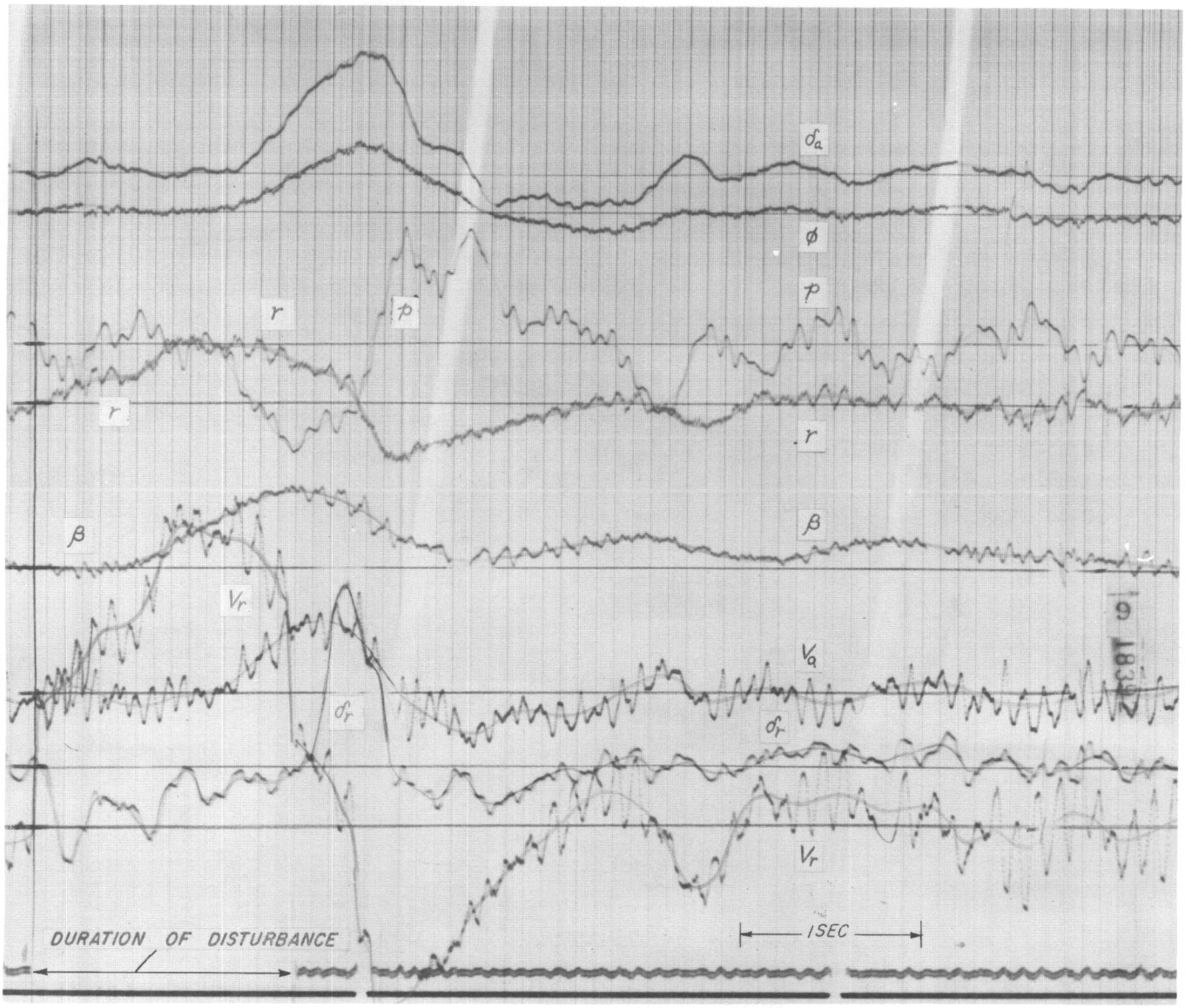
Figure 41

Contrails

PT-26 RUN 102-2e

$\alpha = 28°$

RUDDER DISTURBANCE .53



THIS LENGTH EQUALS :

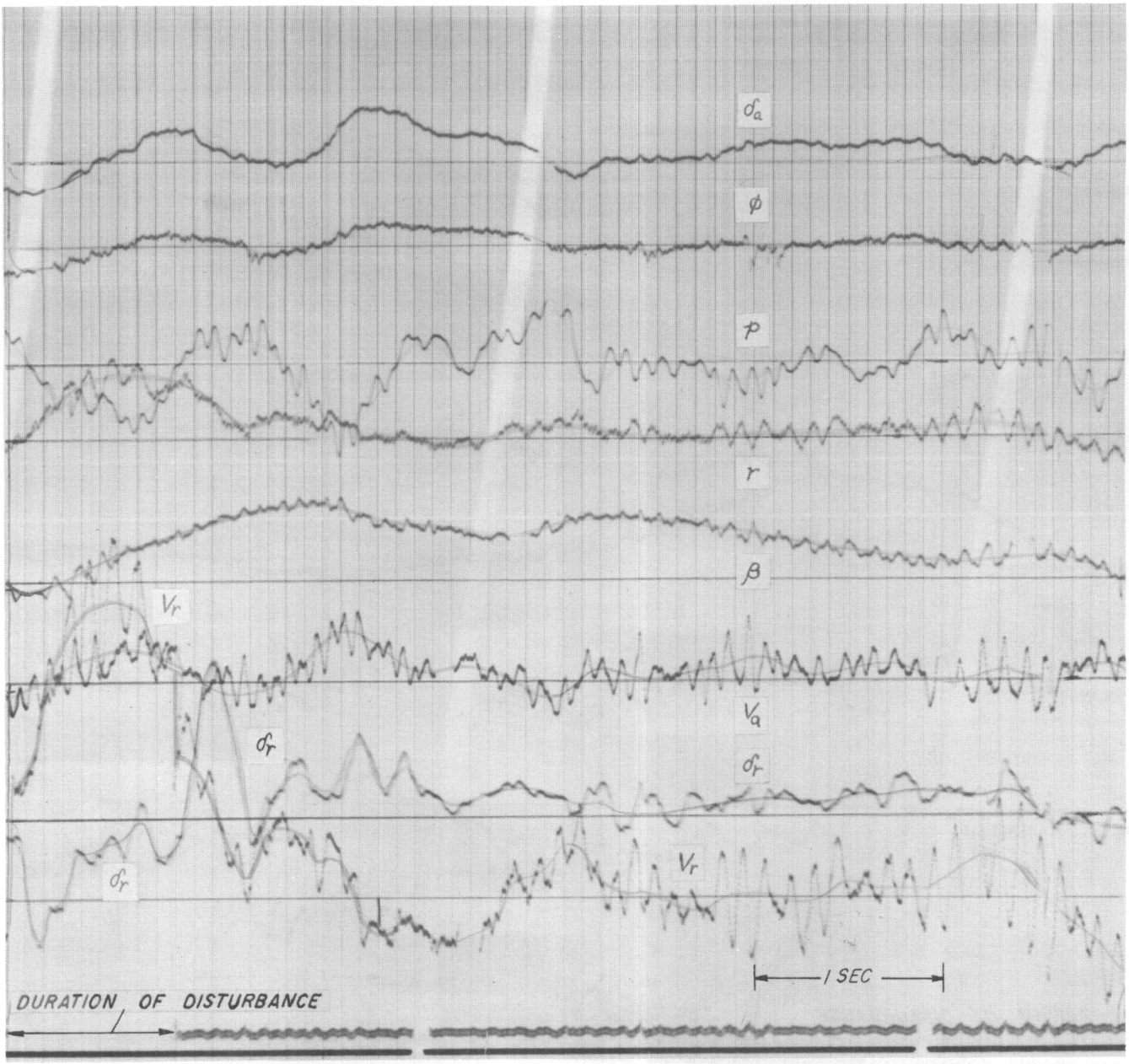
p + 9.24°/SEC	r — 9.92°/SEC
ϕ −10°	δ_a — 47.2°
β + 4.72°	δ_r — 14.6°

Figure 42

PT – 26 RUN 102 – 4a

α = 28°

RUDDER DISTURBANCE .60



THIS LENGTH EQUALS :

p + 9.24°/SEC	r — 9.92°/SEC
ϕ — 10°	δ_a — 47.2°
β + 4.72°	δ_r — 14.6°

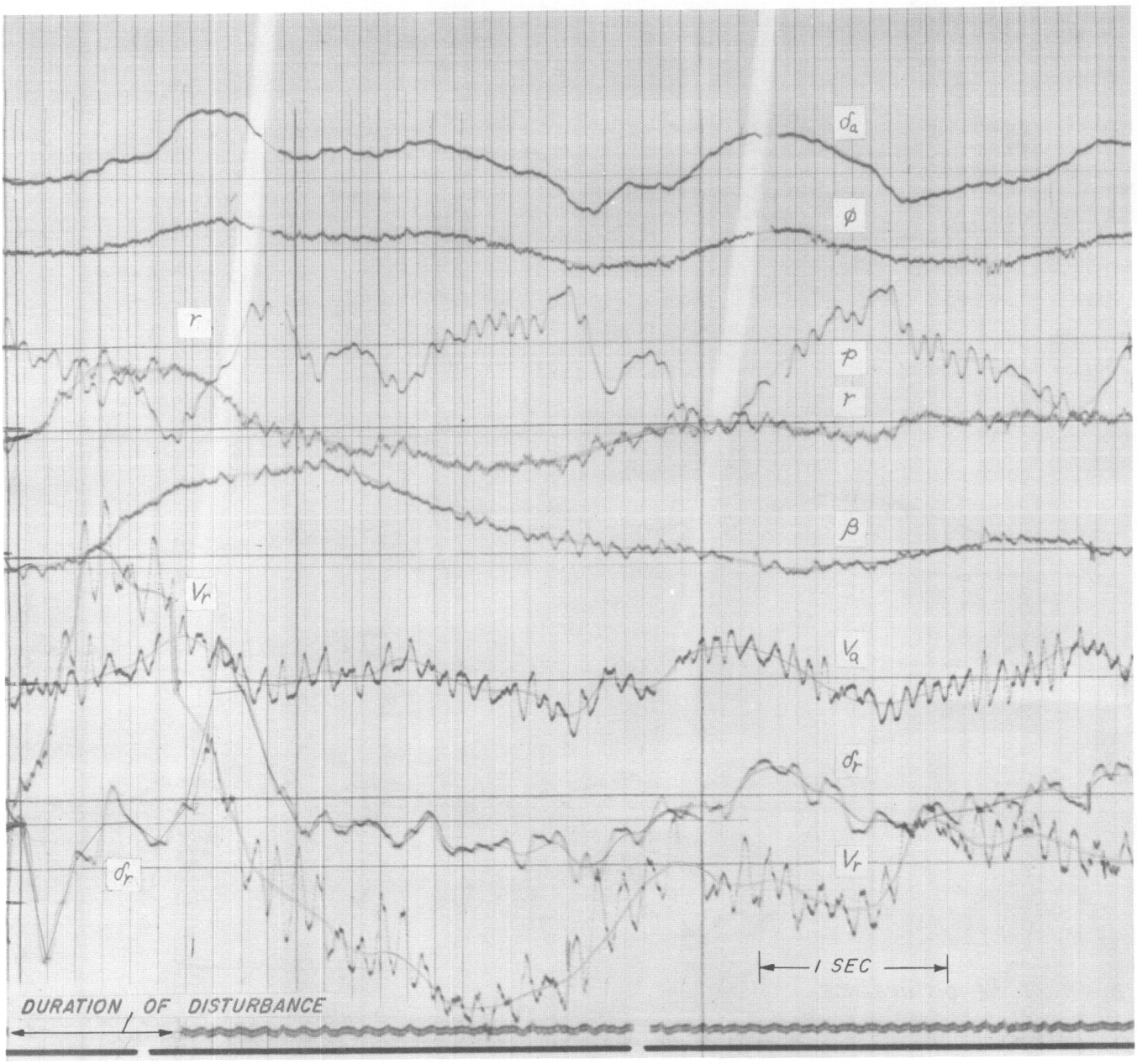
Figure 43

Contrails

PT-26 RUN 102-4b

$\alpha = 28°$

RUDDER DISTURBANCE .60



THIS LENGTH EQUALS :

