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Contract No. Nonr-1675(00)

DUCTED PROPELLER ASSAULT TRANSPORT

Preliminary Structural Analysis

Report No. D181-945-007

15 May 1956

BELL Aircraft CORP.

[redacted]

BELL *Aircraft* **CORPORATION**
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REPORT NO. D181-945-007

PRELIMINARY STRUCTURAL ANALYSIS

DATE 15 May 1956

Form G650C Rev 655

AFFIX TO DOCUMENT

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By direction of
Chief of Naval Research (Code 466)

Report No. D181-945-007

FOREWORD

Contract Nonr 1675 (00) was awarded to Bell Aircraft Corporation by the Office of Naval Research under sponsorship of the Army Transportation Corps. This is one of a series of five study contracts let to investigate the application of various schemes to the design of Vertical Take-off and Landing (VTOL) or Short Take-off (STO) Assault Transport Aircraft.

The particular field of investigation at Bell Aircraft is the application of ducted propeller propulsion systems to the design of aircraft capable of performing the Assault Transport mission. The results of the investigation are presented in the following listed reports:

TITLE	REPORT NUMBER
Summary Report	D181-945-001
Design Report	D181-945-002
Survey-State-of-the-Art	D181-945-003
Performance	D181-945-004
Stability and Control	D181-945-005
Duct and Propeller Analysis	D181-945-006
Preliminary Structural Analysis	D181-945-007
Standard Aircraft Characteristics	D181-945-008

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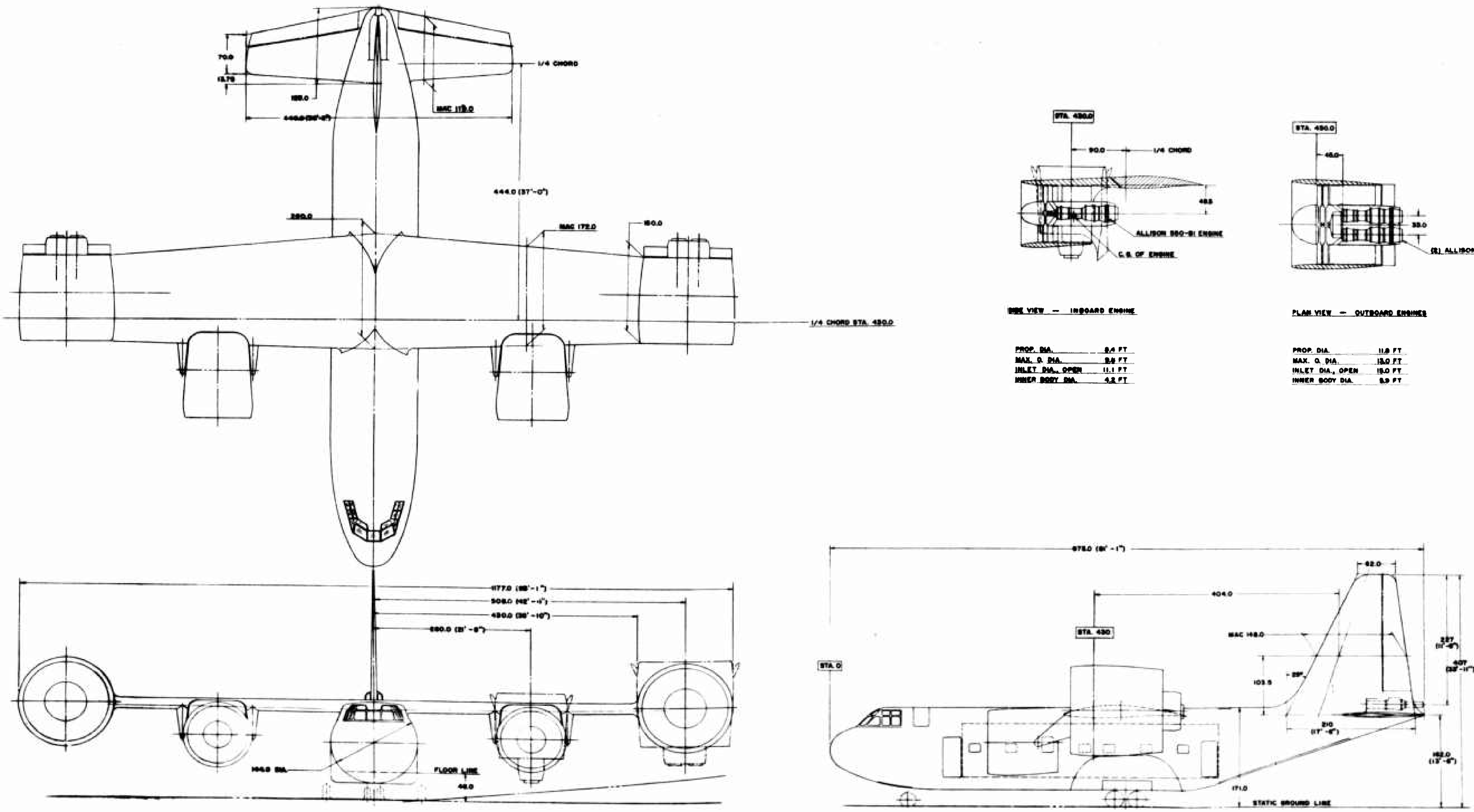
REFERENCES

