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Summary.—The investigation of gust alleviator effectiveness is limited to an analysis of statistical measurements of c.g. accelerations. The measured alleviation is much smaller than was initially expected and in some cases is even negative. Theoretical analysis, supported by experiment, indicates that the loss of gust alleviator effectiveness is mainly due to the large pitching moment contributed by the ailerons. The aircraft with gust alleviator in operation suffers a considerable loss of stability and calculations show that with increasing gust length alleviator effectiveness decreases and eventually becomes negative. Airframe flexibility also has some detrimental effect.

The effectiveness of the alleviator in terms of wing-root bending stress alleviation is considered to be more favourable, but no experimental data are yet available.

1. *Introduction*.—The programme of experimental testing of the gust alleviator fitted to the *Lancaster* aircraft required some 20 to 30 hours flying in turbulent air. It was estimated that the programme could be completed more quickly and economically by taking the aircraft to Libya and flying it there. Subsequent experience in Libya has shown that the decision taken was a sound one, for during three weeks stay about 40 hours experimental and a total of 57 flying hours were done.

During that time not only the statistical investigation of the gust alleviator effectiveness was made, but also some basic investigation of the aircraft response to natural and simulated gusts. Full analysis of the data obtained is still proceeding and will be reported when complete. The present report gives the preliminary analysis of the gust alleviator investigation with the object of giving a general appraisal of the alleviator without unnecessary delay.

2. *Description of Gust Alleviator*.—2.1. *Principle*.—The alleviator is basically a device for reducing the effective lift slope of the wing. Incidence is measured some distance ahead of the aircraft and symmetrical aileron deflections are made in the sense to reduce the wing load produced by any change in incidence. The reduction in lift slope due to the alleviator, neglecting the effect of response is a_2k, where a_2 is the lift slope due to aileron deflection (based on wing area) and k is the amount of upward aileron deflection produced by unit positive change in incidence†. The proportion of alleviation in gust load, again neglecting the effect of response, is a_2k/a, where a is the wing lift slope, and this quantity is termed the static alleviation.

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* R.A.E. Tech. Note Aero. 2244, received 16th February, 1954.

† The actual signal from the incidence detector is the differential pressure between two pitot heads, and this is the quantity used to control the aileron position. Consequently both k and static alleviation vary with forward speed and in fact are proportional to square of speed.

The alleviation of the gust load can be measured as the reduction in the normal accelerations experienced by the aircraft in gusty weather. The structural significance of the gust alleviator however lies in the reduction of the wing-root bending stresses. The centre of the alleviating load due to aileron is situated more outboard than the centre of the wing load and hence the ailerons are more effective in decreasing the root bending moment than in decreasing the total load. For the *Lancaster* aircraft the alleviation in the root bending moment is about four times larger than the alleviation in the c.g. acceleration.

It was realised that aircraft response would have some effect upon the alleviation actually obtained. The most important contribution is the pitching moment due to aileron deflection. This is in the sense to reduce the alleviation of load but it was hoped the larger gusts encountered would be of sufficiently short duration to prevent the pitching moment from having an appreciable effect. Calculations showed that for a single gust a length of the order of 30 chords is necessary if the effect of pitching is to eliminate entirely the alleviator effectiveness. Very little is known about the gust structure (*i.e.*, gust shapes and distribution) and no theoretical calculations can predict the behaviour of the alleviator in turbulent air. The main object of the flight tests was to establish experimentally the gust alleviator effectiveness.

2.2. *Method of Operation.*—The alleviator which was designed and installed by Boulton Paul Ltd. is essentially an error-actuated servomechanism, in which the input is the differential pressure applied to a twin-pitot pitchmeter mounted ahead of the aircraft nose (*see* Figs. 1 and 2) and the output consists of symmetrical movements of the normal aileron surfaces. These movements are made in addition to the usual anti-symmetrical movements by the pilot.

It is not proposed to describe the construction of each part of the servo in detail in this note, but a brief outline will be given. A detailed description can be found in Ref. 1.

A block diagram of the servo is shown in Fig. 3.

The gust detecting pitchmeter consists of two pitot-tubes inclined at 120 deg to each other and mounted at the front of a cone on the nose of the aircraft (Fig. 2). The pitchmeter is 31 ft ahead of the aircraft c.g. and pipes from each pitot are led into the aircraft nose where they feed an inductance-type pressure pick-up.

The output from the pressure gauge θ_1 (Fig. 3) is converted to a d.c. signal which is then fed through a high-pass filter circuit to a mixing box which is also supplied with a d.c. signal proportional to the aileron deflection output θ_8. The filter circuit was designed to eliminate slow changing input signals, such as occur during speed changes and pilot manoeuvres, while still passing the more rapid gust signals. Accordingly, a cut-off frequency of $0 \cdot 5$ cycles/second was aimed at.

From the mixing box the error signal θ_3 is first put through a resistance-capacitance phase advance circuit. This circuit was included to compensate for the time lags inherent in other components of the servo loop and thus to improve the stability of the servo.

After this stage the d.c. signal θ_4 is modulated and used to drive an a.c. hysteresis motor which operates the valve of an hydraulic powered control unit. This unit is of a type extensively used for aircraft flying controls (Ref. 2) and is essentially a variable stroke swash-plate pump supplying hydraulic fluid to a jack. The jack movement θ_8 provides symmetrical aileron deflections proportional to the input pressure signal θ_1.

Anti-symmetrical aileron movements by the pilot's wheel are also possible by means of an additional hydraulic power-unit not shown in Fig. 3. These deflections are in addition to the gust demand and are independent of the latter, except that the total travel of either aileron is limited to $\pm$ 16 deg.

2.3. *Controls of the Servo System.*—A number of controls are available on the alleviator for purposes of varying the characteristics. They are as follows:

(*a*) The pitchmeter can be tilted relative to the aircraft longitudinal datum. This is done by a hydraulic actuator and permits adjustment of the pressure signal during zero-gust conditions (*b*) The closed-loop gain control is used to vary the final steady state ratio of aileron deflection per unit of pressure differential. This can be used to vary the amount of designed alleviation of gust loads

(*c*) The input signal filter can be switched in or out

(*d*) The open-loop gain can be varied. This alters the dynamic response characteristics of the servo, an increase in gain being accompanied by a more rapid response. This gain is referred to as servo stiffness

(*e*) Either of two phase-advance circuits in the closed loop can be chosen. One of these is a single-stage resistance-capacitance network while the other is two-stage providing more phase advance.

3. *Description of Instrumentation.*—3.1. *Scope.*—The aircraft is extensively instrumented for studying both the alleviator and the aircraft response to gusts, while counting equipment is fitted for determining the average effectiveness of the alleviator in reducing strains and accelerations in long periods of flying in turbulent air.

3.2. *Continuous Trace Recorder.*—A Miller 16-channel galvanometer recorder is fitted for gust response measurements. The following quantities are measured:

(i) Normal accelerations at the c.g., nose and each wing tip. Inductance pick-ups are used with an amplification stage. Filters are used between amplifier and galvanometer, and from calibration the amplitude response was found to be cut by 5 per cent at 3 cycles/second.

(ii) The bending moment at each wing root, using strain-gauges on each flange of each spar. Amplifiers are used between the pick-ups and galvanometers

(iii) Differential pressure at the nose pitchmeter and at two additional pitchmeters near the wing tips. Strain-gauge pick-ups are used with amplifiers

(iv) Aileron angles, gust servo output and pilot's wheel position, using desynn transmitters

(v) Elevator angle and nose pitchmeter deflection, also using desynn transmitters

(vi) Rate of pitch, using a desynn transmitter on a spring-restrained gyro

(vii) Incidence ahead of the nose, using a desynn transmitter on a vane. This vane, together with a nose pitchmeter are shown in Fig. 2.

The channels using amplifiers were calibrated during each flight; the accelerations and strains by comparing readings in steady turns with those obtained from a remote-indicating Barnes accelerometer, and the pressures by comparing with visual gauge readings.

The channels using desynn transmitters were calibrated only on the ground. The desynns were used as part of a d.c. bridge circuit, the supply voltage being controlled by an observer in the aircraft. By this means it was hoped that day-to-day calibration changes were avoided.

3.3. *Counting Accelerometer and Strain-Gauges.*—The accelerometer is mounted at the aircraft c.g. and is designed to operate a number of electrical pulse counters. A separate counter operates each time a given acceleration threshold is passed. A detailed description of the instrument is provided in Ref. 3. The pattern used in the *Lancaster* differed from the standard version inasmuch as the deflection of the mass is used to make contact between a brush and various electrical contacts rather than to deflect the counters mechanically.

The counting strain-gauge has been described in Ref. 4. Two gauges are used on the *Lancaster*, one mounted on the front wing spar and one on the tail spar. The quantities measured are the deflections of the wing and tail spar flanges, and as with the accelerometer the results are reproduced on a number of electrical counters.

Both the acceleration and strain counters are mounted in an auto-observer and photographed by an electrically operated camera actuated by an electrical timing device. This unit is capable of taking shots at pre-set intervals of between a half and five minutes and simultaneously switching the alleviator alternately on and off. Switching off the alleviator is attained by short-circuiting the electrical output from the pitchmeter pick-up.

4. *Work Prior to Trip to Libya.*—Approximately 18 months was spent on the preliminary work. Great difficulties were encountered in keeping the instrumentation, the gust alleviator and the aircraft serviceable simultaneously. As a result a very small amount of experimental flying was done. Progress was further delayed due to difficulty in finding sufficiently gusty conditions when the aircraft was serviceable.

The preliminary flying had indicated that the gust alleviator was not decreasing the gust loads as theoretically expected. It was suspected that one of the reasons for lack of alleviation was the loss of aircraft stability and the large pitching movements with gust alleviator on. In order to cut down the aircraft plus gust alleviator response to low frequency disturbances and to improve the handling of the aircraft with gust alleviator on, a filter circuit was incorporated into the electronic link. The frequency response curves without and with filter are shown in Fig. 4. Due to the design characteristics of the electronic link, the frequency response with filter as shown in Fig. 4 was the best obtainable. Pilots commented favourably on the improvement of the aircraft handling with the filter switched in (Appendix III).

It was thought that some improvement of the electronic link could be obtained by increasing the stability of the closed loop. A two-stage phase advance circuit was therefore designed and inserted in the electronic link with the object of increasing the phase advance given by the original single stage unit. The system could be operated either with single-stage or with two-stage phase advance, the circuits being connected via a selector switch. Fig. 5 shows the gust alleviator frequency response curves for both cases. The introduction of the two-stage phase advance decreased the time lag of the gust alleviator from about $0 \cdot 19$ sec to about $0 \cdot 125$ sec and enabled stiffness of the open loop of the electronic link to be increased without running into instability.

5. *Statistical Measurements of Gust Alleviator Effectiveness.*—Figs. 8 to 12 show a few typical examples of the counting accelerometer recordings. Number of encountered accelerations is plotted against magnitude of acceleration. Acceleration was measured in intervals of $0 \cdot 1g$. Two curves are shown, one giving the number of accelerations with gust alleviator OFF, the other the number of accelerations with gust alleviator ON. The time intervals for gust alleviator ON and alleviator OFF were about 30 seconds. After each flight a small sample of the accelerometer record was analysed, so that the subsequent flight programme could be modified in the light of the experience gained.

The behaviour of the electronic link was not entirely satisfactory and it was found necessary to re-set it before each flight. Each setting of the electronic link was checked by static and frequency response calibrations on the ground. The results of static calibrations are summarised in Fig. 6, where each point represents one calibration. The calibrations were made for various electronic link arrangements, but as can be seen from Fig. 6 the static calibration depends only on the setting of the external gain and is reasonably consistent. An example of frequency response calibration is given in Fig. 7, which shows clearly typical day-to-day variations in the dynamic characteristics of the electronic link for nominally constant setting.

Fig. 8 shows the acceleration distribution for the gust alleviator setting giving 19 per cent of static alleviation. Details of the electronic link setting are given on the top of the figure. On the same figure the alleviation due to gust alleviator is plotted against gust magnitude (curves for the initial sample and final results are both given). Alleviation is defined as

$$\frac{\text{number of accelerations with alleviator off} - \text{number with alleviator on}}{\text{number with alleviator off}} .$$

It can be seen that for this particular test the number of the accelerations when alleviator is ON is larger than the number when alleviator is OFF, *i.e.*, the alleviation is negative. The mean alleviation for the whole test is -9 per cent.

Fig. 9 shows the acceleration distribution for a similar alleviator setting, except that the static alleviation was decreased to 12 per cent. The test shows positive alleviation, the mean value for the test being $+5$ per cent.

Figs. 10 and 11 show the records of the counting accelerometer for two different forward speeds. The setting of the alleviator was the same as in Fig. 9, but the static alleviation, which increases with increasing forward speed, was 19 per cent for 180 knots and 9 per cent for 120 knots. The results show negative alleviation at 180 knots and zero alleviation at 120 knots.

Fig. 12 shows the results of flight with reversed action of the gust alleviator. The static alleviation was -25 per cent and the mean alleviation obtained experimentally was also -25 per cent.

Figs. 8 to 12 show only a few typical results of the statistical investigation. The total number of records on which the analysis was based was 15, representing some 20 hours of flying.

It can be seen, that all the records have the same peculiarity in the shape of the measured alleviation curves. It appears that the gust alleviator is more effective in reducing the number of gust loads of small magnitude than the number of large gust loads. Indeed, every record shows an increase of the alleviator effectiveness with decreasing magnitude of the encountered acceleration. It must be remembered that the number of large accelerations is very small in comparison with the number of small accelerations; neverthelesss this phenomenon seems to be a genuine one.

The results of the statistical investigation are summarised, somewhat arbitrarily, in Fig. 13, where the mean alleviation measured in each test is plotted against the static alleviation. No attempt has been made to discriminate between different electronic link settings, but analysis of the available data and theoretical considerations point to the static alleviation as the main parameter.

It can be seen from Fig. 13 that measured alleviation is not a well-defined function of static alleviation, but the general trend can be observed. For small values of static alleviation the measured alleviation is positive, but with increasing static alleviation the measured alleviation decreases and eventually becomes negative. No well-defined curve can be drawn through the experimental points and the curve shown in Fig. 13 is only suggested as the probable characteristic of the aircraft with gust alleviator. The decrease of the measured alleviation with the increase of static alleviation can be explained by the loss of aircraft stability and it is shown in Appendix I that the static margin decreases with increasing static alleviation and becomes zero for about 30 per cent of static alleviation with a corresponding drop in the value of the manoeuvre margin. The handling difficulties of the aircraft with small stability in conjunction with large changes of longitudinal trim due to alleviator action may well result in extra accelerations being induced by the pilot, which are counted as gust loads. The scatter in Fig. 13 can be explained by the effect of shape of the encountered gusts. It is shown in Appendix II that the alleviator effectiveness decreases with increasing gust length, thus the measured alleviation is a function of the shape of the gust.

6. *Aircraft Response to 'Simulated Gust Alleviation'*.—The response of aircraft with gust alleviator to a 'simulated gust' was measured using the Miller recorder. The gust input was simulated by tilting the detector head through the required angle and at a predetermined rate. In this way the pressure fed into the servo was equivalent to a flat-topped gust. Figs. 14 and 15 show the analysis of the records obtained for two gust lengths, $H = 2$ and $H = 9 \cdot 4$ in chords. Only recorded quantities relevant to the present discussion are shown, others such as rate of roll, wing root stresses, etc., are omitted for clarity. A sample of the complete record is shown in Fig. 16, and this sample corresponding to gust length $H = 2$ is analysed in Fig. 14. In both cases the elevator angle was constant.

At the bottom of Fig. 14a the equivalent gust at the detector head is shown, which is the recorded tilt of the head expressed as equivalent gust velocity at a given forward speed. In this particular case the simulated gust velocity was 20 ft/sec and the gust length $\bar{H} = 2$ chords. The next curve shows the pressure recorded at the pressure pick-up. There is a time lag of the order of $0 \cdot 03$ sec between the tilt of the detector head and build-up of pressure at the pressure pick-up. This time lag is due to a lag in the pipes joining the detector head and the pick-up. It should be noted that after about $0 \cdot 5$ sec the pressure begins to increase still further as the result of aircraft pitching motion.

The output from the servomotor, shown in the next curve, is lagging behind the nose pressure by about $0 \cdot 125$ sec, which is the time lag of the gust servo. In this particular case the output from the servomotor is sluggish and less than would be expected from the static calibration. The simulated gust is rather sharp ($\bar{H} = 2$), but the input rate is below the maximum design rate of servo and the gust servo should follow the input signal. This confirms the general impression that the gust alleviator response to sharp gusts was less as compared with initial ground calibrations.

The recorded aileron angle is lagging behind the servomotor by about $0 \cdot 02$ sec, probably due to the time lag of the hydraulic part of the gust alleviator and due to flexibility of the control run.

The acceleration measured at the c.g. of the aircraft due to aileron deflection builds up initially very slowly and oscillates appreciably. The initial time lag in the c.g. acceleration curve may be attributed either to the Wagner effect on the aileron lift or to wing flexibility effects. It is generally assumed that the Wagner effect on the lift of trailing-edge controls is negligibly small. Examination of the wing-tip acceleration curve (Figs. 14 and 15) fully verifies this assumption, as the wing-tip acceleration builds up instantaneously with the aileron deflection. It is thought that the effect of the aileron inertia on the wing-tip acceleration is negligibly small. The oscillatory character of the c.g. and wing-tip accelerations is due to wing oscillation mainly in the fundamental bending mode and after the initial disturbance, the wing-tip and c.g. accelerations are 180 deg out of phase, and the frequency of oscillations is about 3 c.p.s. The initial time lag in the build-up of c.g. acceleration is however larger than that due to the fundamental bending mode.