p + 9.24°/SEC	r — 9.92°/SEC
ϕ −10°	δ_a — 47.2°
β + 4.72°	δ_r — 14.6°

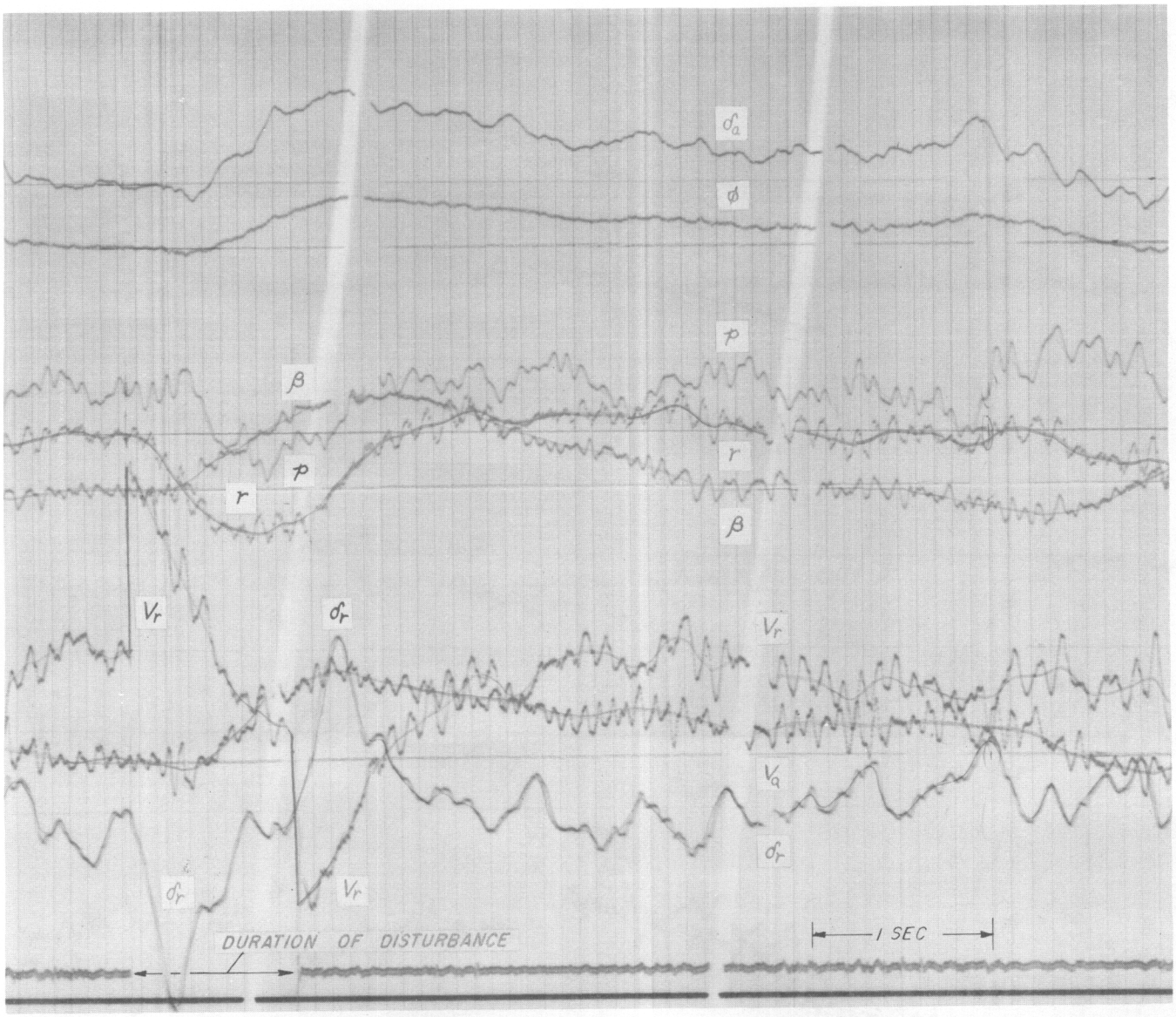
Figure 44

Contrails

PT-26 RUN 105-11b

$\alpha = 28°$

RUDDER DISTURBANCE .70



THIS LENGTH EQUALS :

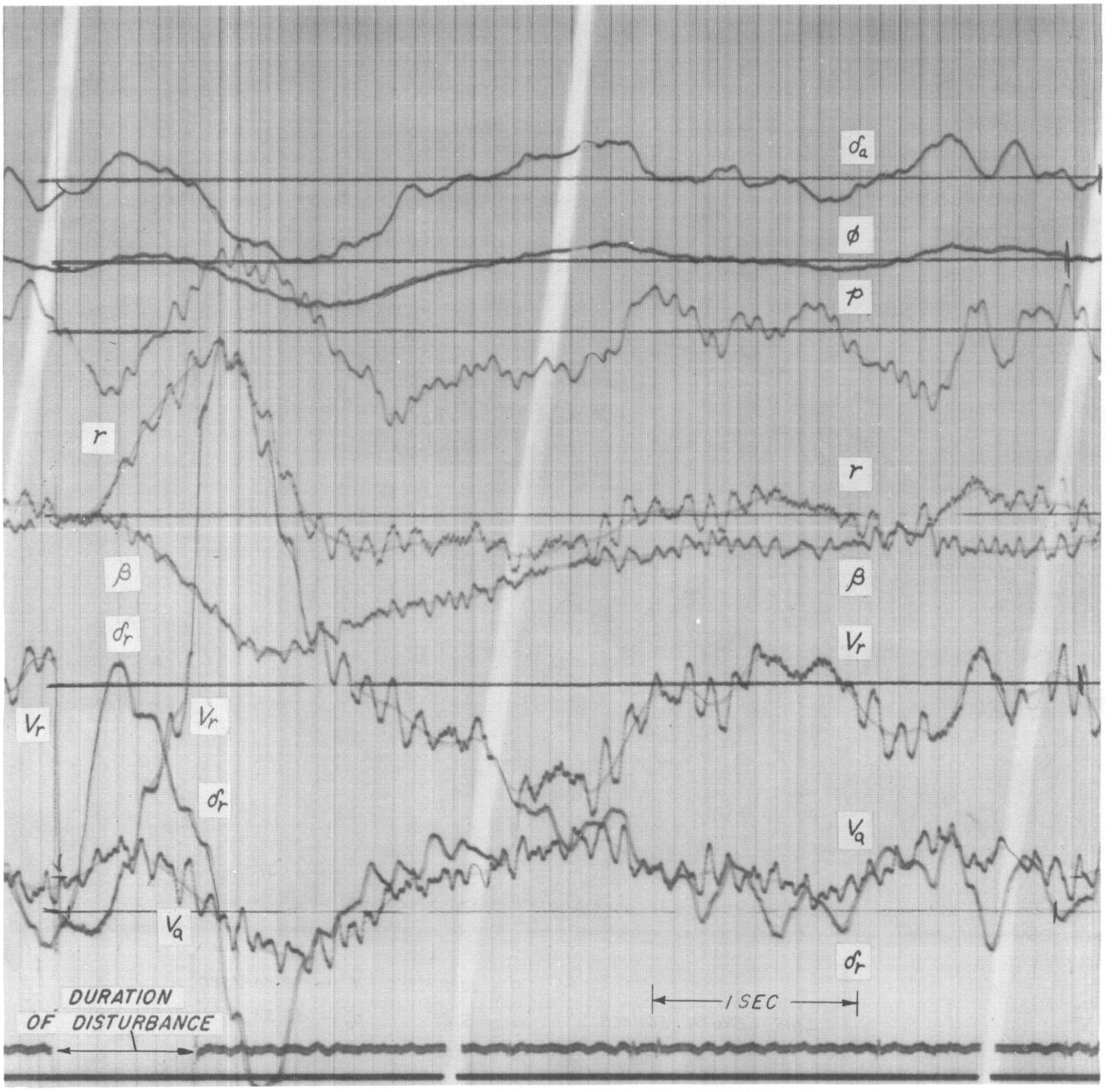
p + 11.2°/SEC	r + 8.75°/SEC
ϕ − 9.2°	δ_a − 50.5°
β + 4.62°	δ_r − 15.3°