1. MIL-A-8629 (Aer), "Military Specification Airplane Strength and Rigidity," dated 28 August, 1953.
2. MIL-S-8698 (ASG), "Military Specification Structural Design Requirements, Helicopters", dated 1 July, 1954.
3. Bell Aircraft Corporation Report No. 02-984-035, "Optimum Stresses in Structural Elements," by H. S. Wolko, dated 10 January 1956



Dwg. No. D181-960-009: Four-Duct Allison 550-B1 Tilting Engine Configuration

I. SUMMARY

The VTOL Ducted Propeller Assault Transport, configuration D181-960-009, is analyzed and described from a structural standpoint in this report. Structurally the transport is conventional in both arrangement and construction with the exception of the rotating ducts and engines. The location of these ducts and engines in the wing influences wing structural design because of the large mass of the units and also because of the high thrust values inherent in the VTOL aircraft. The fuselage is constructed similarly to other transport types, and has large cut-out areas for cargo loading and landing gear.

Appropriate additions or modifications have been made to the current conventional aircraft conditions in MIL-A-8629 (Aer) (Ref. 1) in order to provide structural criteria for ducted propeller aircraft. The gust condition is found to be the most critical flight condition; however, the wing structure is primarily designed by the vertical take-off and taxiing conditions. Preliminary loads are provided for the various loading conditions to permit estimation of required structural sizes and materials.

The estimated weight and balance calculations contained in this report are consistent with the preliminary structural weight data employed in the analysis.

II. INTRODUCTION

A preliminary structural analysis, including design criteria, structural description, loads, and estimated weights is presented in this report for the ducted propeller assault transport, D181-960-009 configuration. The structural design is predicated on the standard Navy requirements set forth in the Military Specification MIL-A-8629 (Aer) Airplane Strength and Rigidity, with deviations and alterations necessary for this VTOL aircraft which are discussed in Section III along with definition of critical loading conditions. Emphasis is placed on outlining criteria and load conditions, and presenting a description of the primary structural design as affected by the loading conditions and mission requirements for this particular configuration.

A description of the structural design required to satisfy the various loading conditions is summarized in Section IV. Preliminary load calculations are presented in the following Section V to supply a basis for selection of approximate structural member sizes and materials. The concluding Section VI contains a weight and balance table which is estimated from the preliminary structural drawings and the results of the preliminary structural analysis.

III. BASIC STRUCTURAL DESIGN CRITERIA

A. General

The applicable design specification is MIL-A-8629(Aer) (Ref. 1). The assault transport under investigation is classed as type VR and is designed for a maximum symmetrical flight maneuver load factor of 3.0. However, the airplane strength level for flight is partially determined by gust considerations which require a design limit load factor of 3.2. The wing basic structure is primarily designed by the vertical takeoff condition at a vertical load factor of 1.75. Because of the large concentrated weights of the ducts and engines, the wing lower surface may be designed by the ground taxiing condition requiring a load factor of 1.67. The airplane is designed for vertical landing only, at a load factor of 2.67. Since a maximum oleo deflection of 10 inches is provided, an estimated oleo efficiency of 75 percent is required. The available horsepower and thrust delivered by the six Allison 550-B1 engines limits the maximum level flight speed of the normal configuration to 400 knots EAS. Since the engines (ducts) are designed to be rotated in an approach to vertical landing, the design conditions applicable to the conventional landing-approach will be applied to the transition configuration (when the engines are rotated from the normal horizontal attitude). Therefore, for structural design purposes the landing approach limit speed is selected as the maximum design speed for the transition. This criterion seems reasonable since it is anticipated that the rotation of the engines will commence at about 120 knots in the typical approach for a vertical landing. The landing-approach limit speed is 175 knots.

In load calculations involving the transition, it must be considered that the engine thrust vertical component and the aerodynamic lift on the rotated ducts will be providing additional components to the lift given by the conventional surfaces.

All control systems are unassisted by power boost mechanisms. However, longitudinal control during hovering or at low speeds is provided by the jet thrust from a J-85 engine located in the airplane aft fuselage in the region of the vertical tail. Lateral and directional controls are provided by controllable surfaces located in the aft portion of the outboard ducts. These surfaces when deflected are loaded by pressures resulting from the ducted propeller exit velocities.

The unique design of an aircraft with rotating engines necessitates consideration of certain design conditions not covered by the applicable specification. These will be pointed out in the following paragraphs and sections.

B. Basic Data

1. Airplane Geometry (Ref. BAC D181-960-009)

ITEM	AIRFOIL SECTION	M.A.C. Inches	$C_{L\alpha}$ M = O	S* Sq. Ft.	MAXIMUM DEFLECTION
Wing	NACA 64A-412	172.0	.075/deg.	1240	
Vert. Tail	NACA 65A-008	148.0	.045/deg.	232	
Horiz. Tail	NACA 65A-008	130.0	.059/deg.	310	
Rudder					±30°
Elevator					±25°
Aileron					±20° 15°**

The engines can be rotated at a maximum angular velocity of 15°/sec.
* Area is that used in determining $C_{L\alpha}$
**In vertical position with split flaps, 15° deflection per flap.

