MR No. A4K04

A-78

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WARTIME REPORT

ORIGINALLY ISSUED

November 1944 as
Memorandum Report A4K04

FLYING QUALITIES OF A HIGH-SPEED BOMBER WITH A DUAL PUSHER PROPELLER AFT OF THE EMPENNAGE AS ESTIMATED FROM WIND-TUNNEL TESTS OF A 1/8-SCALE POWERED MODEL

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MEMORANDUM REPORT

for the

Air Technical Service Command, U.S. Army Air Forces

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SUMMARY

Flying qualities of a high-speed bomber with a dual pusher propeller aft of the empennage, computed from the results of wind-tunnel tests of a 1/8-scale powered model, are presented in this report. The flying qualities are evaluated with respect to the stability and control requirements of the Air Technical Service Command, U.S. Army Air Forces.

For the characteristics investigated, the airplane has satisfactory flying qualities except for the following:

(a) With aft center of gravity, low stick-free longitudinal stability, flaps up, and possible instability, flaps down

(b) High elevator control forces in landing with forward center of gravity

(c) Insufficient elevator control in take-off

(d) Low rudder-free directional stability

INTRODUCTION

Estimates of the flying qualities of a high-speed bomber with a dual pusher propeller aft of the empennage have been made. The analysis, made at the request of the

Air Technical Service Command, U.S. Army Air Forces, is based on the results of wind-tunnel tests of a 1/8-scale powered model in the 7- by 10-foot wind tunnel at the Ames Aeronautical Laboratory. The flying-qualities requirements of reference 1 have been used as a criterion of a satisfactory airplane.

DESCRIPTION OF THE AIRPLANE

The airplane is a three-place light bomber. Major airplane dimensions are listed in table I, and a three-view drawing of the airplane is shown in figure 1. Figure 2 is a line diagram of the wing, and figure 3 is a line diagram of the tail. The airplane is of unconventional design in that it has a dual pusher propeller aft of the empennage. Each set of three propeller blades is gear-driven by one of two engines submerged in the fuselage. The airplane has sealed internal balance control surfaces (figs. 2 and 3 for cross-sectional views), double-slotted partial-span flaps, and a tricycle landing gear that retracts into the fuselage. The small split flap on the wing adjacent to the fuselage (fig. 2) operates in conjunction with the landing gear in that it is retracted when the gear is up and deflected to 40° after the gear is extended. This linkage is necessary in order to provide a flap between the double-slotted flaps and the fuselage and still allow the double-slotted flap to be operated for any position of the gear. The vertical tail extends both above and below the fuselage with the lower half also acting as a propeller guard.

RESULTS AND DISCUSSION

The flying qualities of the airplane are presented and discussed according to the Army Air Forces stability and control requirements (reference 1). Aileron control characteristics were not analyzed since the model tests did not include a determination of the aileron characteristics. The dynamic stability was not investigated since the necessary data were not available from wind-tunnel tests.

The prototype airplane is equipped with an elevator nose balance adjustable between the limits of $0.45c_e$ and $0.50c_e$ geometric balance corresponding to effective balances of $0.40c_e$ and $0.44c_e$, respectively (as estimated by the

Douglas Aircraft Company). The rudder has a geometric nose balance of $0.47c_r$ corresponding to an effective balance of $0.396c_r$. The estimated control-surface effective balance is less than the airplane geometric balance because of the effects of cutouts for hinges, cutoffs for cover-plate ribs, and leakage through drainage holes. The elevator control forces presented herein are for $0.443c_e$ effective balance which was thought to be nearly the correct balance needed for satisfactory control forces in accelerated flight. The pedal forces presented are for both $0.396c_r$ effective balance (corresponding to the airplane) and $0.34c_r$ effective balance, which is just about the proper balance needed for satisfactory pedal forces. The values of effective balance for which results are presented are believed to be accurate to within 1 percent of the control-surface chord.

The computations of flying qualities presented herein are based on the assumption that all control surfaces are rigid and mass-balanced and that no friction exists in the control system. The control-system mechanical advantages used (given in table I) include cable stretch.

The paragraph-numbering system of reference 1 has been retained in the discussion below for easy cross reference.

Longitudinal Stability and Control

<u>E-1b(3) Elevator-fixed stability</u>.- The longitudinal characteristics in steady flight were determined for the conditions of flight defined in the following table:

Condition	Power	Flaps and gear	Gross weight (lb)	Center-of-gravity position (percent M.A.C.)	Fig. no.
Climb	Military	Retracted	25,000 25,000	25 35	4
Glide	Zero thrust $T_c = 0$	Retracted	25,000 25,000	25 35	4
Approach	50 percent rated	Extended 50°	25,000 25,000 21,500	20 25 35	5
Landing	Zero thrust $T_c = 0$	Extended 50°	25,000 25,000 21,500	20 25 35	5

The results are presented as the variation of elevator deflection and stick force with indicated airspeed. The glide condition ($T_c = 0$, fig. 4) was substituted for the cruise condition of the Army's requirements (75 percent rated power) because this airplane, being of unconventional design, is less stable power off than power on.

Elevator-fixed stability, as shown by the variation of elevator deflection with indicated airspeed, exists for all the flight conditions investigated (figs. 4 and 5). As the most aft center of gravity is at 35 and 36 percent mean aerodynamic chord with flaps and gear up and flaps and gear down, respectively, the stick-fixed stability is such as to satisfy the Army's requirements.

<u>E-1b(3) Elevator-free stability</u>.- Elevator-free stability exists in the climb and glide conditions (fig. 4) throughout

the speed range investigated[1] for all speeds down to 123 miles per hour ($0.7\,V_R/C_{max}$). With the airplane trimmed at 341 miles per hour, which corresponds to level flight with military power, the gradient of control-force variation with indicated air-speed for the aft center-of-gravity location (35 percent M.A.C.) is low. This will tend to give inadequate control-centering characteristics unless some device, such as a bungee, is installed to increase the gradient.

The model test results show an unstable control-force variation existing above 130 miles per hour for the landing and approach conditions with the airplane trimmed for zero control force at $1.4V_{s_a}$ (fig. 5). This unstable variation is due to a stall of the horizontal tail of the model which resulted in high hinge moments at the lower lift coefficients. (The tail angle of attack α_T becomes more negative as C_L decreases. Tail stall occurs at $C_L \cong 1.0$, flaps at 50°, where $\alpha_T \cong -12°$.) The results of tests of an isolated tail surface at full-scale Reynolds number indicate that the tail angle of attack at which the elevator stalls will be more negative so that the instability existing between 130 and 225 miles per hour (the limit design speed with flaps deflected) will be removed for center-of-gravity positions at 20 and 25 percent mean aerodynamic chord. With the center of gravity at 35 percent mean aerodynamic chord, however, elevator-free instability will exist above approximately 160 miles per hour. Thus the requirement for elevator-free stability at speeds from allowable V_{max} down to $1.2V_{s_a}$ for the approach condition and down to V_{s_a} for the landing condition will be met only with the center of gravity forward of 25 percent mean aerodynamic chord.

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[1]All control forces in this report are based on model hinge moments obtained at a Mach number of 0.25 or less. For this reason the forces in figure 4 in the high-speed range may be somewhat in error. It is recommended that reference be made to the high-speed wind-tunnel tests of a model of this airplane discussed in reference 2.

E-1c(2) Elevator control-force gradient.- The requirement for elevator control forces in accelerated flight is one of the critical requirements with regard to elevator balance. Control forces on the airplane in steady turning flight with flaps and gear retracted have been determined using a true speed of 345 miles per hour at an altitude of 10,000 feet with the propeller operating at zero thrust ($T_c \cong 0$). This corresponds to turns from trimmed steady level flight at 10,000 feet with normal rated power (except for a slight difference in T_c which has a negligible effect on the final result). The computed and allowable variation of force with acceleration is presented in figure 6 for two different gross weights and center-of-gravity positions. The allowable control-force gradients are based on a limit load factor of 4 for gross weights up to 25,000 pounds and varying from 4 at 25,000 pounds to 2.67 at 35,000 pounds.

The gradients of control force per unit normal acceleration obtained are approximately linear (fig. 6). For both gross weights, with the center of gravity at 25 percent mean aerodynamic chord, the gradient is below the allowable maximum; with the center of gravity at 35 percent mean aerodynamic chord, the gradient is below the allowable minimum. Reduction of the effective balance to $0.43c_e$ will make the gradient (center of gravity at 35 percent M.A.C.) just equal to the allowable minimum.

A summary of the accelerated flight control-force gradients (fig. 7) shows the variation of control force per unit normal acceleration with center-of-gravity position. The maneuvering neutral point (where stick force per g = 0) is at 37 percent mean aerodynamic chord. Due to the closely balanced control surfaces, the magnitude of the induced effects on elevator hinge moments is small so that the center-of-gravity position for zero control force in turns and the level-flight stick-free neutral points are very nearly coincident.

E-1c(3) Elevator control in landing.- The airplane lands with the flaps deflected 50°. Tests of the model in the presence of a ground plane for the determination of the landing characteristics of the airplane indicated abnormal lift characteristics when the flap deflection exceeded 40°. Further tests at increased Reynolds number (obtained through increased stream turbulence) indicated the airplane lift characteristics will be normal with flaps deflected 50°. In order to obtain an indication of the landing characteristics of the airplane for the full flap deflection of 50°, tests were made of the model with the flaps deflected 30° and 40°. These results have been converted to elevator deflection and control force as a function of contact speed and are presented in figures 8 and 9. The computations have been made for the combinations of gross weights and center-of-gravity positions listed in the following table. The resulting control forces and elevator deflections required to reach maximum ground angle have also been tabulated.

Gross weight (lb)	Center-of-gravity position (percent M.A.C.)	Flap deflection (deg)	δ_e required to reach maximum ground angle (deg)	Control force at maximum ground angle (lb)
21,500	35	30 40	3 4	4 7
25,000	25	30 40	11 13	7 27
25,000	20	30 40	16 19	13 74

An extrapolation of the results indicates that with full flap deflection of 50° and with the center of gravity in the normal landing position (36 percent M.A.C.) the elevator will be capable of holding the airplane off the ground at its maximum ground angle without exceeding a control force of 50 pounds. In fact, the control forces may be considered rather light. When the airplane is in the landing condition (flaps at 50°) with full forward center of gravity at 20 percent mean aerodynamic chord, the control requirement will be met; but the force required will be more than double the allowable limit of 50 pounds.

Results of tests on an isolated tail surface at full-scale Reynolds number indicated that for full flap deflection of 50° the control forces in landing with 20 and 25 percent center of gravity will be in the order of 135 pounds and 34 pounds, respectively. With normal landing center of gravity of 35 percent mean aerodynamic chord, the control forces will be well within the allowable 50 pounds (approx. 7 lb).

The break that occurs in the elevator-deflection curve for flaps at 40° (fig. 9) is due to the unstalling characteristics of the model horizontal tail. With decrease in contact speed, the tail angle of attack becomes less negative and a resulting increase in elevator effectiveness occurs. The break in the control-force curves on figures 8 and 9 occurs for the same reason. On figure 8 the tail angle of attack is just at the point where the stall progression is beginning to show on the hinge-moment characteristics, but the stall has not progressed enough to show on the elevator-effectiveness characteristics.

E-1c(4)(b) Elevator control in take-off.- Control characteristics during the take-off run have been determined for a range of center-of-gravity positions corresponding to those on the airplane from 20 to 35 percent mean aerodynamic chord and for gross weights of 20,000, 25,000 and 35,000 pounds. In the computations a coefficient of rolling friction of 0.012 was used corresponding to take-off from a concrete runway. The results, presented in figure 10, show the taxiing speed at which maximum up-elevator (25°) will raise the nose wheel. From this figure it is evident that take-off attitude cannot be attained at or below $0.8V_{s_a}$ for any gross weight or center-of-gravity position. With forward center of gravity and normal gross weight (20 percent M.A.C. and 25,000 lb), take-off attitude can be secured only above speeds of 105 miles per hour as compared to the speed equal to $0.8V_{s_a}$ of 70.4 miles per hour. The greater part of the moment which the elevator must overcome in order to raise the nose wheel is a result of the center of gravity being located so far forward of the main gear. The main gear is located at approximately 55 percent mean aerodynamic chord.

E-1(c)(5) Elevator trimming control.- The elevator trim tab is capable of reducing the elevator control force to zero at all speeds in level flight with flaps and gear retracted for any power setting or center-of-gravity position, and at speeds below $1.4V_{s_a}$ with flaps and gear extended with power off ($T_c = 0$) and forward center of gravity. Therefore, the longitudinal trimming control will satisfy the requirement of reference 1.

E-1c(6) Trim changes due to flaps and power.- The effect on trim of varying the power and flap setting is shown in figure 11. With the tab set for trim at $1.4V_{s_a}$ (V_i = 123 mph) with flaps and gear retracted and military power, the change in control force at constant speed due to varying the power or flap setting is within the allowable limit of 50 pounds. Thus the requirement of reference 1 is satisfied.

Lateral and Directional Stability and Control

E-2b(1)(b) Rudder-fixed directional stability.- The directional characteristics in steady sideslips were determined for the following conditions of flight:

Attitude	Speed at sea level (mph)	Flaps	Gear	Fig. no.	$(\partial\delta_r/\partial\beta)_{C_n = 0}$ Zero thrust	Military power
$1.2V_{s_g}$	129	Up	Up	12	1.9	1.9
Climb	174	Up	Up	12	1.8	1.8
$0.7V_{max}$	218	Up	Up	12	1.8	1.8
$1.2V_{s_a}$	106	Down 50°	Down	13	1.4	1.6

The results are presented for zero thrust and military power as the variation of rudder deflection and pedal force with sideslip angle.