Figure 45

PT – 26 RUN 107 – 3b

$\alpha = 28°$

RUDDER DISTURBANCE .70



THIS LENGTH EQUALS :

p + 11.2°/SEC	r + 8.75°/SEC
ϕ – 9.2°	δ_a – 50.5°
β + 4.62°	δ_r – 15.3°

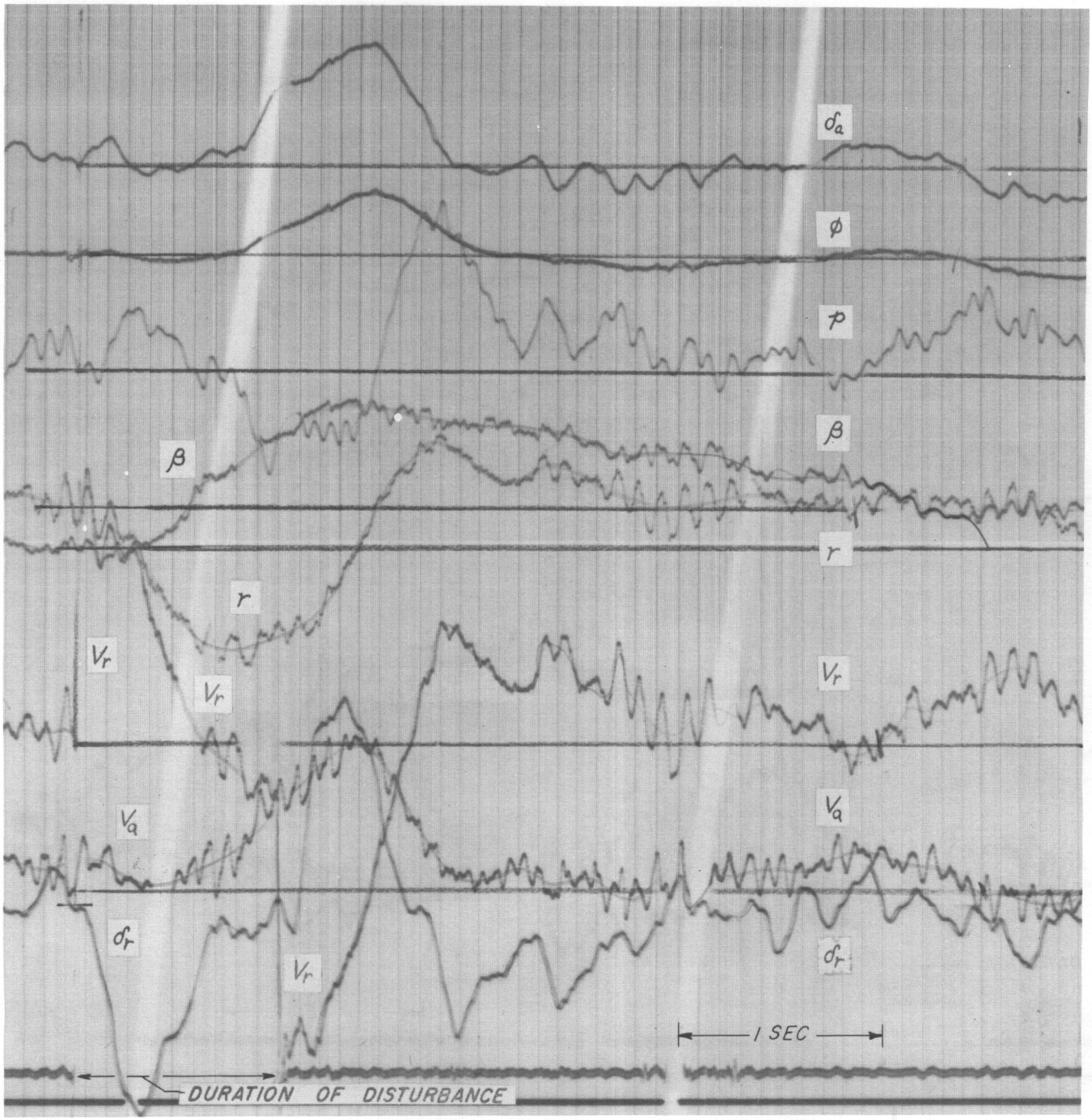
Figure 46

Contrails

PT – 26 RUN 107 – 3e

α = 28°

RUDDER DISTURBANCE .70



THIS LENGTH EQUALS :

p + 11.2°/SEC	r + 8.75°/SEC
φ – 9.2°	δ_a – 50.5°
β + 4.62°	δ_r – 15.3°

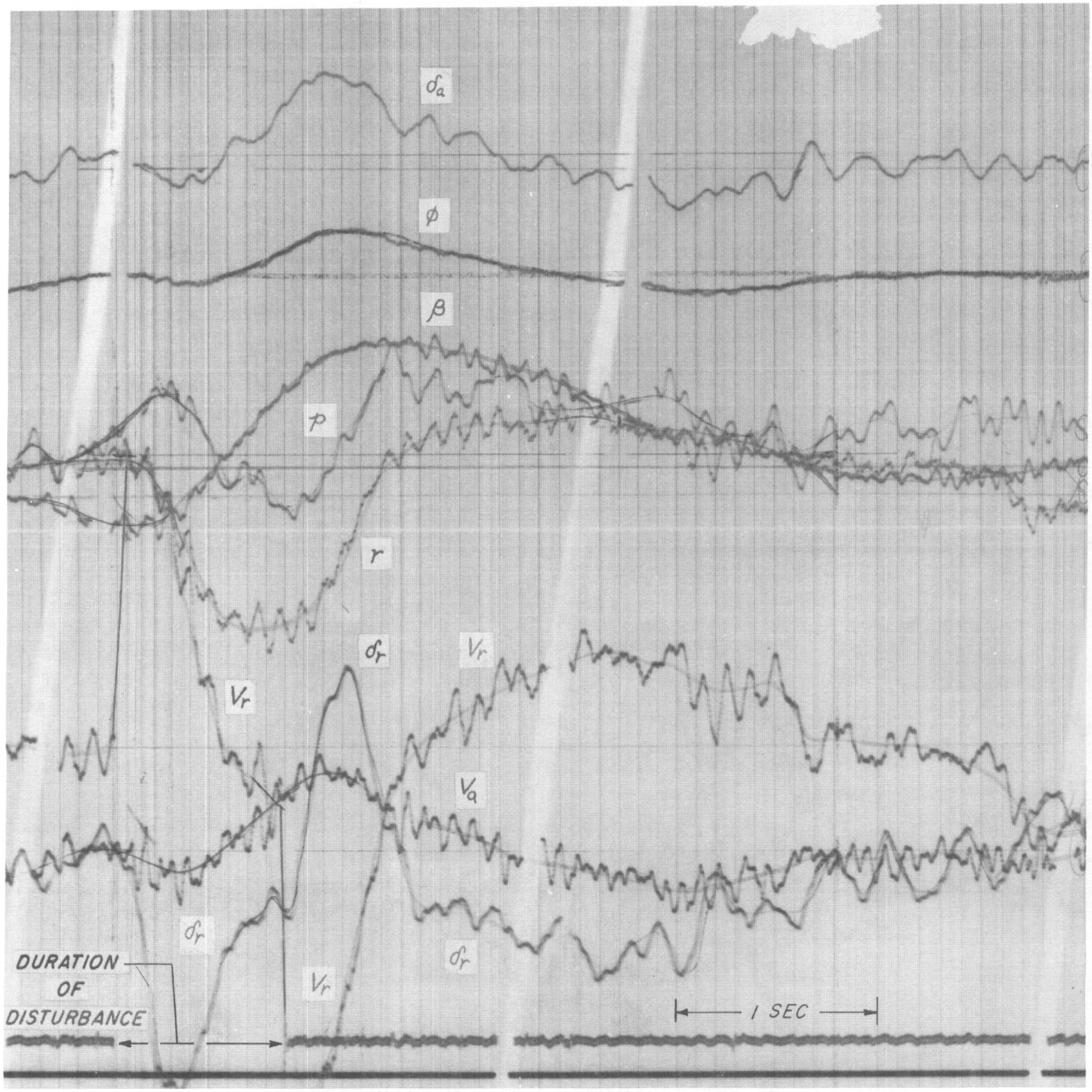
Figure 47

Contrails

PT-26 RUN 105-6c

$\alpha = 28°$

RUDDER DISTURBANCE .80



THIS LENGTH EQUALS :

p + 11.2°/SEC	r + 8.75°/SEC
ϕ − 9.2°	δ_a − 50.5°
β + 4.62°	δ_r − 15.3°

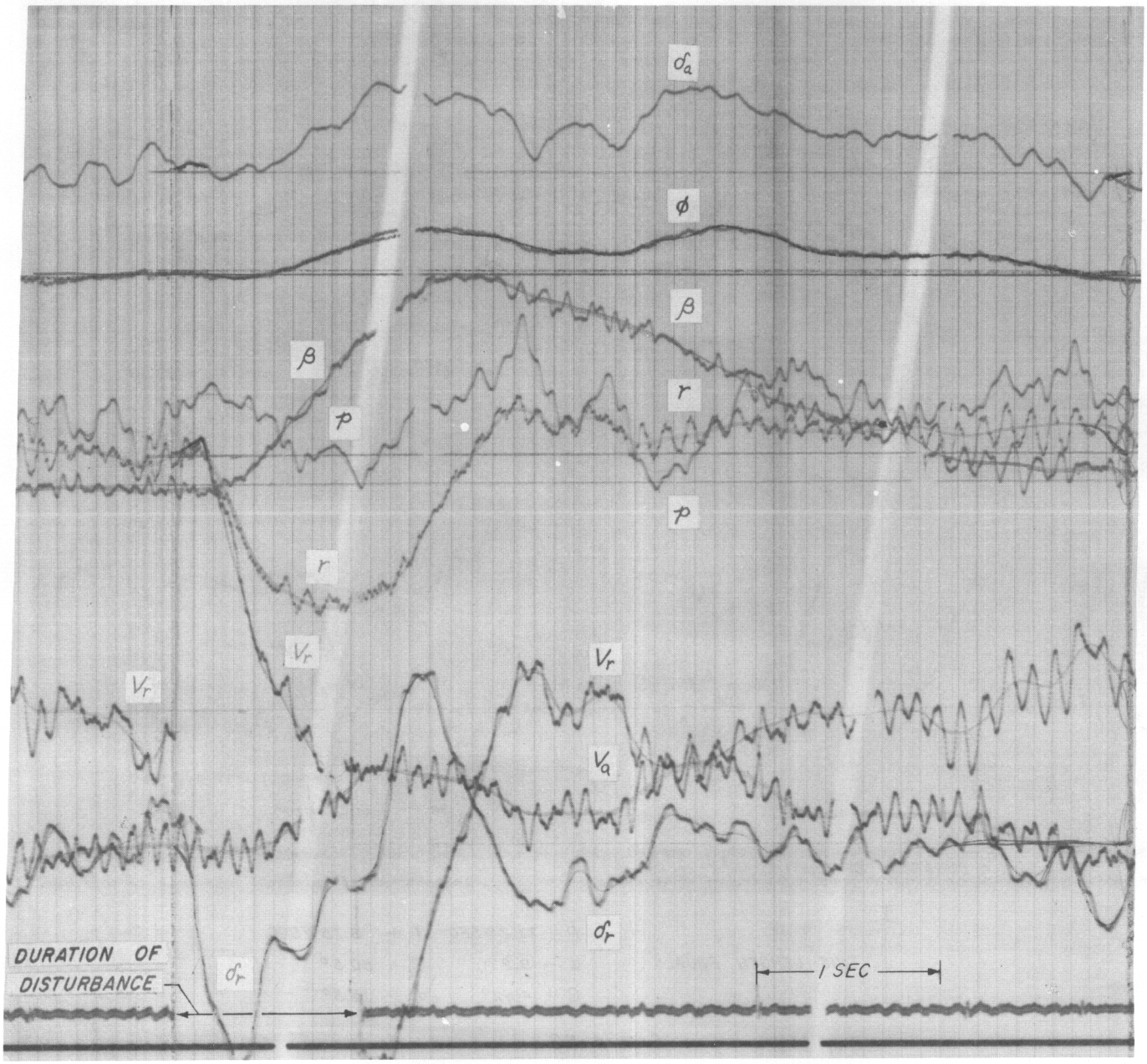
Figure 48

Contrails

PT-26 RUN 105-6d

$\alpha = 28°$

RUDDER DISTURBANCE .80



THIS LENGTH EQUALS :