2. Weight and C.G.

The most forward C.G. location is estimated to be at 17% MAC. with the airplane empty. The most aft C.G. is estimated to be at 17.6% MAC. at maximum gross weight. For structural design purposes a one percent C.G. position tolerance is to be applied to these limits. However, in all preliminary structural calculations the C.G. location is taken as 17.6% MAC. The estimated weights used for structural analysis are presented in Table I.

TABLE I: STRUCTURAL GROSS WEIGHTS

CONDITION	WEIGHT (lbs.)	C.G. % MAC	I_x (Roll) Slug Ft.^{2}	I_y (Pitch) Slug Ft.^{2}	I_z (Yaw) Slug Ft.^{2}
Empty	42,000	17.0	854,000	213,000	1,011,000
Minimum Flying	45,000				
Basic Landing	58,000				
Basic Flight	61,000				
Basic Takeoff	68,000	17.6	873,000	293,000	1,112,000

3. Design Speeds

Conventional Flight:

V_H = 400 Knots EAS (Maximum level flight speed with maximum available thrust)

$V_L = 1.2\ V_H$ = 480 Knots EAS (Ref. Table I of MIL-A-8629 Aer)

V_S = 100 Knots

For Gust Condition: $V = 350$ Knots EAS (Maximum level flight speed with normal rated power)

Landing Approach Speed: $V = 1.75\ V_S = 175$ Knots (Ref. Table I of MIL-A-8629 Aer)

Transition:

The maximum speed for transition is taken to be the landing-approach speed. For structural design purposes, the ducts (engines) are assumed rotated at any possible angle of attack at this design speed of 175 knots.

C. Symmetrical Flight Conditions

1. Symmetrical Maneuver Without Pitching Acceleration

The vertical load factors, gross weights, and design configurations are given in the following table:

Configuration	G.W.	Factor	
		+	−
Basic Flight	61,000	3.0	1.0
Basic (Maximum Takeoff)	68,000	2.5	0
Landing-Approach	61,000	2.0	0

The pitching velocities, engine speeds, and design airspeeds are as specified in Section 3.4.2 of MIL-A-8629(Aer). The control surfaces are in positions such that the airplane pitching acceleration is zero.

The transition configuration is also considered and is designed for the same gross weight, maneuver factors, pitching velocities, engine speeds, and design airspeeds as the landing-approach configuration. For preliminary design purposes, the ducts (engines) are considered rotated at only two different angles of attack, (1) the angle of maximum lift, and (2) the angle of maximum drag.

2. Symmetrical Maneuver with Pitching Acceleration

Only the basic configuration is considered. The flight conditions of item (1) are combined with pitching accelerations (maximum of 2 rad/sec^2) as specified in Section 3.4.2 of MIL-A-8629(Aer). The maneuver load factors of item (1) are maintained.

3. Gust

The airplane is subjected to a 50 f.p.s. gust while in straight and level flight at an airspeed of 350 knots at normal rated power. Only the basic configuration is applicable.

Assuming the 61,000 lb. gross weight to be critical, the following vertical load factors are developed:

Wing loading at 1g = 49.1 p.s.f.

S_W = 1240 Sq. Ft.

V_H = 350 Knots

U_e = 50 fps

a = slope of lift curve = .075 per degree or 4.3 per radian based on wing area of 1240 sq. ft. (This includes lifting on outboard and inboard ducts)

$$\frac{\Delta n_g}{K_g} = \frac{50(350)(4.3)}{498\ (49.1)} = 3.08$$

$$\mu = \frac{122{,}000}{(.002378)(32.2)(14.33)(4.3)(1240)} = 20.8$$

$$K_g = .70$$

Therefore $\Delta n_g = (3.08)(.70) = 2.16$

TOTAL $n_g = 3.16$

Design Limit Load Factors are 3.2 and -1.2

4. <u>Vertical Takeoff</u>

The engines (ducts) of the airplane are rotated in a vertical attitude, and there is no forward or vertical motion of the airplane. An abrupt change in propeller pitch at takeoff is considered to produce momentarily a 50 percent increase in standard day engine thrust. The maximum rated power is applicable to this condition so the total vertical thrust developed is 1.5 times the maximum available thrust. The resulting vertical load factor is

$$n_z = \frac{1.5 \text{ x max. thrust}}{W}$$

D. <u>Roll Maneuvers</u>

1. <u>Rolling Pull-Out</u>

The airplane configuration is the basic. The vertical load factors at the applicable gross weights are 0.8 of the load factors given in item (1). The engine speed and thrust, and the design airspeeds are as specified in Section 3.4.2.3 of MIL-A-8629(Aer). The airplane is considered in a constant rolling maneuver, resulting from aileron deflection, at the specified speed and load factor. For preliminary purposes the maximum rolling velocity may be taken as $\phi = \frac{1.4V}{b}$ rad/sec (based on Military Specification MIL-8785, "Flying Qualities of Piloted Airplanes") where V is airspeed in f.p.s. and b is total wing span in feet). Maximum rolling velocity at 480 knots is

$$\dot{\phi} = \frac{1.4}{105.2}(810) \qquad 1.0 \text{ rad/sec}$$

An unbalanced wing loading of 125 percent - 75 percent is assumed for structural design.

The airplane is also considered in an accelerating roll. The maximum acceleration is assumed to occur at a rolling velocity of one-half the maximum roll values determined at the applicable airspeeds mentioned above. Engine speed and thrust are combat rated. In lieu of detailed flight load analysis prescribed in MIL-A-8629(Aer) assume full instantaneous aileron deflection to compute rolling moment for establishing maximum rolling accelerations (neglect damping).