The results of the theoretical estimate of the rigid aircraft response to the ' simulated gust alleviation ' are also shown in Fig. 14. The estimated and measured values of normal acceleration are in good agreement, but the values of measured rate of pitch are larger than calculated ; it should be noted however that the c.g. position used in calculations is slightly different from the actual c.g. position.

The time history of c.g. acceleration without the gust alleviator and due to a gust equal to that simulated, was estimated using the method of Ref. 5, *i.e.*, neglecting pitching motion. The Wagner effect and the time delay between gust hitting the detector head and the wing were taken into account. The c.g. acceleration of the flexible aeroplane was roughly estimated assuming the presence of a fundamental bending mode only. The results are plotted in Fig. 14b together with the measured c.g. acceleration due to gust alleviator replotted from Fig. 14a. It can be seen that for a rigid aircraft, the maximum of the gust alleviator load occurs roughly at the same instant as the maximum of the load due to the gust itself. The flexibility of the wing delays the gust alleviator response, decreasing considerably the alleviation for the first load peak and actually increasing the load for subsequent peaks. The wing flexibility also increases the difficulty of proper timing between the encountered gust and gust alleviator response, at least for short gusts.

An attempt was made to estimate the c.g. and wing-tip accelerations for a flexible aircraft using the loading function calculated for the rigid aircraft. The calculations were only approximate, using a semi-rigid method and assuming wing bending in the fundamental mode only (Ref. 10). The wing frequency used in calculations was $3 \cdot 08$ c.p.s., and was obtained experimentally for this particular aircraft. It is interesting to note that this frequency is lower than the $3 \cdot 2$ c.p.s. measured during ground resonance tests (Ref. 6). The natural damping in this mode and at 150 knots forward speed was $\zeta = 0 \cdot 06$. This value was obtained from some unpublished previous measurements of wing damping for a range of forward speeds. The shape of wing bending in the fundamental mode was taken from the ground resonance test (Ref. 6). The results of these calculations are shown in Fig. 17, together with the experimental curve. It will be seen that the estimated accelerations of the wing tip agree roughly with the measured ones, except that the frequency of the measured oscillations is somewhat lower. Examination of the original record, Fig. 16, and others not presented here, shows that initially the oscillations are of lower frequency and subsequently they settle down to a frequency corresponding to the fundamental mode. It is not understood if this phenomenon is due to non-linearity of wing oscillation with large amplitude, or due to superimposition of some other modes of wing bending. The calculated c.g. accelerations (Fig. 17) are of the same amplitude as the measured ones, but show an initial lag larger than that predicted. This seems to indicate, that the wing bending due to aileron load is of more complicated pattern than that assumed in the present, simplified calculations.

The response of the aircraft to a simulated gust of greater length, $H = 9 \cdot 4$, is shown in Fig. 15. The gust alleviator setting and the aircraft conditions were identical with those of Fig. 14. The simulated gust magnitude and rate are smaller than for $H = 2$ and the response of the aileron's movement is better. The c.g. acceleration due to aileron deflection follows the ailerons more faithfully, but the delay caused by the wing flexibility is still evident. The estimated and measured c.g. accelerations of the rigid aircraft are in good agreement and the measured rate of pitch is again higher than the estimated one.

The effect of pitching is very pronounced for longer gusts and considerably decreases the alleviating load, otherwise available from ailerons. This is shown in Figs. 14b and 15b, where c.g. acceleration due to aileron displacement corrected to no-pitch condition, is plotted.

7. *The Alleviation of Wing-Root Bending Moment.*—A gust alleviator with moving ailerons as the gust alleviating device is more effective in reducing the wing-root bending moment than in reducing c.g. acceleration. At the same time it is expected that the tailplane loading is increased. The counting strain-gauges installed at wing root and at tailplane front spar were intended for statistical investigation of the gust alleviator effectiveness simultaneously with the counting accelerometer measurements. However the counting strain-gauge records are not reliable enough to be included in the present note. Due to persistent zero drift of the electronic link the ailerons were changing their zero position throughout each flight, causing alterations in the spanwise lift distribution and hence in root bending moment under $1g$ flight conditions. It is hoped, that an approximate evaluation of the gust alleviator effectiveness in decreasing root bending moment will be possible by careful selection of the records obtained. This however requires a considerable amount of work and no results are yet available.

The wing-root bending moment due to simulated gust alleviation as obtained from the full record of Fig. 16, is plotted in Fig. 18, together with c.g. acceleration and aileron deflection. The wing-root bending moment is given in terms of acceleration producing equivalent bending moment in a steady pull-out with ailerons in zero position. It can be seen that the root bending moment, for the rigid aircraft, due to aileron deflection is equivalent to a root bending moment due to approximately $0 \cdot 45g$ normal acceleration in a steady pull-out, while the C.G. acceleration is only slightly more than $0 \cdot 1g$. This indicates that the ailerons are about four times more effective in relieving the root bending moment than in decreasing c.g. accelerations due to gusts. A further point of interest is that the root bending moment does not lag behind the aileron deflection and for the first two oscillations is well out of phase with c.g. acceleration.

8. *Other Experimental Investigations Using Miller Recorder.*—Apart from the statistical measurements of the gust alleviator effectiveness and the aircraft response to a simulated gust, an extensive investigation was made of aircraft dynamics with and without the gust alleviator using Miller recorder. The following records were obtained:

(i) Aircraft response in gusty air with and without the gust alleviator.

(ii) Aircraft frequency response to a sinusoidal pressure signal fed into the gust alleviator.

(iii) An attempt was made to measure the aircraft frequency response to a sinusoidal elevator movement.

The records were taken to provide experimental data against which the analytical study could be checked. It is believed that it was the first time in this country that an attempt was made to measure the frequency response of an aircraft in flight. The analysis of the records obtained will take a considerable time and is not included in the present note.

9. *Measurement of Factors Controlling the Alleviator Performance*—9.1. *Wing Lift Slope.*—This quantity is required for estimation of aircraft response to a given gust and was measured under steady conditions by flying at a series of speeds and recording the change in aircraft attitude by means of a bubble inclinometer. Position error corrections were applied to the A.S.I. readings. The first measurements were all made in level flight, but as it was thought that the slipstream might have an appreciable effect on lift slope the tests were repeated at constant power and engine speed, a correction for rate of change of altitude being applied to the attitude measurements when deriving incidence. No appreciable difference in lift slope as determined in both conditions was detected.

By consideration of the total lift on the aircraft it can be shown that:

$$a = \frac{dC_L}{d\alpha} - \frac{S_T}{S} a_{1\,T}\left(1 - \frac{d\varepsilon}{d\alpha}\right) - \frac{S_T}{S} a_{2\,T} \frac{d\eta}{d\alpha}$$

where $dC_L/d\alpha$ is the rate of change of total lift coefficient with measured incidence. Using previously measured values of $a_{2\,T}$ and $d\eta/d\alpha$ (from trim curves) and also using an estimated value of $a_{1\,T}(1 - d\varepsilon/d\alpha)$ it was found that:

$$a = \frac{dC_L}{d\alpha}$$

to within 5 per cent.

The value of lift slope so determined was:

$$a = 4{\cdot}8\,.$$

9.2. *Lift Slope Due to Ailerons.*—This was measured, relative to wing lift slope by recording the change of aircraft incidence with change in gust aileron angle at constant speed.

Incidence was again measured with a bubble inclinometer and gust aileron was applied by rotating the pitot pitchmeter with the alleviator in operation. Aileron and elevator angles were recorded on the Miller oscillograph, all the tests being made at 150 knots A.S.I.

Again, by consideration of the total aircraft lift the ratio of lift slope due to ailerons to wing lift slope is given by:

$$\frac{a_2}{a} = -\frac{d\alpha}{d\xi}\left\{1 + \frac{a_{1\,T}}{a}\frac{S_T}{S}\left(1 - \frac{d\varepsilon}{d\alpha}\right)\right\} - \frac{a_{2\,T}}{a}\frac{S_T}{S}\frac{d\eta}{d\xi}\,.$$

where $\dfrac{d\alpha}{d\xi}$ is the rate of change of incidence with mean aileron angle (measured)

$\dfrac{d\eta}{d\xi}$ is the rate of change of elevator angle to trim with mean aileron angle (measured).

Once again using measured values of $a_{2\,T}$ and estimated ones for $a_{1\,T}$ and $d\varepsilon/d\alpha$ it was found that to a very close approximation:

$$\frac{a_2}{a} = -\frac{d\alpha}{d\xi}\,.$$

Flight measurements of α against ξ at a constant speed of 150 knots A.S.I. are plotted in Fig. 19. It can be seen that aileron effectiveness a_2/a is constant at $0 \cdot 1$ for up-going ailerons but falls markedly for down-going ailerons.

9.3. *Pitching Moment Due to Ailerons.*—This is found by measuring the amount of elevator required to trim out a given amount of gust aileron deflection at constant speed.

The pitching-moment coefficient due to ailerons (based on tail-arm l) is given by:

$$m_\xi = \frac{S_T}{S} a_{2\,T} \frac{d\eta}{d\xi} .$$

A curve of elevator angle to trim against mean aileron angle at a constant speed of 150 knots A.S.I. for a c.g. position at $0 \cdot 26\bar{c}$ is given in Fig. 20. It will be seen from this that, in common with aileron effectiveness, pitching moment due to ailerons is constant for up-going ailerons but falls considerably for down-going ailerons.

9.4. *Upwash at Pitot Pitchmeter.*—A series of level runs at varying speeds was made and the change in aircraft attitude compared with the change in incidence at the pitchmeter. The incidence change was found by noting the angular change of the pitchmeter required to maintain zero differential pressure between pitots.

Over the lift coefficient range $0 \cdot 2$ to $0 \cdot 9$ the upwash factor $- d\varepsilon/d\alpha$ was between $0 \cdot 08$ and $0 \cdot 10$.

9.5. *Pressure-incidence Slope of Pitot Pitchmeter.*—The aircraft was flown at a constant speed of 150 knots A.S.I. and with the alleviation off (in order to maintain constant wing incidence) the pitchmeter head was rotated in stages, the head setting and pitot differential pressure being measured at each step.

The results are plotted in Fig. 21, and from this curve it will be seen that $dP/d\alpha$ is constant over the operating range of the pitchmeter.

9.6. *Static Alleviation Factor.*—The static alleviation factor for the system, defined as the static fractional reduction in wing lift slope is given by:

$$-\frac{a_2}{a}\frac{d\xi}{dP}\frac{dP}{d\alpha}$$

where P is the pitot differential pressure (upper pitot minus lower pitot)

α is the pitchmeter head elevation (positive for a nose-up position).

Now the terms (a_2/a) and $(dP/d\alpha)$, which are aerodynamic in nature, have already been measured in flight. The term $(d\xi/dP)$, which is the static sensitivity of the servo must also be measured if the static alleviation is to be determined.

Extensive ground measurements of $(d\xi/dP)$ were made at various servo gain settings, but frequent flight checks were also made. This was done by moving the pitchmeter head in stages and recording pressure and aileron angles at each step.

10. *Discussion.*—The statistical investigation described in section 5 shows that the gust alleviator does not produce the gust alleviation expected. There are several contributory causes which are briefly discussed below. In each case the loss of alleviation is a function of gust shape and distribution. Very little is known about the true gust shapes and distributions and at this stage it is difficult to estimate numerically the contribution of each cause to the loss of alleviation. Experimental studies of the gust alleviator are difficult due to randomness of the encountered gusts. This problem could be considerably simplified if a 'standard gust' of known strength and shape were available. Some limited search for such a natural 'gust funnel' has been made

without success but it is still hoped that one can be found. The existence of a gust funnel would be very useful not only for the study of gust alleviators, but for the study of gust loads on aircraft with swept and delta wings.

10.1. *The Effect of Pitching.*—The gust alleviator, in its present form, has a large destabilizing effect on the aircraft and the theoretical investigation (Appendix I) shows loss of both static and manoeuvre margins. The loss of stability is proportional to the static alleviation. For the position of c.g. used in the theoretical calculations ($0 \cdot 26\bar{c}$) the static margin becomes zero at 33 per cent alleviation and manoeuvre margin at 47 per cent alleviation. It is known that the decrease of stability increases the gust loads (Refs. 7, 8 and 9) and makes the aircraft handling more difficult. An attempt was made to improve this matter by incorporating a filter into the circuit of the electronic link to cut down the gust alleviator response to low frequency disturbances. Unfortunately, with the existing electronic link design, no very effective filter could be designed and the best that could be obtained, was to cut down the static response by about 50 per cent. The frequency response measurements have shown however that the overall effectiveness of the filter was even less than initially expected.

Experimental evidence appears to substantiate the above argument on the effect of aircraft stability on gust loads. The gust alleviator is more effective for small values of static alleviation (Fig. 13). With static alleviation of 12 per cent the actual mean alleviation is positive and the average value appears to be about 5 per cent; for larger values of static alleviation, the measured alleviation instead of increasing is actually decreasing and for a static alleviation of 25 per cent is negative. It must be remembered that Fig. 13 is based on all the accelerometer counts and the positive mean alleviation does not necessarily imply that large loads would be decreased in proportion (*see* Figs. 8 to 12). This phenomenon was more widely discussed in section 5, but unfortunately no explanation can be offered in the present state of our knowledge.

Some rough calculations of the gust alleviation factor (Appendix II) for the *Lancaster* aircraft with gust alleviator were made. Wagner effect was neglected and the aircraft was assumed rigid. The main object of these calculations was to find the effect of pitching motion on the gust alleviator effectiveness and the neglect of the Wagner effect has therefore only a secondary influence. It was found that full alleviation could be obtained only for a sharp-edge gust, where there is no time for pitching motion to develop. For longer gusts the gust alleviator effectiveness decreases rapidly and for gust length of 29 chords no gust alleviation can be produced, irrespective of the static setting of the gust alleviator. For gusts longer than 29 chords the action of the gust alleviator increases the load due to the encountered gust. This short analysis points to the importance of pitching motion effect on gust loads. Very little is known about gust structure, but if the aircraft encountered many long gusts during the statistical investigation, it is clear that the effectiveness of the gust alleviator would be negligible, if not negative. It must be remembered that the flight records show rather larger rates of pitch than those predicted by theory. In this case the loss of gust alleviator effectiveness with increasing gust length would be even more rapid.

10.2. *The Effect of Wing Flexibility.*—It was shown in section 6 that due to wing flexibility the growth of the c.g. acceleration due to deflection of ailerons is considerably delayed. This delay is more noticeable for short gusts than for the long ones. For a gust length of two chords (Fig. 14b) the wing oscillation induced by the gust alleviator was almost in phase with the oscillation induced by the gust itself. Only the first peak of the oscillating gust load was slightly decreased, all the subsequent peaks being actually increased by the action of the gust alleviator. It may be deduced that wing flexibility decreases the gust alleviator effectiveness at least for short gusts and possibly, with appropriate gust shape, can even reverse its action.

It should be noted that the above argument is based on c.g. acceleration measurements and does not necessarily hold for the alleviation of the root bending moment.

The delay in the acceleration growth due to flexibility could be reduced by introducing more advance in the aileron response, but the practicability of this suggestion is rather doubtful. It is very difficult to be more definite on this problem, without a better knowledge of gust shape and a careful analysis of the gust alleviator loads on the flexible aircraft. It appears however that this difficulty of alleviating short gusts would be common to all systems using alleviating devices situated at the outboard part of the wing.

10.3. *Undamped Wing Oscillation with Gust Alleviator On.*—The gust alleviator is a servo with positive feed-back which becomes unstable for large values of internal gain (stiffness of servo). The gust alleviator was set on the ground so that the internal gain of the electronic link was 6 decibels below the instability boundary. This setting was recommended by the manufacturers and with it the system had a ' dead-beat ' response without any signs of oscillation. In flight however the system showed signs of instability. Following a sudden gust, the wing developed an oscillation, which sometimes persisted for a very long time. On some occasions these oscillations were felt to a pronounced extent in the cockpit and to a greater extent in the nose of aircraft. They were approximately of wing fundamental frequency and of quite alarming amplitude. In order to reduce this oscillation it was necessary to decrease the internal gain of the servo by at least a further 3 decibels. It is thought that this phenomenon was increasing the number of accelerations counted with gust alleviator on and the impression was certainly obtained that the gust alleviator increased the roughness of flight.

This effect, as already discussed in section 6, can be partly explained by the lag in the aileron response for sharp gusts which results in a wing oscillation of larger amplitude than that without gust alleviator. This effect of wing flexibility may increase the initial amplitude of wing oscillation, but should not affect wing damping. The oscillations experienced in flight however were very persistent and on some occasions showed definite lack of damping.