p + 11.2°/SEC	r + 8.75°/SEC
ϕ − 9.2°	δ_a − 50.5°
β + 4.62°	δ_r − 15.3°

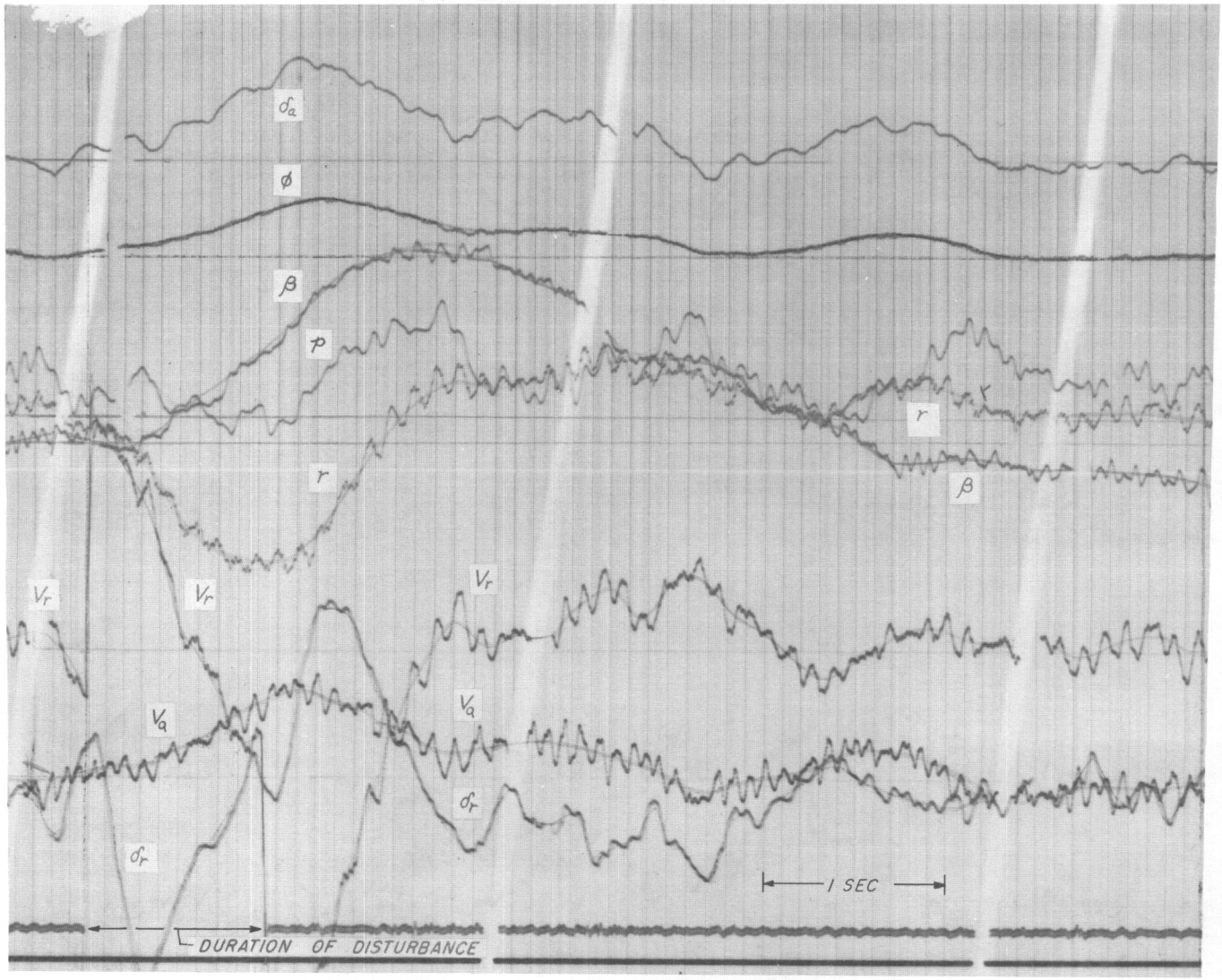
Figure 49

Contrails

PT-26 RUN 105-6e

$\alpha = 28°$

RUDDER DISTURBANCE .80



THIS LENGTH EQUALS :

p + 11.2°/SEC	r + 8.75°/SEC
ϕ − 9.2°	δ_a − 50.5°
β + 4.62°	δ_r − 15.3°

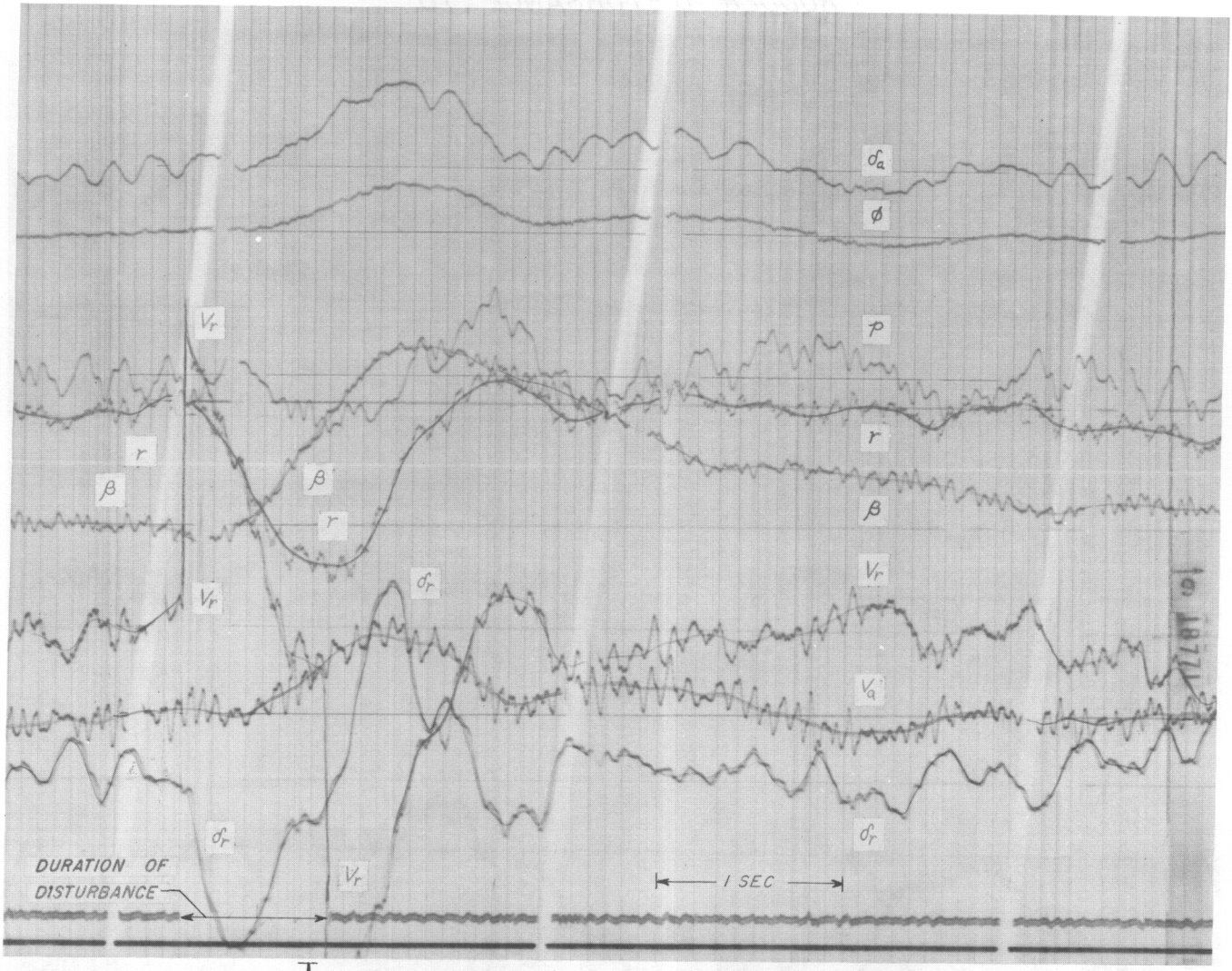
Figure 50

Contrails

PT-26 RUN 105-11h

α = 28°

RUDDER DISTURBANCE .90



THIS LENGTH EQUALS:

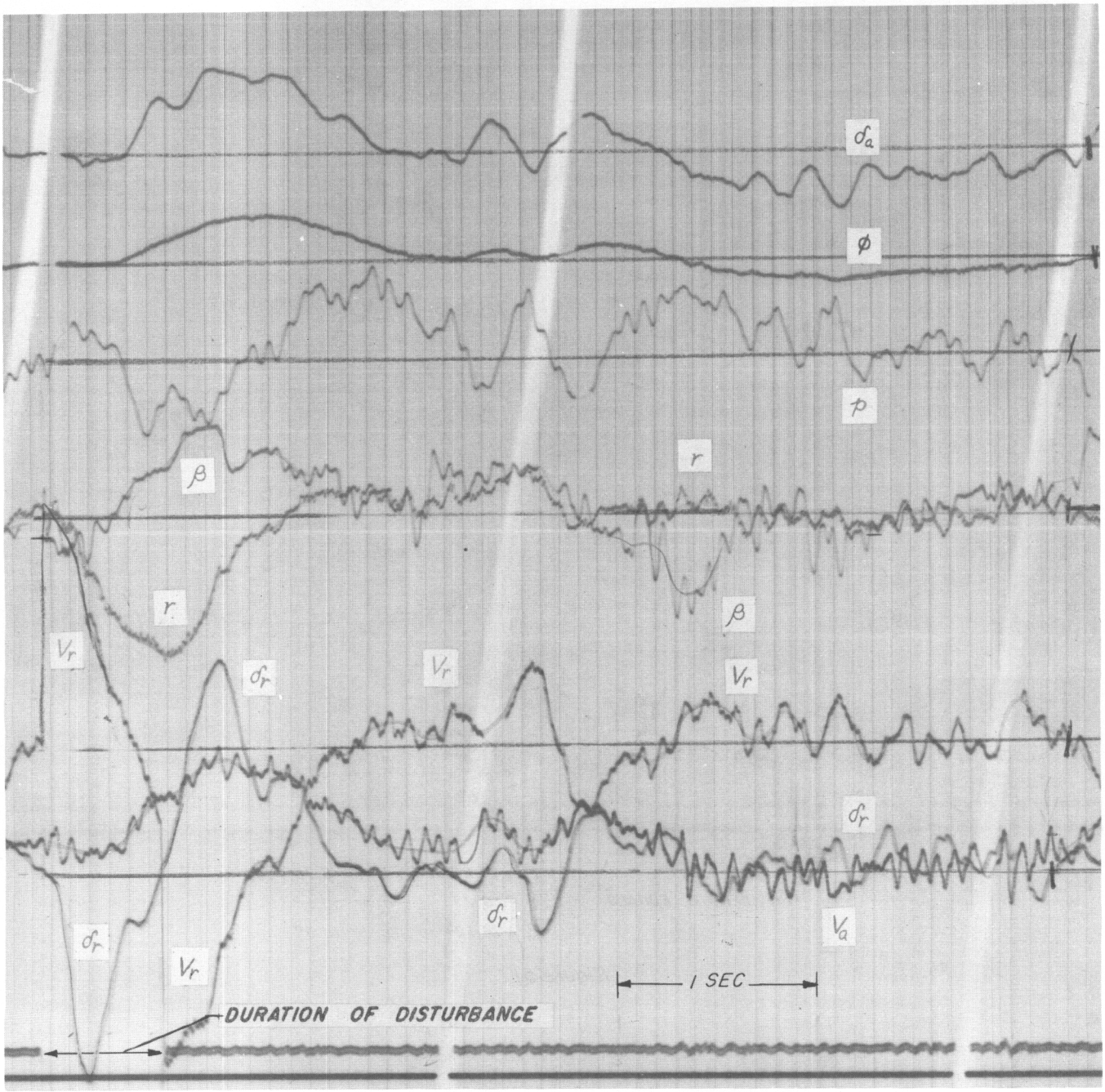
p + 11.2°/SEC	r + 8.75°/SEC
ϕ − 9.2°	δ_a − 50.5°
β + 4.32°	δ_r − 15.3°

