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<th>MODEL NO.</th>
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INTRODUCTION

This report presents the aerodynamic data necessary for a complete determination of the structural flight loads on the airplane. Included are the effects of compressibility, power and configuration changes such as pylon tanks on, ramp deflected, flaps extended, etc. Data specifically for this airplane are obtained from the results of extensive low- and high-speed wind tunnel tests conducted by Lockheed. Additional data as necessary are from NACA publications and other Lockheed investigations.

Many configurations were tested in the development of this airplane. Some were made necessary by the type of test to be run; others by design changes or by stability and/or control requirements. In each case, the data most applicable have been selected for presentation in this report.
GENERAL

The C-130 is primarily a military cargo airplane of conventional arrangement. It is powered by four propeller-driving Allison T-56 gas turbine engines. The airplane is a high-wing design with standard empennage. It incorporates hydraulically boosted primary controls, Lockheed-Fowler flaps for landing and take-off. Loading and unloading are accomplished by means of a ramp at the rear of the fuselage which can be opened for airdrops. Paratroops may be dropped from personnel doors at the rear of the fuselage and are protected from air blast by means of shields immediately forward of the doors. Main landing gear is contained in fairings on each side of the fuselage.

Basic data applicable to the airplane and describing its geometric characteristics are presented on the following pages. A dimensioned three-view drawing of the airplane is presented in Fig. 1. Physical parameters necessary to the purpose of this report are summarized in Table I, along with those items of performance which affect structural flight loads determination. In Fig. 2 are presented curves showing military and normal rated power available at all speeds for several altitudes. Fig. 3 presents curves which summarize design speeds at all altitudes in level flight and dive.
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<td>Span, b</td>
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<td>Taper Ratio, λ</td>
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<tr>
<td>Tip, i_t</td>
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<td>Parameter</td>
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<td>Tab Deflection Limits, $\delta_{at}$</td>
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<td>Maximum Deflection, $\delta_f$</td>
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<td><strong>Horizontal Tail</strong></td>
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<td>Span, $b_t$</td>
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<td>Root (Sta. 0)</td>
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<td>Tip (Sta. 313)</td>
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<td>Stabilizer incidence, $i_s$</td>
<td>-1.75°</td>
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<tr>
<td>Tail Length, $l_t$ (0.25 $c_t$ to 0.25 $c_t$)</td>
<td>523.6 in.</td>
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<td>Tail Volume, $V_t$</td>
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## Elevator

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<tr>
<td>Deflection Limits, $\delta_e$</td>
<td>$+15^\circ$, $-40^\circ$</td>
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<tr>
<td>Tab Area, $S_{et}$</td>
<td>15.3 sq.ft.</td>
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<td>11.0 in.</td>
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<tr>
<td>Tab Deflection Limits, $\delta_{et}$</td>
<td>$+30^\circ$</td>
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<table>
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<td>Span, $b_v$</td>
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<td>Aspect Ratio, $A_v$</td>
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<tr>
<td>Mean Chord, $c_v$</td>
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<td>Taper Ratio, $\lambda_v$</td>
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<td>Chords</td>
<td></td>
</tr>
<tr>
<td>Root</td>
<td>250. in.</td>
</tr>
<tr>
<td>Tip</td>
<td>74. in.</td>
</tr>
<tr>
<td>Straight Element, 75% Chord</td>
<td>F.S. 1126.5</td>
</tr>
<tr>
<td>Location of .25 $c_v$</td>
<td>F.S. 1039.6</td>
</tr>
<tr>
<td>Tail Volume, $\bar{V}_v$</td>
<td>.0552</td>
</tr>
</tbody>
</table>

## Rudder

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Area, $S_r$</td>
<td>75.0 sq.ft.</td>
</tr>
<tr>
<td>Mean Chord, $c_r$</td>
<td>114.5 in.</td>
</tr>
<tr>
<td>Deflection Limits, $\delta_r$</td>
<td>$+35^\circ$</td>
</tr>
<tr>
<td>Tab Area, $S_{rt}$</td>
<td>5.0 sq.ft.</td>
</tr>
</tbody>
</table>
Tab Mean Chord, $c_t$ 10.0 in.
Tab Deflection Limits, $\delta_t$ (for trim) $\pm 25^\circ$

Maximum Lift Coefficients

| Clean Airplane | 1.60 |
| Take-off Flaps | 2.00 |
| Landing Flaps | 2.60 |

Minimum Lift Coefficients

| Clean Airplane | -1.0 |

Design Weights and Load Factors

**YC-130**

| Design Gross Weight | 110,530 lbs. |
| Max. Load Factors, $V < V_H$ | +3.0, -1.0 |
| Max. Load Factors at $V_D$ | +2.5, -0.75 |
| Max. Take-Off Weight | 127,200 lbs. |
| Max. Load Factors | 2.0 |
| Design Landing Weight | 99,000# |
| Max. Alt. Land. Wt. (8 ft/sec. sink speed) | 107,500 lbs. |

**C-130A**

| Design Gross Weight | 106,000 lbs. |
| Max. Load Factors, $V < V_H$ | +3.0, -1.0 |
| Max. Load Factors at $V_D$ | +2.5, -0.75 |
| Max. Take-Off Weight | 124,200 lbs. |
| Max. Load Factors | 2.0 |
| Design Landing Weight | 96,000# |
| Max. Alt. Landing Wt. (8 ft/sec. sinking speed) | 105,000 lbs. |
### Design Velocities (Sea Level)

#### YC-130

- Landing Flap Extension: 155 knots
- Take-Off Flap Extension: 183 "
- Landing Gear Door Opening: 166 "
- Cargo Door Opening: 130 "
- Level Flight: 290 "
- Dive: 348 "

#### C-130A

- Landing Flap Extension: 145 knots
- Take-Off Flap Extension: 183 "
- Landing Gear Door Opening: 166 "
- Cargo Door Opening: 130 "
- Level Flight: 287 "
- Dive: 345 "
LOCKHEED C-130

SPEED SUMMARY

**Note:**

\[ V_D = 1.2 \times V_A \]