2. Roll with Unsymmetrical Thrust

This maneuver is applicable to the transition configuration at a gross weight of 61,000 pounds. The airplane is considered initially in straight and level flight at a speed of 175 knots while the ducts (engines) are in a vertical position or at angle of maximum drag. Any one engine is considered suddenly inoperative producing a rolling acceleration. The operative engines are delivering takeoff power. The rolling acceleration is assumed to occur at zero rolling velocity or in combination with a rolling velocity of 0.2 rad/sec. (This rolling velocity is 1/2 the value determined from $\dot{\phi} = \frac{1.4V}{105.2}$).

3. Hovering Roll

The airplane is in the transition configuration in a wings level horizontal attitude with the ducts (engines) rotated in the vertical position. There is no forward motion of the airplane. A rolling maneuver results from a sudden full displacement of the split flaps in one outboard duct. The engines are delivering takeoff power.

This condition is also investigated for rolling resulting from sudden power loss of one engine. In this case there is no lateral control displacement.

E. Other Flight Conditions

1. Steady Angle of Sideslip

This condition is considered for the landing-approach and transition configurations only and is specified in Section 3.4.2.5 of MIL-A-8629(Aer). The design airspeeds are all values up to 175 knots for both the landing approach and the transition configurations. In the case of the transition configuration the engines (ducts) are considered rotated at only the maximum lift angle and the maximum drag angle. For preliminary purposes the maximum yawing velocity is 2.0 rad/sec.

2. Yawing

The airplane is considered in the basic, landing approach, and transition configurations. The design airspeeds are all values up to 240 knots for the basic configuration and all values up to 175 knots for the landing-approach and transition configurations. The airplane is initially in straight and level flight when the rudder is abruptly displaced to its maximum position. The rudder is held at this displacement until the resultant maximum (dynamic) angle of sideslip is attained. This angle is determined by applying a dynamic overshoot factor of 1.5 to the steady state yaw angle. (The dynamic factor of 1.5 is used in lieu of detailed dynamic condition calculations, and is based on preliminary estimates of actual dynamic response.) The yawing acceleration resulting from this maneuver is considered at zero yawing velocity and in combination with a yawing velocity of 1.0 rad/sec (to be verified by calculation).

For preliminary purposes, when the transition configuration is investigated, the ducts (engines) are considered rotated at only the maximum lift angle and the maximum drag angle.

3. Unsymmetrical Thrust Plus Side Gust

The airplane is in the basic configuration. Two engines in one of the outboard ducts are considered to be inoperative and their propellers set at minimum drag. (Emergency operation on one engine in this duct may result in 2 engines out.) The airplane is trimmed for level, straight flight at zero sideslip with the specified unsymmetrical thrust. The airspeed is the maximum attainable under these conditions when all operative engines are delivering normal-rated power and is taken to be 240 knots. A side gust of 50 f.p.s. is encountered.

4. Unsymmetrical Horizontal Tail Loads

The airloads on the horizontal tail are distributed unsymmetrically at 70%-30% as well as symmetrically.

F. Ground Conditions

1. Taxiing at Takeoff

The airplane is taxiing for takeoff and the engines (ducts) are rotated in any position. There is no wing lift and the engine power is any value between idling and maximum takeoff. The vertical load factor is

$$n_z = 1.67$$

2. Symmetrical Landing

A design sinking speed of 8 f.p.s. is used for vertical landings. This sinking speed is required by both ref. 1 for conventional aircraft, and ref. 2, for helicopters. The airplane is not designed for conventional horizontal landings. There is no airplane horizontal motion and the engines (in a vertical attitude) are considered to provide a vertical thrust of 2/3 W. The basic landing gross weight of 58,000 pounds is applicable.

The design ground reaction factor is 2.0 and the maximum oleo deflection is 10 inches. In order not to exceed the design load factor under the 8 fps sinking speed, the oleo must be designed for an estimated efficiency of 75 percent. The airplane landing load factor is 2.67.

3. Side Drift Landing

This condition, as specified in Section 3.5.1.6 of MIL-A-8629(Aer), is applicable for the vertical landing with ducts (engines) in a vertical attitude. The engine thrust is the same as in item (2).

4. Standing at Rest, Engines Vertical

The airplane is resting on the ground with its engines rotated in a vertical position. Any two or four engines are in operation so as to load the airplane symmetrically. These operating engines are assumed to deliver 1.5 times their standard day maximum rated power and thrust due to abrupt change in propeller pitch. This condition may result in critical duct and engine loads.

G. Engine and Duct Criteria

The unusual design and location of power plant installations necessitates careful consideration of the design criteria for these items. Loads imposed on the ducts and engines may design basic wing structure, as well as the ducts and engines and their supports. No arbitrary criteria is utilized.

The design load factors, and angular velocities and accelerations are compatible with the flight and ground conditions. The corresponding engine torque, gyroscopic moments and airloads are considered with each design condition.

Rigidity of the ducts in the region of the propeller blades is to be provided such that the blades do not strike the ducts at any time during operation.