Figure 51

PT – 26 RUN 107 – 4f

$\alpha = 24°$

RUDDER DISTURBANCE .70



THIS LENGTH EQUALS :

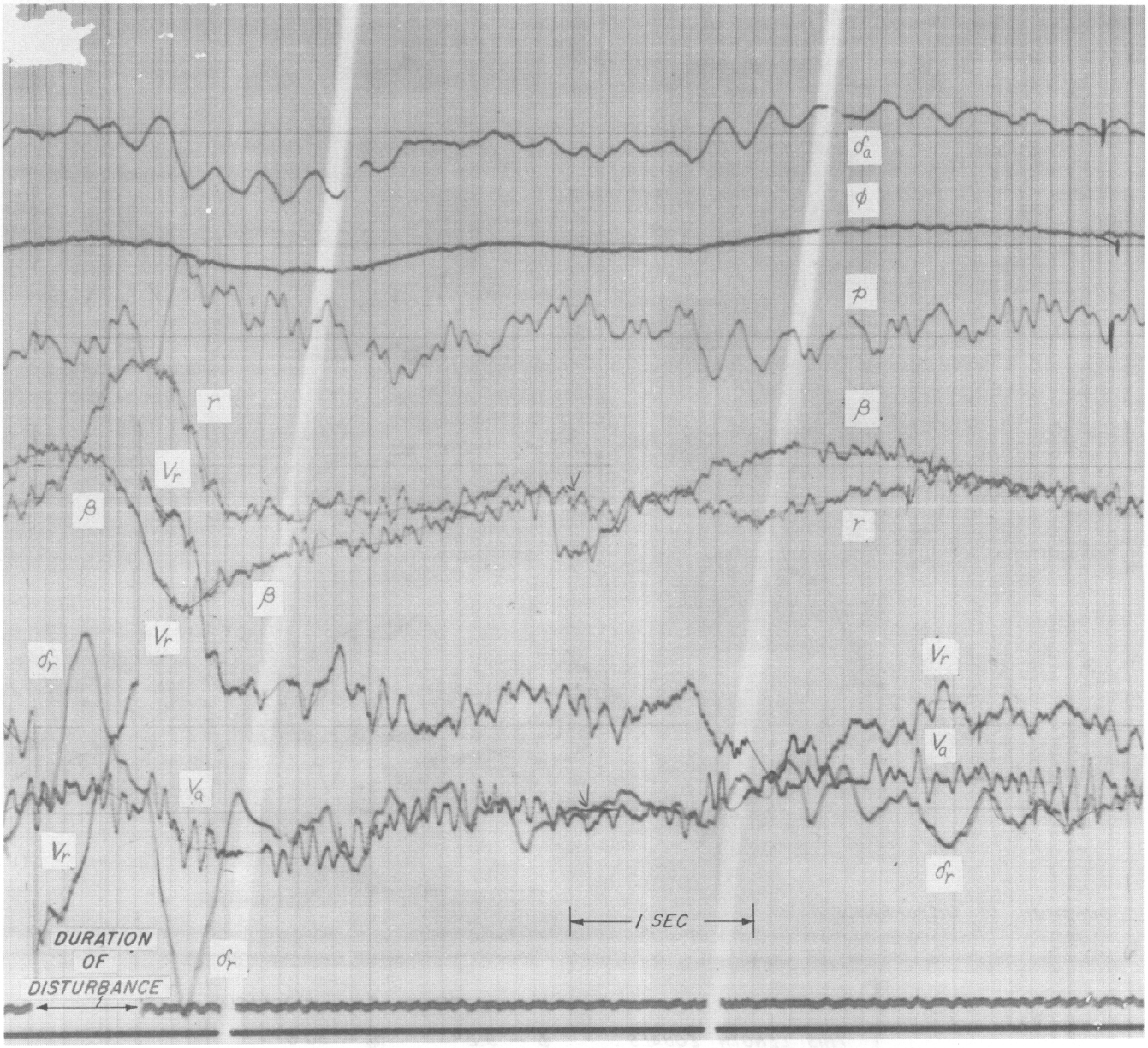
p + 11.2°/SEC	r + 8.75°/SEC
ϕ – 9.2°	δ_a – 50.5°
β + 4.62°	δ_r – 15.3°

Figure 52

PT-26 RUN 107-4h

$\alpha = 24°$

RUDDER DISTURBANCE .70



THIS LENGTH EQUALS :

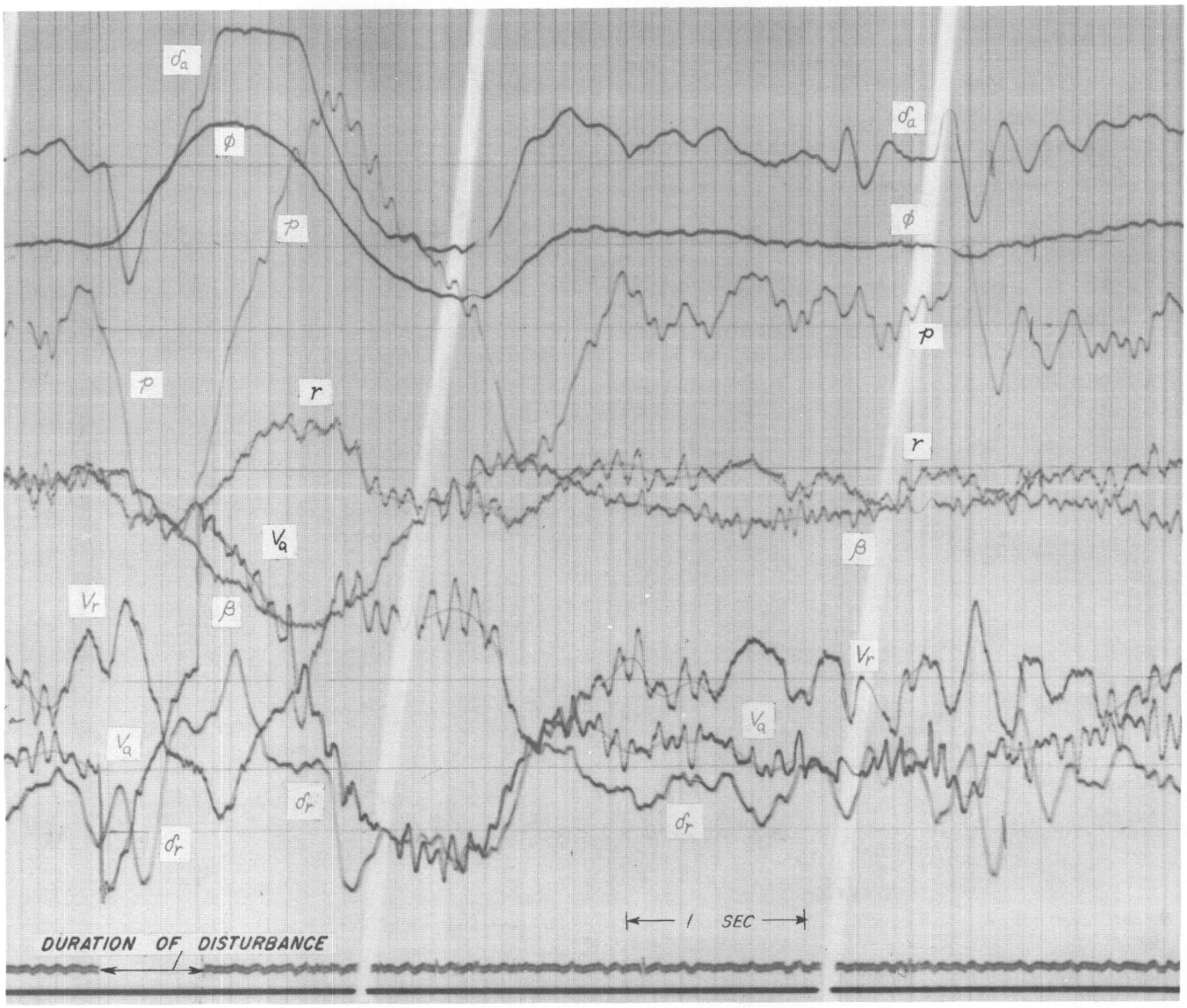
p + 11.2°/SEC	r + 8.75°/SEC
ϕ − 9.2°	δ_a − 50.5°
β + 4.62°	δ_r − 15.3°

Figure 53

PT – 26 RUN 107-5h

$\alpha = 24°$

AILERON DISTURBANCE .23



THIS LENGTH EQUALS:

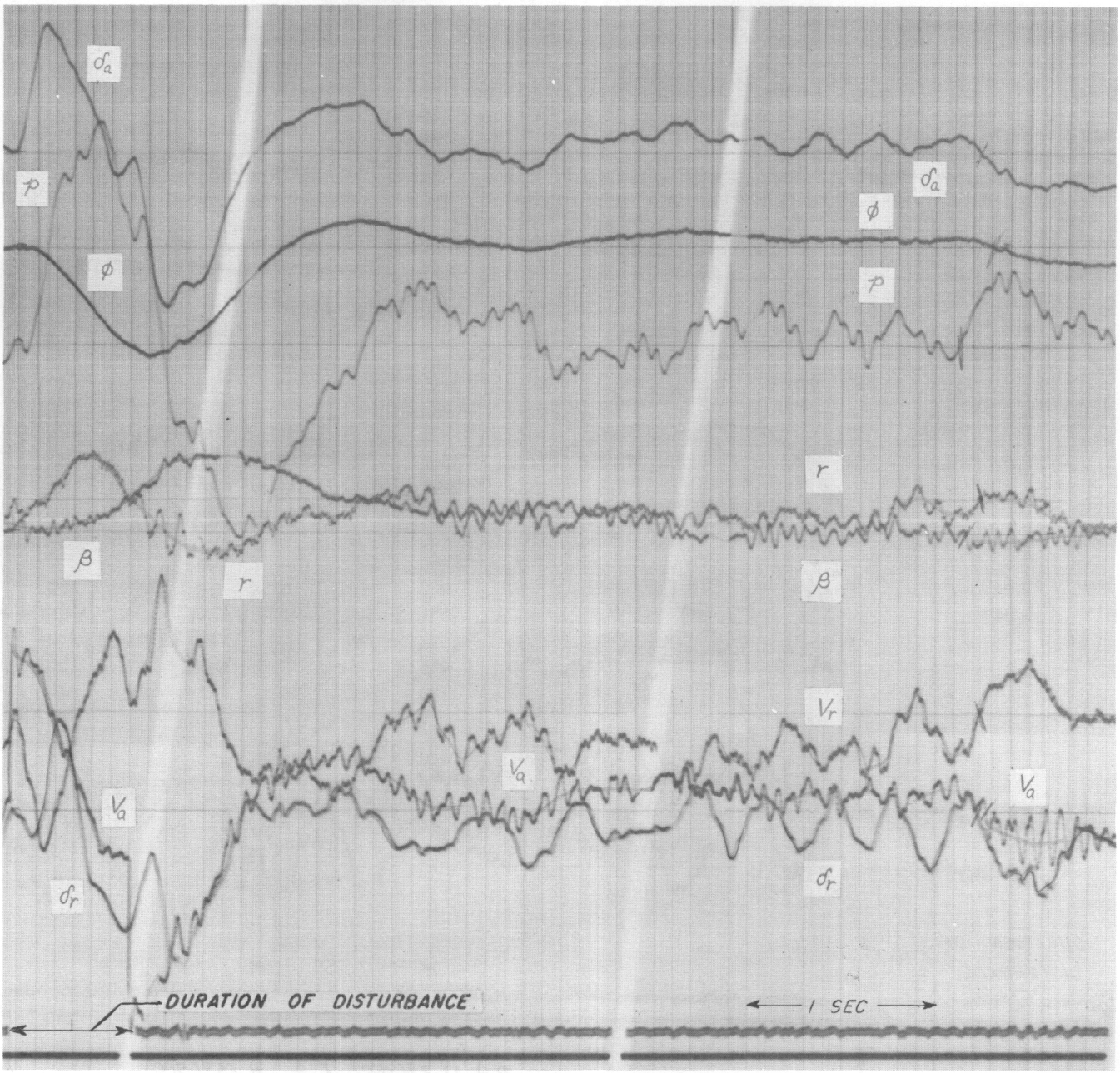
p + 11.2°/SEC	r + 8.75°/SEC
ϕ – 9.2°	δ_a – 50.5°
β + 4.62°	δ_r – 15.3°

Figure 54

PT – 26 RUN 107-5e

$\alpha = 24°$

AILERON DISTURBANCE .23



THIS LENGTH EQUALS :

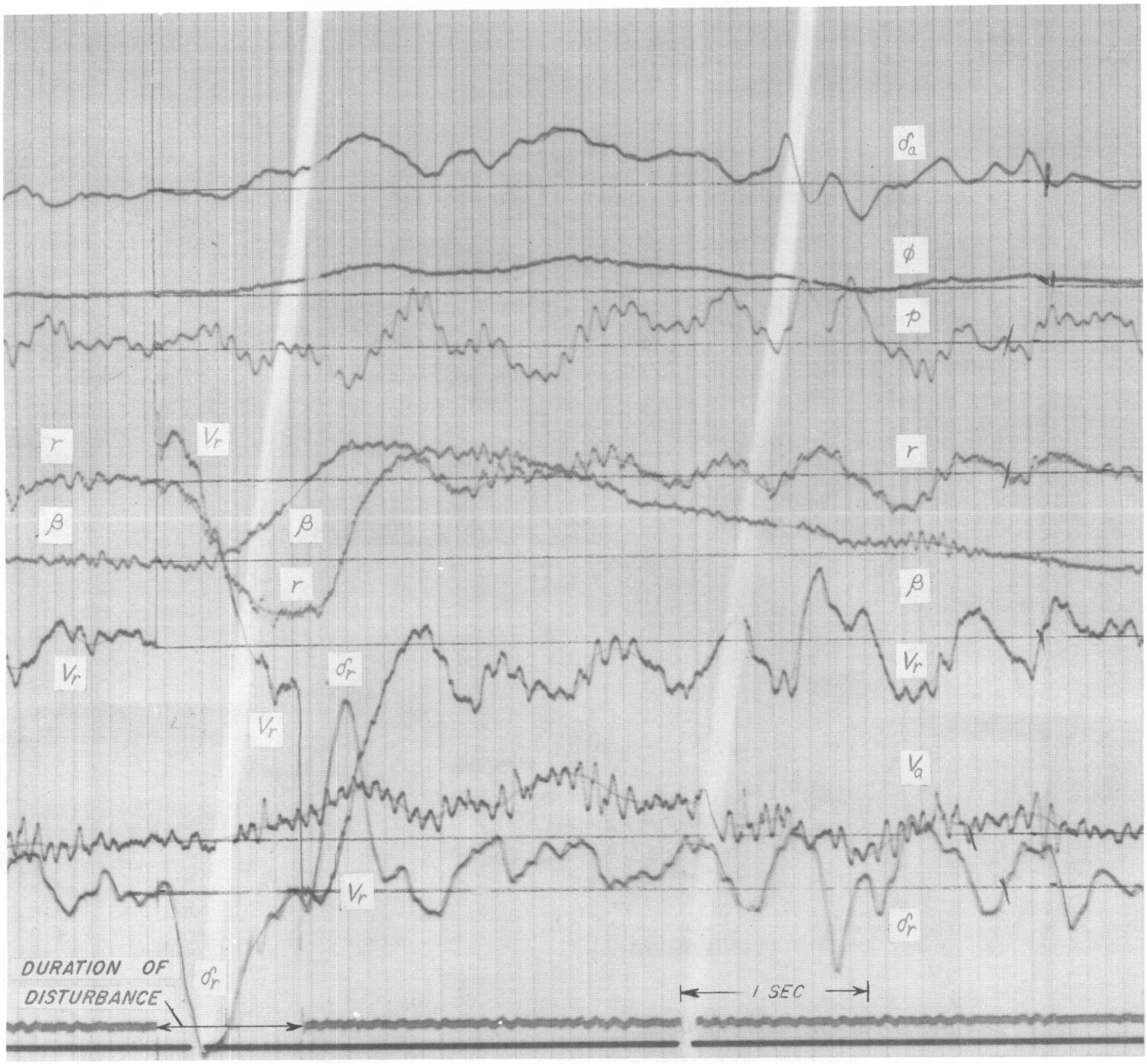
p + 11.2°/SEC	r + 8.75°/SEC
ϕ – 9.2°	δ_a – 50.5°
β + 4.62°	δ_r – 15.3°

Figure 55

PT-26 RUN 107-7b

$\alpha = 20°$

RUDDER DISTURBANCE .70



THIS LENGTH EQUALS :

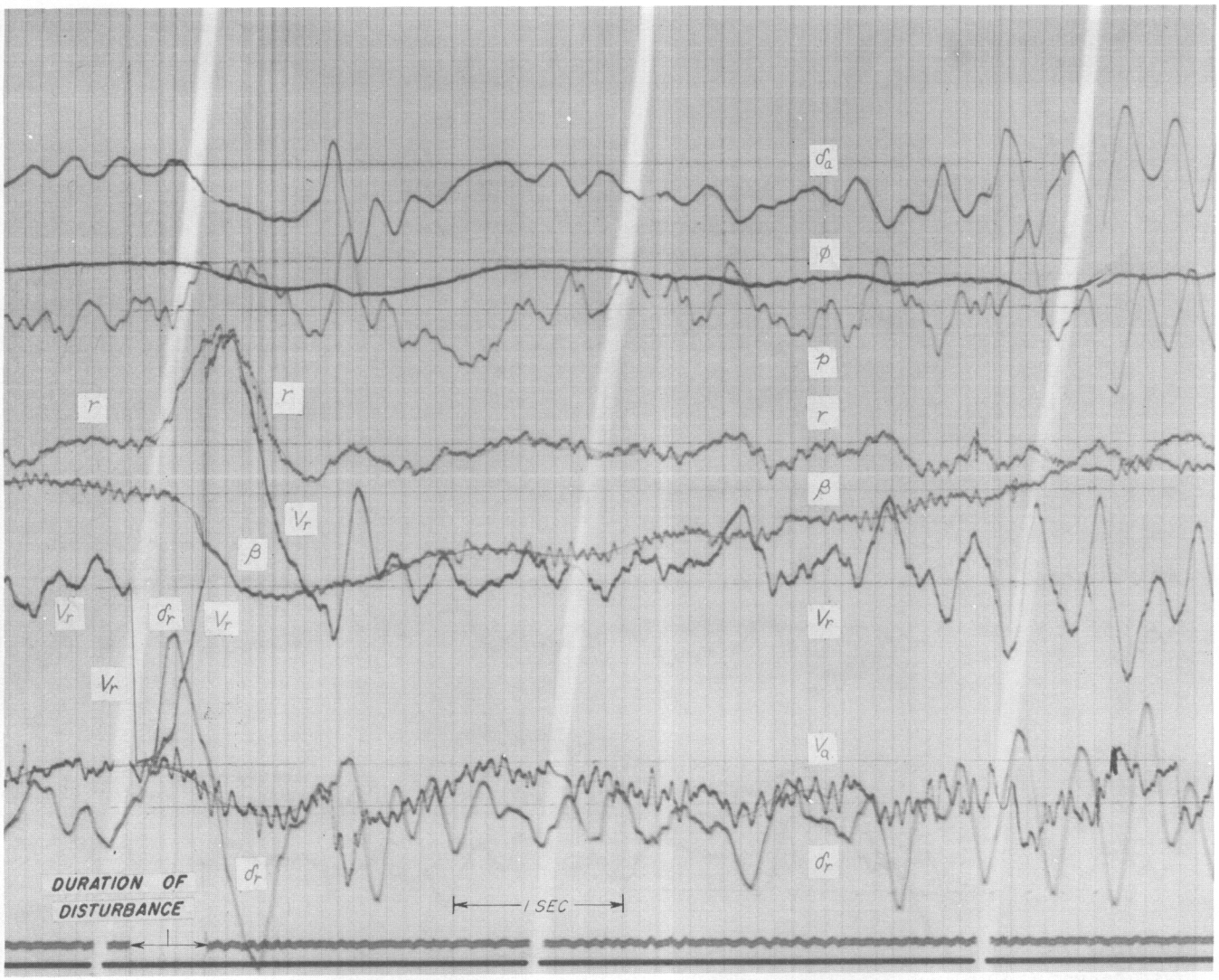
p + 11.2°/SEC	r + 8.75°/SEC
ϕ − 9.2°	δ_a − 50.5°
β + 4.62°	δ_r − 15.3°