**Fig. 2**
MILITARY AND NORMAL RATED POWER AT SEVERAL ALTITUDES

Four Engines Operating

\[ C_D \]

\[ 0.015 \quad 0.02 \quad 0.03 \quad 0.04 \quad 0.05 \quad 0.06 \quad 0.07 \quad 0.08 \quad 0.09 \quad 0.1 \quad 0.2 \quad 0.3 \quad 0.4 \quad 0.5 \]

\[ 10 \text{,}000 \text{ FT} \]
\[ 20 \text{,}000 \text{ FT} \]
\[ 30 \text{,}000 \text{ FT} \]

\[ \text{TRUE AIRSPEED - KNOTS} \]
### TABLE II

**AERODYNAMIC SIGN CONVENTIONS**

<table>
<thead>
<tr>
<th>Item</th>
<th>Positive When</th>
</tr>
</thead>
<tbody>
<tr>
<td>Lift, $L \left( C_L \right)$</td>
<td>Up</td>
</tr>
<tr>
<td>Drag, $D \left( C_D \right)$</td>
<td>Aft</td>
</tr>
<tr>
<td>Thrust, $T \left( C_A \frac{T}{qS} \right)$</td>
<td>Forward</td>
</tr>
<tr>
<td>Pitching Moment, $M \left( C_m \right)$</td>
<td>Nose Up</td>
</tr>
<tr>
<td>Angle of Attack, $\alpha$</td>
<td>Nose Up</td>
</tr>
<tr>
<td>Side Force, $Y \left( C_Y \right)$</td>
<td>Right</td>
</tr>
<tr>
<td>Yawing Moment, $N \left( C_n \right)$</td>
<td>Nose Right</td>
</tr>
<tr>
<td>Angle of Yaw, $\psi$</td>
<td>Nose Right</td>
</tr>
<tr>
<td>Rolling Moment, $L \left( C_L \right)$</td>
<td>Right Wing Down</td>
</tr>
<tr>
<td>Elevator Angle, $\delta_e$</td>
<td>Trailing Edge Down</td>
</tr>
<tr>
<td>Rudder Angle, $\delta_r$</td>
<td>Trailing Edge Left</td>
</tr>
<tr>
<td>Aileron Angle, $\delta_a$</td>
<td>Trailing Edge Down</td>
</tr>
</tbody>
</table>
LONGITUDINAL AERODYNAMIC CHARACTERISTICS

Aerodynamic data in pitch for the airplane and the airplane build-up are obtained principally from the low-and-high-speed wind tunnel tests of refs. 1, 2 and 3. Data on individual components not tested by Lockheed are obtained largely from publications of the NACA, specific references being noted as used. The airplane is conventional in all geometric and aerodynamic respects, presenting no unusual problems not capable of solution by standard methods.

AIRFOIL SECTION DATA

Airfoil section data for all lifting surfaces, together with effects of deflection of flaps or control surfaces are presented in figs. 4 through 11. These data are obtained from NACA publications, refs. 8, 9, 11-13, 15-21 and unpublished tests of the Lockheed XP2V-1.

AIRPLANE LIFT CHARACTERISTICS

Low Speed. -- Low speed lift characteristics for the airplane build-up are presented in figs. 12 and 13. These data are determined directly from the low speed wind tunnel tests, corrected for Reynolds' No. effects.

Power Effects. -- Effects of power on the lift characteristics of the airplane with and without horizontal tail are given in figs. 15 through 19. These data are determined by interpolation and extrapolation from low speed wind tunnel tests. It will be noted that both lift curve slope and maximum attainable lift coefficient are increased with increasing power coefficient. The manner of obtaining
power effects is discussed in the section on pitching moment characteristics below.

**Mach Effects.** -- Effects of compressibility on airplane build-up lift characteristics are shown in fig. 14. High speed wind tunnel tests show that all configurations which include the wing have a lift curve slope variation with Mach No. which is identical on a percentage basis, and that the incremental change in zero lift angle of attack is the same. Thus, these two simple curves are sufficient to describe the Mach effects on lift. These curves apply only to power-off cases; power effects should be ascertained from the low speed curves and added unchanged.

**AIRPLANE DRAG CHARACTERISTICS**

**Low Speed.** -- Drag characteristics for the airplane build-up are presented in figs. 20 and 21. These data are determined from the low speed wind tunnel tests, corrected for effects of Reynold's No. on lift.

**Mach Effects.** -- Effects of compressibility on the drag of the airplane in four stages of build-up are shown in figs. 22 and 23. Data are presented in the form of drag coefficient increments at several values of lift coefficient, to be added to low speed drag. These curves are obtained from an analysis of the low and high speed wind tunnel data.

**AIRPLANE PITCHING MOMENT CHARACTERISTICS**

**Low Speed.** -- Pitching moment characteristics for the airplane build-up are presented in figs. 24, 25 and 26. It is noted that the addition of the vertical tail to the wing-fuselage combination adds
an appreciable and approximately constant, nose-down increment which will contribute to fuselage bending in the same manner as horizontal tail load. Additional configurations tested, for which no data are presented, include pylon tanks on and cargo doors in various positions. In testing with the complete airplane, addition of the pylon tanks caused a nose-down trim shift of $\Delta C_m = -0.016$ while opening the upper cargo door caused a nose-up trim shift of $\Delta C_m = +0.014$ and the lower, or ramp door caused a similar shift of approximately $\Delta C_m = +0.0015$ per degree of door opening.

Mach Effects. -- Pitching moment coefficient curves are plotted for several Mach Numbers for each of several configurations in the airplane build-up, and presented in figs. 27 through 31.

Power Effects. -- Wind tunnel tests with power were made at very low tunnel speeds to enable attainment of large power coefficients with the model motors used. Due to the very low Reynold's Numbers of such tests, the absolute magnitude of the results are not deemed to be as trustworthy as those run at higher speeds. The incremental effects due to power are considered to be entirely reliable, and are used by adding them to proper basic power-off test results from normal tunnel speed runs. This procedure has been followed in the analysis of all power-on data presented herein.

Effects of power-on airplane pitching moment characteristics with and without horizontal tail and for various flap settings are presented in figs. 32 through 36. It will be noted that the effects of power are destabilizing for both configurations and all flap settings.
These power effects are absolute in value with respect to speed; to
determine pitching moment at high speeds, the power-off value should
first be ascertained for the proper Mach No. and an increment due to
power obtained from these low speed curves to be added.

ELEVATOR CHARACTERISTICS

Elevator effectiveness is given in fig. 37, showing change
in airplane pitching moment at constant lift coefficient. Values are
presented for several lift coefficients and for flaps retracted
and extended. Effects of Mach No. on elevator effectiveness are given
in fig. 38.