The explanation of this instability could probably be found in a coupling between the fundamental bending mode of the aircraft and the gust alleviator. In the mode of this oscillation the displacements and velocities of the wing tips and fuselage nose are believed to be 180 deg out of phase, *i.e.*, when wing tips are moving up, the nose moves down*. The downward movement of the nose gives a gust detector signal corresponding to an up-gust and produces upward deflection of the ailerons proportional to the downward velocity of the nose. If no time lag existed in the gust alleviator system this would give a lift on the outboard part of the wing proportional to the instantaneous wing-tip velocity in bending and the sense to increase wing damping. There is however a time lag between the input pressure at the detector head and the aileron deflection. The frequency of the first bending mode was found to be of the order of 3 c.p.s., *i.e.*, $\frac{1}{6}$ sec time lag is necessary to bring the aileron lift 180 deg out of phase and produce negative damping. The actual time lag of the gust alleviator was $0 \cdot 19$ sec with single-stage phase advance and $0 \cdot 125$ sec with two-stage phase advance at the frequency of 3 c.p.s. The oscillations were actually more noticeable with the single-stage phase advance circuit where the time lag of $0 \cdot 19$ sec agrees very well indeed with time lag of $\frac{1}{6} = 0 \cdot 167$ sec necessary to induce negative damping of the fundamental bending mode.

In order to check the above qualitative explanation of wing oscillation, the aircraft was flown with the pressure pipes leading to the gust detector pick-up reversed. This of course reversed the action of the gust alleviator, but might be expected to increase wing damping. The test fully confirmed the above argument and it was possible to increase the internal gain of the electronic link until instability of the gust servo was obtained. In this condition the ailerons oscillated at the gust servo frequency (higher than the wing frequency) but it was not possible to induce coupling between wing bending oscillation and gust alleviator oscillation. The investigation was limited to qualitative observations and no systematic measurements of the wing bending damping were taken.

* This is actually in disagreement with ground resonance test (Ref. 6) and the mode suggested here is rather that of the *Lincoln* aircraft. However, the aircraft used in the present test differs appreciably from the standard *Lancaster* regarding the mass distribution along the fuselage axis, and the shape of fuselage flexing in the fundamental mode will differ appreciably from the shape established by the ground resonance test.

10.4. *The Effect of Non-uniform Spanwise Gust Distribution.*—It has been suggested that the present gust alleviator system using one central detector head may not be fully effective in a non-uniform gust distribution. The argument is, that if the aircraft meets an up-gust at its wing tips and simultaneously a down-gust at its centre-line, the action of the ailerons would increase the gust load on the outboard part of the wing. To a first approximation however this phenomenon would only affect the bending moment of the outer wing. The bending moment at the root and the c.g. acceleration in a gust of this shape are small and even with the load due to negative alleviation would be smaller than in a uniform gust. It is felt that the problem of non-uniform spanwise gust distribution does not enter into the present investigation and the lack of expected alleviation cannot be attributed to it. This was to a certain extent verified experimentally by the test with reversed action of the gust alleviator. The results plotted in Fig. 12 show negative alleviation of the expected magnitude, which seems to indicate the absence of the effect of non-uniform spanwise gust distribution.

10.5. *Application to High-speed Aircraft.*—The analysis of the aircraft and gust servo response to a short gust made in section 6, shows that the main contributions to the time lag of the gust alleviator are from the electronic link (0·125 sec) and from lag between the detector head and pressure pickup (0·03 sec). The lag of hydraulic servos was only 0·02 sec and the aerodynamic lag of ailerons was negligibly small. The application of the present alleviator to a fast aircraft flying at say 600 knots would require a system with a time lag of about a quarter of the present value, *i.e.*, 0·04 sec. This estimate based on speed alone tends to be pessimistic for modern types of aircraft where the wing loading may be expected to be considerably greater than on the *Lancaster*. Increase of wing loading has the effect of increasing the time taken for the gust load to build up to a maximum. An example of this effect is shown in Fig. 22, obtained from an unpublished part of Ref. 5. If we assume a gust load lag of only two chords, this would increase the permissible gust servo lag by about 0·02 sec (assuming a chord of 10 ft), the total lag then being 0·06 sec.

There is no basic difficulty in placing the pressure pick-up directly behind the pressure head, thus eliminating entirely the pipe lag. If the existing hydraulic system is retained, then only the electronic link would have to be redesigned to reduce the time lag from 0·125 sec to 0·04 sec.

The above very rough analysis, indicates that there is no fundamental difficulty in the application of the present gust alleviating system to a high-speed aircraft.

10.6. *Hydraulic Servos.*—It may be of general interest to put on record the satisfactory behaviour of the hydraulic servos. These servos supply the power to operate the ailerons and form a part of the gust alleviator system.

11. *Conclusions.*—(i) The gust alleviator in the present form does not produce the expected alleviation of the c.g. accelerations due to gust

(ii) It is thought that the gust alleviator gives more relief in the wing-root bending moment but due to faulty behaviour of the electronic link of the gust alleviator this cannot at present be established with any degree of certainty

(iii) The gust alleviator response to the simulated gust agrees fairly well with theoretical prediction and would probably give the required alleviation if no pitching motion of aircraft occurred and the wings were sensibly rigid

(iv) Theoretical analysis shows that the effectiveness of the gust alleviator decreases with increasing gust length and for very long gusts the action of the gust alleviator is reversed

(v) Analysis of flight records shows that wing flexibility decreases gust alleviator effectiveness in alleviating the c.g. accelerations particularly for short gusts. The alleviation of wing-root bending moment is apparently not affected by wing flexibility

(vi) A coupling between the aircraft fundamental mode and the gust alleviator may decrease the damping of this mode. It is clear that the flexibility of aircraft structure has a great effect upon the loads produced by the gust alleviator

(vii) It is felt that from the point of view of crew and passenger comfort the alleviation of the aircraft pitching motion as well as the accelerations due to gusts are equally important and it is interesting to note that during the flight with reversed action of the gust alleviator the general impression was of a much smoother ride, in spite of the accelerometer records showing the contrary

(viii) A gust alleviator, similar to the gust alleviator on the *Lancaster* aircraft is being tested in America. A C–47 aircraft is equipped with a Douglas alleviator, where the wing bending deflection operates the ailerons. No reports on these tests are yet available, but information received from America quotes 8 to 9 per cent alleviation in root bending moment but only 1 per cent alleviation on c.g. acceleration

(ix) The assumption of no aerodynamic lag in the lift of the trailing-edge control surface was confirmed by experiment, within the limits of experimental error

(x) There is no fundamental difficulty in the application of the present gust alleviating system to a high-speed aircraft

(xi) *The final conclusion* is, that the gust alleviating system using movable ailerons can produce positive and reliable alleviation only if the trim changes due to aileron deflection are much smaller, or perhaps even better are of opposite sign to those now experienced. Even then, some loss of alleviation is to be expected due to airframe flexibility, but this, it is believed, is of secondary importance.

12. *Further Work on Gust Alleviation*.—Full analysis of all flight records, which is not yet completed, will give a better understanding of the gust alleviator behaviour, particularly regarding wing-root bending stresses. The present analysis has shown however that the loss of gust alleviator effectiveness is mainly due to the large pitching response of the aircraft. The problem of the effect of aircraft pitching on the gust loads with and without the gust alleviator and on the comfort of passengers appears important enough to warrant further theoretical and experimental study. The present aircraft, *Lancaster* ME.540, is well suited for further development of the existing gust alleviator and it is proposed to incorporate into the elevator circuit a servo, similar to the servo used in the aileron circuit to which the gust signal will be fed simultaneously with that to the ailerons. The amplitude and phase of elevator deflection will be variable to facilitate study of the effect of pitching motion. As the maximum elevator deflection to counteract the aileron pitching moment is only of the order of 3 deg for full aileron travel the proposal appears to present no serious difficulties. An aircraft equipped with such a gust alleviator will provide a very useful tool for the experimental investigation of the effects of the two independent variables *i.e.*, lift slope and pitching moment, on the gust loads. It is now felt that in continuous turbulent air the pitching response of the aircraft has a much larger effect than was hitherto believed*.

The experimental investigation proposed would not only supply information for the best combination of the parameters for the ideal gust alleviator, but would also enable a basic study to be made of aircraft behaviour in turbulent air for a range of aircraft stability parameters.

It should be noted that the present type of gust alleviator will be ineffective on swept-back wing aircraft, as on such aircraft the pitching moment due to ailerons is very large†. Tests where the effects of changes in lift and pitching moment due to a gust can be investigated separately should provide information on the desirable characteristics of the required lift and pitching moment control. When this primary design information is available an investigation of the best form of aerodynamic control will clearly be required and it is suggested that the use of spoilers will merit serious consideration. For swept wings it may be necessary to sacrifice a proportion of alleviation by the use of inboard controls.

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* A new approach to gust load analysis, proposed by Press and Mazelsky[11] used with a measured gust spectrum, also seems to indicate the important effect of the aircraft pitching response on the gust loads in continuous turbulent air.

† Some unpublished investigations on a 35-deg swept-back fighter aircraft show that the pitching moment due to ailerons $\partial C_m/\partial \xi$ is roughly equal to the pitching moment due to elevator $\partial C_m/\partial \eta$.

LIST OF SYMBOLS

a		Wing lift slope
a_2		Lift slope due to ailerons, based on wing area
a_{1T}		Tail lift slope
a_{2T}		Tail lift slope due to elevator
B		Coefficient of stability equation (damping)
b	=	$\frac{a_2}{2}\frac{\lambda}{\mu}k$, Auxiliary coefficient
C		Coefficient of stability equation (stiffness)
$\bar{c}$		Mean wing chord
D		Denotes differentiation with respect to aerodynamic time
g		Acceleration of gravity
H		Gust length in chords
H_n		Static margin, stick fixed
H_m		Manoeuvre margin, stick fixed
h_T		Tailplane contribution to static margin of aircraft
i_B	=	$(k_B/l)^2$, Pitching moment of inertia coefficient
K	=	$\frac{\text{max. accel.} \times W/g}{\frac{1}{2}\rho USau_g}$, Gust alleviation factor
k		Gearing coefficient between gust input and aileron deflection
k_B		Radius of gyration, in pitching
l		Tail length
l_1		Distance between gust detector and c.g. of aircraft
M_ξ		Pitching moment due to ailerons, per radian
m_w		Pitching-moment coefficient due to vertical velocity
$m_{\dot{w}}$		Pitching-moment coefficient due to downwash delay
m_q		Pitching-moment coefficient due to pitching velocity
m_ξ		Pitching-moment coefficient due to aileron deflection
P		Pressure difference measured by gust detector head
p		Operator in Laplace transform
q		Pitching velocity
S		Wing area
S_T		Tail area

s		Dimensionless distance, in chords
$\hat{t}$	$=$	$\mu l/U$, Unit of aerodynamic time
U		Forward velocity of aircraft
u		Gust velocity
u_g		Maximum gust velocity
W		Weight of aircraft
w		Velocity increment along z-axis
Z_ξ		Lift due to aileron deflection, per radian
z_w		Force coefficient due to vertical velocity
z_q		Force coefficient due to pitching velocity
α		Wing incidence
ε		Downwash angle (measured at tail)
η		Elevator angle
λ	$=$	l_1/l, Dimensionless distance between gust detector and aircraft c.g.
μ	$=$	$\dfrac{W}{g\rho Sl}$, Mass parameter
μ_g	$=$	$\dfrac{2W}{g\rho S\bar{c}a} = \dfrac{2}{a}\dfrac{l}{\bar{c}}\mu$, Gust mass parameter
ν	$=$	$-m_q/i_B$, Rotary damping coefficient
ξ		Mean angle of ailerons
ρ		Air density
τ		Aerodynamic time
τ_s		Time constant of gust servo (in aerodynamic time units)
χ	$=$	$-\dfrac{\mu m_{\dot{w}}}{i_B}$, Downwash damping coefficient
ω	$=$	$-\dfrac{\mu m_w}{i_B}$, Static stability coefficient

The *dashed symbols* denote a given quantity as affected by the gust alleviator.

The Laplace transform of a given function is denoted by a *bar above* this function.

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APPENDIX I

The Effect of Gust Alleviator on Aircraft Stability

The change of airflow incidence α at the detector head produces an aileron deflection $-\xi$. The servo-mechanism operating the ailerons has a time lag, and it is assumed that the total time lag between receiving signal at the detector head and the deflection of ailerons is τ_s in units of aerodynamic time. From frequency response measurements of the gust servo (Fig. 5) it was found that this time lag is constant for a wide range of frequencies, and in the present analysis it is assumed that $\tau_s =$ constant, and the gust servo is a system with time lag only and its transfer function has a form:

$$\bar{\xi} = -\bar{\alpha}k\frac{1}{\tau_s p + 1}. \qquad (1)$$

The coefficient k is a constant describing a gearing between the input α at the gust detector head and the output of the aileron angle ξ. The numerical value of k depends on the amount of alleviation required. The Laplace transform of a given function is denoted by a bar above this function.

The change of incidence α at the detector head is due to a vertical velocity of aircraft w, and the pitching velocity q, and is given by:

$$\alpha = \frac{w}{U} - \frac{ql_1}{U} \qquad (2)$$

where U is the forward velocity of the aircraft, assumed to be constant, and l_1 is the distance between the head and aircraft c.g. (Fig. 23).

The aileron deflection ξ produces a force and a moment which can be written:

$$\left.\begin{aligned} \xi.Z_\xi &= +\tfrac{1}{2}\rho U^2 S a_2 . \xi \\ \xi.M_\xi &= -\tfrac{1}{2}\rho U^2 S l m_\xi . \xi \end{aligned}\right\} \qquad (3)$$

where a_2 is aileron lift slope based on total wing area S, and m_ξ is a pitching-moment coefficient based on wing area S and tail length l. The values of both coefficients were obtained experimentally.

Substituting equations (1) and (2) into (3) the lift and moment due to aileron are obtained in operational form:

$$\left.\begin{aligned} \overline{\xi.Z_\xi} &= -\tfrac{1}{2}\rho U^2 S a_2\left(\frac{\bar{w}}{U} - \frac{\bar{q}l_1}{U}\right)\frac{k}{\tau_s p + 1} \\ \overline{\xi.M_\xi} &= \tfrac{1}{2}\rho U^2 S l m_\xi\left(\frac{\bar{w}}{U} - \frac{\bar{q}l_1}{U}\right)\frac{k}{\tau_s p + 1} \end{aligned}\right\}. \qquad (4)$$

The standard equations of motion of the aircraft in the pitching plane, neglecting changes in forward velocity, and written in dimensionless form are:

$$\left.\begin{aligned} (D - z_w)\frac{w}{U} - \left(1 + \frac{z_q}{\mu}\right)q\hat{t} &= 0 \\ -\left(\frac{\mu m_{\dot{w}}}{i_B}D + \frac{\mu m_w}{i_B}\right)\frac{w}{U} + \left(D - \frac{m_q}{i_B}\right)q\hat{t} &= 0 \end{aligned}\right\}. \qquad (5)$$

The equations of motion including the gust alleviator are obtained by adding the forces and moments due to alleviator in dimensionless form and derived from equation (3) to the forces and moments of equation (5). The equations of motion with gust alleviator written in operational form and introducing the following symbols:

$$\omega = -\frac{\mu m_w}{i_B} \quad \text{static-stability coefficient}$$

$$\nu = -\frac{m_q}{i_B} \quad \text{rotary-damping coefficient}$$

$$\chi = -\frac{\mu m_{\dot{w}}}{i_B} \quad \text{downwash-damping coefficient}$$

are as follows:

$$\left.\begin{aligned} \underbrace{(p - z_w)\frac{\bar{w}}{U} - \left(1 + \frac{z_q}{\mu}\right)\bar{q}\hat{t}}_{\text{aircraft}} \quad & \underbrace{-\frac{a_2}{2}\left(\frac{\bar{w}}{U} - \frac{\bar{q}l_1}{U}\right)\frac{k}{\tau_s p + 1}}_{} = 0 \\ (\chi p + \omega)\frac{\bar{w}}{U} + (p + \nu)\bar{q}\hat{t} \quad & +\frac{m_\xi}{2}\frac{\mu}{i_B}\left(\frac{\bar{w}}{U} - \frac{\bar{q}l_1}{U}\right)\frac{k}{\tau_s p + 1} = 0 \end{aligned}\right\} \quad (6)$$

aircraft — gust alleviator

Let us denote the ratio of the detector head arm l_1 to the tail-arm l by $\lambda = l_1/l$. Remembering that the unit of the aerodynamic time $\hat{t} = \mu l/U$, by solving the equation (6) the characteristic equation is obtained. The characteristic equation can be written in the form:

$$\tau_s p^3 + (1 + \tau_s B)p^2 + (B' + \tau_s C)p + C' = 0 \quad (7)$$

where B and C are the well-known constants of the stability equation neglecting the changes in forward speed, and are given by:

$$\left.\begin{aligned} B &= -z_w + \nu + \left(1 + \frac{z_q}{\mu}\right)\chi \\ C &= -z_w \nu + \left(1 + \frac{z_q}{\mu}\right)\omega \end{aligned}\right\} \quad (8)$$

The dashed coefficients B' and C' are coefficients similar to B and C but modified by the presence of the gust alleviator. The formulae for these coefficients are as follows:

$$\left.\begin{aligned} B' &= B - \left[\frac{m_\xi}{2}\frac{\lambda}{i_B} + \frac{a_2}{2}\left(1 + \frac{\lambda}{\mu}\chi\right)\right]k \\ C' &= C + \left[\left\{z_w\frac{\lambda}{i_B} + \frac{\mu}{i_B}\left(1 + \frac{z_q}{\mu}\right)\right\}\frac{m_\xi}{2} - \left(\nu + \frac{\lambda}{\mu}\omega\right)\frac{a_2}{2}\right]k \end{aligned}\right\} \quad (9)$$

The amount of the static gust alleviation is defined as the decrease in the value of the effective lift slope due to gust alleviator. It can be shown from equation (6) that the effective value of the derivative z_w denoted z_w' is given by:

$$z_w' = z_w + \frac{a_2}{2}k \quad (10)$$

and by putting $z_w = - a/2$ the relationship between coefficient k and amount of static alleviation can be established:

$$z_w' = -\frac{a}{2}\left(1 - \frac{a_2}{a}k\right) = z_w\left(1 - \frac{a_2}{a}k\right) \quad \ldots \quad (11)$$

where the expression $(a_2/a)k$ is the amount of static alleviation. The value of a_2/a was measured experimentally.