Rudder-fixed directional stability, as shown by the variation of rudder deflection with sideslip angle in figures 12 and 13, exists for any flap and power condition. The rudder-deflection curves obtained are linear throughout the sideslip-angle range. For this airplane the value of $\partial\delta_r/\partial\beta$ is quite high, as may be noted from the preceding table.

E-2b(1)(c) Rudder-free directional stability.- The variation of rudder-pedal force with sideslip angle in figures 12 and 13 indicates that rudder-free stability exists for all conditions of flight investigated with $0.396c_r$ effective balance. However, the gradient of force vs sideslip angle is low with the propeller operating at zero thrust in the climb condition (fig. 12) and with flaps and gear down (fig. 13). Reduction of the effective balance to $0.34c_r$ will increase the rudder-free stability and still keep the pedal forces, developed in meeting the rudder-control requirements, within the allowable limit of 180 pounds (as will be discussed later).

E-2b(2) Rolling moment due to sideslip (dihedral effect).- A summary of the dihedral effect is given in the following tabulation which lists the stability derivative $C_{l'_\beta}(\partial C_{l}'/\partial_\beta)$ for three flight conditions:

Attitude	α_u (deg)	Flaps	Gear	$C_{l'_\beta}$	C_{n_β}	$C_{l'_\beta}/C_{n_\beta}$	Power condition
$1.2V_{s_g}$	6	Up	Up	-0.00137	0.00137	-1.00	Propeller removed
				-.00121	.00162	-.75	$T_c = 0$
				-.00130	.00183	-.71	$T_c = 0.46$
Climb and $0.7V_{max}$	2	Up	Up	-.00102	.00118	-.86	Propeller removed
				-.00120	.00158	-.76	$T_c = 0$
				-.00115	.00171	-.67	$T_c = 0.2$
$1.2V_{s_a}$	-1	Down 50°	Down	-.00110	.00088	-1.25	Propeller removed
				-.00122	.00125	-.98	$T_c = 0$
				-.00151	.00147	-1.03	$T_c = 0.75$

The values of $C_{l'_\beta}$ indicate positive dihedral effect existing for all flight conditions.

For the purpose of comparison with other airplanes the table above includes values of the ratio $(C_{l'_\beta}/C_{n_\beta})$. At the higher speeds with flaps up, the ratio for this airplane is approximately 0.70.

E-2b(3) Side force due to sideslip.- The variation of side force with angle of sideslip has a positive slope (right bank accompanying right sideslip) for all flight conditions, as shown by the curves of angle of bank required in steady sideslips in figures 12 and 13.

E-2c(1)(a) Rudder to overcome adverse aileron yaw.- Reference 1 requires that the rudder control be sufficient to overcome the adverse aileron yaw in abrupt aileron rolls at $1.2V_{s_a}$ and $1.2V_{s_g}$. Since the model tests did not include measurements of the aileron characteristics, the adverse yaw in aileron rolls has been estimated by the methods of reference 3. The computed results, given in the following table, show that the rudder control is sufficient to overcome the maximum adverse aileron yaw with flaps and gear up.

Condition	Adverse C_n	Control necessary to counteract adverse yaw at zero sideslip		
		Rudder deflection (deg)	Rudder pedal force (lb)	
			0.34 effective balance	0.396 effective balance
$1.2V_{s_g}$, flaps and gear up, $T_c = 0$ C_L=1.05, V_i=153 mph 35,000 lb gross weight	0.0121	12	176	147
Same as above but with military power	.0121	12	180	151
$0.7V_{max}$ at SL, flaps and gear up, $T_c = 0$ C_L=0.37, V_i=218 mph 25,000 lb gross weight	.0043	4.8	44	24

Methods of estimating adverse yaw with flaps down are not available, but normally the rudder-control requirement is easier to meet with flaps down. Therefore, the rudder control will everywhere be adequate to overcome the adverse yaw resulting from an abrupt full deflection of the ailerons. The pedal forces required to overcome the adverse aileron yaw at $1.2V_{s_a}$ as given in the above table are, for the critical speed, just equal to the allowable limit of 180 pounds when the effective balance is $0.34c_r$. With $0.396c_r$ effective balance, as on the prototype airplane, the forces are below the allowable limit.

E-2c(1)(b) Rudder control in landing.- Reference 1 requires the rudder control to be sufficient to maintain straight ground paths in landing in cross winds of up to $0.2V_{s_a}$ at 90° to the flight path without exceeding a pedal force of 180 pounds. The maximum angle of sideslip in cross winds to which the rudder must be capable of trimming occurs when the forward velocity of the air is a minimum. Consequently,

$$\beta_{max} = \tan^{-1} \frac{0.2V_{s_a}}{\text{landing speed}} = \tan^{-1} \frac{17.6 \text{ mph}}{94 \text{ mph}} \quad \left(\begin{array}{c}\text{at 25,000 lb} \\ \text{gross weight}\end{array}\right)$$

$$\beta_{max} = 10.5^\circ$$

Figure 13 indicates that at 106 miles per hour with flaps at 50° and with zero thrust, about 15° of rudder are necessary to perform the maneuver. The corresponding pedal forces (fig. 13) for $0.34c_r$ and $0.396c_r$ effective balance are 62 and 53 pounds, respectively. The rudder will therefore be adequate to maintain directional control in cross-wind take-offs and landings without exceeding pedal forces of 180 pounds.

E-2c(3) Rudder and aileron trimming control.- Trim is accomplished on the rudders and ailerons by means of a spring trim device controllable from the cockpit. The strength of the springs will determine the adequacy of the rudder and aileron trimming control.

Summary Table of Estimated Flying Qualities

A summary table of the flying qualities of the airplane is presented in figure 14. This table, besides summarizing the above discussion, also includes the stall characteristics and neutral-point variation with flap and power setting.

CONCLUSIONS

For the conditions investigated, the airplane has the following flying qualities evaluated in terms of the Army Air Forces requirements for stability and control.

1. The static longitudinal stability is adequate except for the following:

(a) Low elevator-free stability exists in the climb and glide conditions for the aft center-of-gravity position (35 percent M.A.C.) with the airplane trimmed at high speed in level flight.

(b) Elevator-free instability exists in the landing and approach conditions above 130 miles per hour which is traceable to a stall of the model horizontal tail in the wind-tunnel tests. At full-scale Reynolds number, the horizontal-tail stall will occur at a greater negative angle of attack in which case this instability will not be present at speeds below 225 miles per hour (design V_{max} with flaps deflected) for center-of-gravity positions at 20 or 25 percent mean aerodynamic chord. With the aft center of gravity (35 percent M.A.C.) elevator-free stability will exist at full-scale Reynolds number above approximately 160 miles per hour.

2. The control-force gradient in turns will be satisfactory if the elevator effective balance is adjusted to $0.43c_e$.

3. The elevator control is sufficient to land the airplane but the control force required with the forward center of gravity (20 percent M.A.C.) and full flap deflection (50°) is excessive.

4. Take-off attitude cannot be attained at $0.8V_{s_a}$ for any gross weight or center-of-gravity position.

5. The elevator trim tab is adequate at all normal flight speeds.

6. Trim changes due to changes in power or flap setting are satisfactorily small.

7. Directional stability, rudder fixed, is satisfactory. Directional stability, rudder free, will be satisfactory if the rudder effective balance is of the order of $0.34c_r$.

8. The rolling moment due to sideslip is everywhere in the proper direction.

9. The side-force characteristics are satisfactory.

10. The rudder control is adequate to overcome adverse aileron yaw and to maintain straight ground paths in cross-wind take-offs and landings.

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REFERENCES

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2. Hall, Charles F., and Mannes, Robert L.: Longitudinal Characteristics and Aileron Effectiveness of the XB-42 Airplane From High-Speed Wind-Tunnel Tests. NACA CMR, Sept. 1944.

3. Pearson, Henry Adolph: Theoretical Span Loading and Moments of Tapered Wings Produced by Aileron Deflection. NACA TN No. 589, 1937.

TABLE I.- SPECIFICATIONS OF THE AIRPLANE
(All dimensions in feet)

Classification according to Army specifications	Class II
Engines (two - operated side by side for driving dual propeller)	Allison V-1710-93 (E 11)
Gear ratio	0.361
Ratings (each)	bhp/rpm/alt
War emergency power	1500/3000/SL
Take-off power	1325/3000/SL
Military power	1325/3000/SL to 1200/3000/22,500
Maximum rated power	1050/2600/SL to 20,000
Propeller	Hamilton Standard Pusher

	Front (r.h.)	Rear (l.h.)
Blades (3 each)	2C15B1-24	2C15B2-24
Diameter	13.167	13.0

Loading conditions	Gross weight (lb)	l_w	C.G. position* (Percent M.A.C.)
Design	25,000	45	25
Attack	25,000	45	25 to 35
Bomber	34,580	62.2	25.3
Landing	21,500	38.8	36

Tail lengths

25 percent M.A.C. of wing to 26 percent M.A.C. of horizontal tail	19.91
25 percent M.A.C. of wing to 26 percent M.A.C. of vertical tail	20.28

Over-all dimensions

Length	53.64 ft
Height	18.79 ft

*Vertical c.g. location for design condition is 6.8 percent mean aerodynamic chord above fuselage reference line or 13.3 percent mean aerodynamic chord below thrust line (assumed same for other conditions).

TABLE II

GENERAL GEOMETRIC DIMENSIONS OF THE AIRPLANE

(All dimensions in feet)

Item	Wing	Horizontal tail	Vertical tail
Area	554.6	139.28	86.98
Span	70.5	25.0	17.4
Mean Aerodynamic chord	8.56	5.78	5.29
Aspect ratio	8.96	4.49	3.48
Taper ratio	0.333	0.57	--
Geometric twist	2.07° (Washout)	--	--
Dihedral from reference plane	4.0°	0°	--
Incidence from reference plane	0	0	--
Incidence from α_{L_o}	--	1.89°	--
Section profile (constant)	Douglas C-17	Douglas F1	Douglas H
Maximum percent thickness	17.02	13.45	15.55
Root chord	11.83	7.17	6.33
Tip chord	3.94	4.08	4.25*
Percent chord line straight	85	65	60

*Dimension given is for upper vertical. Lower vertical has irregularly shaped bumper on tip.

TABLE III

MOVABLE SURFACES OF THE AIRPLANE

(All dimensions in feet)

Item	Ailerons	Elevators	Rudders Upper	Rudders Lower	Double-slotted flap	Split flap
Area	-----	-----	-----	----	[a]57.98	11.66
Area aft hinge line	26.34	37.50	12.54	8.99	------	------
Span	23.62	20.34	6.25	4.72	29.82	[b]2.88 [c]3.42
Percent balance	0.43	0.45 to 0.50	0.47		------	------
Percent chord[d]	22	35	40		25	34.7
Percent span	33.5	81.5	63		42.3	[b]4.08 [c]4.85
$S_{aft}\ t_{aft}$	31.34	71.6	26.08	16.57	------	------
Control travel	±16°	10° down 25° up	±20°		50° down	40° down
F/HM	[a,f]0.142	[e]0.60	[e]1.008		------	------
Area aft hinge line affected by balance	23.78	34.42	----	----	------	------
Trim tab area	------	3.38	----	----	------	------
Tab travel	------	10° up 20° down	----	----	------	------

[a]Does not include vane.
[b]Measured along hinge line.
[c]Measured along trailing edge.
[d]Ratio of chord aft of hinge line to total surface chord.
[e]These values are for cockpit control motions which allow for cable stretch.
[f]Wheel moment/hinge moment.