Elevator hinge moment data are presented in figs. 39 and
40 for flaps extended and retracted. Values are given in each case
for airplane angles of attack of zero and six degrees. From elevator
floating characteristics, it has been ascertained that hinge moment for
angles of attack greater than six degrees is the same as at six degrees;
all variation of hinge moment with angle of attack is for angles less
than six degrees, as noted on the figures, for all flap positions.
Characteristics of the control and boost system are shown in figs. 77,
78 and 79, including a plot of the variation of pilot force with eleva-
tor hinge moment.
Wing Airfoil Section Lift Characteristics

with Various Flap Extensions

\[ \text{C}^2 \]

\[ \text{Angle of Attack} \quad \text{Deg.} \]

\[ \begin{align*}
&\text{C}^2_0 \\
&0.5 \\
&1.5 \\
&2.5 \\
&3.5 \\
&4.0
\end{align*} \]

\[ \begin{align*}
&\text{0°} \\
&5° \times 18^\circ \\
&10° \times 20^\circ \\
&15° \times 24^\circ \\
&20° \times 28^\circ \\
&25° \times 32^\circ \\
&30° \times 36^\circ
\end{align*} \]
Lockheed C-130 Wing: Airfoil Section Drag Characteristics with Various Flap Extensions
LOCKHEED C-130

WING AIRFOIL SECTION PITCHING MOMENT CHARACTERISTICS WITH VARIOUS FLAP EXTENSIONS

Fig. 7
HORIZONTAL TAIL AIRFOIL SECTION

CHARACTERISTICS

NACA 23012 INVERTED

\[ C_l \]

\[ \alpha - \text{ANGLE OF ATTACK - DEG.} \]

\[ C_d \]

\[ C_m \]
LOCKHEED C-130

HORIZONTAL TAIL AIRFOIL SECTION

PITCH CHARACTERISTICS DUE TO ELEVATOR DEFLECTION

LOW SPEED

\[(\Delta C_{m,25})_\alpha \quad \Delta C_L \quad \Delta C_{m,25}\]

ELEVATOR ANGLE \(\delta_e\) - Deg.
LOCKHEED C-130

VERTICAL TAIL AIRFOIL

SECTION CHARACTERISTICS

NACA 64A015 AIRFOIL

\[ C_{m_{25}} = 0 \]

\[ C_{L} \]

\[ C_{D} \]

\[ \text{Angle of Attack } \sim \text{ Deg} \]

\[ 0 \quad 4 \quad 8 \quad 12 \quad 16 \]

\[ 0 \quad 0.004 \quad 0.008 \quad 0.012 \quad 0.016 \quad 0.020 \]
Effect of Aileron Deflection on Wing Airfoil Section Characteristics

\[ \Delta C_{L \alpha} \]

\[ \Delta C_{m_{\alpha}} \]

\( \delta_{\alpha} \sim \text{DEG} \)

\( \delta_{\alpha} \sim \text{DEG} \)

\( E = 0.28 \)

Fig. 11
EFFECT OF AIRPLANE COMPONENTS ON LIFT CHARACTERISTICS

LOW SPEED

$C_l$

$M = 0.23$

WING
WING AND FUSELAGE
WING NACELLES AND FUSELAGE
WITH AND WITHOUT VERTICAL AND DORSAL AIRPLANE... (UNTRIMMED)

$\alpha_{fri.} \text{ DEG.}$

Fig. 12
LOCKHEED C-130

AIRPLANE LIFT CHARACTERISTICS
WITH VARIOUS FLAP EXTENSIONS

POWER OFF

\[ C_L \]

\[ \delta_f = 0^\circ \]

\[ \delta_f = 18^\circ \]

\[ \delta_f = 36^\circ \]

AIRPLANE (UNTRIMMED)
AIRPLANE LESS TAIL

ANGLE OF ATTACK - DEG.

Fig. 13
LOCKHEED C-130

VARIATION OF LIFT CURVE SLOPE AND ZERO LIFT INTERCEPT WITH MACH NO

\[ \frac{C_{L\alpha}}{C_{L\alpha}_0} \]

NOTE: THESE CURVES APPLY TO ALL CONFIGURATIONS EXCEPT THE FUSELAGE ALONE WHICH SHOWS NO VARIATION IN THIS RANGE.

MACH NO

\[ \Delta \alpha L_0 \]

MACH NO

Fig. 14
Lockheed C-150

Effects of power on lift

Characteristics of airplane

Less horizontal tail

Flaps up

Low speed

Fig. 15.
 Lockheed C-130

Effects of Power on Lift

Characteristics of Airplane

Less Horizontal Tail

Take-off Flaps

$\alpha = 15^\circ$

$C_L$ vs. $\theta$

$C_L$

Take-off Flaps $\alpha = 15^\circ$

$C_D$

$C_L$

$0^\circ - 4^\circ$

Fig. 16
EFFECTS OF POWER ON LIFT

CHARACTERISTICS OF AIRPLANE

LESS HORIZONTAL TAIL

LANDING FLAPS - \( \delta = 36^\circ \)

\( C_{L} \):
- \( 2.0 \)
- \( 1.6 \)
- \( 1.2 \)
- \( 1.0 \)
- \( 0.8 \)
- \( 0.4 \)
- \( 0.0 \)

\( \alpha - \deg. \):
- \( -12 \)
- \( -8 \)
- \( -4 \)
- \( 0 \)
- \( 4 \)
- \( 8 \)
- \( 12 \)
- \( 16 \)

FIG. 17
Lockheed C-130

Power Effects on Airplane

Lift Characteristics

Low Speed

\( \Delta \theta = 0^\circ \)

\[ C_L \]

\[ 0 \rightarrow 2.0 \rightarrow 2.4 \rightarrow 2.8 \]

\[ 0 \rightarrow 4 \rightarrow 8 \rightarrow 12 \rightarrow 16 \rightarrow 20 \]

Angle of Attack - Deg.

Fig. 18
EFFECT OF POWER ON AIRPLANE

LIFT CHARACTERISTICS

FLAPS DOWN 36°

$C_L$ vs. $\alpha$ for $C_\text{E}$:
- $C_\text{E} = 0$: $C_L = 3.6$
- $C_\text{E} = 0.2$: $C_L = 3.2$
- $C_\text{E} = 0.4$: $C_L = 2.8$
- $C_\text{E} = 0.6$: $C_L = 2.4$
- $C_\text{E} = 0.8$: $C_L = 2.0$

Fig. 19
Airplane Drag Build-Up

Wing
Fuselage
Wing plus Fuselage
Airplane less Empennage
Airplane plus External Fuel Tanks

* NOTE: Fuselage Drag Coefficients are plotted against lift coefficients for the wing.

Fuselage Combination

Fig. 20
AIRPLANE DRAG RISE

\[ \Delta C_D \]

\[ \text{MACH NO.} \]
EFFECT OF AIRPLANE COMPONENTS
ON PITCHING MOMENTS

LOW SPEED

$C_L$

FUSELAGE
WING
WING PLUS FUSELAGE
AIRPLANE LESS EMPENNAGE
AIRPLANE LESS HORIZ. TAIL

* FUSELAGE $C_m$ is plotted against $C_L$ for wing plus fuselage, and is not affected by compressibility in the applicable Mach range.
LOCKHEED C-130

AIRCRAFT CORPORATION

DATE 6-16-52
CHECKED BY

REPORT NO 2062

AIRPLANE-LESS-TAIL PITCHING

MOMENT FOR VARIOUS FLAP DEFLECTIONS

FULL FLAPS \( \delta_f = 36^\circ \)
HALF FLAPS \( \delta_f = 18^\circ \)
FLAPS UP \( \delta_f = 0 \)

\[ C_{m} \]

\[ C_{L} \]
LOCKHEED C-130

AIRPLANE PITCHING MOMENT CHARACTERISTICS FOR VARIOUS FLAP POSITIONS

Low Speed

\[ C_m = -1.75 \text{ for } \delta = 0^\circ \]