Discussion.—The present discussion is limited to the stick-fixed conditions, but the flight measurements on a *Lancaster* aircraft (unpublished) show no appreciable difference between stick-fixed and stick-free static margins.

Assuming that time lag of the gust servo is small, the term with τ_s in the equation (8) can be omitted, and a simplified stability equation with the gust alleviator is obtained:

$$p^2 + B'p + C' = 0 . \quad \ldots \quad (12)$$

The value of the coefficient C' which is proportional to the manoeuvre margin is decreasing with the increase of the gust alleviation. Using relationship:

$$C' = \frac{a}{2}\frac{\mu}{i_B}\frac{\bar{c}}{l} H_m . \quad \ldots \quad (13)$$

The value of manoeuvre H_m can be calculated for the range of alleviation. The aircraft and alleviator data used in the calculations are given in Table A and the manoeuvre margin plotted against gust alleviation as a percentage is shown in Fig. 23. In the same figure the loss of static margin is shown. The static margin was calculated from the following formulae:

$$\left.\begin{aligned} m_w' &= -\frac{a}{2}\frac{\bar{c}}{l} H_n \\ m_w' &= m_w - \tfrac{1}{2} m_\xi k \end{aligned}\right\} \quad \ldots \quad (14)$$

where the pitching derivative m_w' denotes this derivative as affected by the gust alleviator.

It can be seen that the increase of the gust alleviation decreases both static and manoeuvre margins and for about 33 per cent alleviation the static margin becomes zero, and further increase of alleviation to about 47 per cent reduces to zero the manoeuvre margin.

When the time lag of the gust servo became available as the results of frequency response measurements, it was thought advisable to repeat the stability calculations taking this time lag into account, using the full stability equation (7). The time lag was found to be of the order of $0 \cdot 2$ sec, which in aerodynamic time units gave $\tau_s = 0 \cdot 1$.

The results of calculations are given in Fig. 24 where the real parts of the stability roots are plotted against the alleviation in percentage. Two sets of curves are shown, broken curves for stability roots without servo lag, and continuous curves for stability roots with servo lag taken into account.

It can be seen, that the servo lag has a small effect on the aircraft stability, and the aircraft becomes dynamically unstable for alleviation in excess of 47 per cent., *i.e.*, when the manoeuvre margin becomes negative.

The interesting effect of the servo lag, is the coupling between the servo and aircraft resulting in a new mode of oscillations for alleviation larger than 27 per cent.

TABLE A

Data Used in the Calculations of Stability with the Gust Alleviator—Stick Fixed (Lancaster ME.540)

$W =$	49,000 lb	Aircraft weight
$S =$	1,300 sq ft	Wing area
$S_T =$	237 sq ft	Tail area
$\bar{c} =$	$12 \cdot 7$ ft	Wing mean chord
$l =$	$37 \cdot 4$ ft	Tail arm
$l_1 =$	$31 \cdot 0$ ft	Gust detector arm
c.g.	at $0 \cdot 26\bar{c}$	
$H_n =$	$0 \cdot 105$	
$i_B =$	$0 \cdot 125$	Moment of inertia coefficient
$\lambda = l_1/l =$	$0 \cdot 83$	
$\mu = \dfrac{W}{g\rho Sl} =$	$13 \cdot 3$ ($\mu_g = 16 \cdot 3$) ..	Mass parameter
$U =$	150 knots	Forward speed
$\hat{t} = \dfrac{\mu l}{U} =$	$1 \cdot 96$ sec	
$a =$	$4 \cdot 8$	Wing lift slope
$a_{1T} =$	$3 \cdot 2$	Tail lift slope (flight measured)
$a_{2T} =$	$1 \cdot 45$	Elevator lift slope (metal elevators measured)
$\dfrac{a_2}{a} = \dfrac{\text{effective aileron lift slope}}{\text{wing lift slope}} = 0 \cdot 1$		Measured in flight
$\dfrac{d\xi}{d\eta} =$	-5	Measured in flight; to assess the pitching-moment of ailerons
$m_\xi = \dfrac{S_T}{S} a_{2T} \dfrac{d\eta}{d\xi} = -0 \cdot 053$	.. ..	Pitching-moment coefficient of ailerons

Stability derivatives calculated from above data

$z_w = -2 \cdot 4$	$z_q = 0$
$m_w = -0 \cdot 0855$	$\omega = 9 \cdot 1$
$m_q = -0 \cdot 291$	$\nu = 2 \cdot 33$
$m_{\dot{w}} = -0 \cdot 011$	$\chi = 1 \cdot 17$

APPENDIX II

The Effect of Pitching Moment of Ailerons on the Gust Alleviator Effectiveness

Theory.—The deflection of ailerons due to gust alleviator action produces pitching moment in a sense increasing the gust load. This decreases the gust alleviator effectiveness, especially for longer gusts, when there is enough time for the aircraft to develop large changes in pitch. The object of the present calculations is to estimate the loss of gust alleviator effectiveness due to pitching motion. The following simplifying assumptions have been made:

(*a*) The aircraft is rigid

(*b*) The stick is held fixed throughout the manoeuvre. (The results of calculations made under this assumption are directly applicable to stick-free conditions, as for this particular aircraft there is practically no change in stability on freeing the stick)

(*c*) The unsteady lift functions are assumed to be constant and equal to unity. This assumption gives for short gusts an appreciable error in the estimation of the absolute values of the gust alleviation factor. In the present analysis we are interested only in the relative changes in the value of gust alleviation factor due to the gust alleviator action and from this standpoint the Wagner effect can be neglected. In the numerical calculations, the order of the error due to omission of Wagner effect is estimated, and is shown in Fig. 26

(*d*) The aerodynamic force and moment due to gust are split up into two parts: firstly, force and moment acting on the aircraft less tail and secondly, force acting on the tailplane. The lift of the aircraft less tail is acting at the aerodynamic centre of aircraft less tail, which is assumed to coincide with c.g. position $(0 \cdot 26\bar{c})$. The lift of the tail acts at the quarter-chord point of the tailplane

(*e*) Before entering the gust the aircraft flies straight and level. The forward speed is constant throughout the manoeuvre. The vertical gust velocity is uniform along the span of the wing at any instant

(*f*) The encountered gust is a flat-topped gust of a length H mean chords and a maximum velocity u_g ft/sec

(*g*) The lag in the gust servo and its effect on the aircraft *free* motion are neglected (*see* Appendix I)

(*h*) It is assumed that the gust alleviator setting is ideal, *i.e.*, the ailerons deflect simultaneously with gust hitting the wing.

The gust shape is defined by equation (Fig. 25)

$$\left.\begin{aligned} u &= u_g \frac{s}{H} \quad \text{for} \quad 0 \leqslant s \leqslant H \\ \text{and} \qquad u &= u_g \qquad \text{for} \quad s \geqslant H \end{aligned}\right\} \quad \ldots \quad \ldots \quad (\text{II}.1)$$

where u = gust velocity at any instant, in ft/sec

u_g = maximum gust velocity, in ft/sec

s = distance travelled by the aircraft, in mean chords $\bar{c}$

H = gust length, in mean chords.

The aircraft response due to gust is calculated as a sum of the response due to gust force and moment acting on the aircraft less tail and the response due to gust force action on the tail.

Aircraft Response Due to Gust Hitting the Wing.—It is assumed that lift is produced by the wing only and that the centre of pressure coincides with the c.g.

Wing lift due to vertical up-gust:

$$Z_g = -\tfrac{1}{2}\rho U u_g \frac{\mu l}{H\bar{c}} S a \tau \quad \ldots \quad (\text{II}.2)$$

where τ is the aerodynamic time, and the gust shape equation (II.1) can be expressed as a function of τ:

$$u = u_g \frac{s}{H} = u_g \frac{\mu l}{H\bar{c}} \tau \,. \quad \ldots \quad (\text{II}.3)$$

Lift due to deflection of ailerons (*see* Appendix I):

$$\xi \,.\, Z_\xi = +\tfrac{1}{2}\rho U u_g \frac{\mu l}{H\bar{c}} S a_2 k \tau \,. \quad \ldots \quad (\text{II}.4)$$

Moment due to deflection of ailerons (*see* Appendix I):

$$\xi \,.\, M_\xi = -\tfrac{1}{2}\rho U u_g \frac{\mu l}{H\bar{c}} S l m_\xi k \tau \,. \quad \ldots \quad (\text{II}.5)$$

The expressions given by (II.2), (II.4) and (II.5) are the forcing functions acting on the aircraft due to gust, free motion of the aircraft being given by the equations of motion developed in Appendix I, equation (6).

The full equations of motion of the aircraft due to a linearly increasing gust hitting the wing alone, are written in dimensionless form (term z_q is neglected):

$$\left.\begin{aligned}
&\overbrace{\underbrace{\left(D+\frac{a}{2}\right)\frac{w}{U} - q\hat{t}}_{\text{basic aircraft}} \underbrace{- \tfrac{1}{2}ka_2\left(\frac{w}{U}-\frac{ql_1}{U}\right)}_{\text{gust alleviator}}}^{\text{aircraft free motion}} &&= \overbrace{\tfrac{1}{2}(\underbrace{a}_{\text{wing}} \underbrace{- ka_2)}_{\text{gust alleviator}} \frac{u_g}{U}\frac{\mu l}{H\bar{c}}\tau}^{\text{gust forcing function}} \\
&(\chi D+\omega)\frac{w}{U} + (D+\nu)q\hat{t} + \tfrac{1}{2}km_\xi\frac{\mu}{i_B}\left(\frac{w}{U}-\frac{ql_1}{U}\right) &&= \tfrac{1}{2}km_\xi\frac{\mu}{i_B}\frac{u_g}{U}\frac{\mu l}{H\bar{c}}\tau
\end{aligned}\right\} \quad \ldots \quad (\text{II}.6)$$

Let us denote by dashed symbols the stability derivatives as affected by the gust alleviator.

$a' = a\left(1 - \frac{a_2}{a}k\right)$, wing lift slope with gust alleviator.

The term $\frac{a_2}{a}k$ is referred to as *the static alleviation*

$\omega' = \omega + \Delta\omega = \omega + \frac{m_\xi}{2}\frac{\mu}{i_B}k$ static stability coefficient

$\nu' = \nu - \frac{m_\xi}{2}\frac{\lambda}{i_B}k$ rotary damping coefficient

χ downwash damping coefficient unaffected by the gust alleviator.

In addition let us put:

$$b = \frac{a_2}{2} \frac{\lambda}{\mu} k .$$

Using the above defined coefficients the equations of motion are written in a more concise form:

$$\left.\begin{aligned} \left(D + \frac{a'}{2}\right)\frac{w}{U} - (1 - b)q\hat{t} &= \frac{a'}{2} \frac{u_g}{U} \frac{\mu l}{H\bar{c}} \tau \\ (\chi D + \omega')\frac{w}{U} + (D + \nu')q\hat{t} &= \Delta\,\omega \frac{u_g}{U} \frac{\mu l}{H\bar{c}} \tau \end{aligned}\right\} \quad \ldots \quad \ldots \quad \ldots \quad \text{(II.7)}$$

Solving equations (II.7) the c.g. acceleration and aircraft pitching velocity due to gust acting on the wing alone can be obtained.

Aircraft Response Due to Gust Hitting the Tailplane.—The gust velocity when it reaches the tailplane is modified by the downwash from the wing. Neglecting Wagner effect, the gust velocity at the tailplane is:

$$u\left(1 - \frac{d\varepsilon}{d\alpha}\right)$$

and hence the pitching moment due to gust hitting the tailplane is:

$$M_T = -\tfrac{1}{2}\rho U S_T l a_{1T} u \left(1 - \frac{d\varepsilon}{d\alpha}\right) \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad \text{(II.8)}$$

and the dimensionless pitching-moment coefficient due to tail:

$$m_T = \frac{M_T}{\rho S l U^2} = -\frac{a_{1T}}{2} \frac{S_T}{S} \frac{u}{U}\left(1 - \frac{d\varepsilon}{d\alpha}\right) = -\frac{a}{2} \frac{\bar{c}}{l} h_T \frac{u}{U} \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad \text{(II.9)}$$

where $h_T = \frac{a_{1T}}{a}\left(1 - \frac{d\varepsilon}{d\alpha}\right)\frac{S_T l}{S\bar{c}}$ is the tailplane contribution to the static margin of the aircraft.

The tail lift contribution to the gust load is assumed to be negligibly small.

The equations of motion of the aircraft with gust alleviator and due to gust load on the tailplane can be written in analogy to equations (II.7):

$$\left.\begin{aligned} \left(D + \frac{a'}{2}\right)\frac{w}{U} - (1 - b)q\hat{t} &= 0 \\ (\chi D + \omega')\frac{w}{U} + (D + \nu')q\hat{t} &= \frac{a\mu}{2\,i_B} h_T \frac{\mu}{H} \frac{u_g}{U} \tau \end{aligned}\right\} \quad \ldots \quad \ldots \quad \ldots \quad \text{(II.10)}$$

The solution of equations (II.10) gives the c.g. acceleration and aircraft pitching velocity due to a linearly increasing gust acting on the tailplane alone.

Adding the solutions of equations (II.7) and (II.10) with time displacement equal to time for aircraft to travel the tail length distance, the c.g. acceleration and pitching velocity due to gust acting on the whole aircraft can be obtained. The maximum value of the c.g. acceleration gives the gust alleviation factor K, by the definition (Ref. 5):

$$K = \frac{\text{max. acceleration} \times W/g}{\tfrac{1}{2}\rho U S a u_g} . \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad \text{(II.11)}$$

Numerical Calculations.—The numerical calculations of the gust alleviation factor for *Lancaster* ME.540 aircraft with gust alleviator were made using stability parameters from Appendix I, Table A, and for the range of static alleviation from 0 per cent to 40 per cent. The results are shown in Fig. 25 where gust alleviation factor is plotted against gust length. It can be seen that only for zero gust length, the gust alleviator decreases the value of the gust factor by an amount equal to static alleviation. For gusts longer than a sharp-edge gust, the decrease in the value of alleviation factor is progressively less and for a gust length of about 29 chords, the action of the gust alleviator gives no decrease in the gust load. The effect of the gust length on the loss of the gust alleviator effectiveness is shown in Fig. 26. The effectiveness of the gust alleviator was defined as the ratio of the relative decrease in the value of alleviation factor due to gust alleviator at a given gust length to the corresponding decrease at zero length of gust. The relative decrease in the value of gust alleviation factor for $H = 0$ is of course the static alleviation. Gust alleviator effectiveness is independent of static alleviation and for a given aircraft and gust alleviator is a function of the gust length only. It can be seen from Fig. 26 that the alleviator effectiveness decreases almost linearly with increasing gust length and becomes zero at the gust length of 29 chords. For gusts longer than 29 chords, the action of the gust alleviator will result in gust loads higher than those without gust alleviator.

It must be remembered that due to Wagner effect, neglected in the present analysis, the gust load build-up takes a finite time even for a sharp-edge gust. This particular aircraft has to travel about 3 chords before the gust load reaches its maximum for sharp-edge gust. Due to this effect the gust alleviator effectiveness is less than 100 per cent even for sharp-edge gust. The curve of approximate gust alleviator effectiveness including Wagner effect is shown in Fig. 26 by a dotted line. This curve was roughly estimated taking into account a lag between the peaks of gust velocity and the gust load.

APPENDIX III

Gust Alleviator Lancaster ME.540

Pilot's Handling

1. *Introduction.*—This report gives the pilot's impressions of the handling of *Lancaster* ME.540 with gust alleviator installed, during experimental flights made at Idris in October, 1952.

2. *Conditions Relevant to Test.*—The aircraft was a *Lancaster* Mark 3 No. ME.540.

The c.g. position was approximately $0 \cdot 26c$.

The all-up weight was about 49,000 lb during most of the tests.

The elevator and rudder controls were normal, the ailerons were power operated with spring feel. The ailerons were modified and had cord on the trailing edges. The system was controlled by a 3-position lever on the right of the pilot's seat marked ' Manual—Power—Gust '. A push button on the stick centralised the ailerons instantaneously if they ran away.

The gust alleviator system comprised a detector on a probe on the nose of the aircraft; the impulses from this were amplified and fed to the power controls so that an up-gust detected at the nose produced an upward movement of both ailerons together and *vice versa*. The gain and stiffness of the system were adjustable and the frequency response could be altered by means of a filter. With the filter OUT all frequency down to zero was fed through so that any change of flow at the nose due to gusts, speed or incidence changes produced a corresponding change in aileron position. With the filter IN frequencies below $0 \cdot 5$ cycles/second were cut out so that steady changes of flow did not affect the aileron position.