IV. STRUCTURAL DESCRIPTION

A. General

The structural configuration of the D181 assault transport is generally conventional in that aluminum alloy, stringer stiffened shell structure is used for the pressurized fuselage and the lifting surfaces. The fuselage structure contains a number of door and window cutouts, typical of a transport; in particular there is a large cargo loading door in the rear lower surface of the fuselage. All cutouts are longeron reinforced. Unconventional aspects of the structure arise from the ducted fans. Each fan, complete with engine, is carried in a nacelle structure, supported in turn by radial spokes within the ducts. The complete duct assemblies, one at each wing tip and one at the 60% of span station of each wing panel, are hinged about the pitch axis.

In view of the conventional structure, the minimum of stress analysis has been performed, to justify feasibility and the weight estimate. This section therefore contains only a structural description and, where necessary, a brief discussion of the reasons for the structural arrangements.

B. Lifting Surfaces

The lifting surfaces (wing, horizontal and vertical tails) are stringer stiffened covers of 7075-T6 aluminum alloy material with three spanwise shear webs. This type construction is the optimum structure for the low intensity cover loading present in this configuration, and has been selected by the methods of Section VIII Part II of Reference 3.

The wing is made up of two panels of which the structural section, between the front and rear spars, carries through the fuselage. These two panels are spliced together at the airplane centerline by means of match angle fittings. Ribs are provided at the splice to distribute the loads. The wing-fuselage attachment is accomplished by bolted connections at four points. Fittings are provided to distribute the loads to the front and rear spars and a root rib. Fittings and ribs are also provided at the inboard and outboard duct support points to distribute the loads from these ducts into the wing structure. Because of the large masses of the ducts located outboard on the wing, the wing has been designed for compression in both the upper and lower surfaces. Critical conditions are vertical take-off (compression in the upper surface) and taxiing (compression in the lower surface). Ribs have been spaced at 20 inch centers, along the wing span to stabilize the stringers.

The vertical tail is attached to the fuselage by six bolts through fittings which attach to the three spars and a closure rib. The load is distributed to the fuselage by fittings which are fastened to three fuselage frames. The horizontal tail is fabricated as two outer panels which are fastened directly to the fuselage by match angle fittings. The mating fuselage frames provide the stabilizer carry-through structure across the fuselage and supply the bending rigidity required. This is accomplished by providing a web with upper and lower caps across each frame. Large doublers at the stabilizer root collect the stringer loads and concentrate them at the spar caps. Ribs have been spaced along the span of both the fin and stabilizer to stabilize the stringers and also to distribute the concentrated hinge loads from the rudder and elevator.

C. Fuselage

The fuselage is constructed primarily of stringer stiffened skin in 2024-T3 aluminum alloy, stabilized by frames. This construction is again dictated by the low axial loading in the skin, which results from the large depth and breadth of the fuselage. Since the fuselage is pressurized, but is not completely circular at all stations, the stringers are also necessary to carry pressure loads not resisted by skin tension.

The fuselage contains a number of doors and windows, a cutout for the wing, and a large cargo loading door in the lower surface at the rear. Reinforcements around these cutouts are sufficiently extensive that four continuous longerons result. Heavy frames are provided to distribute wing, tail surface and landing gear loads.

The cabin area is designed to maintain 8000 feet pressure altitude at 30,000 feet actual altitude and a domed bulkhead is provided at the rear to terminate the pressurized area. Where the cargo loading door removes a large area of the lower fuselage shell, provision is made in the door fastenings to carry the "bursting" loads due to pressure.

The lower part of each frame in the cabin area forms a deep cross-beam supporting the cargo floor, which is aluminum sheet stiffened by longitudinal angle section stringers. Fuel is carried in a flat lined cell between the cabin ceiling and the top outer skin. Again the cabin ceiling is stringer stiffened to carry the fuel weight, while the cell is vented to cabin pressure, so that pressure loads are carried by the outside fuselage shell.

D. Ducts

The inboard duct contains an Allison 550-B1 engine which is mounted conventionally to two rings in the nacelle-type structure forming the hub. The hub is attached to the duct shroud by a set of four spokes, two of which contain trunions about which the entire duct rotates. These spokes extend some distance along the ducts to provide fairing and fore and aft stiffness. An actuator is provided to rotate the ducts and to maintain them statically during normal operation. The annulus type duct has an airfoil cross-section, and consists of an inner and outer skin separated by ribs. The circular shape is maintained by rings, one of which is placed in the plane of the fan to maintain a small gap between the inner wall of the duct and the fan. This latter is a requirement for high fan efficiency. Two truss type struts with skin fairing support the duct and distribute the loads to the front and center wing spars and supporting ribs. The critical design condition for these struts is a lateral load from the ducts assembly.

The outboard ducts are similar to the inboard ducts except that there are two Allison 550-B1 engines mounted in the hub, and the whole duct assembly is supported by a 10 inch steel tube cantilevered from the wing tip. This tube transmits the loads, through bearings and fittings, directly into the wing structure. It also serves as a torque tube providing the duct rotation. The critical condition for this tube is vertical take-off which imposes the maximum bending moments.

E. Landing Gear

For landing, a tricycle type gear mounted in the fuselage has been provided. The nose gear has a dual wheel and is conventional in design. The main gear consists of two tandem wheeled gears mounted at each side of the fuselage. Each gear is mounted to a single fitting which is hinged to the fuselage, thereby making it possible to fold the gear into the fuselage. Because of this, the side load on the gear, which imposes torque on this fitting, is the critical design condition.