Power Off

\[ \delta_f = 36^\circ \]

\[ \delta_f = 18^\circ \]

\[ \delta_f = 0^\circ \]

\[ C_m = 0.25 \]
LOCKHEED C-130

PITCHING MOMENTS AT SEVERAL MACH NOS. FOR WING

\[ C_L \]

\[ M_1 \]

- .23
- .5
- .6
- .67
- .7

\[ C_m \]

\[ .12 \quad .08 \quad .04 \quad 0 \quad .04 \quad .08 \quad .12 \quad .25 \]

\[ .2 \quad .4 \quad .6 \quad .8 \quad 1.0 \quad 1.2 \quad 1.4 \quad 1.6 \]
LOCKHEED C-130

Pitching Moment at Several Mach Nos. for Wing Plus Fuselage

\[ C_m \]

\[ M \]

\[ C_2 \]

\[ 0.7 \]

\[ 0.67 \]

\[ 0.56 \]

\[ 0.23 \]
Pitching Moment at Several Mach Nos. for Airplane Less Empennage
Pitching Moment at Several Mach Nos. for Airplane Less Horizontal Tail

\[ C_m \text{ vs } C_L \]

- Mach Numbers: 0.23, 0.5, 0.6, 0.67, 0.7
LOCKHEED C-130

Pitching Moments at Several Mach Nos for Complete Airplane

\[ C_L \]

\[ C_m \text{.25} \]

\[ M \]

\[ .23, .5, .625, .65, .675, .7 \]
LOCKHEED C-130

EFFECTS OF POWER ON PITCHING

CHARACTERISTICS OF AIRPLANE

LESS HORIZONTAL TAIL

FLAPS UP

LOW SPEED

\[ C_L \]
EFFECTS OF POWER ON Ditching Characteristics of Airplane

LESS horizontal tail

Pre-Off Ramps - $\phi = 18^\circ$
LOCKHEED C-130

EFFECTS OF POWER ON PITCHING CHARACTERISTICS OF AIRPLANE LESS HORIZONTAL TAIL

LANDING FLAPS $\delta_F = 36^\circ$

$C_L$ vs. $C_m$ for different $C_l$ values.

Fig. 34
Power Effects on Airplane

Pitching Moment Coefficients

Low Speed

\[ C_l \]

\[ C_{m.25} \]
EFFECTS OF POWER ON PITCH
CHARACTERISTICS OF AIRPLANE

Fig. 36
ELEVATOR EFFECTIVENESS

Low Speed

\[
(C_{me})_{\alpha}
\]

\[
C_{L} = 0, 0.5, 1.0, 1.5
\]

FLAPS DN ALL C_L'S.

\[\phi_e \sim \text{ELEVATOR ANGLE} \sim \text{DEG.}\]
VARIATION OF ELEVATOR EFFECTIVENESS WITH MACH NO

\[
\frac{L_{\text{elevator}}}{L_{\text{max}}} = \frac{C_{\text{elevator}}}{C_{\text{max}}} \quad \text{versus MACH NO}
\]
Elevator Hinge Moment Characteristics

Low Speed - Flaps Up

\[ S_e = 114.9 \text{ ft} \]
\[ C_e = 2.79 \text{ ft} \]

\[ \alpha = 0^\circ \]
\[ \alpha = 6^\circ \]

\[ \Delta C_{he} \]

\[ S_e \sim \text{Elevator Angle \sim Deg.} \]

\[ \delta_t \sim \text{Tab Angle \sim Deg.} \]
LOCKHEED C-130

ELEVATOR HINGE MOMENT CHARACTERISTICS

Low Speed Flaps Down

\[ S_e = 114.9 \text{ sq. ft.} \]
\[ C_{e} = 2.79 \text{ ft.} \]

\[ \Delta C_{e} \]

\[ \alpha = 0^\circ \]
\[ \alpha = 6^\circ \]

\[ S_e - \text{ELEVATOR ANGLE - DEG.} \]

\[ \Delta \text{ELEVATOR ANGLE - DEG.} \]

Fig. 40
DIRECTIONAL AERODYNAMIC CHARACTERISTICS

Directional aerodynamic characteristics for the airplane and airplane build-up are obtained principally from the test results of refs. 1, 2 and 3. Where data from any other source is used, proper reference is made in the test.

SIDE FORCE CHARACTERISTICS

Variations of side force coefficient with yaw angle for the airplane build-up for low speeds is given in fig. 41. From many test results it has been ascertained that airplane less tail side force characteristics do not change appreciably with increasing Mach No., but that the force on the vertical tail varies according to the Prandtl-Glauert relation. Effect of compressibility on airplane side force and vertical tail normal force has been determined on this basis, and is presented in fig. 47. Effects of power on airplane side force with and without the vertical tail are given in fig. 42 as obtained from wind tunnel test results. As with other power effects, these are not subject to compressibility correction but should be added as increments to corrected power-off values. The manner of determining power effects from tunnel test is discussed in the section on pitching moment characteristics on p. 14.

YAWING MOMENT CHARACTERISTICS

The variation of yawing moment coefficient with yaw angle for the airplane build-up is given in fig. 44 for low speeds. These curves are applicable to all flap settings. As is the case with side force, effects of compressibility are confined to the contribution of the vertical
tail, and the effects of Mach Number on airplane directional stability, power-off, presented in fig. 47 are determined on that basis. Power effects for flaps retracted and extended are presented in figs. 45 and 46 to be used as increments applied to data corrected for compressibility. See the discussion of power effects on pitching moment characteristics, pg. 114, for the manner of obtaining all power effects.

RUDDER CHARACTERISTICS

Rudder effectiveness, including tabs, is presented in fig. 48 for all flap settings. Rudder hinge moment coefficient characteristics are shown in fig. 49, including trim tab effects. The values for zero yaw angle are obtained directly from wind tunnel tests, while effects of yaw angle are inferred from rudder floating characteristics observed in the tunnel and comparison with other similar surfaces. For the range of Mach Numbers in which critical design air loads will be produced, there are no changes in any of the aerodynamic characteristics of the rudder. The characteristics of the rudder control and boost system are shown in figs. 80 through 83, including a plot of the variation of rudder pedal force with rudder hinge moment.
SIDE FORCE CHARACTERISTICS:

LOW SPEED

$\psi$ - YAW ANGLE - DEG.