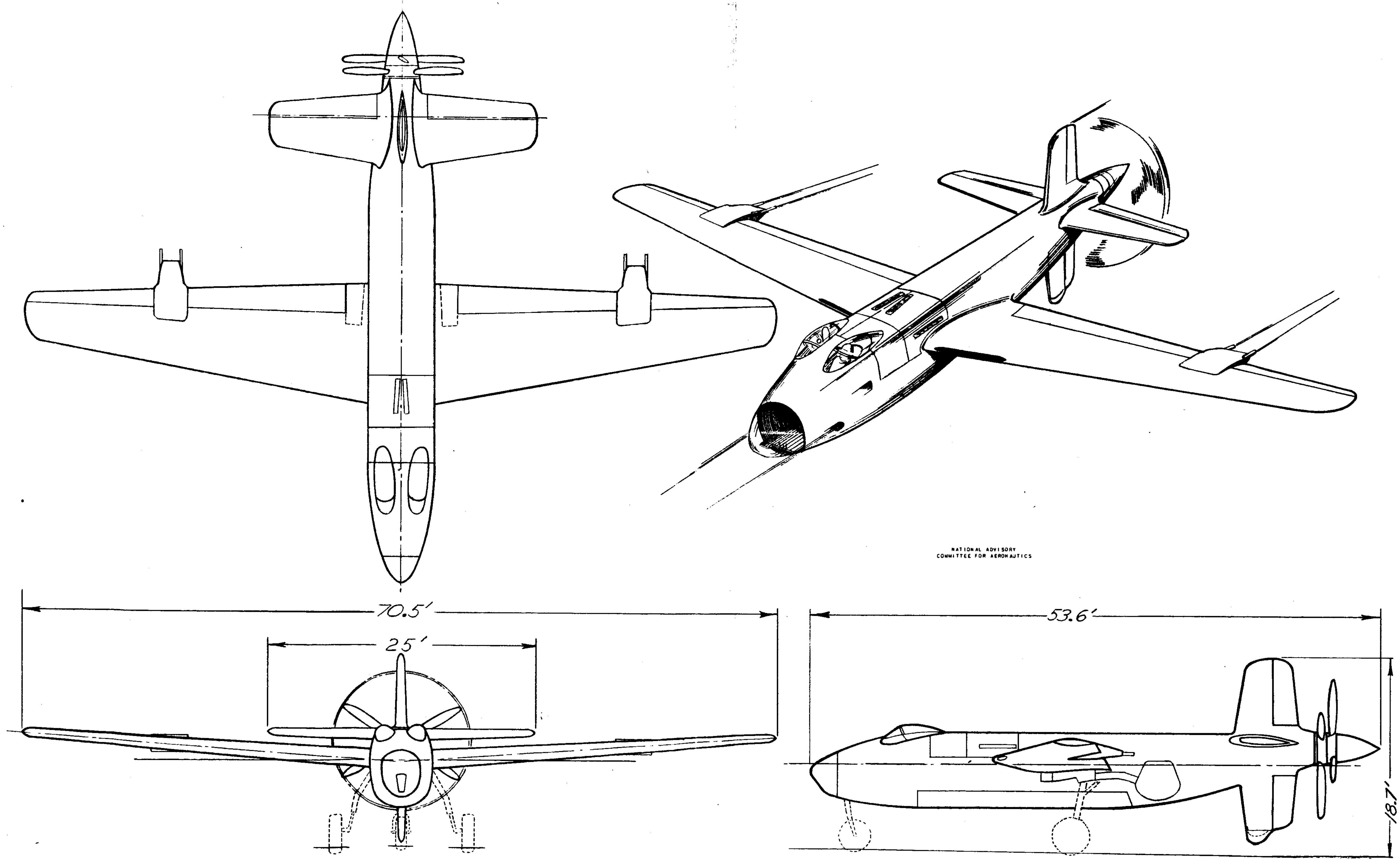
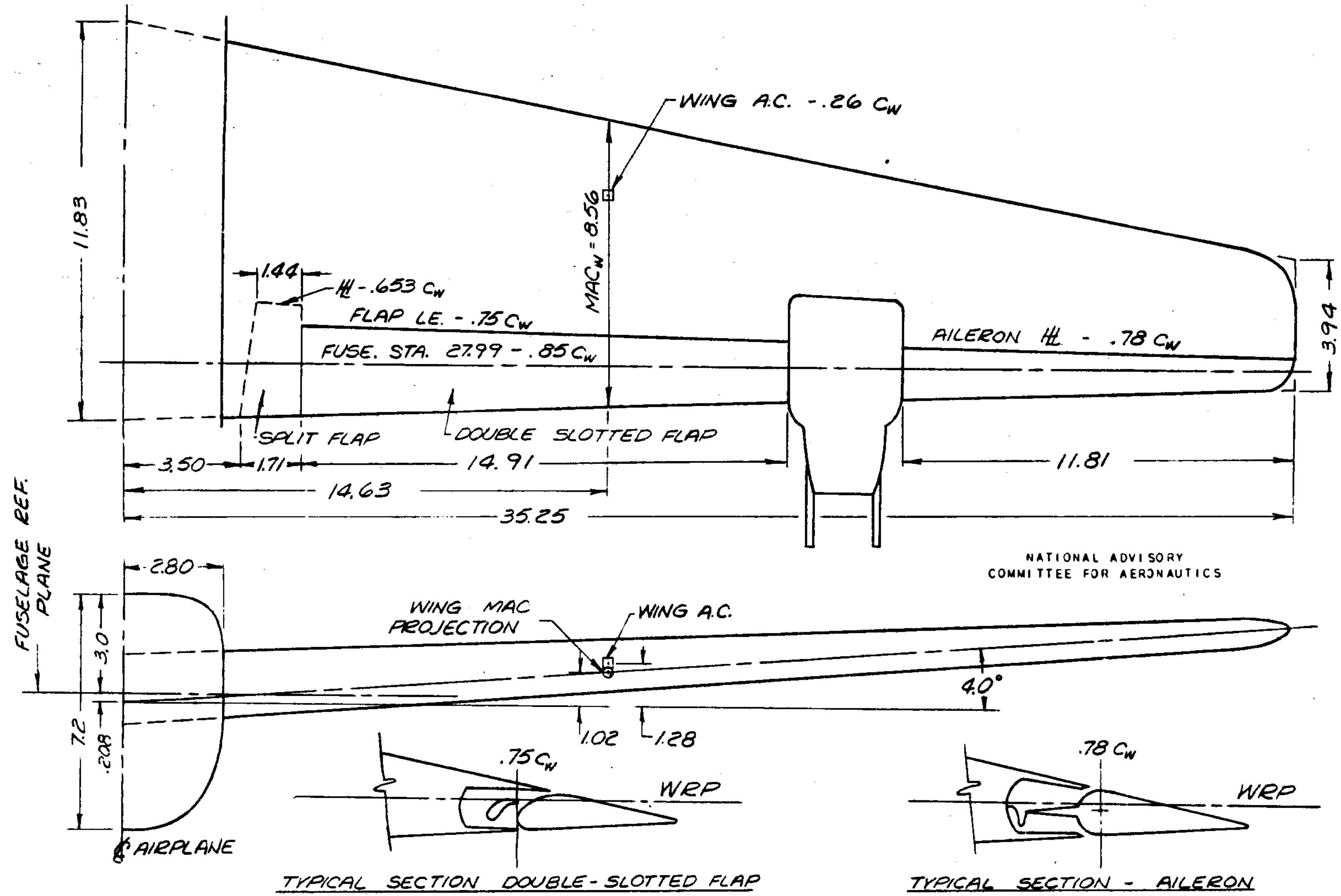
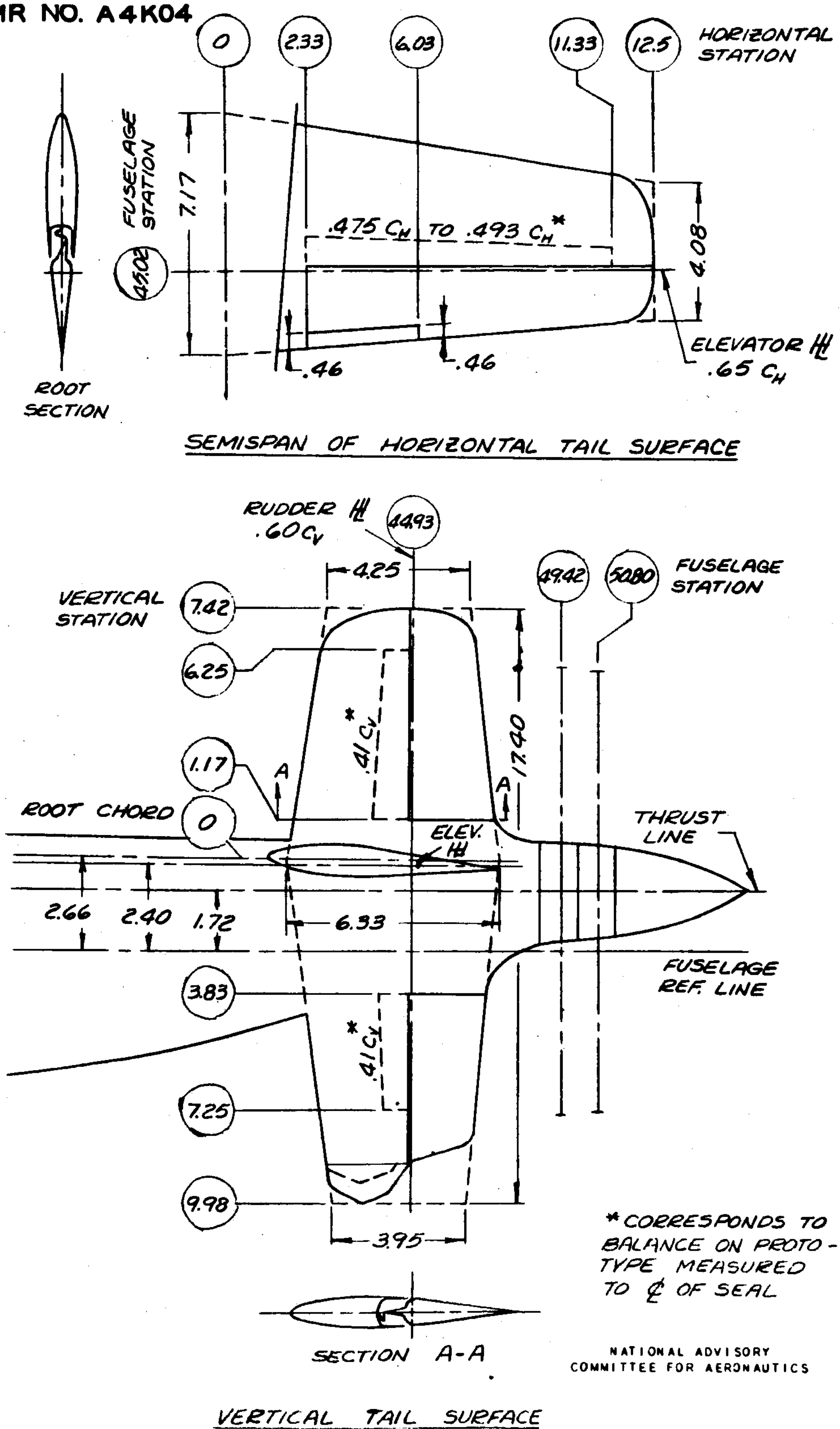


FIGURE 1. THREE-VIEW DRAWING OF THE AIRPLANE.



ALL DIMENSIONS FEET FULL SCALE 1/40 SCALE

FIGURE 2.- LINE DIAGRAM OF WING SEMISPAN OF THE AIRPLANE.



ALL DIMENSIONS FEET FULL SCALE
1/40 SCALE

FIGURE 3- LINE DIAGRAM OF EMPENNAGE OF THE AIRPLANE.

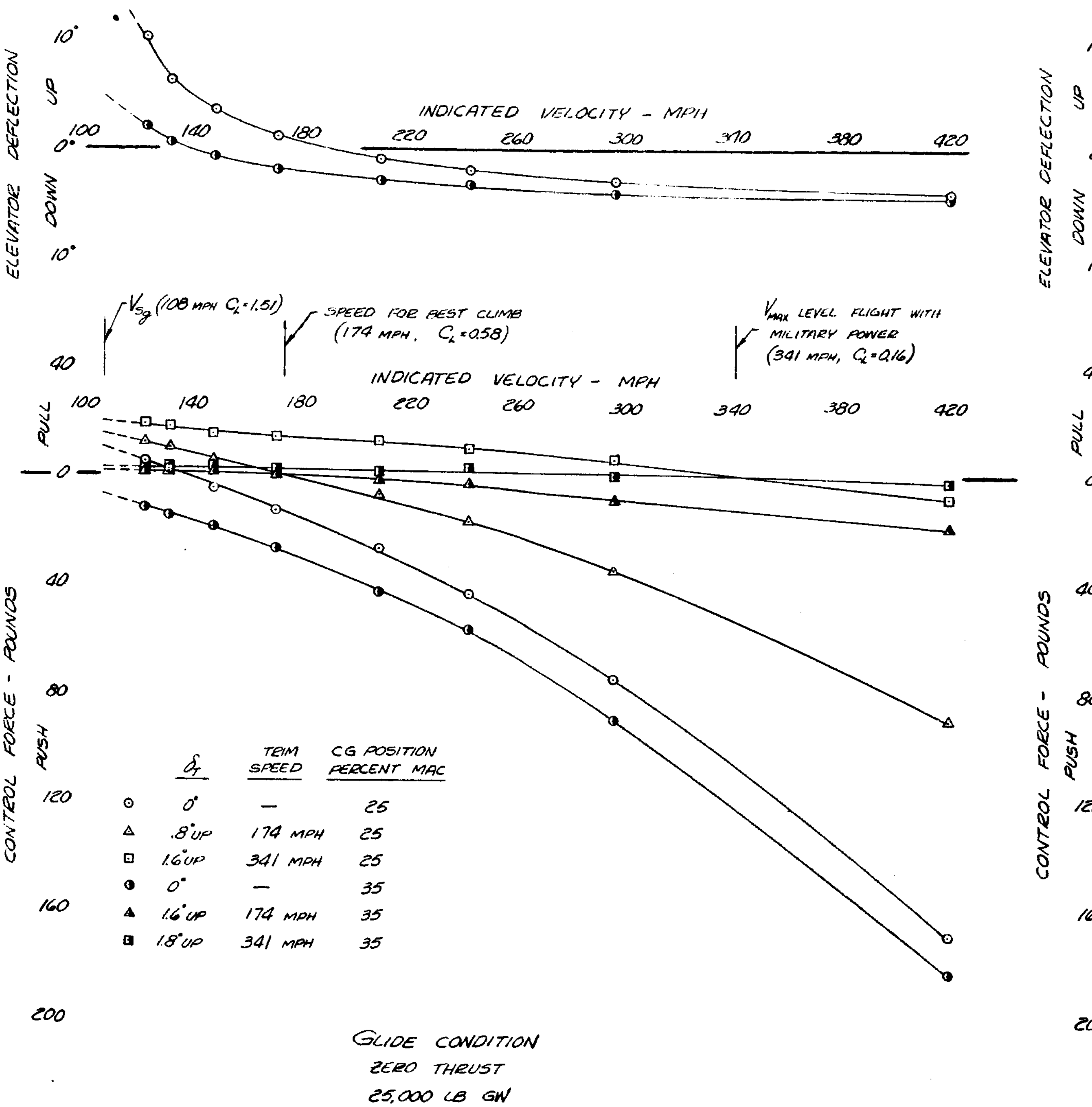

ELEVATOR DEFLECTION
UP
DOWN
10°
0°
10°
INDICATED VELOCITY - MPH
100
140
180
220
260
300
340
380
420
V_{Sg} (108 MPH C_L = 1.51)
SPEED FOR BEST CLIMB (174 MPH, C_L = 0.58)
V_{MAX} LEVEL FLIGHT WITH MILITARY POWER (341 MPH, C_L = 0.16)
INDICATED VELOCITY - MPH
CONTROL FORCE - POUNDS
PULL
PUSH
40
0
40
80
120
160
200
δ_T | TRIM SPEED | CG POSITION PERCENT MAC
0° | — | 25
.8° UP | 174 MPH | 25
1.6° UP | 341 MPH | 25
0° | — | 35
1.6° UP | 174 MPH | 35
1.8° UP | 341 MPH | 35
GLIDE CONDITION
ZERO THRUST
25,000 LB GW