V. DESIGN LOADS

A. Symmetrical Flight Maneuvers

1. Symmetrical Steady-State Balance

The basic configuration steady-state balance loads are calculated in Table II for 3.0g, 1.0g, and -1.0g conditions. The maximum symmetrical maneuver wing airloads are those determined for Conditions I and II. The 1.0g loads presented for Condition IV are utilized in calculating gust loads.

The calculations in Table II are based on aerodynamic parameters estimated by the Aerodynamics Section. (See Figures 1 and 2.) Although the fuselage pitching moment contributes greatly to the airplane balance the resultant lift on the fuselage is considered to be zero. The balance equations used in Table II are as follows.

$$\Sigma M_{CG} = M_W + M_D + M_F - L_t \ell_t = 0$$

$$\Sigma L_Z = -nW + L_W + L_t = 0$$

$$\frac{\Sigma M_{CG}}{\ell_t} + \Sigma L_Z = \frac{M_W + M_D + M_F}{\ell_t} - nW + L_W = 0$$

$$(C_{m\alpha})_{W+D+F} \, (\alpha) \, \frac{\bar{c}}{\ell_t} \, q \times S + (C_{L\alpha})_W \, (\alpha) \, q \times S = nW$$

$$\alpha = nW \div q \times S \left[(C_{m\alpha})_{W+D+F} \left(\frac{\bar{c}}{\ell_t}\right) + (C_{L\alpha})_W \right]$$

$$L_t = (C_{m\alpha})_{W+D+F} \, (\alpha) \left(\frac{\bar{c}}{\ell_t}\right) (q \times S)$$

$$L_W = nW - L_t$$

Table II: STEADY-STATE AND DYNAMIC BALANCING LOADS

NO.	ITEM	OPERATION	CONDITION			
			I	II	III	IV
1.	V_e	EAS in Knots	480	480	400	350
2.	M		.73	.73	.61	.53
3.	n		3.0	-1.0	3.0	1.0
4.	W		61,000	61,000	61,000	61,000
5.	nW		183,000	-61,000	183,000	61,000
6.	q x S	$1481\ (1240)(2)^2$	979,000	979,000	684,000	516,000
7.	$(C_{L\alpha})_W$	Ref. Figure 1	.097/deg	.097/deg	.089/deg	.084/deg
8.	$(C_{m\alpha})_{W+F+D}$	$C_{m\alpha_W} + C_{m\alpha_F} + C_{m\alpha_D}$ Ref. Figure 2	.005/deg	.005/deg	.0055/deg	.0057/deg
9.	$\bar{c}$	M.A.C. in Feet	14.3	14.3	14.3	14.3
10.	ℓ_{HT}		38.1	38.1	38.1	38.1
11.	8 $\bar{c}/\ell_{HT}$	8 × 9 ÷ 10	.0019/deg	.0019/deg	.00206/deg	.00214/deg
12.	α	5 ÷ 6 (11 + 7)	1.89°	-0.63°	2.94°	1.37°
13.	L_t	6 11 12	3500#	-1200#	4100#	1500#
14.	L_W	5 - 13	179,500#	-59,800	178,900	59,500
		DYNAMIC				
15.	I_y		270,000	270,000	270,000	
16.	M	$I \times \ddot{\alpha} = \pm 2.0 \times$ 15	540,000	-540,000	540,000	
17.	ΔL_{HT}	16 ÷ 10	14,200	-14,200	14,200	
18.	L_{HT_D}	13 + 7	17,700	-15,400	18,300	
19.	L_{W_D}	14 - 17	165,300	-45,600	164,700	

FIGURE 1: LIFT CURVE SLOPES

NOTE: Lift curve slopes are based on individual lift surface areas given below.

S_W = 1240 Sq.Ft. (Wing reference area)

S_{HT} = 310 Sq.Ft. (Horizontal tail area)

S_{VT} = 232 Sq.Ft. (Vertical tail area)