- AIRPLANE
- AIRPLANE - LESS - TAIL
- FUSELAGE
EFFECTS OF POWER ON AIRPLANE
SIDE FORCE CHARACTERISTICS

Low Speed

$C_{y}$

$C_{4}$

$\psi$ - DEG.
Lockheed C-130

Effects of Power on Airplane

Side Force Characteristics

Landing Flaps

$C_Y$

$\gamma$ - Yaw Angle - Deg.
LOCKHEED: C-130

YAWING MOMENT CHARACTERISTICS

LOW SPEED

\[ C_n \]

\[ \Psi - \text{Angle of Yaw - Deg.} \]

\[ \text{Airplane} \]

\[ \text{Fuselage Alone} \]

\[ \text{Airplane - Less Tail} \]
EFFECTS OF POWER ON CHARACTERISTICS IN YAW OF AIRPLANE WITH/WITHOUT TAIL

LOW SPEED

$C_A$ vs $\psi$ (deg)

$C_h$ vs $\psi$ (deg)

FLAPS UP & TAKE-OFF FLAPS

TAIL OFF

TAIL ON
EFFECTS OF POWER ON AIRPLANE

YAWING MOMENT CHARACTERISTICS

LANDING FLAPS

\[ \psi - \text{yaw angle} \text{ deg.} \]
VARIATION OF AIRPLANE YAWING CHARACTERISTICS WITH MACH NO.

**Yawing Moment**

\[
\frac{C_{nA}}{C_{nA,23}}
\]

**Side Force**

\[
\frac{C_{yA}}{C_{yA,23}}
\]
RUDDER HINGE MOMENT COEFFICIENTS

$S_r = 75 \text{ sq. ft.}$

$C_r = 44.5 \text{ in.}$

$\Delta C_{hr}$

$C_{hr}$

$S_r - \text{ Rudder Angle - Deg.}$

Eq. 4aw for Power off, low speed

$\psi = 0 - 10^\circ$
LATERAL AERODYNAMIC CHARACTERISTICS

DIHEDRAL EFFECT

Airplane roll due to yaw for all angles of attack and flap settings is given in fig. 50 for power off. Application of power tends to reduce the dihedral effect by approximately \( C_{\alpha} \times 60\% \) for flaps up. At high speeds, the power off curve may be corrected by the Prandtl-Glauert factor to account for compressibility in the Mach No. range wherein critical loads will occur.

DAMPING IN ROLL

The low speed value of the roll damping coefficient is determined on the basis of geometric parameters from ref. 10. The variation of the coefficient with Mach Number above 0.2 is according to the Prandtl-Glauert relation, which is valid in the Mach Number range in which critical air loads will be acting. Fig. 51 presents this data.

AILERON CHARACTERISTICS

Aileron effectiveness for flaps retracted and extended for all speeds is given in figs. 52 and 53. Values are given for various lift coefficients; intermediate values of effectiveness may be obtained by interpolation and extrapolation. Aileron hinge moment coefficients are presented in fig. 54 for all flap settings and all speeds. Values of hinge moment coefficient are given for two airplane angles of attack, from which values for other angles may be determined by interpolation and extrapolation. These hinge moment coefficients have been estimated from the data of refs. 4, 7 and 10.
Characteristics of the aileron control and boost system are given in figs. 84, 85 and 86. In figs 87 and 88 are shown plots of pilot force and aileron hinge moment versus wheel angle for several speeds, showing the effects of the boost ratio and boost cut-off and the relationship between pilot force and aileron hinge moment.
LOCKHEED C-130

**Dihedral Effect**

![Graph showing dihedral effect with axes labeled as follows:](image)

- $C_l$ vs. $\psi$ for dihedral effect.
- $\Delta C_l$ vs. $\phi_r$ for dihedral effect.

**Figure 50**
VARIATION OF AIRPLANE ROLL DAMPING CHARACTERISTICS WITH MACH NO.
Aileron Effectiveness

Clean Configuration

ROLLING MOMENT COEFF. - C_{y}

δ_{a} = Aileron Angle - Deg.

LOCKHEED C-130

LOCKSHEED AIRCRAFT CORPORATION
CALIFORNIA DIVISION

REPORT NO. 9062

MODEL: C-130

FIG. 52
Aileron Effectiveness

Landing Configuration

Rolling Moment Coefficient $C_m$

$a$-Aileron Angle - Deg

$C_l$ - 2.0
$C_l$ - 5
$C_l$ - 10 to 15
$C_l$ - 25
LOCKHEED C-130

Estimated Aileron Hinge Moment Characteristics

\[ \delta \sim \text{Aileron Angle} \sim \text{Deg.} \]
CHORDWISE PRESSURE DISTRIBUTIONS

All chordwise pressure distributions except that over the wing with Fowler flaps extended have been determined from the methods and data of refs. 8, 10 and 24. The curves for the wing section with Fowler flaps extended are obtained from unpublished wind tunnel test data on the Lockheed XP2V-1 and from flight test pressure measurements on the Lockheed Constellation airplane.

WING

Chordwise pressure distributions are presented for upper and lower surfaces at several lift coefficients for the inboard wing airfoil section, for a typical section in the outboard panel and for the wing tip section in figs. 55, 56, 58, 59 and 61. In addition, net unit and basic pressure distributions are shown for the inboard and typical outboard sections in figs. 57 and 60. The pressure distribution over the wing airfoil section ahead of the extended Fowler flap is presented in fig. 63. Flap load coefficients are shown in fig. 70 and discussed on page 88.

Net pressure distribution increment due to deflection of the aileron is given in fig. 62, determined from the data of ref. 25, using the section pitching moment data of fig. 11.

EMPENNAGE

The horizontal tail surface embodies an inverted NACA 23012 airfoil section. Chordwise pressure distributions are presented in figs. 64, 65 and 66 for the inverted airfoil (the normally lower surface now becoming
the upper, and vice versa), for both surface pressures and the net unit and basic distributions. Fig. 67 shows the added net distribution due to elevator deflection. If necessary to determine local surface pressures with elevator deflected, this net distribution may be considered to apply approximately half to each surface. Figures 68 and 69 show the pressure distributions for the vertical tail and the added net contribution due to deflection of the rudder.
Lockheed C-130

Chordwise Pressure Distribution Over Inboard Wing Section

Lower Surface Airfoil Section 64A-318 618

Pressure Coefficient $C_p$

Chord Station $x/c$

Fig. 36
Net Chordwise Pressure Distribution Over Inboard Wing Section
LOCKHEED C-130

CHORDWISE PRESSURE DISTRIBUTION OVER OUTBOARD WING SECTION

UPPER SURFACE AIREFOIL SECTION GAA-415 C=0.8

PRESSURE COEFFICIENT

CHORD STATION 10

0 1 2 4 6 8 10
LOCKHEED C-130

CHORDWISE PRESSURE DISTRIBUTION OVER OUTBOARD WING SECTION

LOWER SURFACE AIRFOIL SECTION

Fig. 59
LOCKHEED C-130

Chordwise Pressure Distribution Over Wing Tip Section

Airfoil Section 64A-412 (A = 0.8)