During periods of counting the alleviator was switched on and off at 30-second intervals by a timing mechanism. The aircraft was instrumentated to count ' *g*'s ' and strains at various positions in the aircraft during the counting periods.

About 42 hours flying were carried out during the three week period.

3. *Tests Carried Out.*—No specific handling tests were carried out. The results given are those obtained during recording and calibrating periods for:

3.1. Ailerons in manual.

3.2. Ailerons in power.

3.3. Ailerons in gust.

3.3.1. General.

3.3.2. Filter OUT.

3.3.3. Filter IN.

3.3.4. Filter OUT, alleviator reversed.

4. *Results of Tests.*—4.1. *Aircraft with Ailerons in ' Manual '.*—The rudder and elevators worked normally and the longitudinal stability was very good. No measurements were made, but the stick force per g was moderate (about 50 lb per g). The change of trim with speed was in a stable sense and fairly large. The aircraft was easy to trim and could be flown for long periods hands off. The ailerons were extremely heavy and at 150 knots less than $\frac{1}{2}$ aileron was obtained at maximum two-handed force (about 80 lb). The aircraft was not flown in this configuration except in steady cruising flight in calm air.

4.2. *Aircraft with Ailerons in 'Power.'*—The ailerons were light and very pleasant; full aileron required about 20-lb force at all speeds (spring feel) and self-centring was good.

4.3. *Ailerons in 'gust'.*—4.3.1. *General.*—The alleviator did not work consistently over long periods. On some occasions the system tended to oscillate and on others gave a low response. There was usually an appreciable change of sensitivity during each counting period (1 to 3 hours).

When both ailerons reached the limit of their travel there was still some control available because the one aileron came away from the stops when the stick was moved.

The push button on the stick cut out the alleviator instantly, leaving the ailerons in power.

At high gain the system tended to oscillate and flap the wings up and down at about 3 per second. This was stopped by pushing the cut-out button.

4.3.2. *Normal operation—filter* OUT.—The response depended upon the gain of the gust amplifier, but in general there was a marked decrease in longitudinal stability. If the stick was pulled back the ailerons went up due to the incidence increase being detected as an up-gust. This produced a nose-up change of trim and a moderate push was needed to lower the nose again. The same thing happened if the speed was decreased; the increased incidence caused the aileron to go up producing a nose up trim change or *vice versa, i.e.*, the aircraft was unstable, stick free. In this condition continuous concentration was needed to keep a steady attitude. The speed and height were not difficult to maintain because the lift slope was flattened and the attitude changes did not produce as much speed and height changes as they otherwise would. The aircraft was flown in bumpy air over the desert for up to three hours at a time, but it was rather tiring.

The alleviator appeared to work satisfactorily except that the trim changes were working in the opposite sense to that required for alleviation of the gust. An up-gust produced an upward aileron movement which tended to alleviate the gust and reduce the *g* but the nose then came up due to the trim change and increased the *g* again.

During periods of 'counting' the alleviator was switched on and off at 30-second intervals and at each change-over there was usually a movement of the ailerons due to creep in the system, the aileron position with the alleviator ON being different to that with it off. This produced a trim change requiring about a 10-lb force at each change-over.

4.3.3. *Filter* IN.—With the filter IN there were no steady changes of trim and no decrease in stability in calm air. The alleviator appeared to work satisfactorily but there was a greater tendency to instability apparent as a magnification of some bumps and a slight increase in wing flapping. The bumps produced some changes of trim which died away in 2 to 3 seconds but still seemed sufficient to counteract most of the initial effect of the aileron movement.

4.3.4. *Gust alleviator action reversed—filter* OUT.—In this condition the aircraft was very pleasant to fly in bumpy conditions and would stay as trimmed with hands off. The aircraft rode the bumps without any need for elevator movement and kept a steady speed very easily. The records showed that there was a 50 per cent increase in *g* due to bumps in this condition, but this was not apparent.

5. *Conclusions.*—The alleviator is unsatisfactory as installed in this aircraft due to:

5.1. The occasional instability in the system and changes in response during flight and from one flight to the next.

5.2. The longitudinal instability with the alleviator on. If the trim change with aileron movement could be counteracted the system would be much more successful. A trim change in the opposite sense is probably desirable.

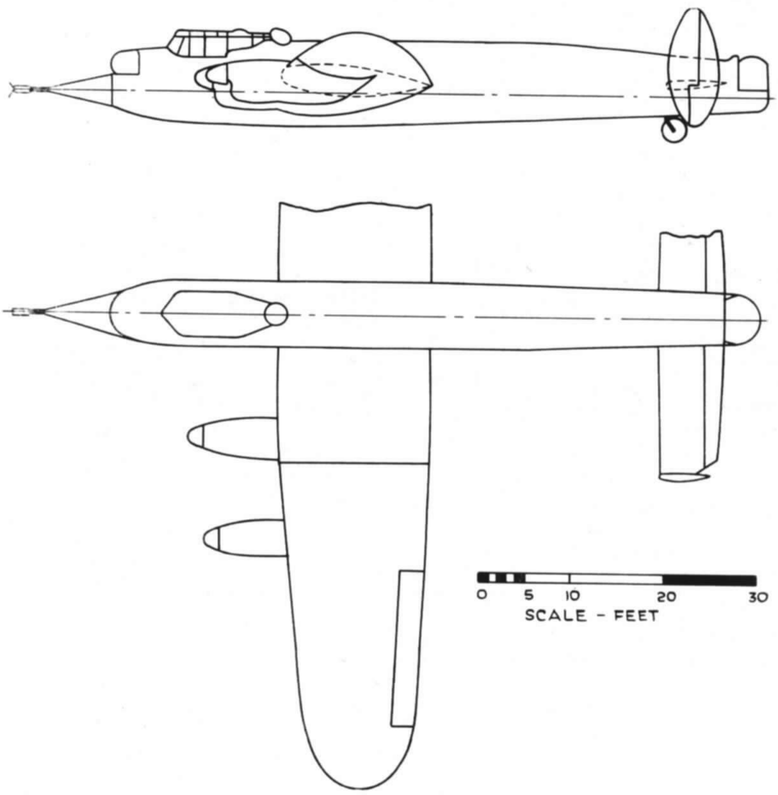


FIG. 1. General arrangement of *Lancaster* ME.540, fitted with gust alleviator.

FIG. 2. Nose of *Lancaster* ME.540 showing the two pitchmeters.

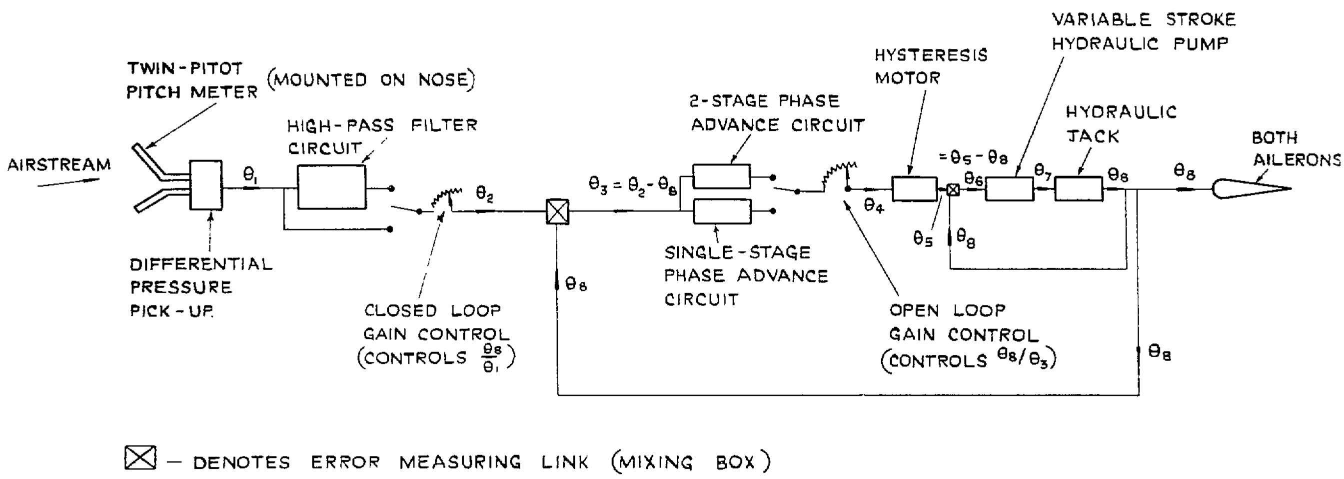


FIG. 3. Diagram of gust alleviator.

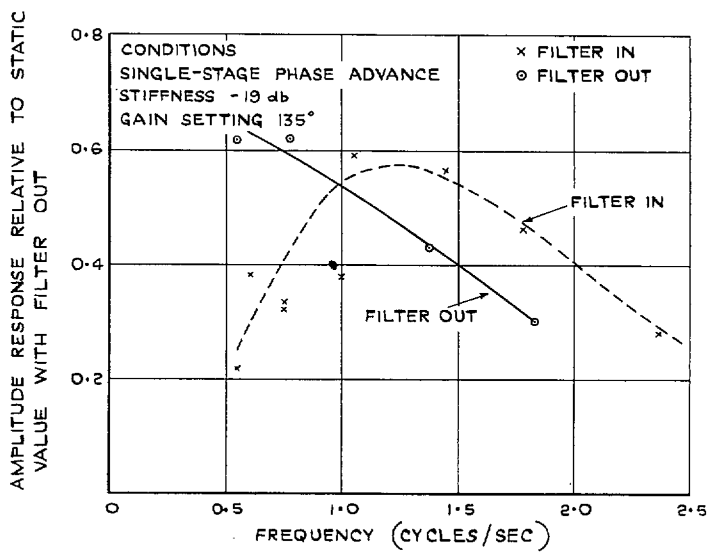
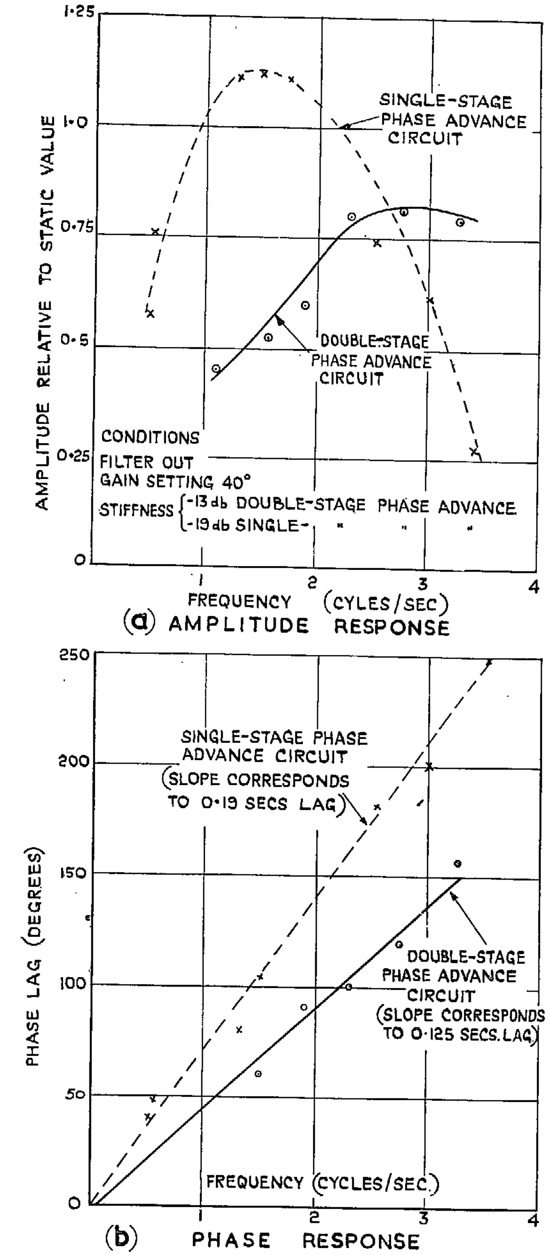


FIG. 4. Effect of filter on amplitude of frequency response of gust servo.

(4039)



FIGS. 5a and 5b. Effect of phase advance circuit on frequency response of gust servo.

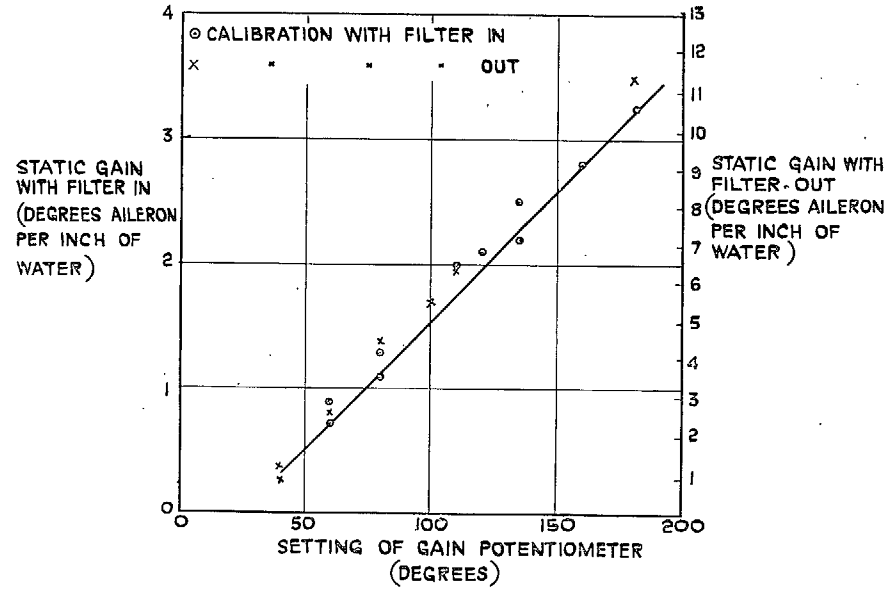


FIG. 6. Summary of static calibrations of gust servo.

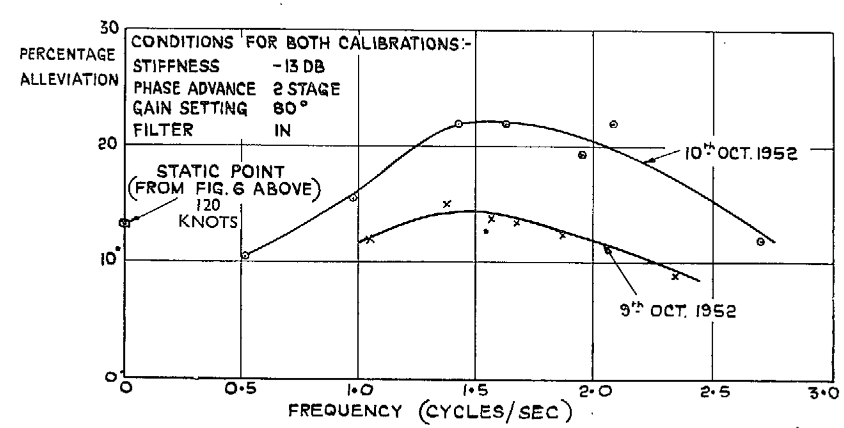


FIG. 7 Typical frequency response curves for gust servo showing day-to-day variation under nominally constant conditions.

B**

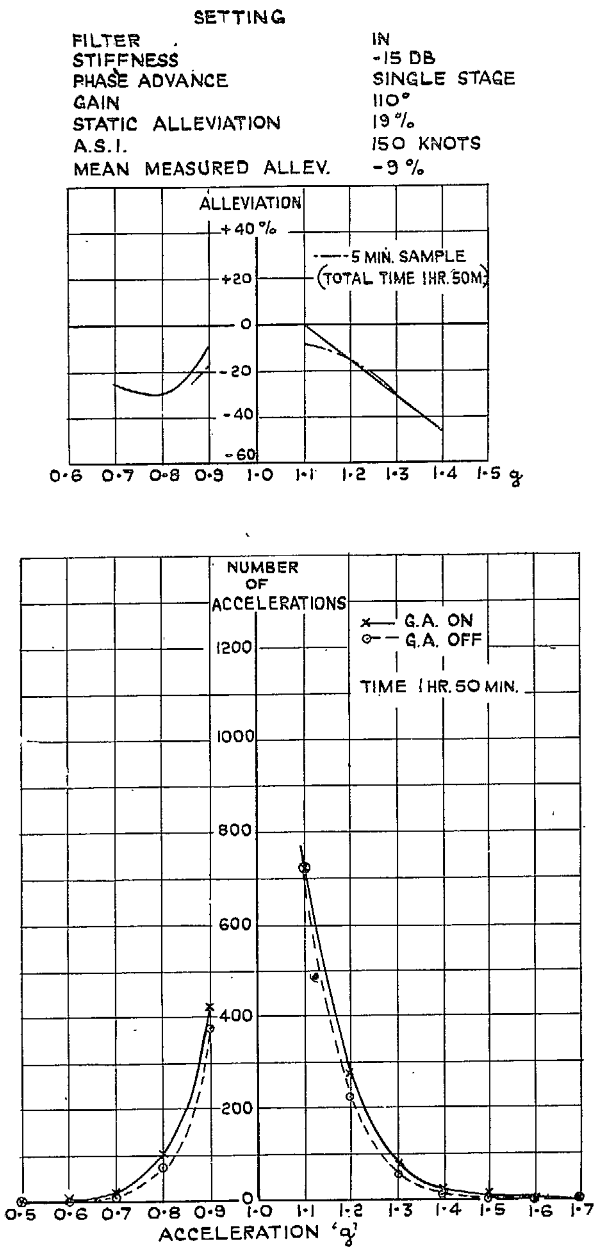


FIG. 8. A typical record of the counting accelerometer. Static alleviation 19 per cent.