FIGURE 4 A

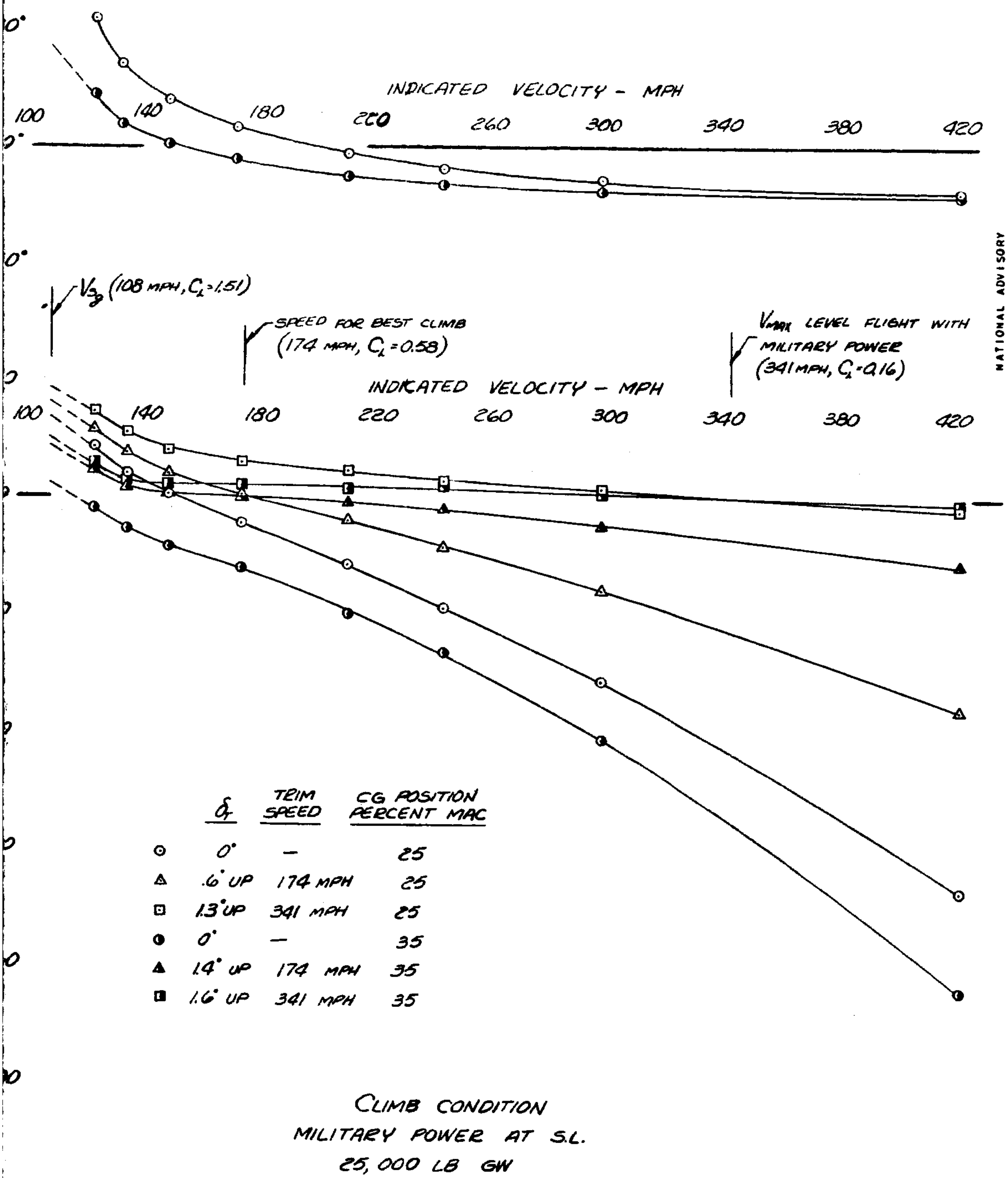


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FIGURE 4.13 ELEVATOR CONTROL CHARACTERISTICS IN STEADY FLIGHT AT SEA LEVEL. GLIDE AND CLIMB CONDITIONS.

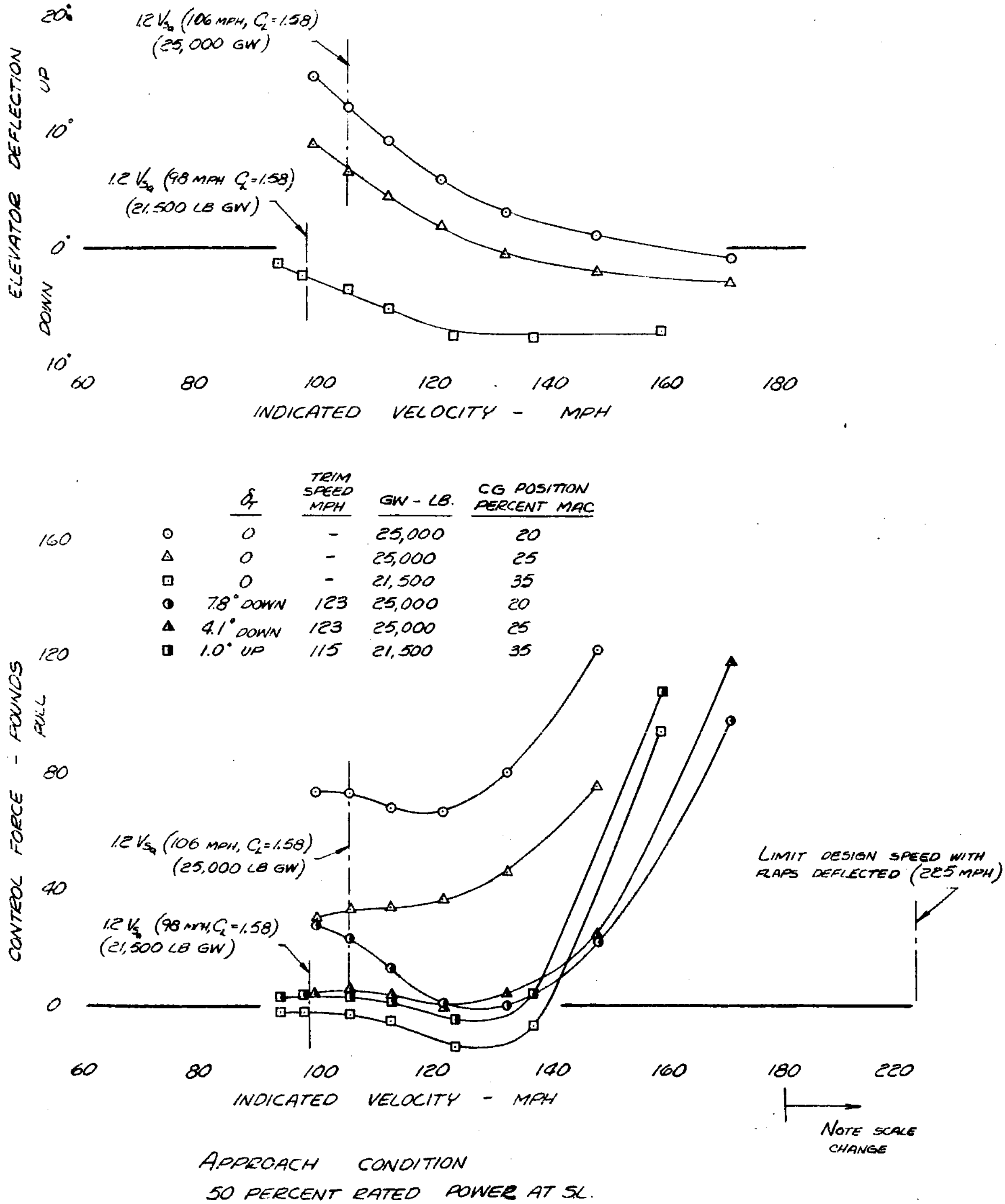

1.2 V_{s_a} (106 MPH, C_L=1.58) (25,000 GW)
1.2 V_{s_a} (98 MPH C_L=1.58) (21,500 LB GW)
20°
10°
0°
10°
UP
DOWN
ELEVATOR DEFLECTION
60
80
100
120
140
160
180
INDICATED VELOCITY - MPH
δ_T
TRIM SPEED MPH
GW - LB.
CG POSITION PERCENT MAC
0 - 25,000 20
0 - 25,000 25
0 - 21,500 35
7.8° DOWN 123 25,000 20
4.1° DOWN 123 25,000 25
1.0° UP 115 21,500 35
160
120
80
40
0
CONTROL FORCE - POUNDS
PULL
1.2 V_{s_a} (106 MPH, C_L=1.58) (25,000 LB GW)
1.2 V_{s_a} (98 MPH, C_L=1.58) (21,500 LB GW)
LIMIT DESIGN SPEED WITH FLAPS DEFLECTED (225 MPH)
60
80
100
120
140
160
180
220
INDICATED VELOCITY - MPH
NOTE SCALE CHANGE
APPROACH CONDITION
50 PERCENT RATED POWER AT SL.