Pressure Coefficient \( p \)

Chord Station \( x \)

Fig. 61.
Net Pressure Distribution

Increment Due to Aileron Deflection

Low Speed

For local surface pressures, divide these net pressures by 2.0 and apply to each surface, adding by principle of superposition.

\[
\delta_a \quad C_n
\]

\[
\begin{array}{c|c|c}
\delta_a & \text{Low Speed} & C_n \\
-5 & -1.28 & \\
-10 & -2.53 & \\
-15 & -3.52 & \\
-20 & -4.27 & \\
-25 & -4.77 & \\
-30 & -5.17 & \\
-35 & -5.47 & \\
-40 & -5.67 & \\
-45 & -5.87 & \\
-50 & -6.07 & \\
\end{array}
\]

\[
\frac{\Delta P}{\delta_a} \quad \text{Pressure Coefficient}
\]

X/C = Chord Station

Fig. 62
WING CHORDWISE PRESSURE DISTRIBUTION WITH FOWLER FLAP EXTENDED

PRESSURE COEFFICIENT

CHORD STATION

Fig. 63
HORIZONTAL TAIL CHORDWISE

PRESSURE DISTRIBUTION

UPPER SURFACE
NACA 23012 INVERTED

PRESSURE COEFFICIENT, C_p

CHORD STATION ~ %C
HORIZONTAL TAIL CHORDWISE

PRESSURE DISTRIBUTIONS

LOWER SURFACE
NACA 23012
INVERTED

PRESSURE COEFFICIENT - C

CHORD STATION - \( \frac{y}{c} \)

Fig. 65
HORIZONTAL TAIL NET CHORDWISE PRESSURE DISTRIBUTION

AEROFOIL SECTION:
NACA 23012 INVERTED

NET PRESSURE COEFFICIENT  \( \frac{p}{p_0} \)

CHORD STATION  \( \% C \)

Fig. 66
Lockheed C-130

Horizontal Tail Net Chordwise Pressure Distribution Due To Elevator Deflection

Low Speed

For local surface pressures divide these net pressures by 2 and apply to each surface. Adding by principle of superposition to pressures due to angle of attack.

Net Pressure Coefficient

<table>
<thead>
<tr>
<th>Angle of Attack (°)</th>
<th>Net Pressure Coefficient</th>
</tr>
</thead>
<tbody>
<tr>
<td>10</td>
<td>2.75</td>
</tr>
<tr>
<td>20</td>
<td>5.53</td>
</tr>
<tr>
<td>30</td>
<td>7.13</td>
</tr>
<tr>
<td>40</td>
<td>8.20</td>
</tr>
</tbody>
</table>

Chord Station %

Fig. 67
Lockheed C-130

Vertical Tail Chordwise Pressure Distribution

Elevator Neutral

Airfoil Section 64A-.015
VERTICAL TAIL NET CHORDWISE PRESSURE DISTRIBUTION DUE TO RUDDER DEFLECTION

LOW SPEED

For local surface pressures, divide these net pressures by 2 and apply to each surface, adding by principle of superposition to pressures due to angle of attack. (V = W)

<table>
<thead>
<tr>
<th>Angle of Attack (°)</th>
<th>Pressure Coefficient (C_p)</th>
</tr>
</thead>
<tbody>
<tr>
<td>5</td>
<td>0.191</td>
</tr>
<tr>
<td>10</td>
<td>0.262</td>
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<tr>
<td>15</td>
<td>0.395</td>
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<tr>
<td>20</td>
<td>0.525</td>
</tr>
<tr>
<td>30</td>
<td>0.807</td>
</tr>
<tr>
<td>40</td>
<td>1.138</td>
</tr>
</tbody>
</table>

NET PRESSURE COEFFICIENT \(C_p\) vs CHORD STATION \(\gamma/L\)

Fig. 69
MISCELLANEOUS AIR LOADS

CARGO DOOR

Hinge moment coefficients for both upper and lower cargo doors are shown in fig. 70 for several different positions of each door. These data were obtained from wind tunnel tests reported in ref. 1.

FOWLER FLAP LOAD COEFFICIENTS

Flap normal and chord force coefficients are presented in fig. 71. These data are determined from extensive two-dimensional wind tunnel tests conducted by Lockheed and summarized in ref. 6. The curves are presented as total force coefficients, although they are developed from pressure distribution measurements and are, therefore, section coefficients. Two curves of chord force coefficient are presented inasmuch as the tests revealed either might be attained, owing largely to gap and condition of the under surface of the wing exposed by extension of the flap. Wing chordwise pressure distributions with the flaps extended are presented in fig. 63.

ENGINE NACELLE PRESSURE DISTRIBUTION

Upper and lower centerline pressures for two airplane angles of attack are presented in fig. 72. These curves are estimated from wind tunnel test data on the nacelle of a Lockheed XP-58 airplane, reported in ref. 24.

PYLON TANK AND STRUT

Aerodynamic characteristics of the pylon tank are presented in fig. 73. These data are estimated from test data of ref. 14 on an
isolated body of similar geometry, with the angles of attack modified to account for the effects of wing upwash. Pylon tank strut aerodynamic properties are determined from simple theory, considering it as a swept-forward wing of low aspect ratio with the airplane wing acting as an end-plate and the pylon tank being a partially effective end-plate. Strut characteristics are shown in fig. 7h.

**PROPELLER SIDE LOADS**

Propeller loads may be determined from the Hamilton Standard equation for propeller side force:

\[
\frac{F}{\alpha q \left( \frac{D^2}{4} \right)} = \frac{0.0256}{110} \left[ \frac{A}{\pi} \right] \left[ \frac{N}{4} \right] \left[ \frac{h}{L} \right] \sin \left( \beta_{.75R} + 10^\circ \right)
\]

where

- \( F \) = propeller side load, lbs.;
- \( \alpha \) = angle of relative wind with respect to axis of rotation, degrees;
- \( q \) = dynamic pressure, \( \frac{V^2}{2} \) lbs. per sq. ft.;
- \( D \) = propeller diameter, 15 ft.;
- \( AF \) = propeller activity factor, 150;
- \( N \) = number of blades, 3;
- \( \beta_{.75R} \) = propeller blade angle at .75 radius, fig. 75.

Substituting the known values in the foregoing gives:

\[
F = 1.58 \alpha' q \sin \left( \beta_{.75R} + 10^\circ \right)
\]

The loads in pitch are computed for an angle of attack, \( \alpha' \), corrected for upwash induced by the presence of the wing such that

\[
\alpha' = 1.32 \left[ \alpha_{FRL} + 0.7^\circ \right]
\]
For propeller loads in yaw, $\alpha = \psi$.

**PARATROOPER SHIELD DOORS**

Paratrooper shield door normal force coefficient variation with door position is presented in fig. 76. A range of values is given to allow for effects of airplane yaw. The curves are estimated from wind tunnel test data on various airplane fuselage dive flaps, principally those of the Lockheed P-80 reported in ref. 5.