Figure 56

PT-26 RUN 107-7e

$\alpha = 20°$

RUDDER DISTURBANCE .70



THIS LENGTH EQUALS :

p + 11.2°/SEC	r + 8.75°/SEC
ϕ − 9.2°	δ_a − 50.5°
β + 4.62°	δ_r − 15.3°

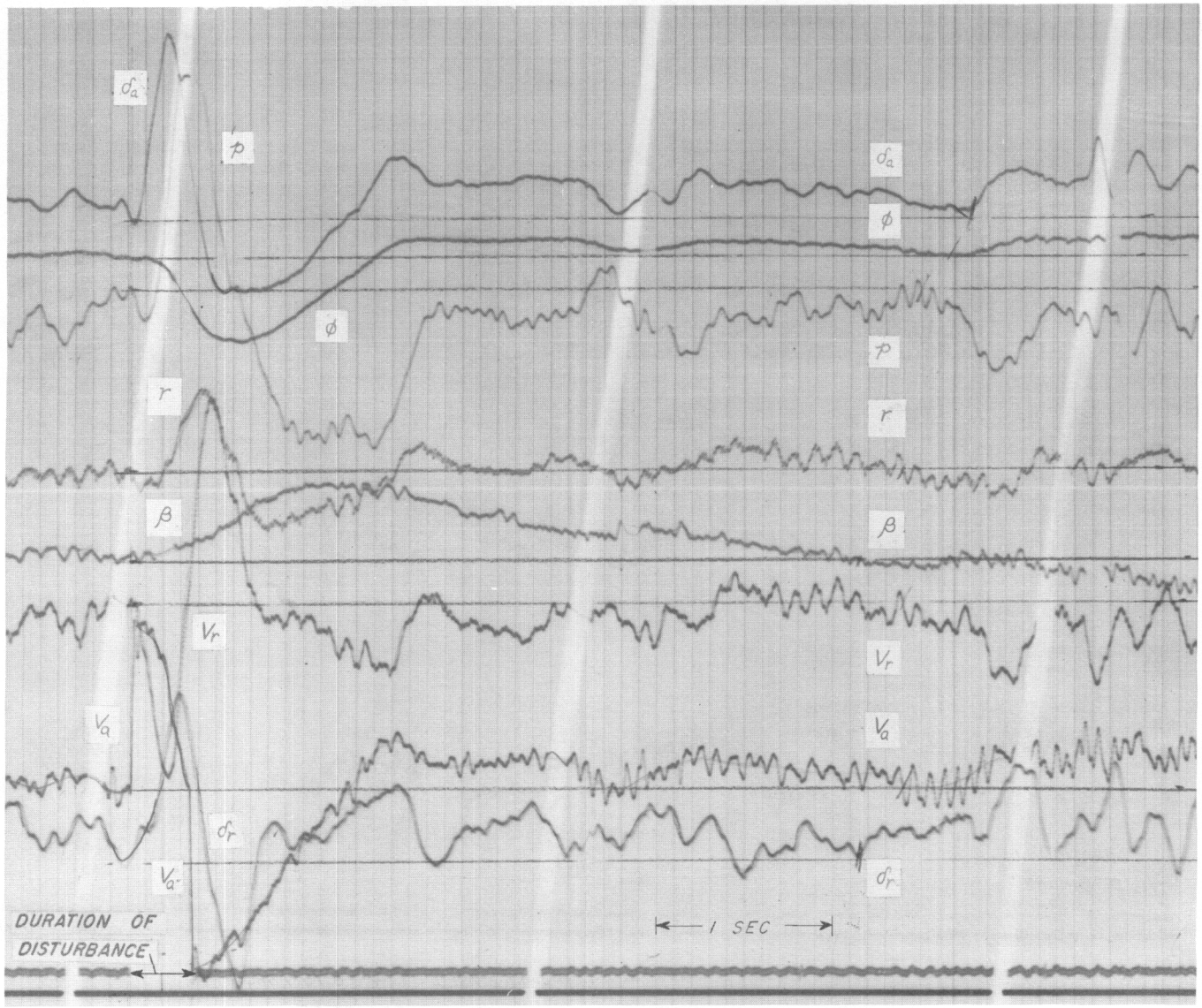
Figure 57

Contrail

PT – 26 RUN 107 – 6g

α = 20°

AILERON DISTURBANCE .23



THIS LENGTH EQUALS :

p + 11.2°/SEC	r + 8.75°/SEC
ϕ – 9.2°	δ_a – 50.5°
β + 4.62°	δ_r – 15.3°

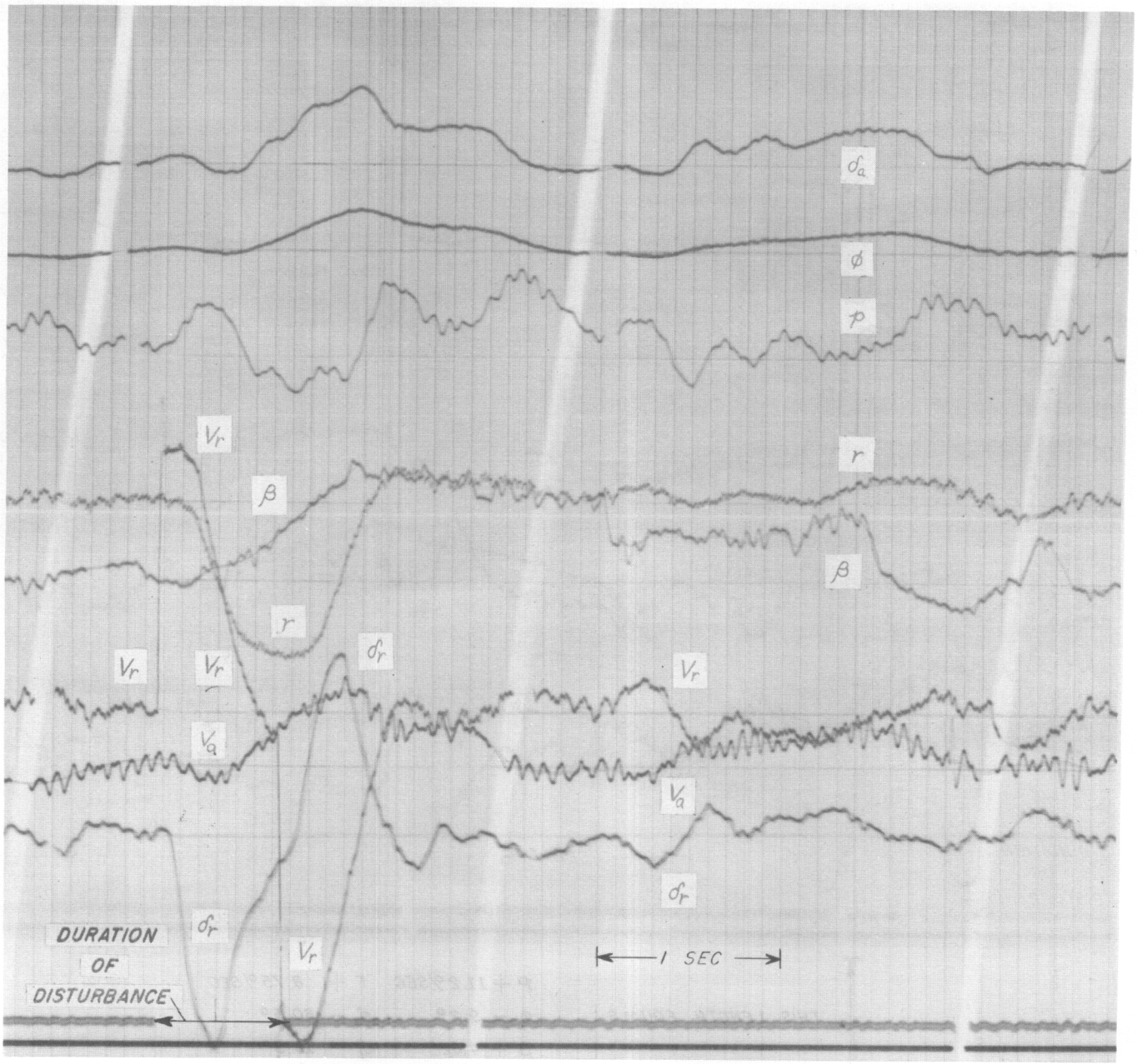
Figure 58

Contrails

PT-26 RUN 107-9g

α = 18°

RUDDER DISTURBANCE .80



THIS LENGTH EQUALS :