FIGURE 5 A

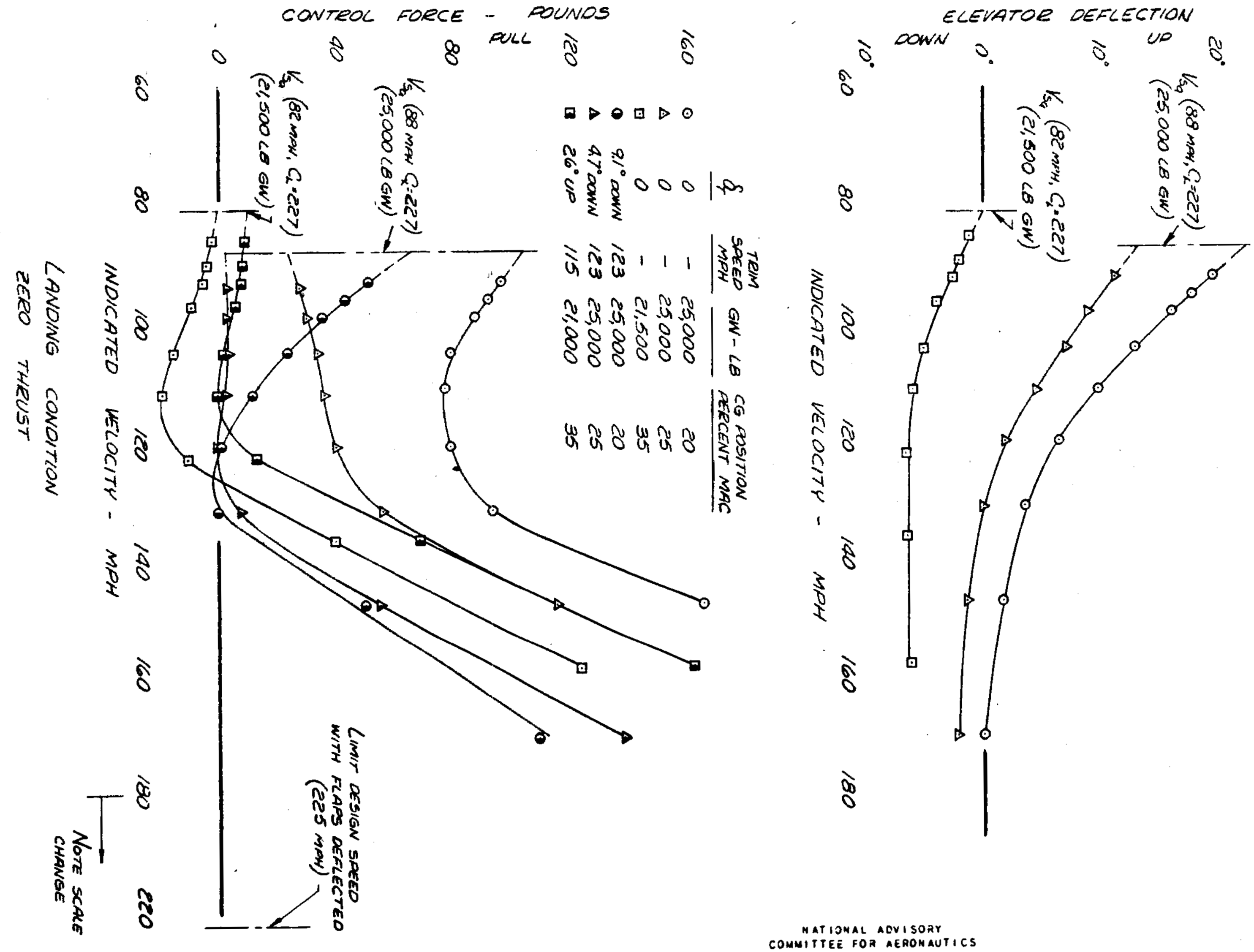
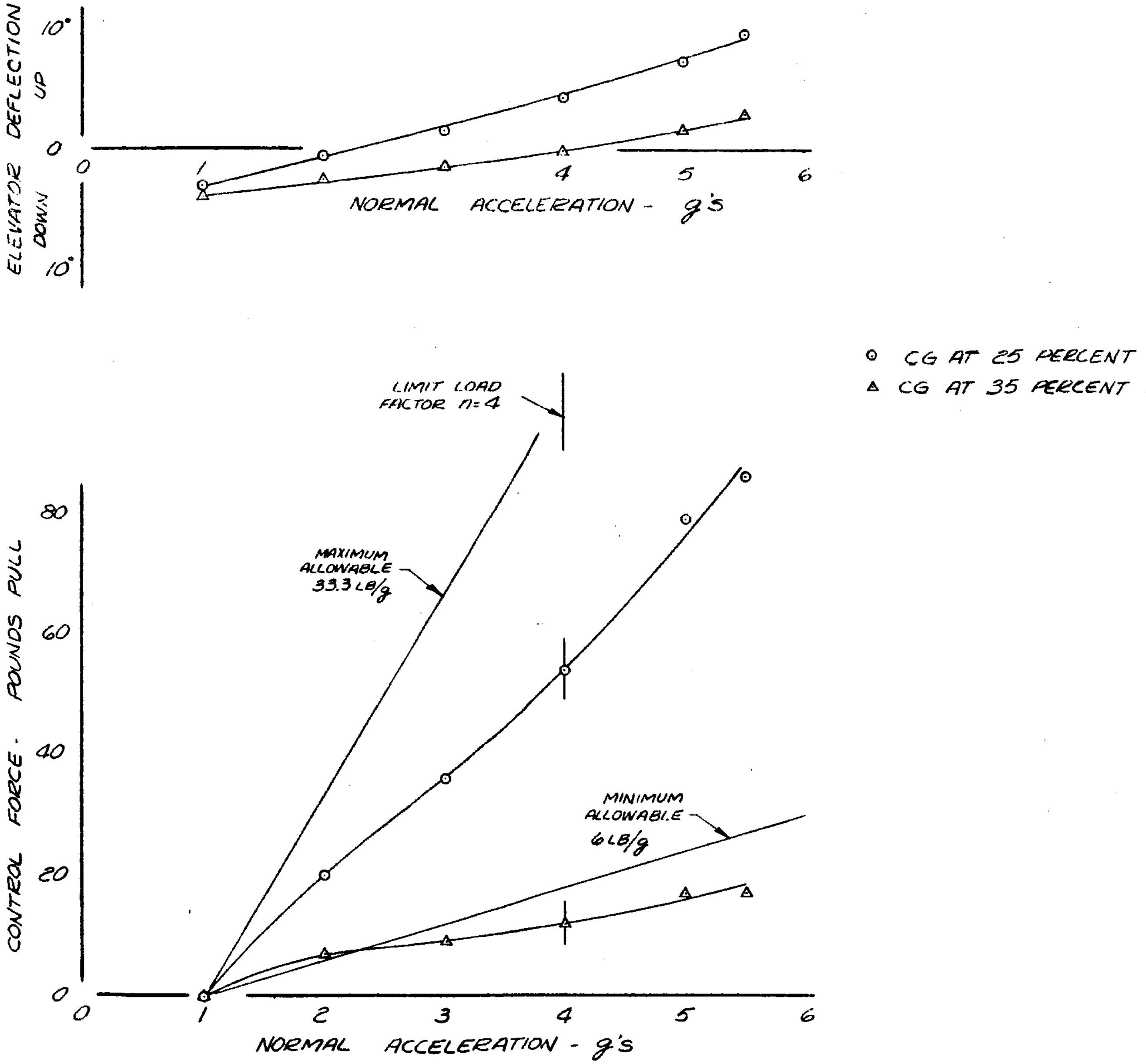


FIGURE 5.B ELEVATOR CONTROL CHARACTERISTICS IN STEADY FLIGHT AT SEA LEVEL.
APPROACH AND LANDING CONDITIONS.

A78



25,000 POUNDS GROSS WEIGHT

FIGURE 6A

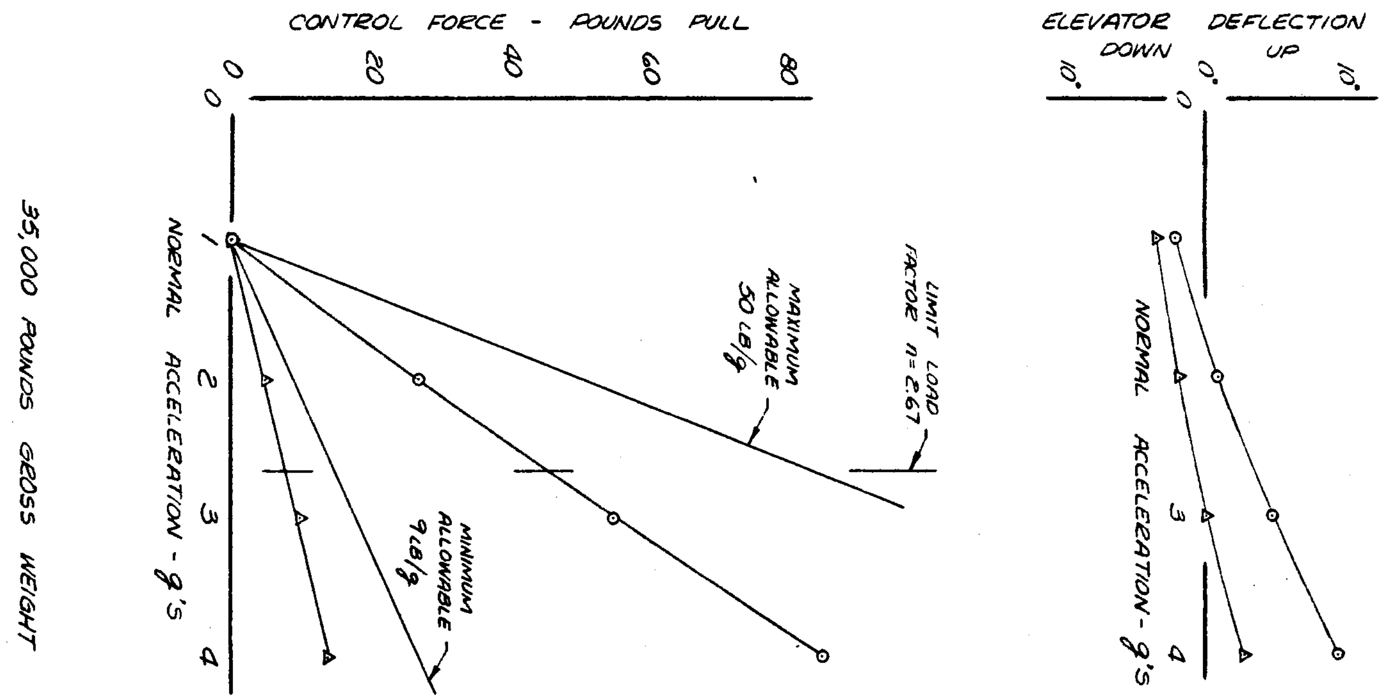


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FIGURE 6.B ELEVATOR CONTROL CHARACTERISTICS IN STEADY TURNING FLIGHT AT 10,000 FT.
FLAPS AND GEAR RETRACTED, PROPELLER OPERATING AT ZERO THRUST ($T_c = 0$),
V_T = 345 MPH AT 10,000 FT.

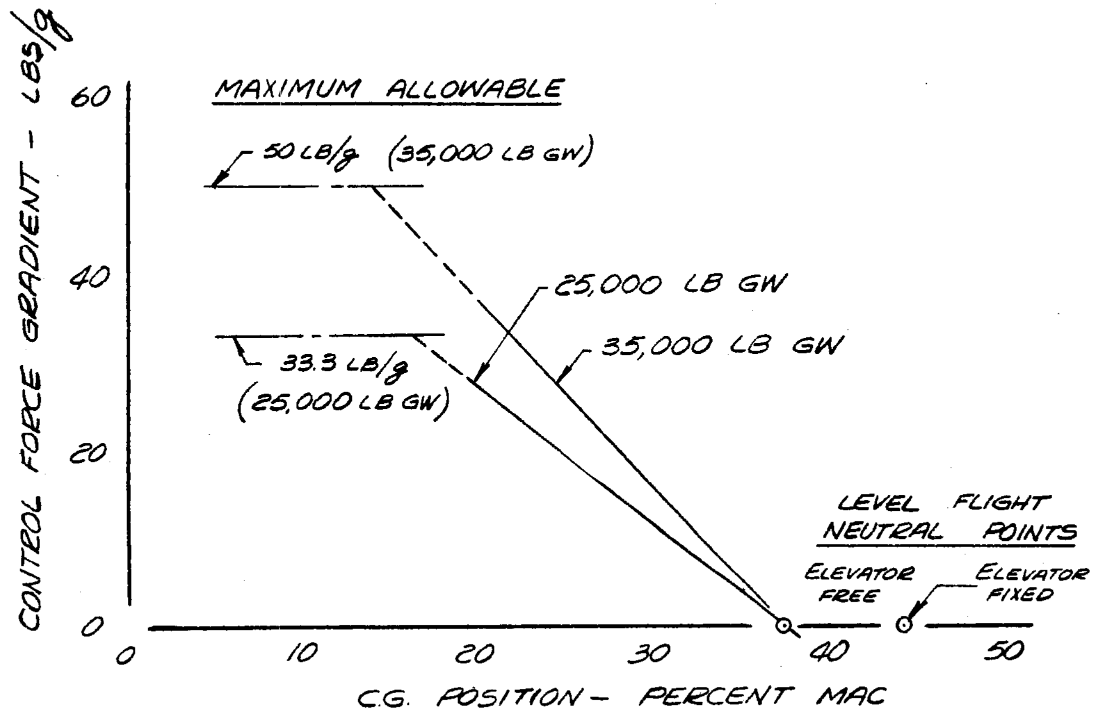

FIGURE 7. VARIATION OF ELEVATOR CONTROL FORCE GRADIENTS IN ACCELERATED FLIGHT WITH C.G. POSITION. FLAPS AND GEAR RETRACTED, PROPELLER OPERATING AT ZERO THRUST ($T_c = 0$). V_T = 345 MPH AT 10,000 FT.

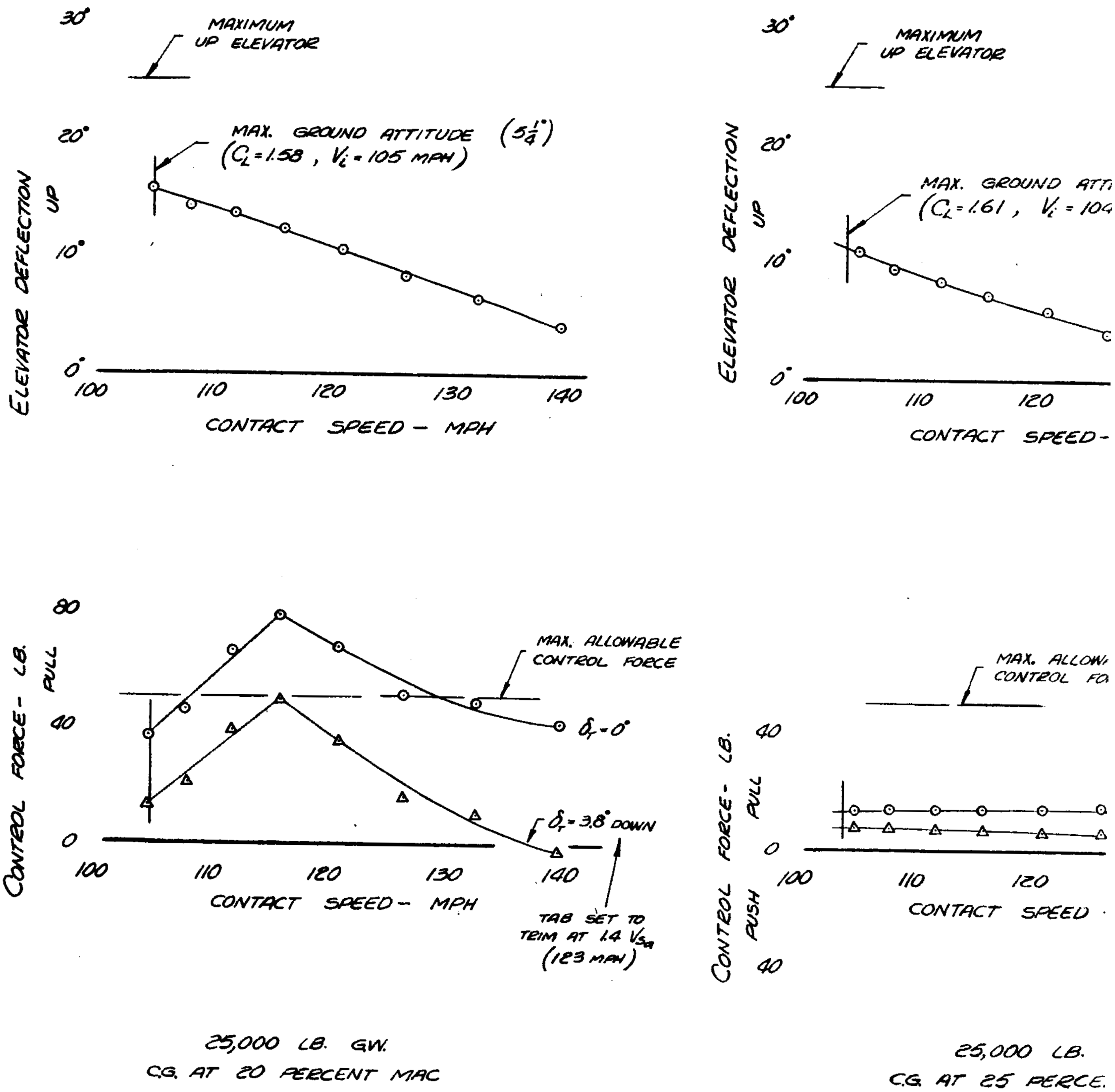
30°
MAXIMUM UP ELEVATOR
20°
MAX. GROUND ATTITUDE (5¼°)
(C_L=1.58 , V_i = 105 MPH)
10°
0°
100
110
120
130
140
ELEVATOR DEFLECTION
UP
CONTACT SPEED – MPH
80
40
0
MAX. ALLOWABLE CONTROL FORCE
δ_T = 0°
δ_T = 3.8° DOWN
TAB SET TO TRIM AT 1.4 V_{S_a} (123 MPH)
CONTROL FORCE – LB.
PULL
25,000 LB. GW.
C.G. AT 20 PERCENT MAC
MAX. GROUND ATT
(C_L=1.61 , V_i = 10
PUSH
25,000 LB.
C.G. AT 25 PERCE