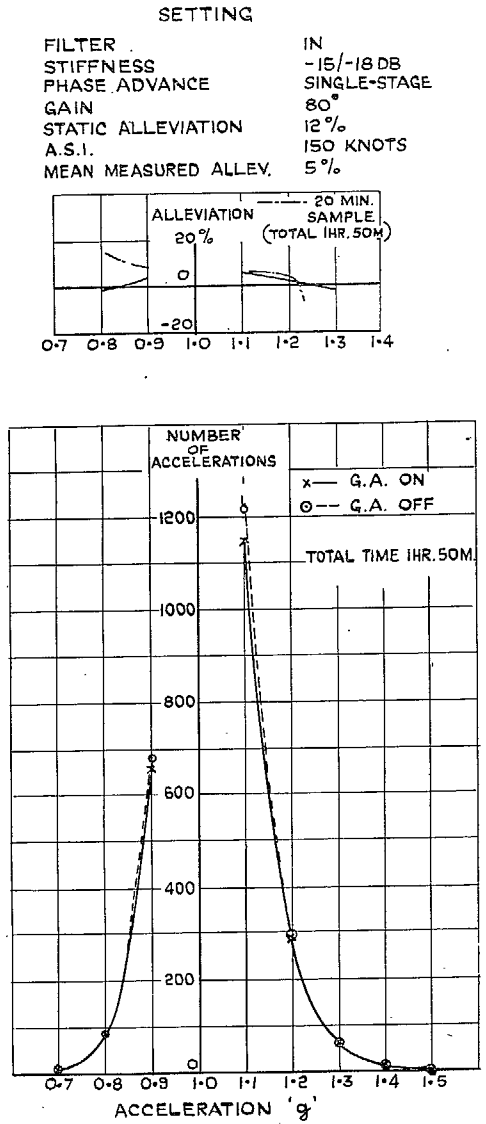


FIG. 9. A typical record of the counting accelerometer. Static alleviation 12 per cent.

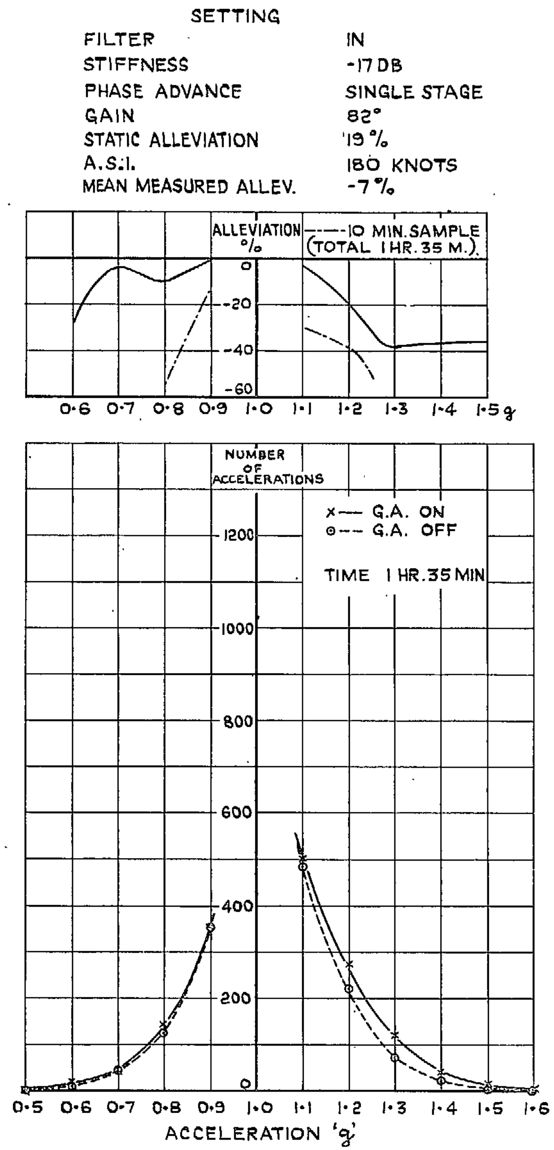


FIG. 10. A typical record of counting accelerometer. Speed 180 knots, static alleviation 19 per cent.

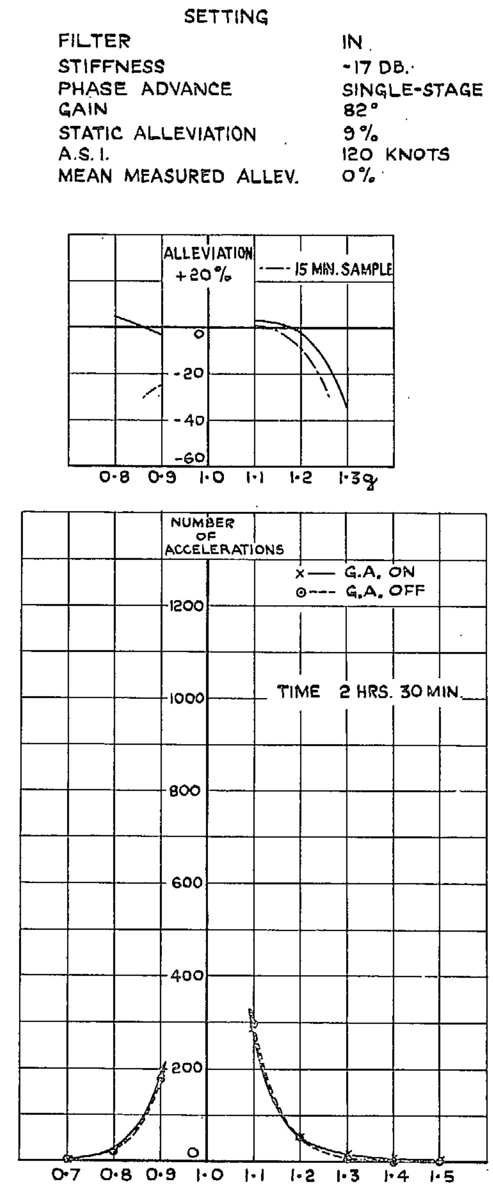


FIG. 11. A typical record of the counting accelerometer. Speed 120 knots, static alleviation 9 per cent.

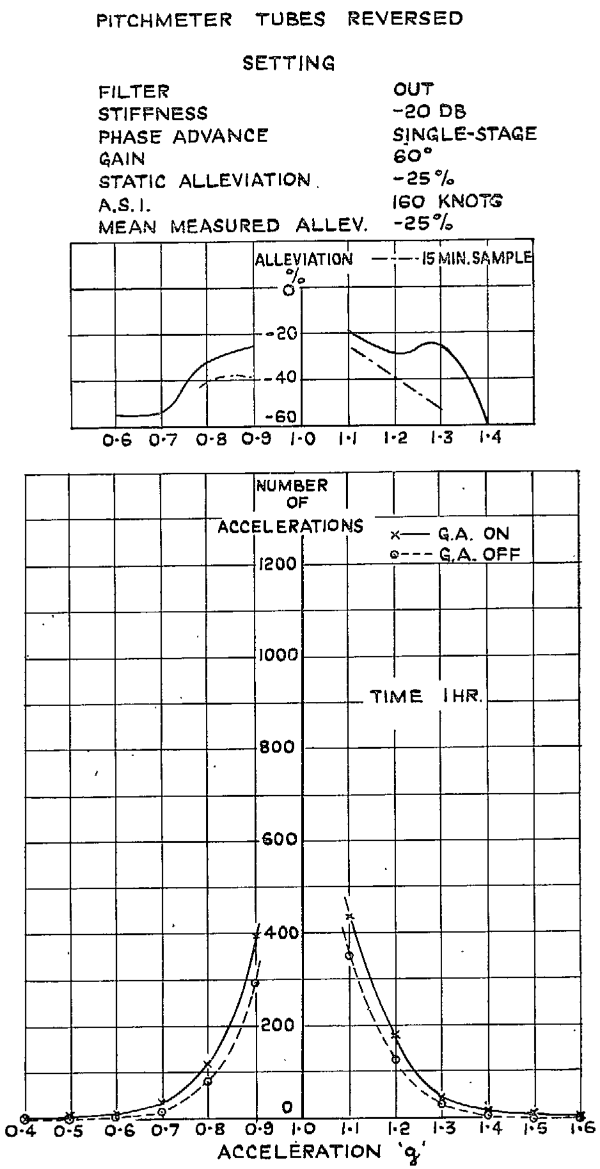


FIG. 12. A typical record of the counting accelerometer. Gust signal reversed, static alleviation —25 per cent.

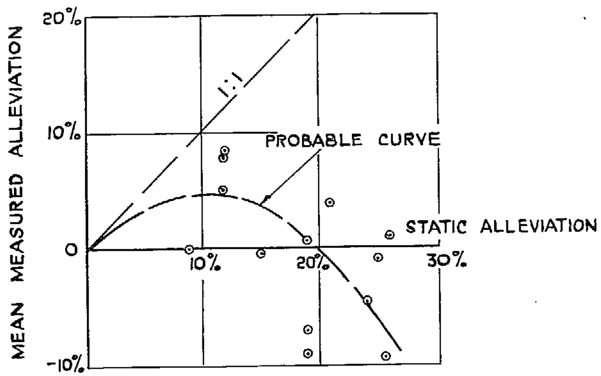


FIG. 13. The effect of static alleviation on the actual mean alleviation as measured in flight.

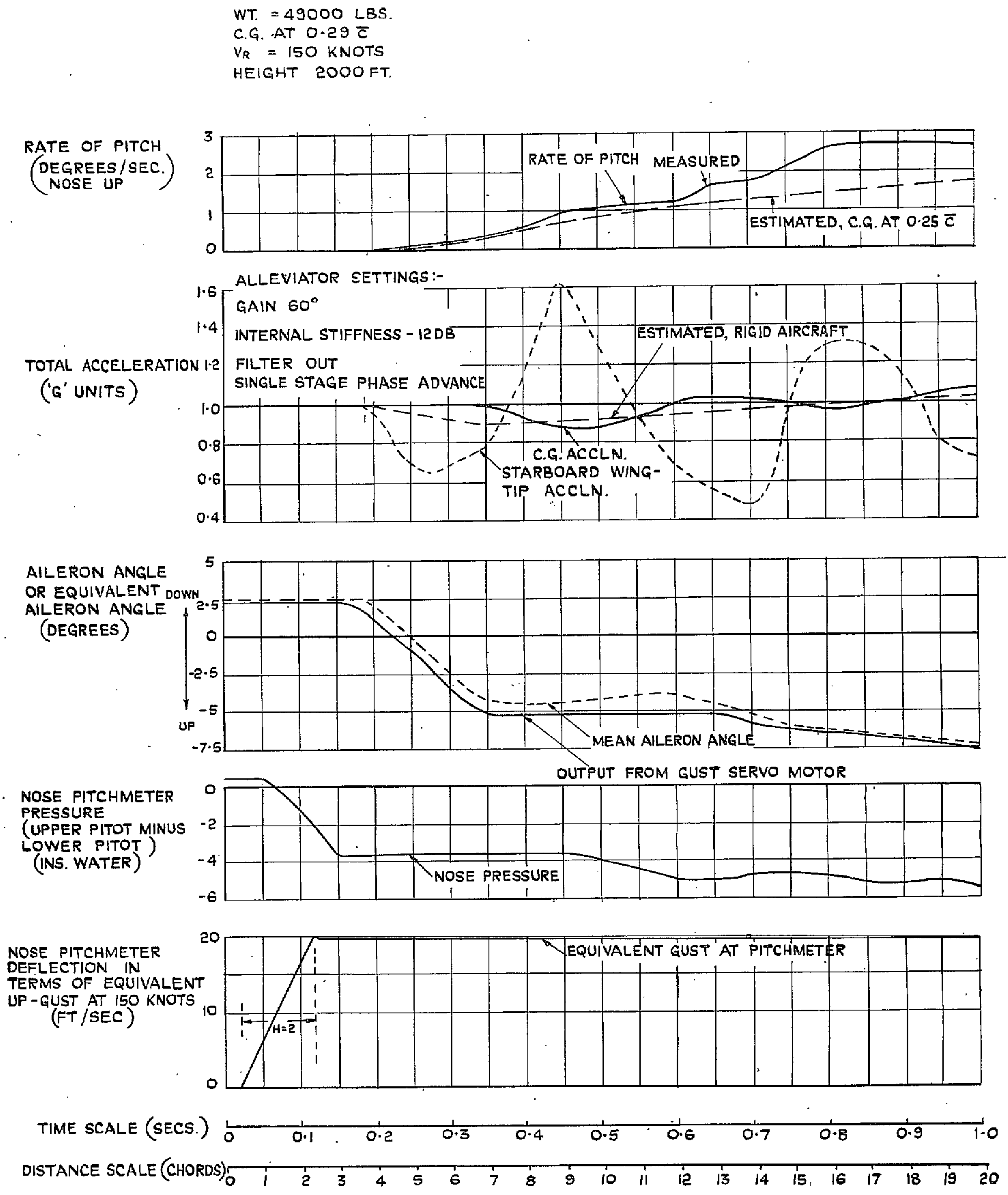


FIG. 14a. Response of aircraft and servo to simulated gust for flat-topped input ($H = 2$). Elevator fixed.

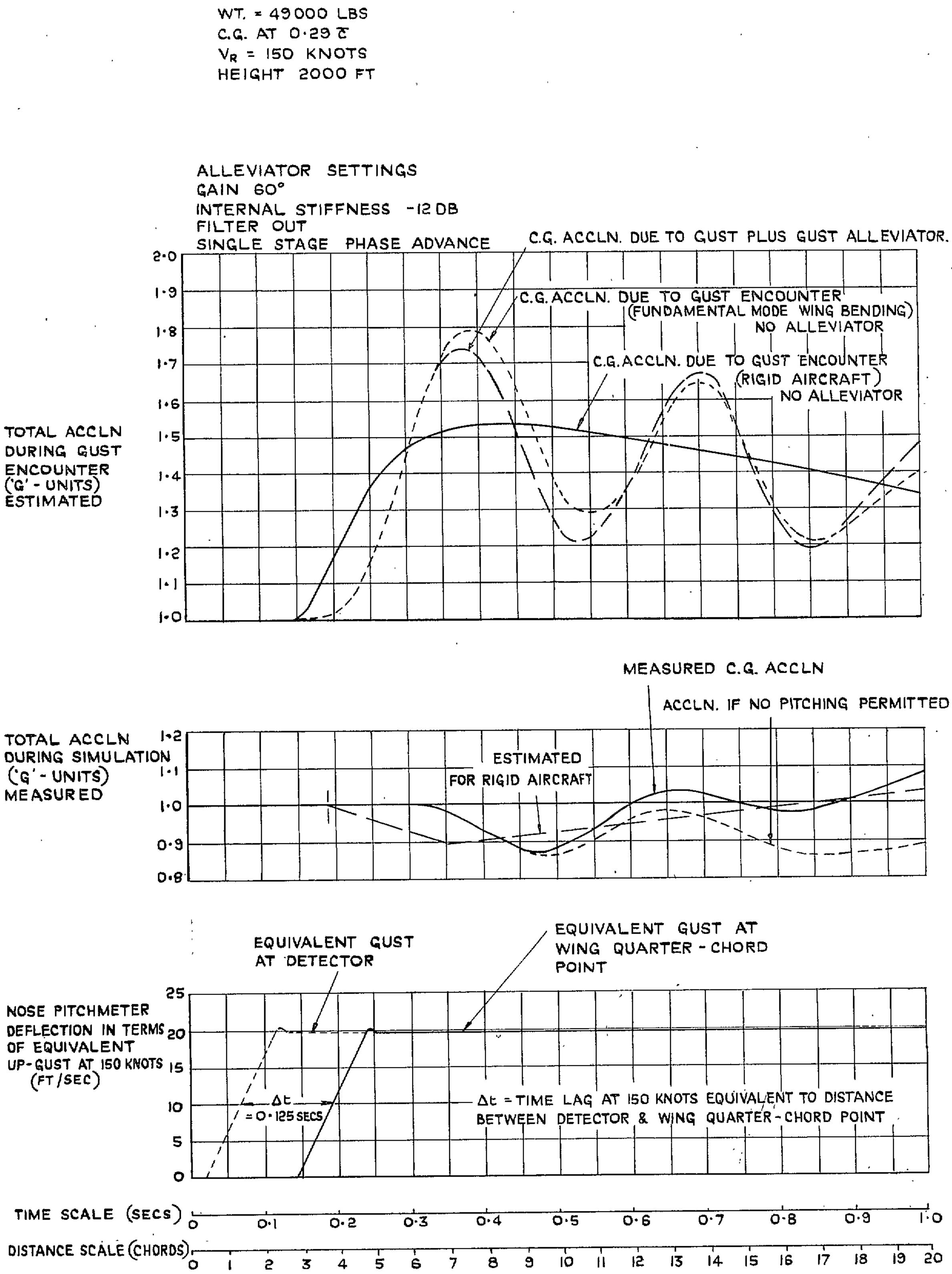


FIG. 14b. Estimated response of aircraft to flat-topped gust and comparison with measured alleviation during corresponding simulation ($H = 2$). Elevator fixed.