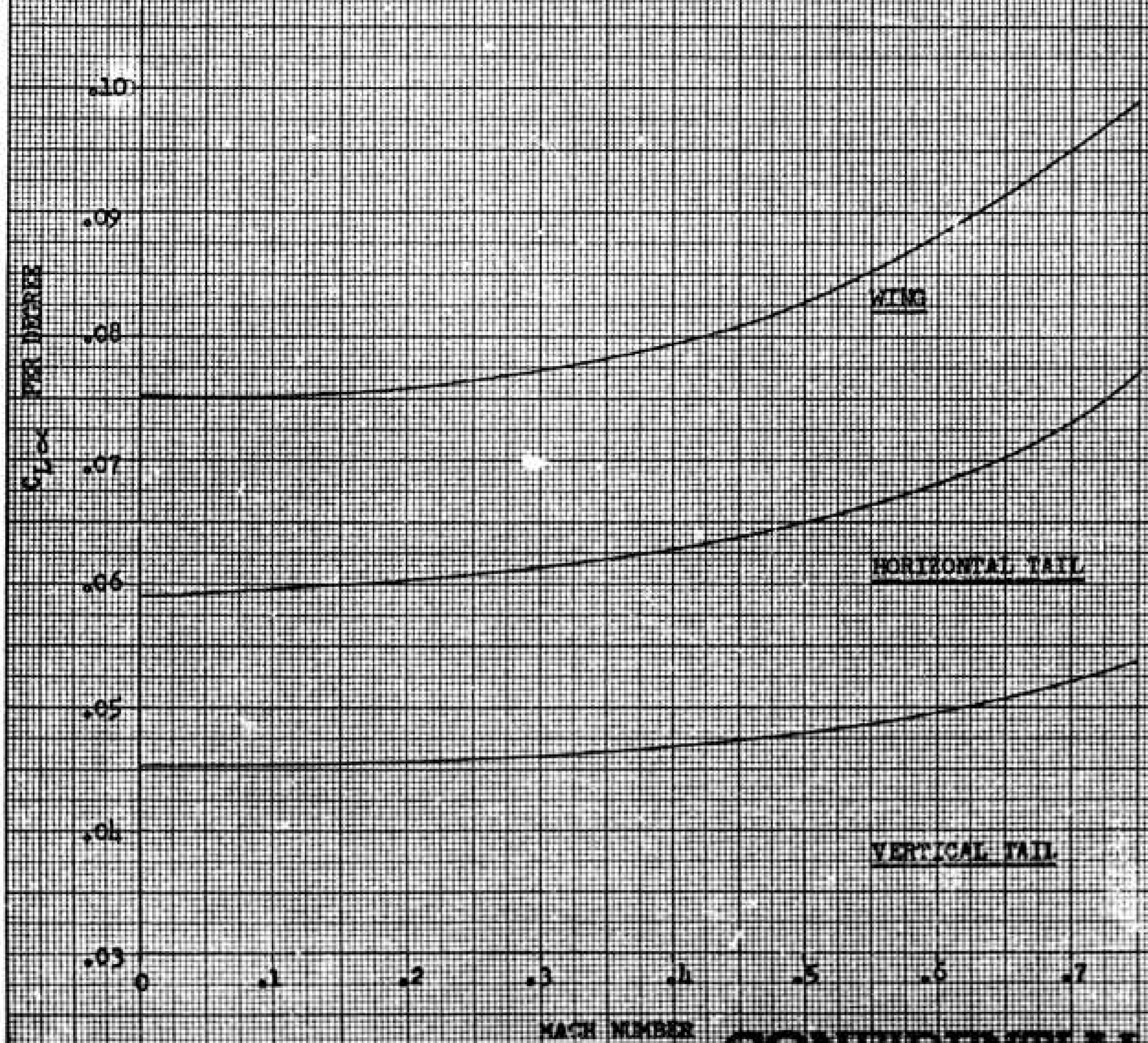
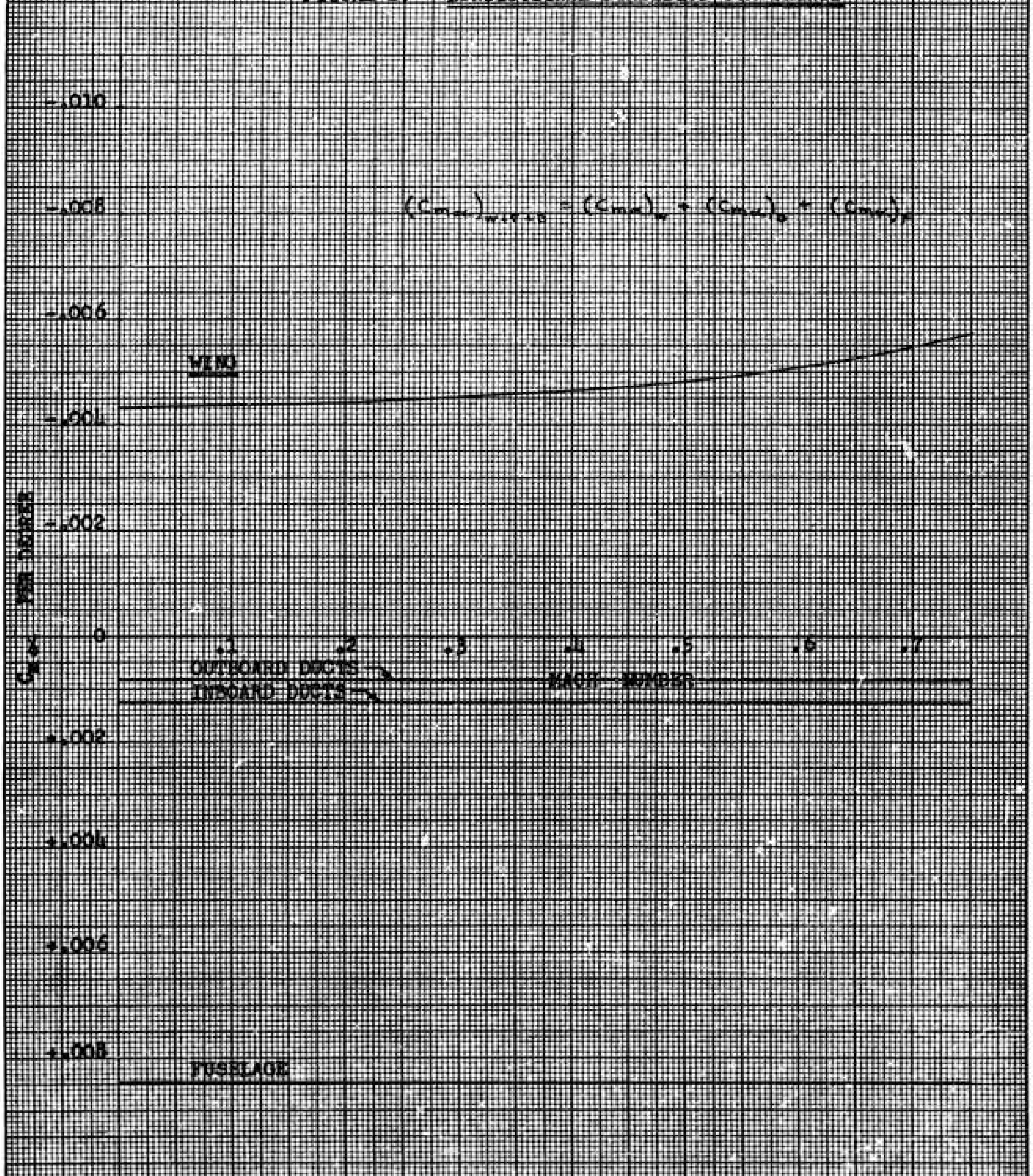