FIGURE 8A

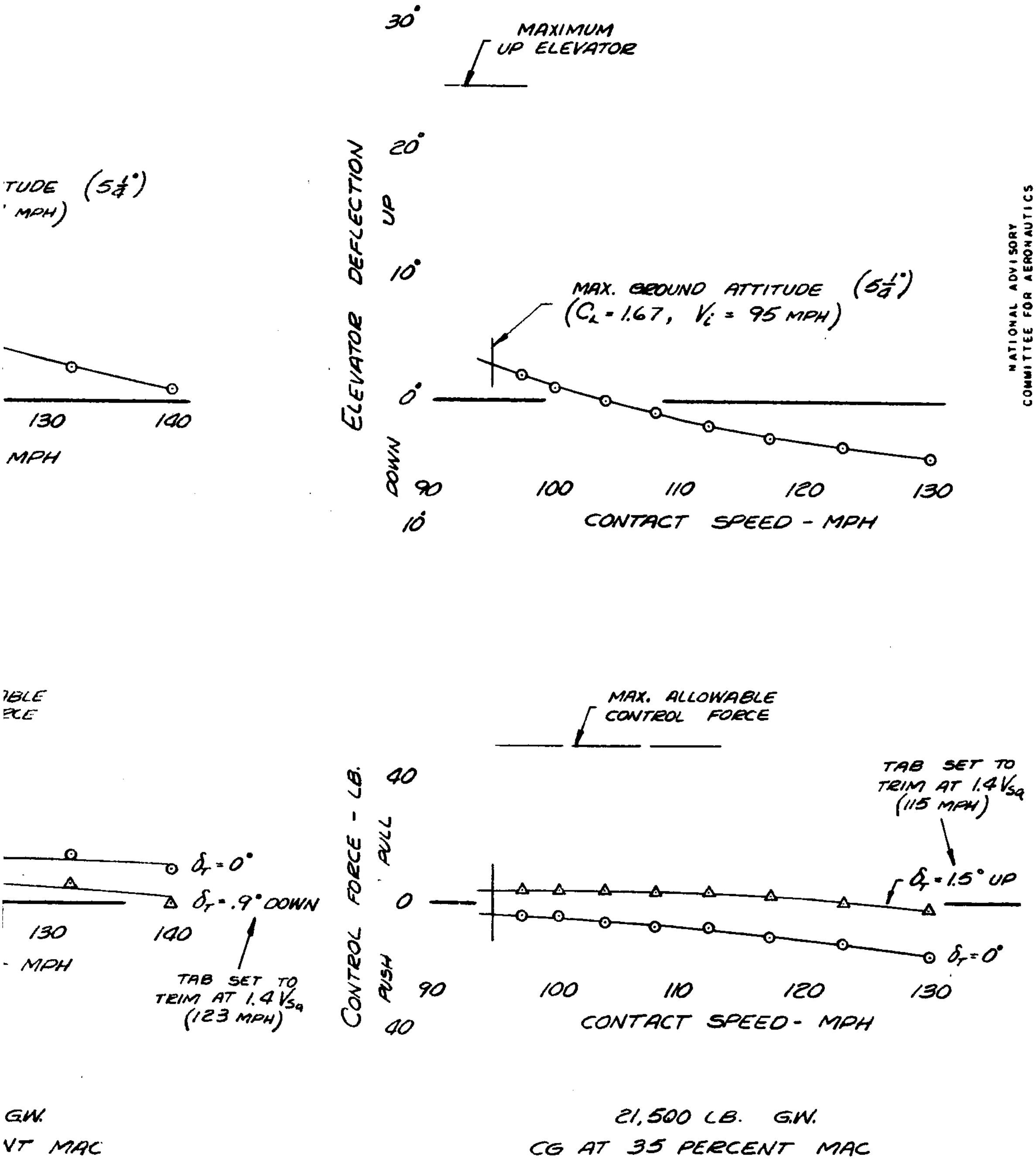

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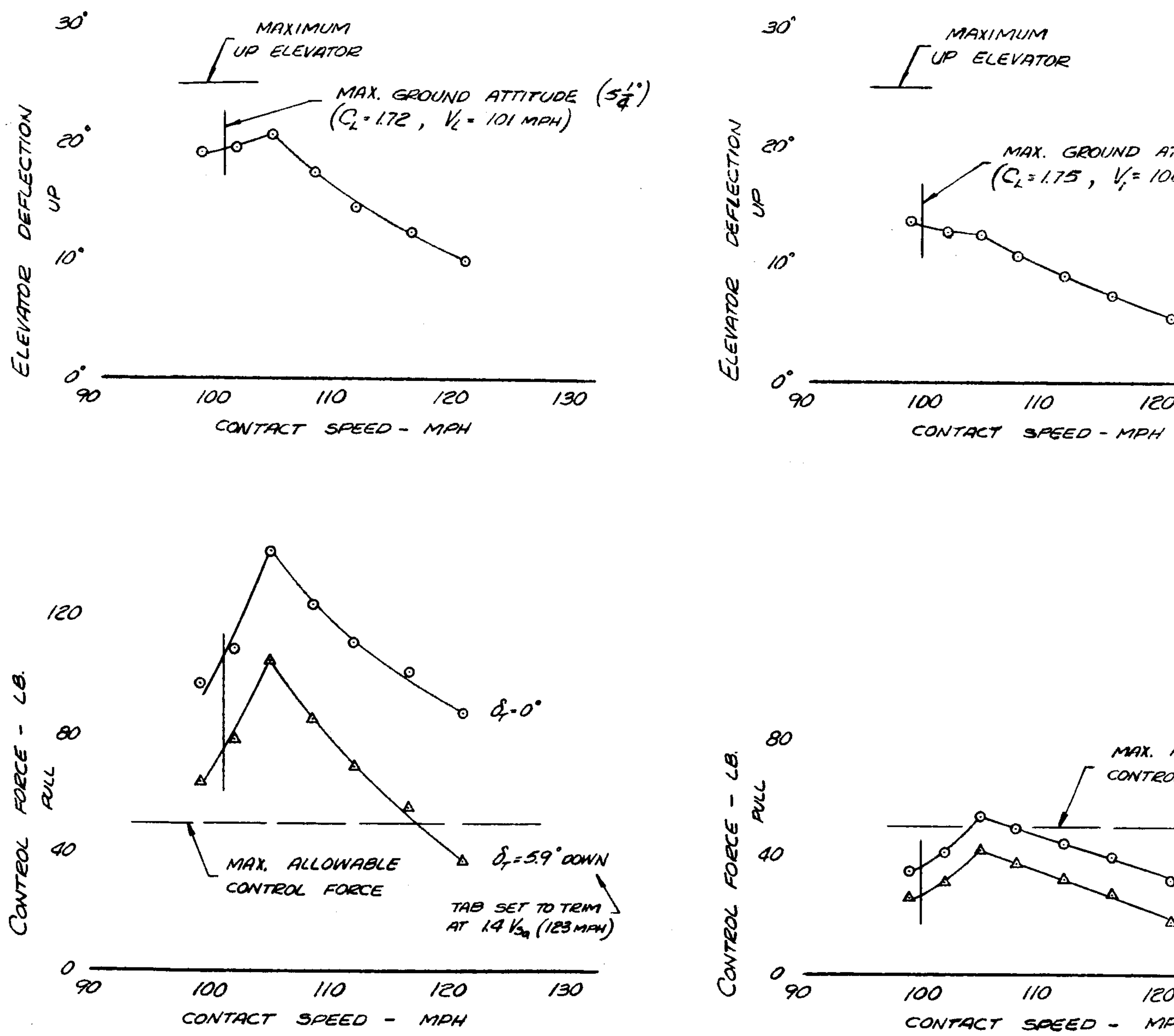
FIGURE 8B ELEVATOR CONTROL CHARACTERISTICS IN LANDING. FLAPS AT 30°, GEAR EXTENDED, PROPELLER OPERATING AT ZERO THRUST (T_c = 0).



25,000 LB. GW.
C.G. AT 20 PERCENT MAC

25,000 LB. GW.
C.G. AT 25 PERCENT M

FIGURE 9A

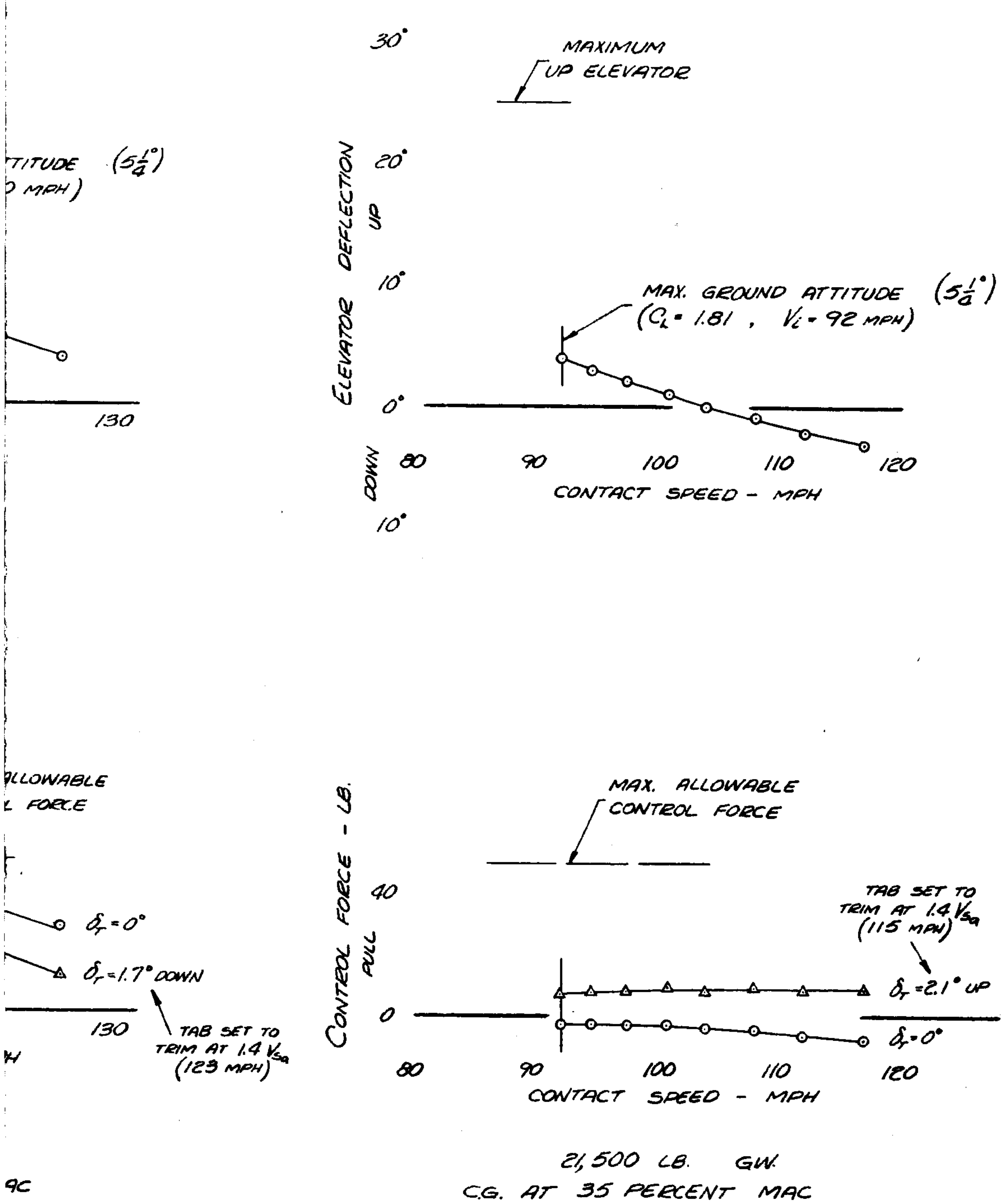


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FIGURE 9.B ELEVATOR CONTROL CHARACTERISTICS IN LANDING FLAPS AT 40°, GEAR EXTENDED, PROPELLER OPERATING AT ZERO THRUST ($T_c \approx 0$)