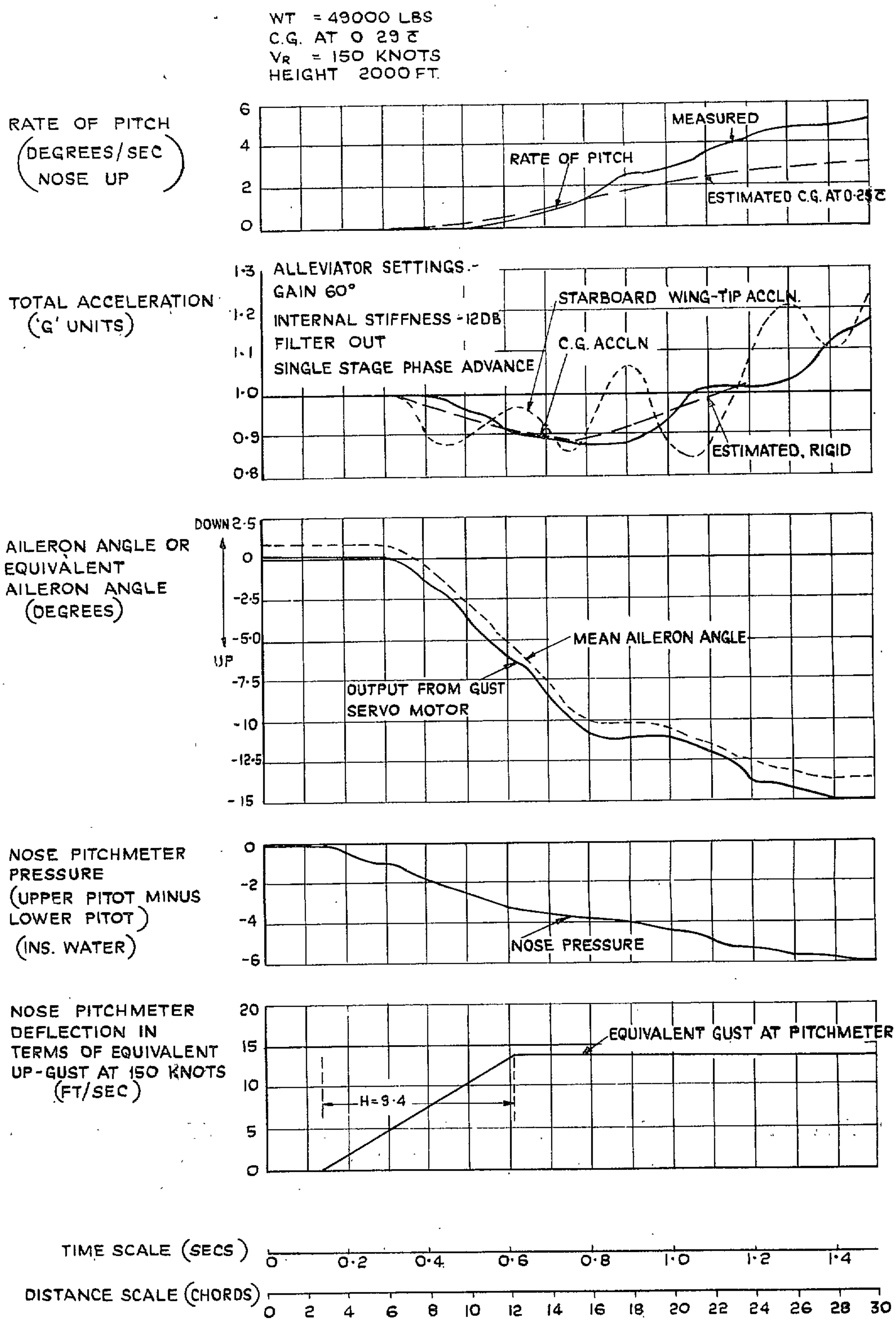


FIG. 15a. Response of aircraft and servo to simulated gust for flat-topped input ($H = 9 \cdot 4$). Elevator fixed.

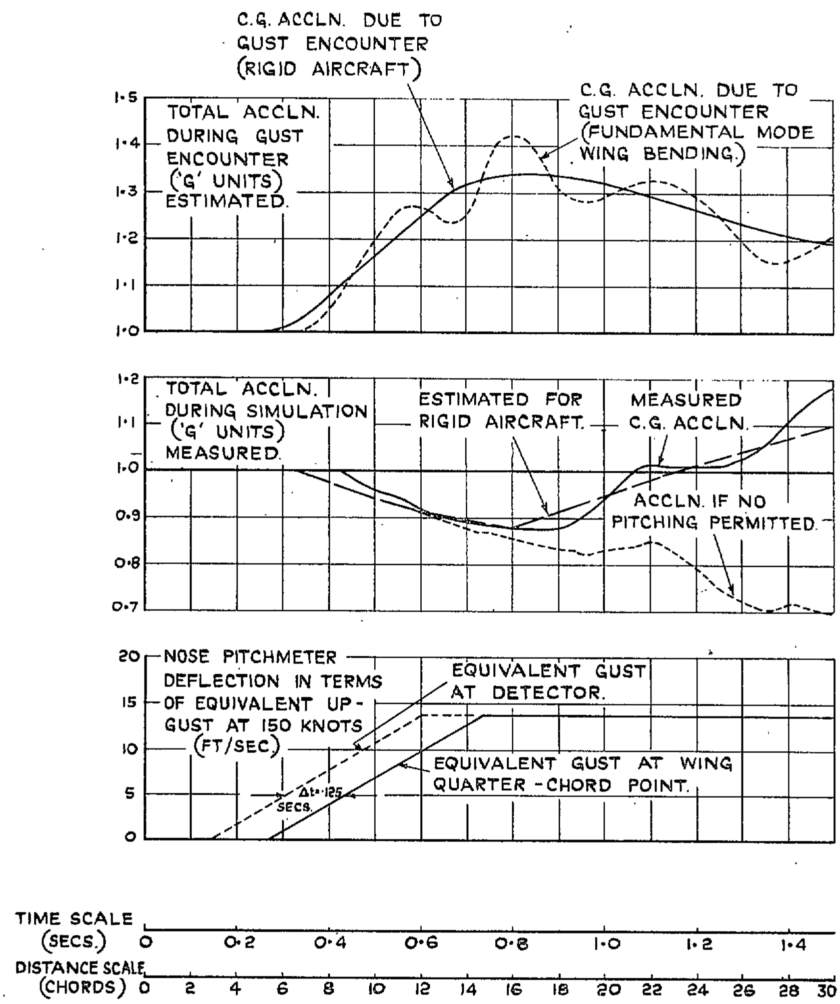


FIG. 15b. Estimated response of aircraft to flat-topped gust and comparison with measured alleviation during corresponding simulation ($H = 9{\cdot}4$). Elevator fixed.

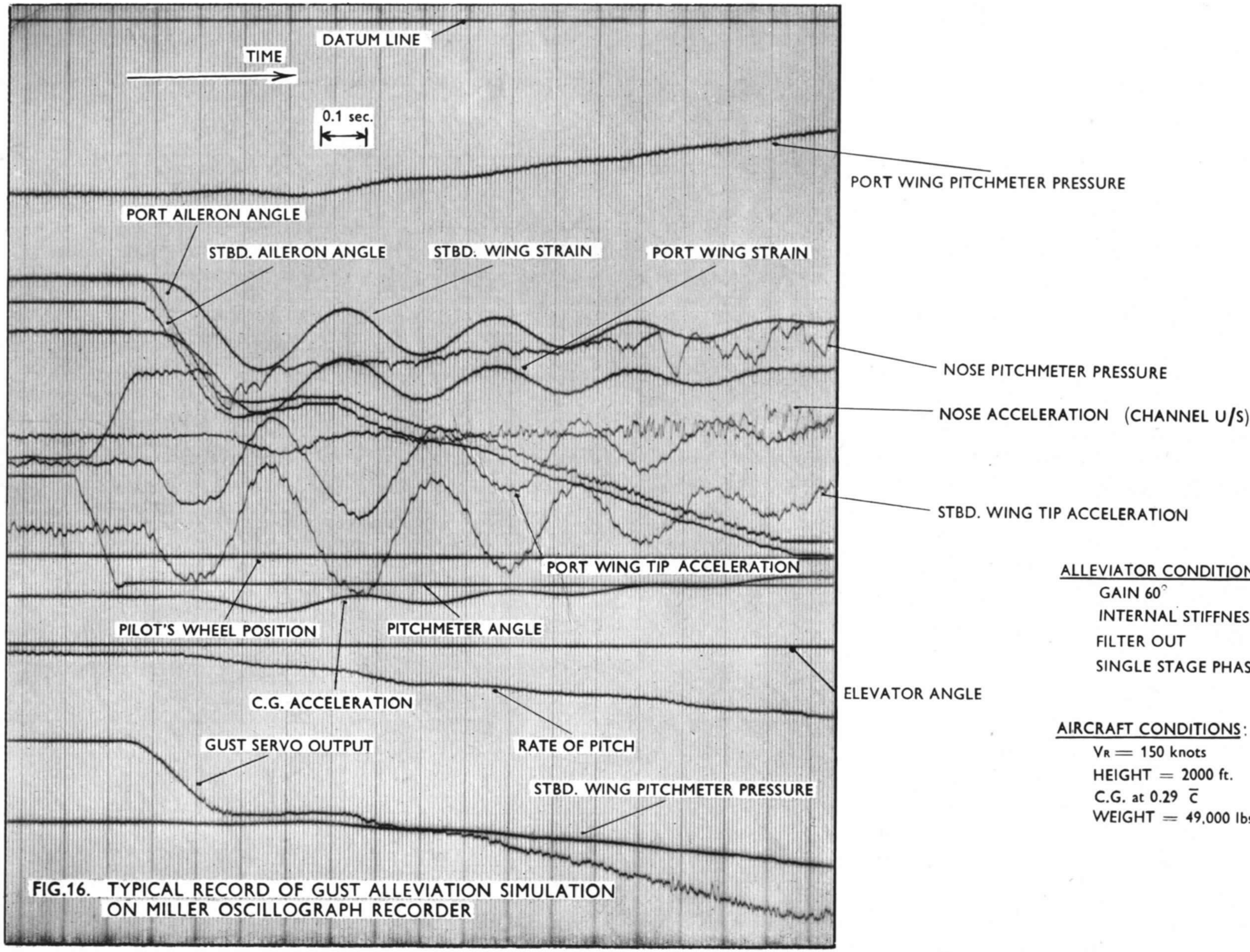


FIG.16. TYPICAL RECORD OF GUST ALLEVIATION SIMULATION ON MILLER OSCILLOGRAPH RECORDER

ALLEVIATOR CONDITIONS:

GAIN 60°
INTERNAL STIFFNESS –12DB.
FILTER OUT
SINGLE STAGE PHASE ADVANCE

AIRCRAFT CONDITIONS:

V_R = 150 knots
HEIGHT = 2000 ft.
C.G. at 0.29 $\bar{c}$
WEIGHT = 49,000 lbs

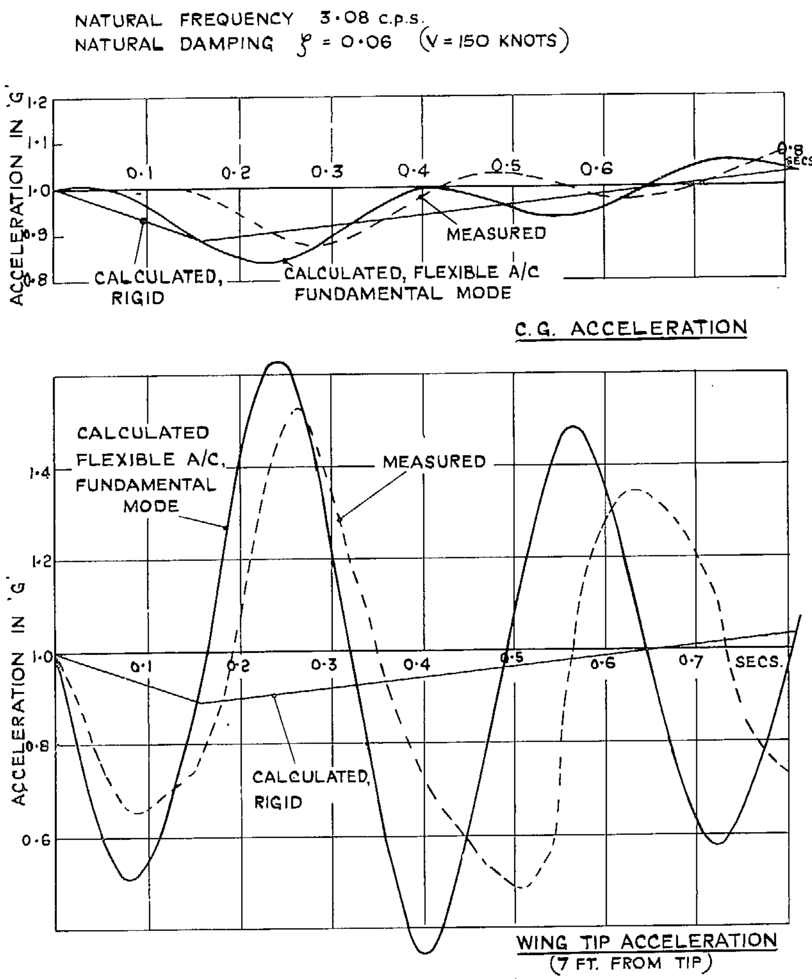


FIG. 17. Comparison of calculated and measured accelerations due to deflection of ailerons.

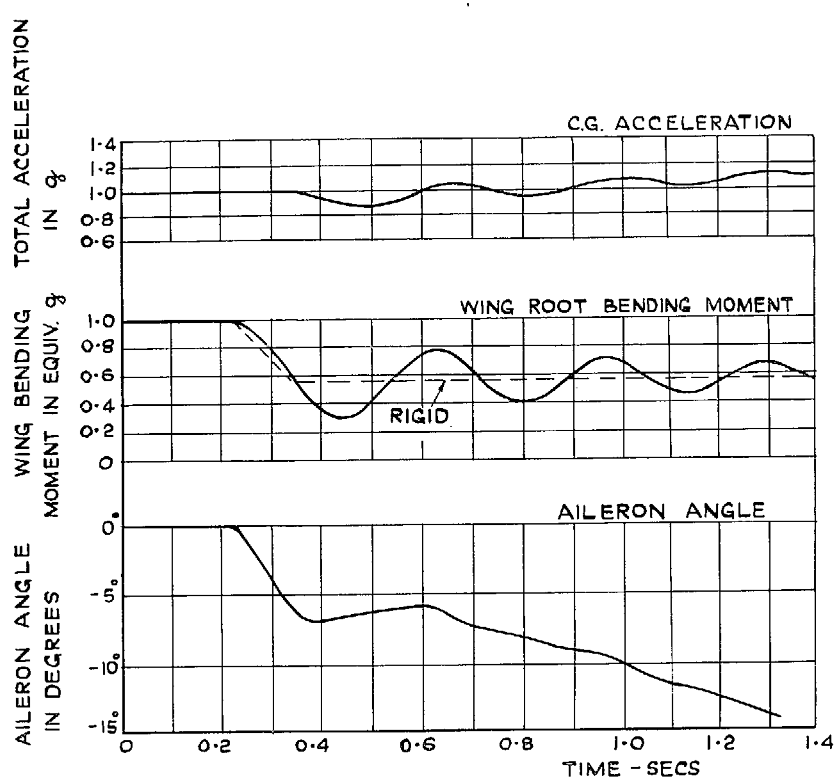


FIG. 18. Wing-root bending moment and c.g. acceleration in response to aileron deflection.

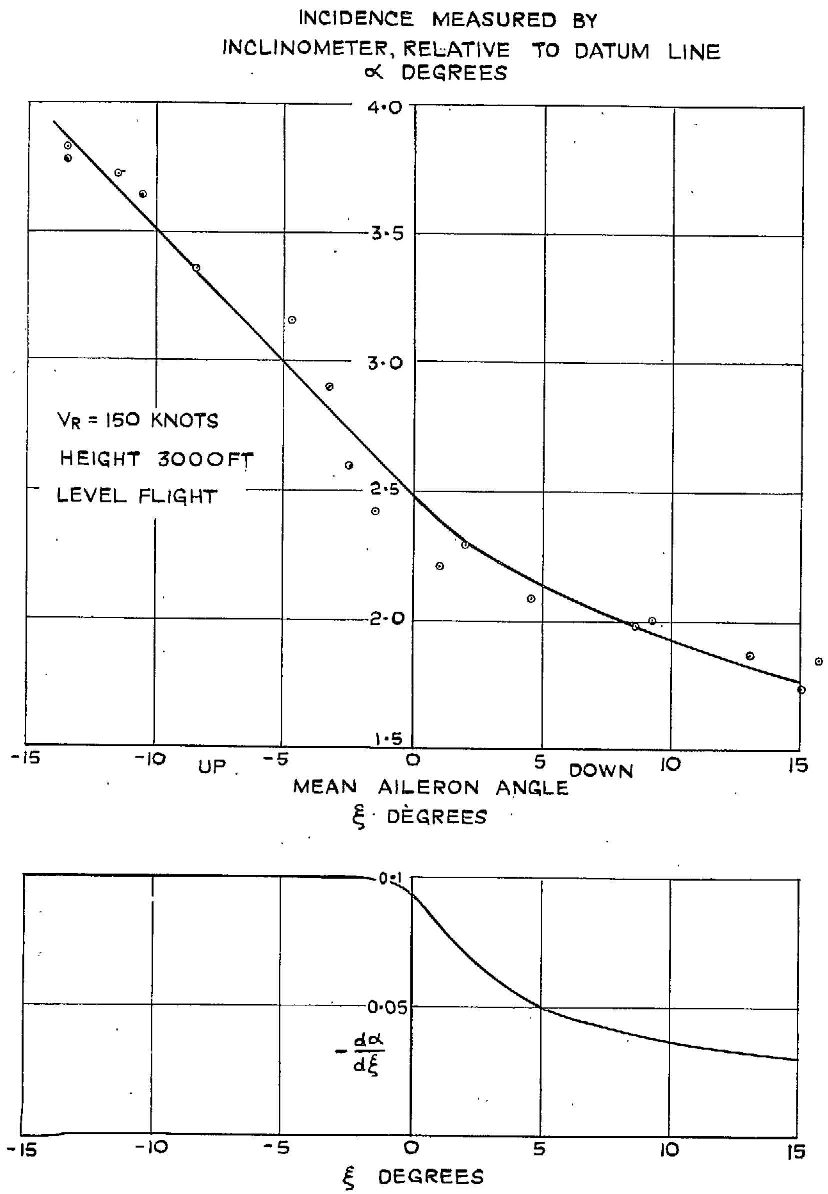


FIG. 19. Change of incidence with various amounts of gust aileron angle applied in steady flight.