p + 11.2°/SEC	r + 8.75°/SEC
ϕ − 9.2°	δ_a − 50.5°
β + 4.62°	δ_r − 15.3°

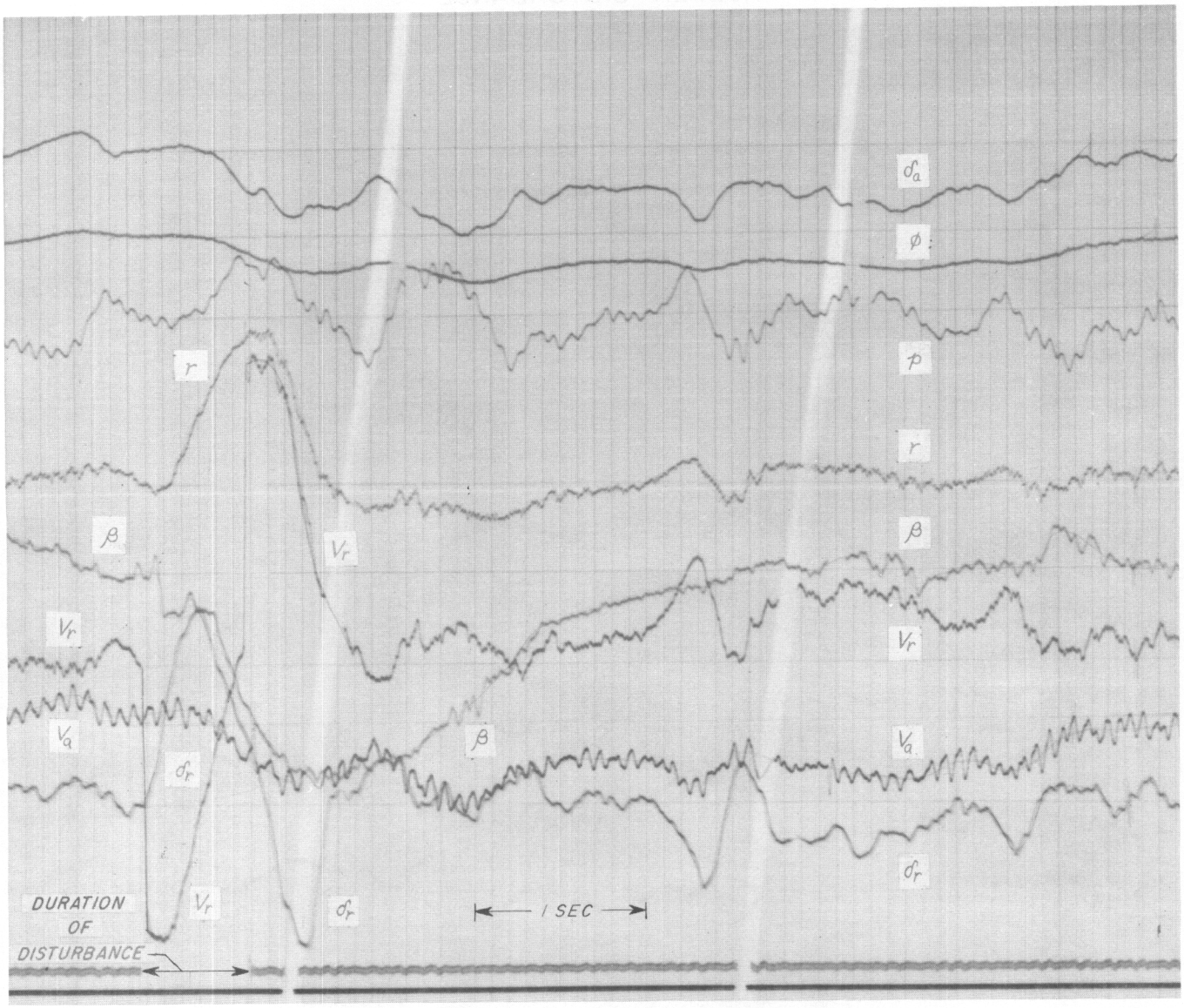
Figure 59

Contrails

PT – 26 RUN 107 – 9i

α = 18°

RUDDER DISTURBANCE .80



THIS LENGTH EQUALS :

p + 11.2°/SEC	r + 8.75°/SEC
ϕ – 9.2°	δ_a – 50.5°
β + 4.62°	δ_r – 15.3°

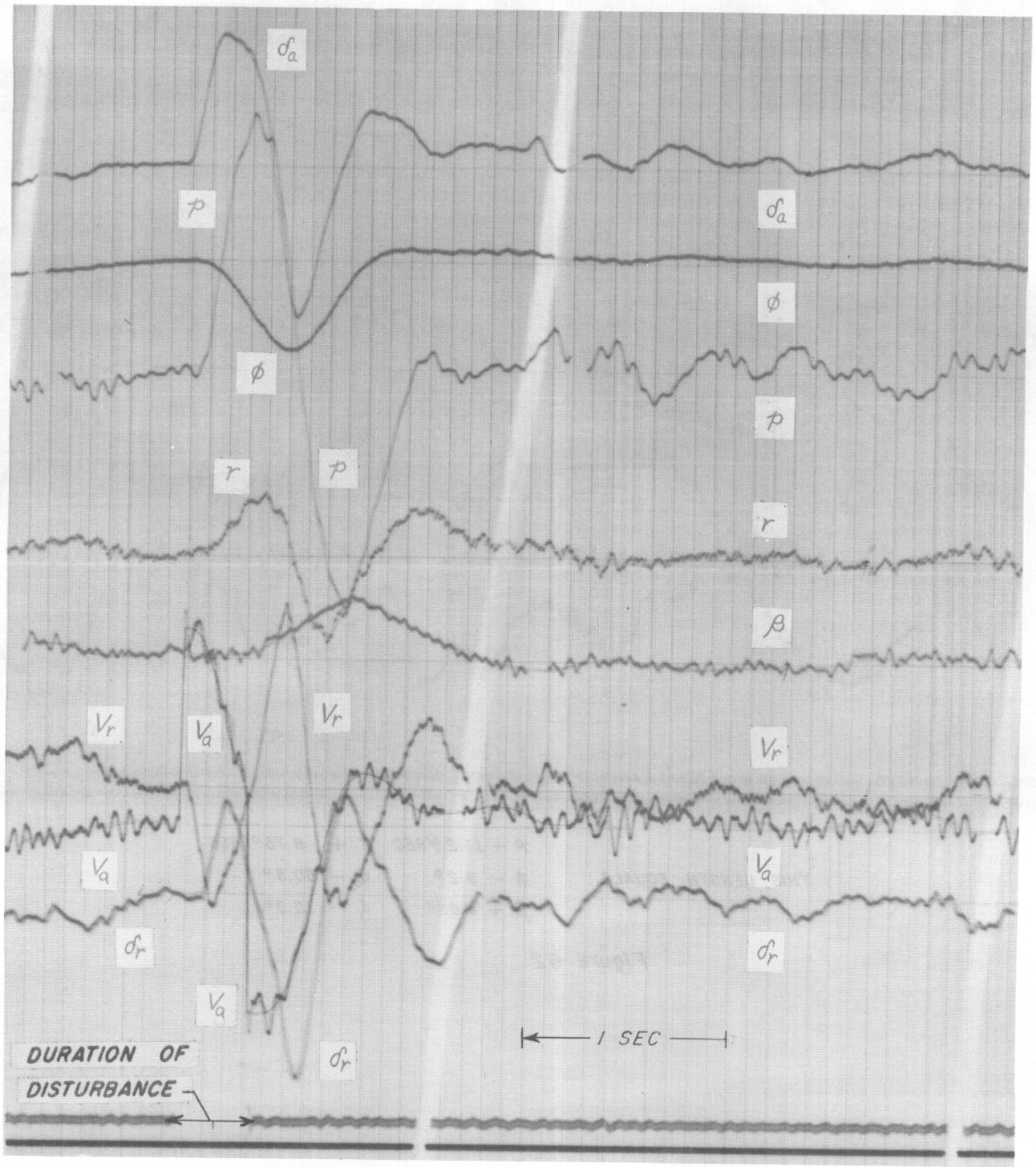
Figure 60

Contrails

PT – 26 RUN 107 – 9a

α = 18°

AILERON DISTURBANCE .23

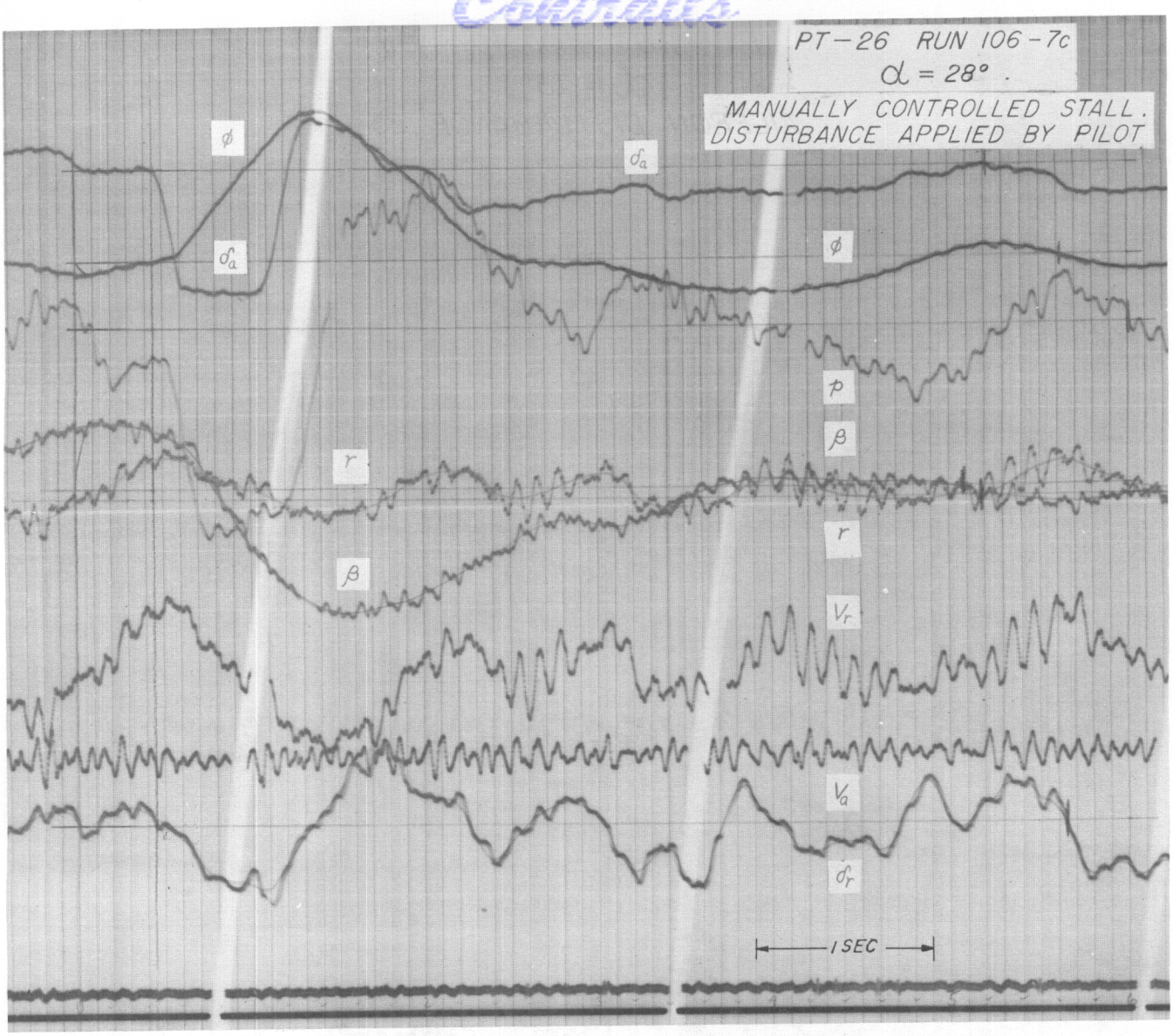


THIS LENGTH EQUALS :

p + 11.2°/SEC	r + 8.75°/SEC
ϕ – 9.2°	δ_a – 50.5°
β + 4.62°	δ_r – 15.3°

Figure 61

Contrails



THIS LENGTH EQUALS :

p + 11.2°/SEC	r + 8.75°/SEC
ϕ − 9.2°	δ_a − 50.5°
β + 4.62°	δ_r − 15.3°

Figure 62