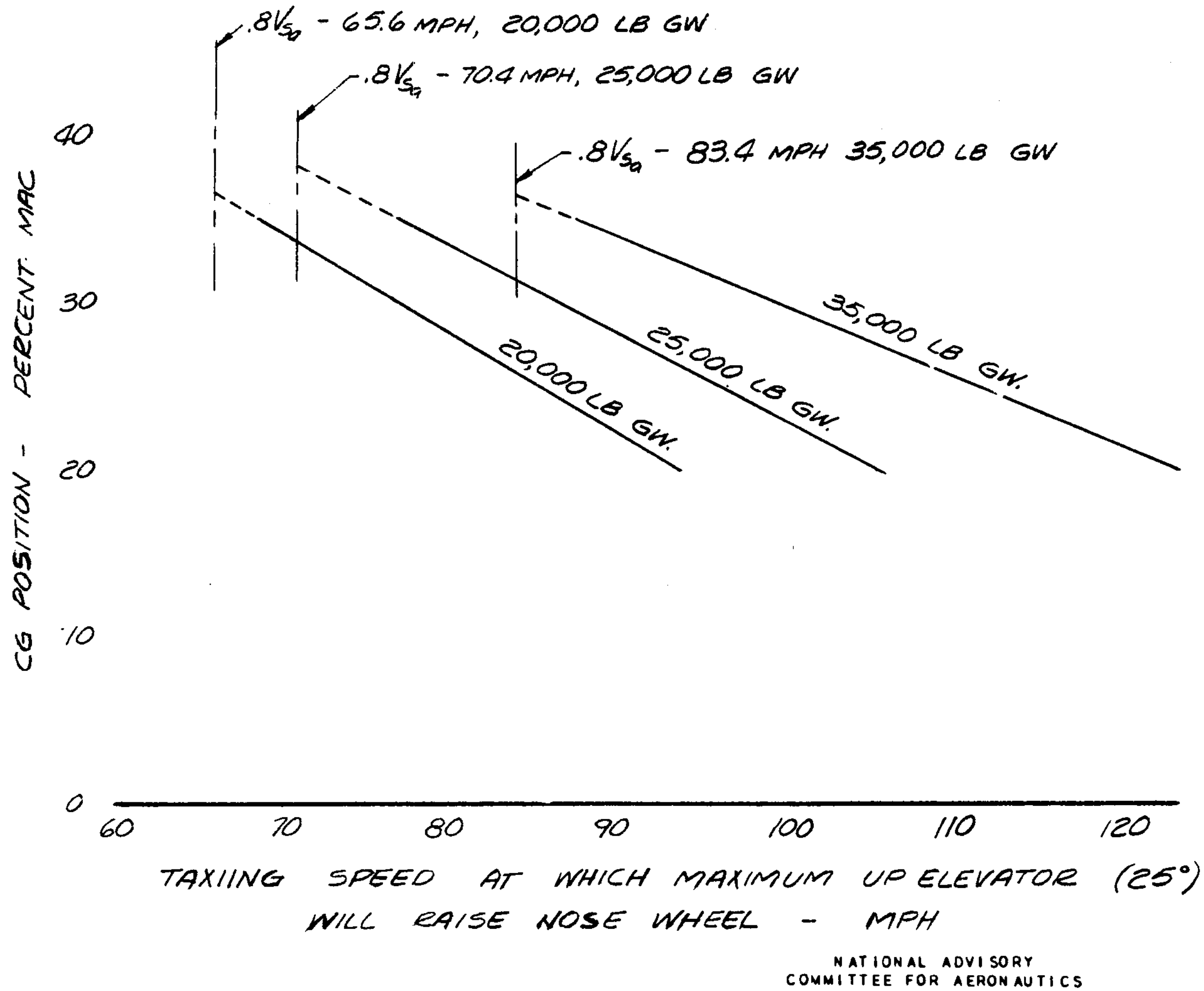

FIGURE 10. TAKE-OFF CHARACTERISTICS OF THE AIRPLANE. FLAPS AT 30°, GEAR EXTENDED PROPELLER OPERATING AT TAKE-OFF POWER AT SL.

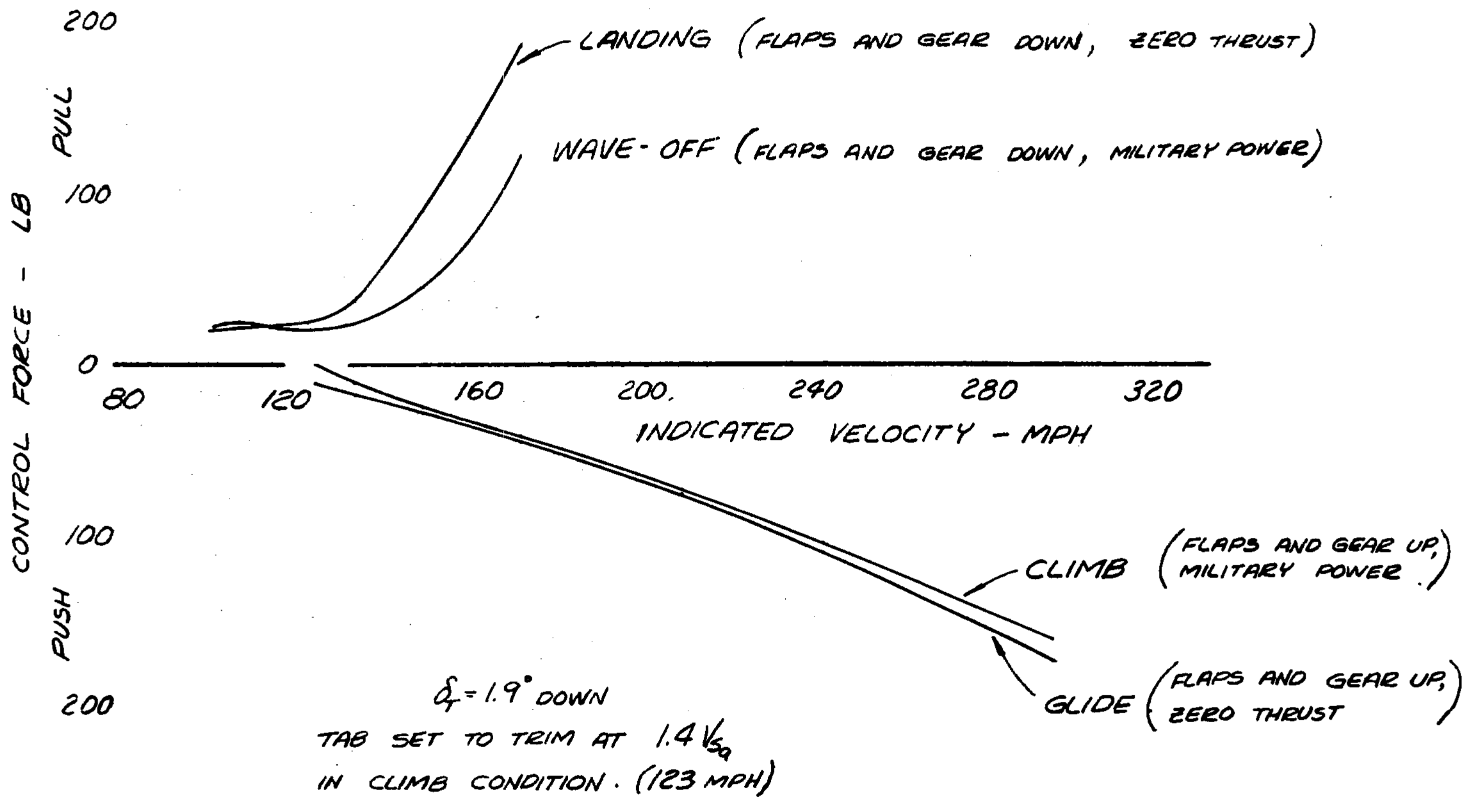

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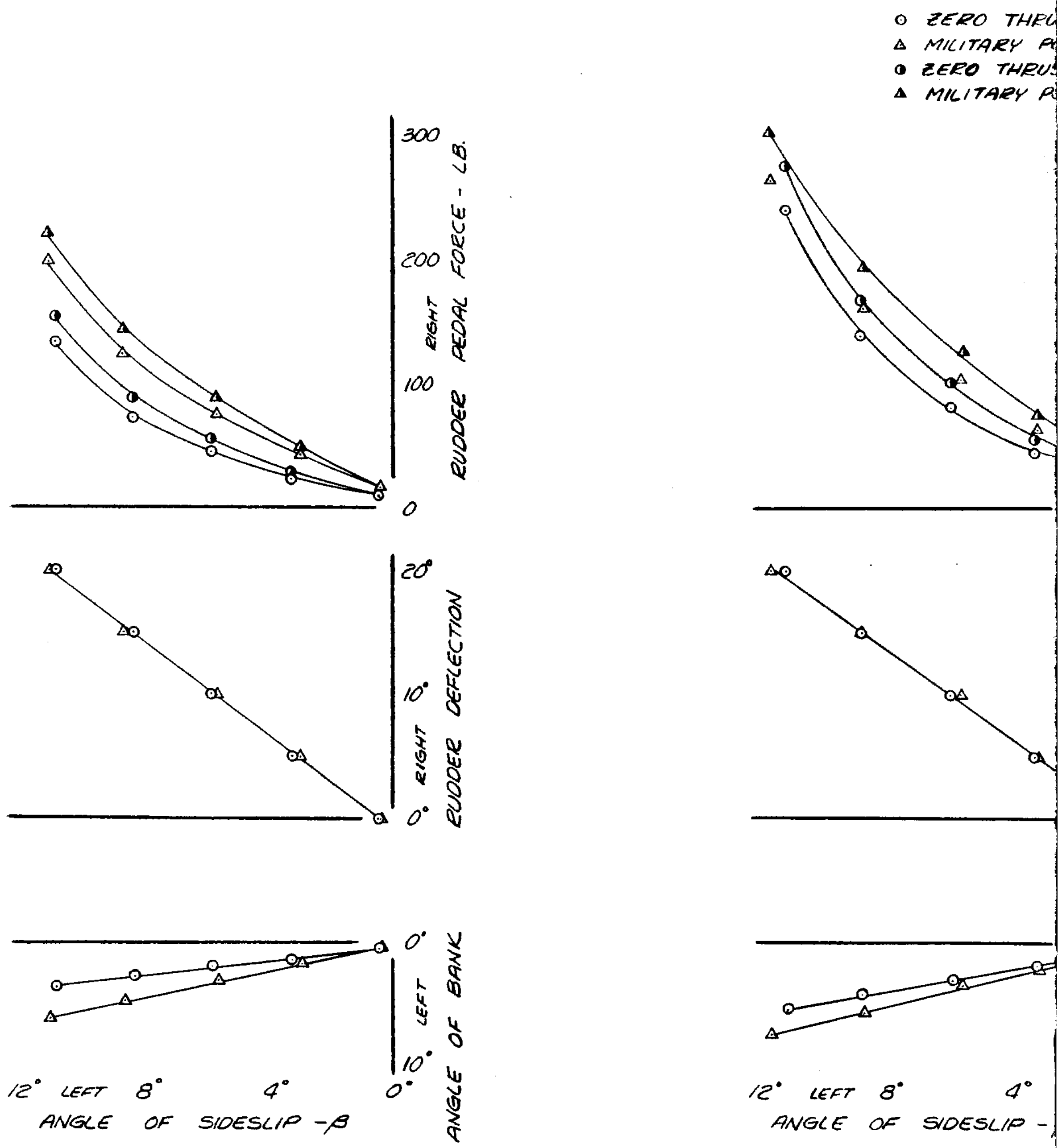
FIGURE 11. TRIM CHANGES DUE TO POWER AND FLAPS. TRIM TAB SET TO TRIM AT $1.4 V_{s_a}$ (123 MPH) IN CLIMB CONDITION. G.W. = 25,000 LB. C.G. AT 25 PERCENT MAC