**NOSE LANDING GEAR DOORS**

The nose gear doors include three panels which, when closed, form part of the fuselage contour. The forward door is rectangular in shape and is hinged along its forward edge so that, in its extended position, it is similar to a fuselage dive-flap. The rear doors are also rectangular in shape and are arranged symmetrically on either side of the airplane centerline. Each of these doors operates from a linkage which, in opening, allows it to drop down from the fuselage a short distance, enough to clear and then to move sideways along the fuselage contour to clear the wheel well. In the closed position, the forward door is subject to fuselage pressure distribution, estimated to vary between limits of $\frac{\Delta p}{q} = -0.2$ to $+0.1$, accounting for angle of attack variations. When fully open, the forward door is subject to an airload equivalent to a pressure coefficient of $\frac{\Delta p}{q} = 1.2$ tending to close it. During extension, the loads are assumed to vary linearly between the foregoing values. The external face of the rear doors are subject to a pressure coefficient of $\frac{\Delta p}{q} = -0.2$ to $0$ in any position. In addition, a ram pressure coefficient of $\frac{\Delta p}{q} = 0.5$ exists between the side of the fuselage and the inner face of the panel.
MAIN LANDING GEAR DOORS

Air load coefficients for the main gear doors are presented in fig. 77. These data are determined from a survey of past design practice and wind tunnel and flight test results on bomb bay and landing gear doors of several Lockheed airplanes.
CARGO DOOR HINGE MOMENTS

MOMENTS FOR RAMP DOOR
RAMP DOOR CLOSED OPEN
OPEN 12° OPEN 22° OPEN 34°
UPPER DOOR CLOSED CLOSED TO OPEN 15°

NOTE: HINGE MOMENT IS ZERO FOR UPPER DOOR IN FULL OPEN POSITION.

Fig. 70
FLAP LOAD COEFFICIENTS

- Check this point with C.P. at A.C.F.
- Use this curve with C.P. at 3 C.F.
- Use both curves for design.
LOCKHEED C-130

ESTIMATED PRESSURE DISTRIBUTION ON NACELLES

\[
\frac{\Delta P}{q} \quad \alpha = 0° \quad \alpha = 16°
\]

FUSELAGE STATION

\[360 \quad 400 \quad 440 \quad 480 \quad 520 \quad 560\]
LOCKHEED C-130

PYLON TANK AERODYNAMIC CHARACTERISTICS

\[ C_l, C_D, C_m = \frac{L \cdot D \cdot C}{q \cdot A} \]
\[ C_m, C_n = \frac{M \cdot N}{q \cdot A \cdot L} \]

\( A \) = FRONTAL AREA, \( L \) = TANK LENGTH

\( C_l \), \( C_D \), \( C_m \), \( C_n \) vs. \( C\) and \( \alpha_{FRL} \) (Deg.)

\( C_c \) and \( C_n \) are given for tank on right side

Fig. 78
Lockheed C-130

Pylon Tank Strut Aerodynamic Characteristics

Coefficients are based on strut area.
Yawing moment about strut quarter-chord point is zero.

Fig. 74
VARIATION OF PROPPELLER BLADE ANGLE WITH SPEED

\[ \beta_{75r} - \text{blade angle at } 75 \text{ radius} \]

\[ V_{T} - \text{true airspeed - knots} \]

\[ h = \text{S.L.} \]
\[ h = 15,000 \text{ FT.} \]
\[ h = 25,000 \text{ FT.} \]
\[ h = 30,000 \text{ FT.} \]
\[ h = 40,000 \text{ FT.} \]
Paratrooper Shield Door

Air Loads

Center of pressure, at 45° aft of front edge

$C_N$ vs. door angle (°)

Measured from closed position

Fig. 73
Main Gear Door Air Loads

For use with full door chord and area.
For hinged section, use loads resulting
from these curves & also from a uniform
pressure equivalent to \( \Delta \rho \) = 0.10

Door Angle - Deg.
(from closed position)

Right Side Door

Oscillatory Limits

Yaw Angle - Deg.

Fig. 77
CONTROL SYSTEM CHARACTERISTICS

All three primary controls are actuated with the aid of hydraulic boosters. In each system, motion of the pilot's control is transmitted through suitable linkages to an arm which operates the booster valve. When this valve is bottomed, which requires only a small fraction of an inch of control movement, further motion is transmitted to the control surface, with the pilot's effort magnified by the action of the boost cylinder. On the prototype airplane, provision has been incorporated in the design for varying the boost ratio. Control system characteristics based on nominal settings are presented in the following pages and curves; final setting for both prototype and production airplanes will be determined during flight testing. The effect of changing boost ratio will, for practical purposes, affect only the magnitude of the pilot force at booster cut-off, decreasing it in the same proportion by which the boost ratio is increased and vice versa. Mechanical advantage between the pilot's control and the control surface remain unchanged and the hinge moment available at booster cut-off is only negligibly affected. The following data summarize the design requirements of the control surfaces, the control systems and the power boosters.

Elevator System

The nominal boost ratio averages approximately 35:1, as shown in Fig. 79, and with the variable feature of the prototype.
mentioned above, can be adjusted to any value between 27:1 and 80:1. Owing to the mechanical design of the system, the boost ratio and the other characteristics vary somewhat with control surface position; exact values of each are given in Figs. 76 and 79. The hydraulic supply to the cylinders has been made adequate to permit elevator movement rates of 40 degrees per second at one-third of the design hinge moment for short intervals and 10 full cycles per minute for continuous movement at approximately one-half of booster capacity.

The foregoing relationships have been crossplotted in Fig. 80 to show the variation of pilot effort with elevator hinge moment for several elevator angles. This plot illustrates the effectiveness of the boost system in aiding the pilot and shows that, for practical purposes, the hinge moment available at maximum booster output is the designing factor; further effort by the pilot produces only a negligible increase.