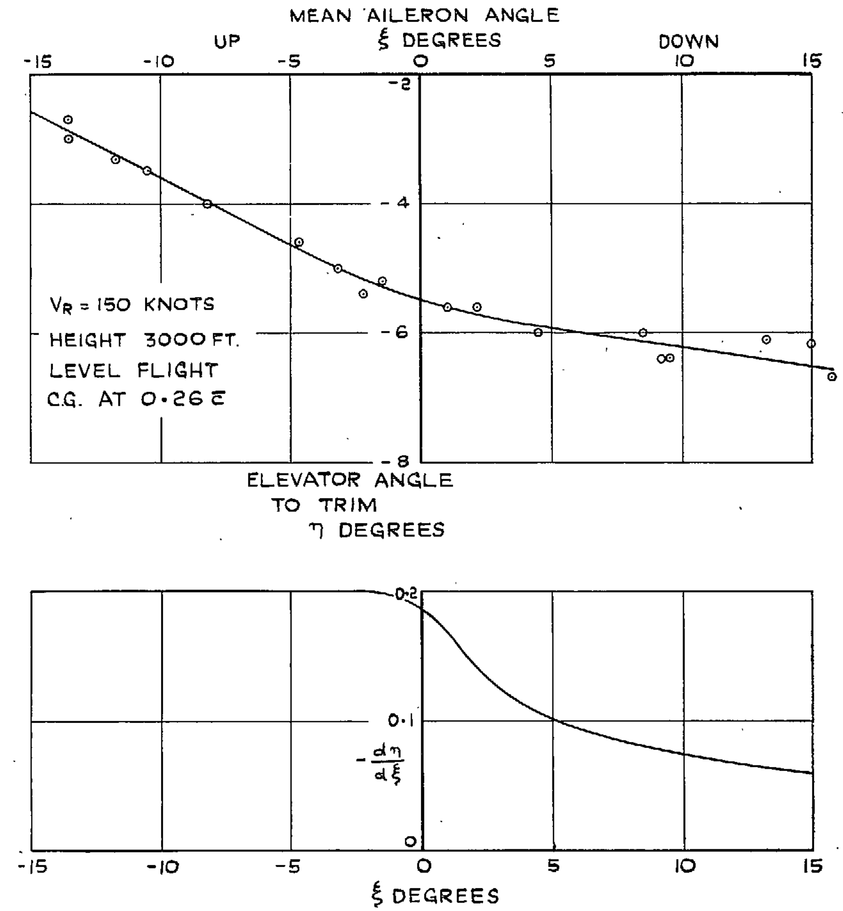


FIG. 20. Elevator angle to trim various amounts of gust aileron angle applied at constant speed.

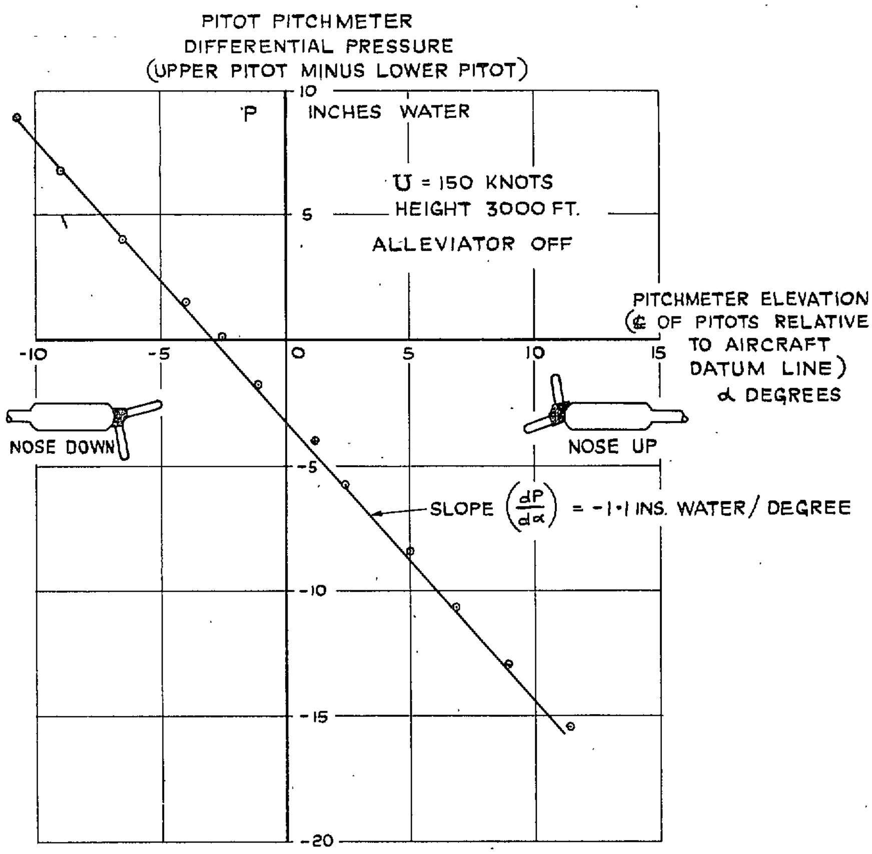


FIG. 21. Pressure—incidence relation for pitot pitchmeter.

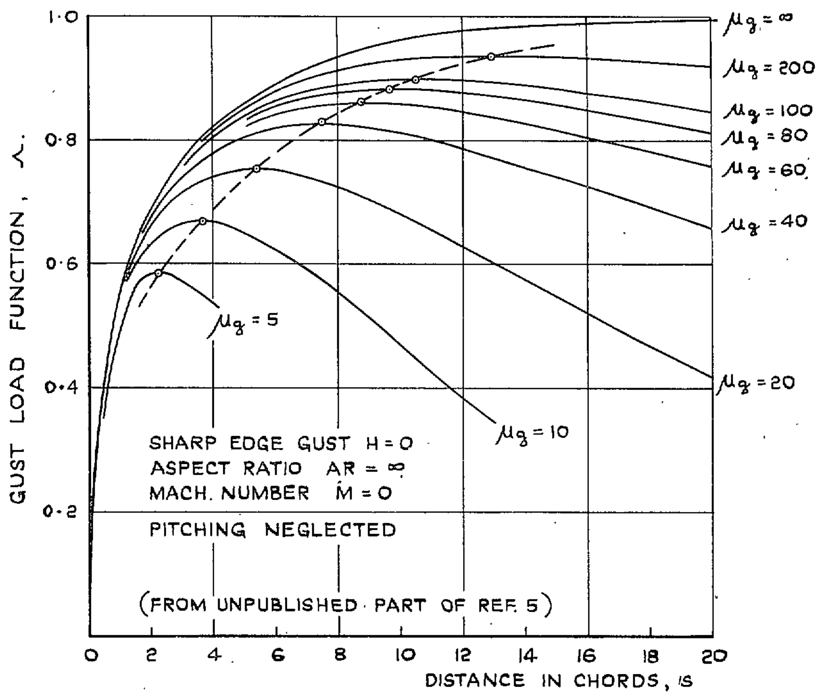


FIG. 22. An example of a gust load build-up as a function of distance travelled in chords, for a range of mass parameters.

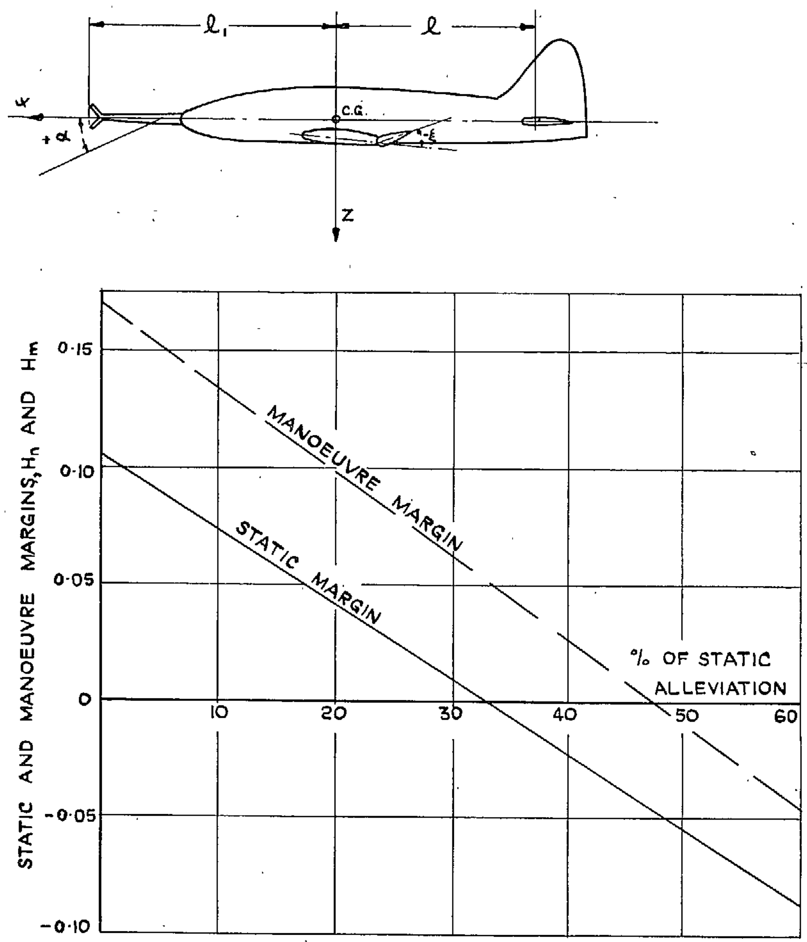


FIG. 23. The effect of gust alleviator on static and manoeuvre margins.

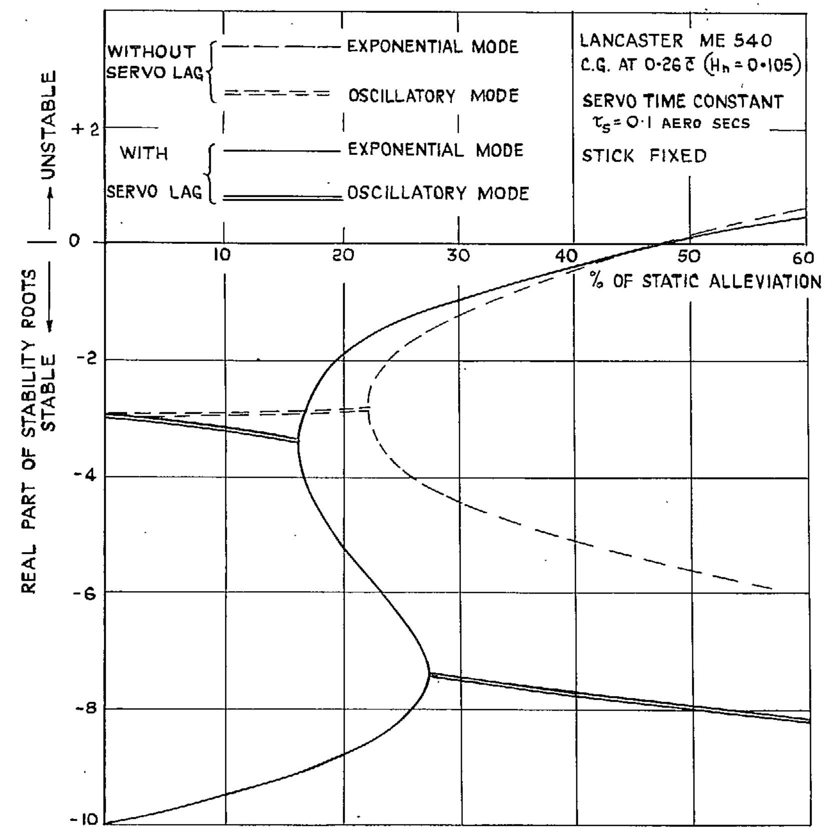


FIG. 24. Real part of stability roots vs. per cent of alleviation. The effect of gust alleviator and servo lag on aircraft stability.

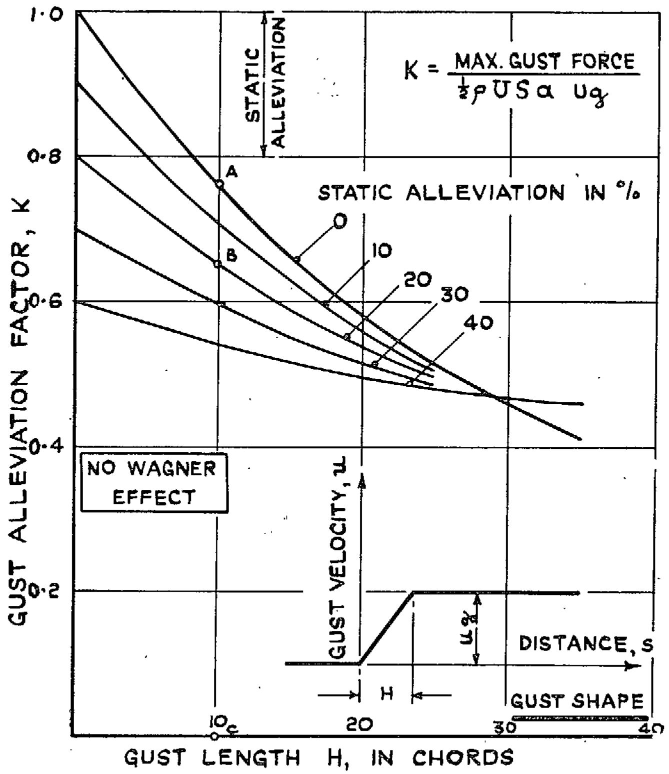


FIG. 25. Gust alleviation factor of the aircraft with gust alleviator, flat-top gust (*Lancaster* ME.540).

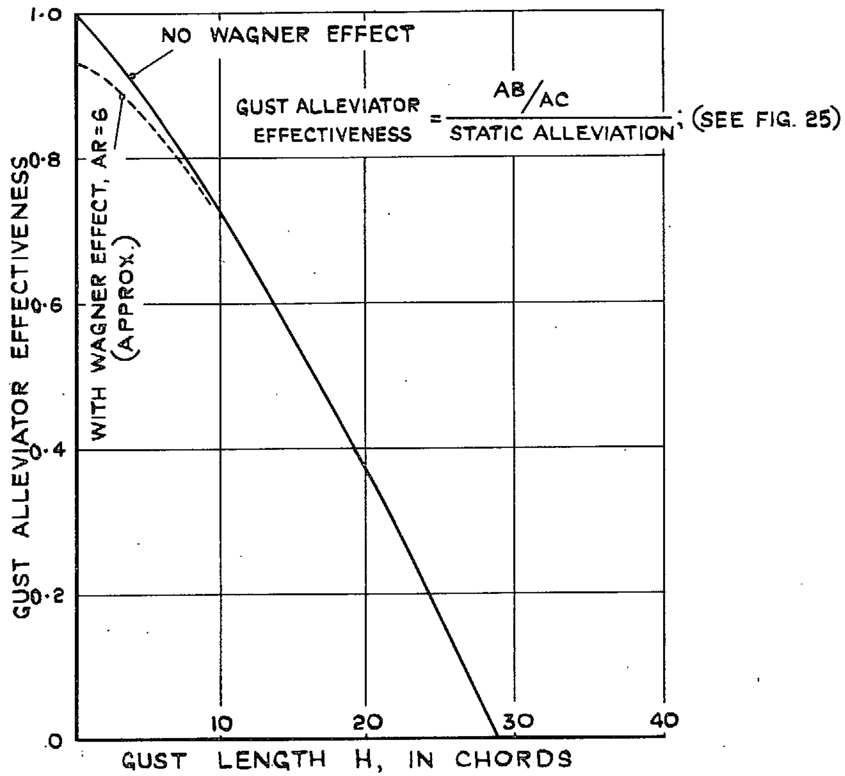


FIG. 26. The effect of gust length on gust alleviator effectiveness (*Lancaster* ME.540).

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