1.2 V_{s_g} - 129 MPH

CLIMB CONDITION - 174

FIGURE 12A

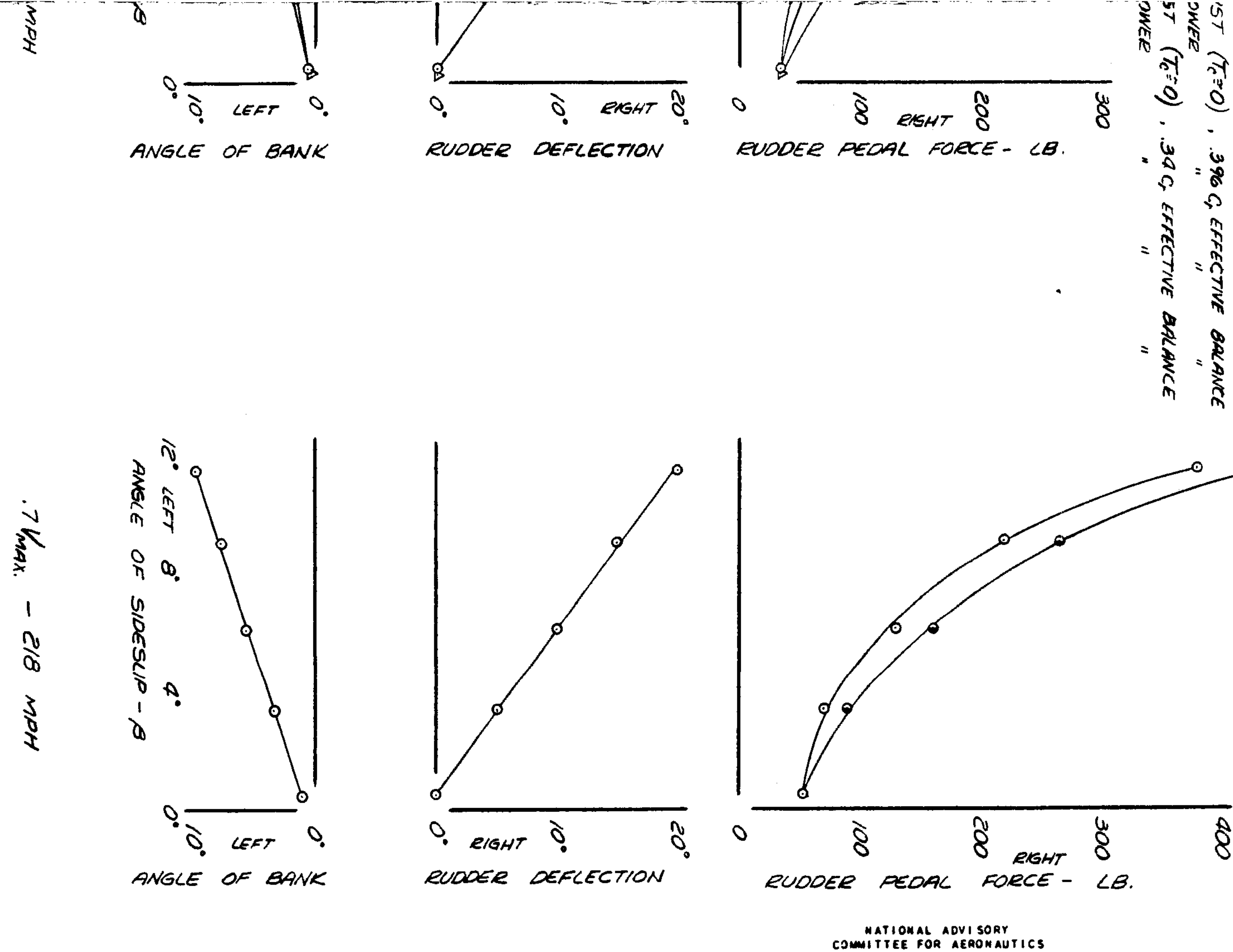
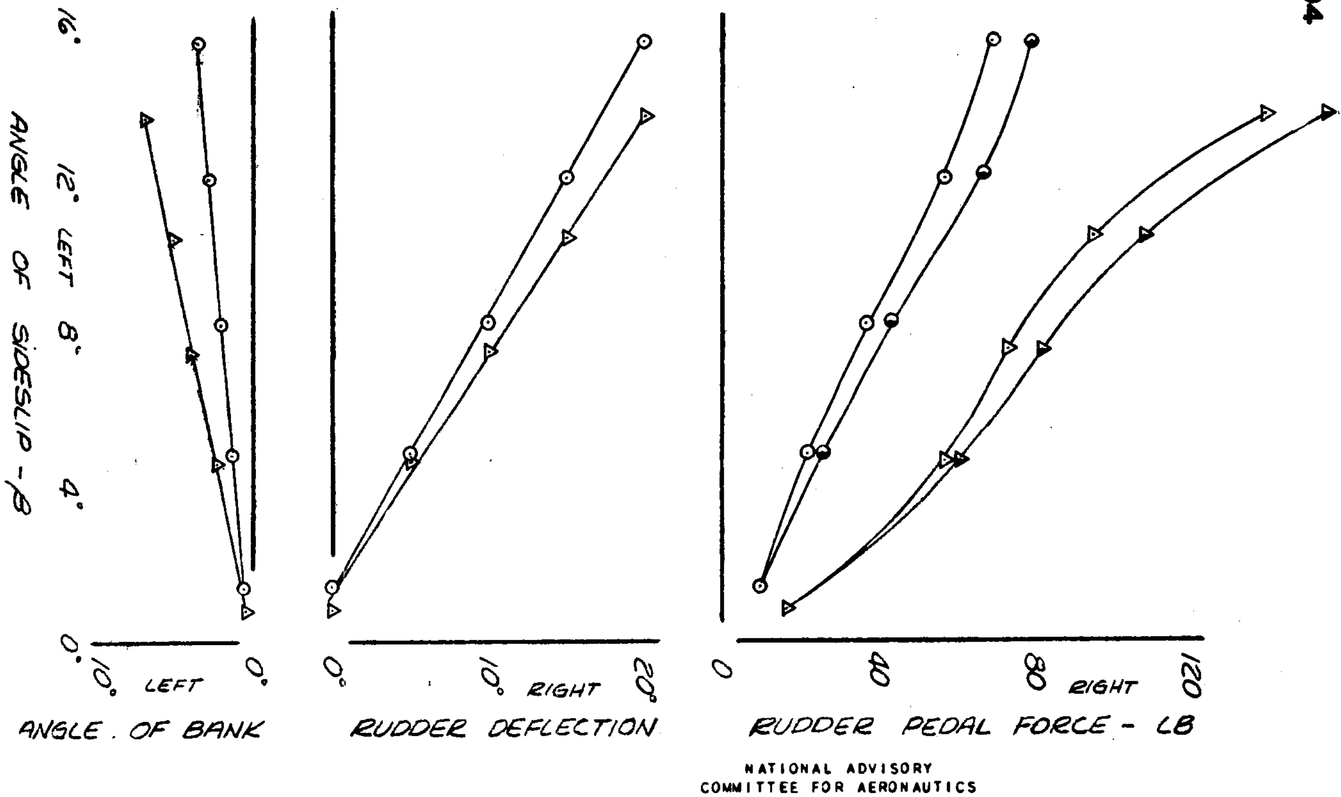


FIGURE 12 B RUDDER CONTROL CHARACTERISTICS IN STEADY SIDESLIPS AT SEA LEVEL.
FLAPS AND GEAR RETRACTED, GW = 25,000 LB., CG AT 25 PERCENT MAC.



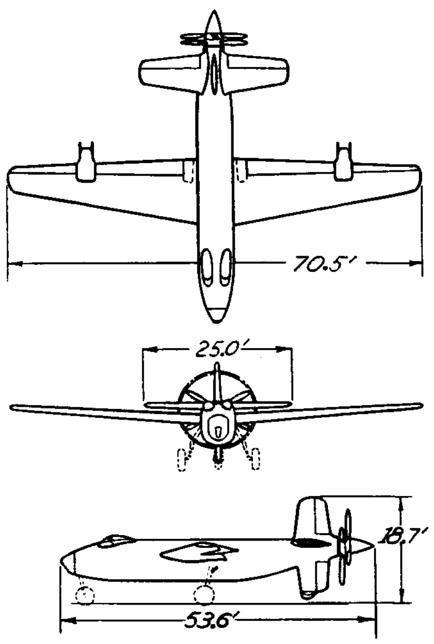
○ ZERO THRUST ($T_c \cong 0$) . .396 C_f EFFECTIVE BALANCE
△ MILITARY POWER " " "
◐ ZERO THRUST ($T_c \cong 0$), .34 C_f EFFECTIVE BALANCE
▲ MILITARY POWER " " "

1.2 V_{sg} - 106 MPH

FIGURE 13. RUDDER CONTROL CHARACTERISTICS IN STEADY SIDESLIPS AT SEA LEVEL. FLAPS AT 50°, GEAR EXTENDED, 25,000 LB. GW., CG AT 25 PERCENT MAC.

SUMMARY OF HANDLING CHARACTERISTICS
NACA WIND TUNNEL PREDICTION
7 x 10 FT. WIND TUNNEL.

AIRPLANE AS TESTED



PERTINENT DETAILS:

Loading conditions:	Gross weight	c.g. range (% M.A.C.)
Design	25,000	25
Attack	25,000	25 to 36
Bomber	34,580	25.3
Landing	21,500	36
Span	70.5 feet	
Mean aerodynamic chord	8.56 feet	
Wing area	554.6 square feet	
Airfoil	Douglas 8-17	
Type flap	double slotted	
Control-surface deflections:		
Elevators	25° up, 10° down	
Ailerons	± 18°	
Rudders	± 20°	
Type of control-surface balance	sealed, internal	
Engines (2) (operated side by side for driving dual propellers)	Allison V-1710-93 (E 11)	
Gear ratio	0.361	
Ratings (each)	bhp/rpm/alt.	
War emergency power	1500/3000/S.L.	
Take-off power	1325/3000/S.L.	
Military	1325/3000/S.L. to 1200/3000/22,500	
Maximum rated power	1050/2600/S.L. to 20,000	
Propeller:	Hamilton Standard pusher	
	Front (RH)	Rear (LH)
Blades (3 each)	2C1561-24	2C1582-24
Diameter	13.167 feet	13.0 feet
Wheel-type control	13-inch diameter	

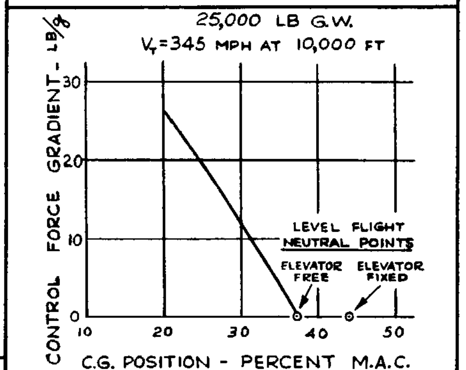
LONGITUDINAL

NEUTRAL POINT LOCATION:

Where the neutral point changed with C_L, the data are presented in coordinate form.

Condition	Stability	Neutral point	C_L Range
Landing	Stick fixed	0.446 M.A.C.	1.4 - 2.0
	Stick free	.365	1.4 - 2.0
Approach	Stick fixed	.460	1.4 - 2.0
	Stick free	.385	1.4 - 2.0
Glide	Stick fixed	.440	0 - .5
	Stick free	.373	0 - .5
Climb	Stick fixed	.440, .451, .460	0, .25, .50
	Stick free	.373, .379, .386	0, .35, .70
Wave-off	Stick fixed	.477	1.4 - 2.0
	Stick free	.410	1.4 - 2.0

MANEUVERING CHARACTERISTICS:



TRIM CHANGES DUE TO POWER OR FLAP VARIATIONS:

Power	Flaps and gear	V_i (mph)	Control force (lb)
Military	Up	123	0 Trim condition
Zero thrust	Up	123	11 push
Take-off	Down	123	32 pull
Zero thrust	Down	123	5 push

25,000 pounds gross weight, center of gravity at 0.25 M.A.C. elevator tab setting 1.9° down

CONTROL IN TAKE-OFF AND LANDING:

Adequate control to land but control forces are excessive with full forward center of gravity (0.20 M.A.C.). Nose wheel cannot be raised at $0.8V_{s_a}$ for any gross weight or center of gravity.

LONGITUDINAL TRIMMING DEVICE:

Adequate for both fore and aft center of gravity throughout the speed range.

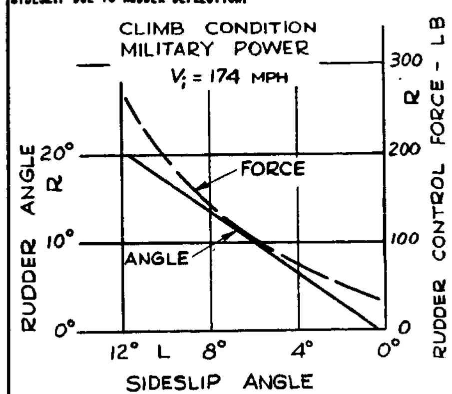
LATERAL

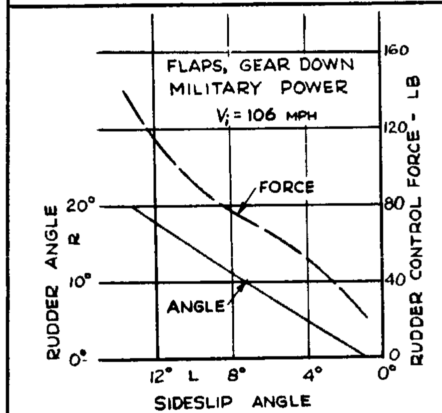
ROLLING MOMENT DUE TO SIDESLIP:

Positive dihedral effect exists for all flight conditions.

DIRECTIONAL

SIDESLIP DUE TO RUDDER DEFLECTION:





RUDDER CONTROL:

Adequate to overcome adverse aileron yaw and to maintain straight ground paths in cross-wind take-offs and landings with less than 180 pounds pedal force.

STALLING

STRAIGHT FLIGHT:

The stall pattern of the test airplane is good. With flaps up, the stall progresses out from about 25 percent of the semispan. With flaps deflected, that part of the wing over the ailerons remains completely unstalled when that part of the wing over the flaps is completely stalled. The flow at the juncture of the wing and fuselage is perfectly smooth for any flap position up to $C_{L_{max}}$

The trim $C_{L_{max}}$ measured on the model at a Reynolds number of 1,200,000 is given in the following table:

Flap position	$C_{L_{max}}$
f^0	1.15
f^{30}	1.78
f^{30} f^{40}	1.83
f^{50}	1.86
f^{50} f^{40}	1.98

NOTE: f^{40} indicates gear and split flap are deflected.

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FIGURE 14. - SUMMARY OF ESTIMATED HANDLING CHARACTERISTICS OF THE AIRPLANE