Rudder System

As with the elevator, the mechanical advantage of the rudder system is determined from the relation between the desired rudder pedal length and travel and the required rudder deflection. Hinge moment output capacity required has been determined to be critical for the case of an engine failure at take-off. A value of 3,450 ft.-lbs. with full rudder has been set, which allows sufficient margin of control for this case and which determines the design of the booster cylinder and linkage to the rudder. Maximum pilot effort associated with this output is approximately 150 lbs. which, with the other factors, sets
the boost ratio. The nominal boost ratio, based on the foregoing, averages approximately 52:1 and, on the prototype, can be adjusted to any value between 26:1 and 78:1 as explained above. This system provides adequate control at low speeds, with vertical tail loads within reasonable limits. At small rudder angles, the system provides over 4,000 ft.-lbs. of hinge moment capacity which, coupled with the hinge moment coefficient and airplane stability characteristics, results in more than sufficient control and unduly large design vertical tail loads in high speed flight. These loads exceed 120 lbs. per sq. ft. of surface, as compared to less than 90 on comparable airplanes. It was ascertained that the penalties associated with the foregoing could be avoided by reducing the hinge moment output of the boost cylinders for all flight with flaps retracted. Thus, for all flight with flaps extended in any amount, the system is as described above while, with flaps retracted, pressure to the boost cylinder is reduced from 3,000 psi to 1,600 psi with a proportionate decrease in hinge moment output. The resulting system provides ample high speed control, and design vertical tail loads are maintained within reasonable limits. Actual values, based on the nominal boost ratio setting, of the several parameters discussed here are given in Figs. 81, 82 and 83. Certain of these relationships have been crossplotted in Fig. 84 to show the variation of rudder pedal force with rudder hinge moment. The effect of the dual output system, dependent on flap position, is clearly demonstrated here. Also, it is readily apparent that pilot effort beyond that for boost limit produces only negligible effects. The hydraulic supply to the boost cylinders has been made adequate to permit rudder movement rates of 40 degrees per second under one-half of the design hinge moment for short intervals and, for continuous movement, 10 full cycles per minute at approximately one-half of design hinge moment.
Aileron System

Design requirements of the aileron system were set to obtain full aileron travel at $V_T \sigma 1/2 = 20$ knots. With the booster designed for this condition, it was found that appreciably greater hinge moments are available at lesser wheel throws, due to the mechanism of the system which produces dissymmetry between up and down aileron travel, so the nominal boost ratio of approximately 50:1 was set so that no more than 80 lbs. pilot effort would be required to attain full booster cutout at any wheel position consistent with the speed range of the airplane. On the prototype airplanes, the boost ratio may be adjusted to any value between 20:1 and 75:1 as found to be desirable during flight testing. The several parameters necessary to determine the relation between aileron hinge moment and wheel force are presented in Figs. 65, 96 and 97. Figs. 88 and 89 show variation of pilot force and aileron hinge moment with wheel angle for several speeds. The hinge moment data of Fig. 54 have been used with the control and boost system characteristics to determine these plots. The hydraulic supply to the cylinders has been made adequate to permit aileron movement ratio of 50 degrees per second at one-quarter of the design hinge moment for short intervals and 15 full cycles per minute for continuous movement at approximately one-half of design hinge moment.

Control Surfaces Tabs

All tabs are electrically driven and are used to trim pilot forces. Design loads are present in the table below for conditions determined to be critical. These hinge moments may be regarded to result from a triangular pressure distribution with its
peak at the tab hinge line. Tab hinge moment coefficients used are plotted in Fig. 90.

<table>
<thead>
<tr>
<th></th>
<th>Tab Area, Sq. Ft.</th>
<th>Tab Chord, In.</th>
<th>Max. Tab Angle, Deg.</th>
<th>Max. Control Surface Angle, Deg.</th>
<th>Limit Design $\dot{H}$, $\dot{V}_h$, In.-Lbs.</th>
<th>Normal Operating $\dot{H}$, $\dot{V}_h$, In.-Lbs.</th>
<th>Minimum Operating Rate, Deg./Sec.</th>
</tr>
</thead>
<tbody>
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<tr>
<td></td>
<td>5.0</td>
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<td>1120</td>
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<td>+15,-25</td>
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<td>+15,-40</td>
<td>5750</td>
<td>2100</td>
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</tbody>
</table>

*Table: Performance specifications for the aircraft.*
ELEVATOR CONTROL SYSTEM

MECHANICAL ADVANTAGE

\[ \delta_e = \text{ELEVATOR ANGLE} \quad \text{DEG.} \]
ELEVATOR BOOST SYSTEM

CHARACTERISTICS

BOOST RATIO

TOTAL HINGE MOMENT AT BOOST CUT-OFF

PILOT FORCE AT BOOST CUT-OFF

$\theta_e$ = ELEVATOR ANGLE = DEG.
VARIATION OF PILOT FORCE WITH ELEVATOR HINGE MOMENT

- 240
- 200
- 160
- 120
- 80
- 40
- 0

Pilot Force ~ Lbs

Hinge Moment ~ 1000 Ft-Lbs

Boost Cut-Off
RUDDER SYSTEM CHARACTERISTICS

A. Mechanical Advantage

B. Boost Ratio

\( \delta_T \sim \text{Rudder Angle} \sim \text{Deg.} \)

\( \delta_T \sim \text{Rudder Angle} \sim \text{Deg.} \)
**PEDAL FORCE AT RUDDER BOOST CUT-OFF**

![Graph showing pedal force at different rudder angles](image_url)

**Flaps Extended**

**Flaps Retracted**

---

**Fig. 82**
LOCKHEED C-130

RUDDER HINGE MOMENT
AT BOOST CUT-OFF

---

\[ \text{Hinge Moment - Fl. Lbs.} \]

\[ \text{Flaps Extended} \]

\[ \text{Flaps Retracted} \]

\[ \theta - \text{Rudder Angle - Deg.} \]

\[ \text{Fig. 63} \]
VARIATION OF PEDAL FORCE
WITH RUDDER HINGE

MOMENT

PEDAL FORCE - LBS.

300
200
100

6 4 2 0 2 4 6

LEFT RUDDER

Boost Cut-off, Flaps Up

Boost Cut-off, Flaps Down

RIGHT RUDDER

Hinge Moment - 1000 ft lbs.

Fig. 64
LOCKHEED C-130

AILERON CONTROL SYSTEM

CHARACTERISTICS

\[ \theta_w \sim \text{WHEEL ANGLE} \sim \text{DEG.} \]

\[ \text{LEFT AIL.} \quad 120 \quad 80 \quad 40 \quad 0 \quad 40 \quad 80 \quad 120 \]

\[ \text{RIGHT AIL.} \]

\[ \text{Boost Ratio} \]

\[ \text{LEFT} \quad \text{RIGHT} \]

\[ 120 \quad 80 \quad 40 \quad 48 \quad 50 \quad 52 \quad 54 \]
Lockheed C-130

Wheel force at boost cut-off

Fig. 87
AILERON HINGE MOMENTS

$W = 100,000$ LBS.
$n = 1.0$
$h = SEA LEVEL$

FLAPS UP
LANDING FLAPS

$V = 308$ KNOTS

$V = 300$
$V = 200$
$V = 150$
$V = 100$

$V = 100$
$V = 150$
$V = 200$
$V = 250$
$V = 300$
$V = 348$

$\theta_w$ - WHEEL ANGLE - Deg.

Boost Cut-off

Fig. 88
NOTE: THIS CURVE IS BASED ON A NOMINAL SETTING OF THE BOOST RATIO ADJUSTMENT AS DETERMINED BY CONTROL REQUIREMENTS.
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MISCELLANEOUS