FIGURE 2: LONGITUDINAL STABILITY COMPONENTS



NOTE: These values are based on Reference Area of 1240 Sq. Ft. and Chord of 172 inches. Moment is about 17.6 percent M.A.C. C.G.

2. Symmetrical Maneuver with Pitching Acceleration

The pitching acceleration of 2.0 radians/sec^2 is added to the steady-state balance so that the incremental balancing moment is in a stabilizing direction (nose down for positive load factors). The basic configuration horizontal tail loads for this maneuver are calculated in Table II for the same conditions that are considered in the symmetrical steady-state balance. A comparison of conditions I and III indicates that slightly higher maneuvering tail loads are developed at lower airplane velocities. However, these loads are not investigated further in this preliminary work for it is obvious that the horizontal tail is designed by the loads developed in the gust condition (see item C).

3. Vertical Take-Off

The maximum available standard day thrust with the engines in the vertical position is 79,300 lbs. A 50 percent increase in this thrust due to abrupt propeller pitch results in a vertical load factor of

$$n_Z = \frac{(1.5)(79,300)}{68,000} = 1.75$$

There is no wind airload accompanying the vertical take-off maneuver; however, the vertical thrust of 119,000 lbs. is applied to the wing.

B. Rolling Pull-Out

The wing unsymmetrical load distribution is considered to be 125 percent - 75 percent. The vertical maneuver load factor is (0.8) (3.0) or 2.4. Therefore the total wing airload is .8 of the steady-state balance wing load shown in Table II for Condition I.

$$L_W = 0.8\ (179,500) = 143,700 \text{ lbs.}$$

$$L_{W_{unsym.}} = (1.25)(0.5)(143,700) = 90,000 \text{ lbs. one side}$$

C. Gust Loads

The 1.0g steady-state condition surface loads, for an airspeed of 350 knots, are taken from Table II. To these are added the Δn for gusts at the corresponding flight condition.

1. Wing Gust Loads

For wing analysis the Δn for gust is applied entirely to the wing.

$F_W = 59{,}500$ Lbs.

$\Delta n = 2.2$

$\Delta nW = (2.2)\ (61{,}000) = 134{,}200$ Lbs.

Total wing gust load

$F_{W_g} = (59{,}500) + 134{,}200$

$= 193{,}700$ Lbs.

2. Horizontal Tail Gust Loads

The 1.0g horizontal tail load is combined with a gust load based upon increase in horizontal tail angle of attack. A horizontal tail gust alleviation factor of 1.1 K_g is assumed.

$K_{HT} = (1.1)\ (.7) = 0.77$

$F_{HT_{1g}} = 1500$ Lbs.

$U = 50$ f.p.s.

$V_T = 591$ f.p.s.

$C_{L\alpha_{HT}} = .066$

$F_{HT_g} = K_{HT_{1g}}\ (U/V)\ 57.3\ (C_{L\alpha})_{HT}\ (q \times S)$

$= 0.77\ (50 \times 57.3/591)(0.066)(1481 \times 0.53^2)(310)$

$= \underline{31{,}800}$ Lbs.

Total $F_{HT_g} = 1500 + 31{,}800 = 33{,}300$ Lbs.

3. Vertical Tail Gust Loads

The vertical tail gust loading is investigated at an airspeed of 350 knots with the initial β = 0 and the maximum gust velocity perpendicular to the vertical tail.

$$F_{VT_g} = (50 \times 57.3/591)(.0485)(1481 \times 0.53^2)(232)$$
$$= 22,700 \text{ Lbs.}$$

The maximum vertical tail gust loading at an airspeed of 240 knots is

$$F_{VT_g} = (50 \times 57.3/405)(.0465)(1481 \times 0.363^2)(232)$$
$$= 14,900 \text{ Lbs.}$$

D. Vertical Tail Loads

Vertical tail loads are analyzed for two different flight conditions, unsymmetrical thrust plus side gust, and yawing.

1. Unsymmetrical Thrust Plus Side Gust

Two outboard engines are inoperative and the airplane is trimmed for straight and level flight at zero sideslip at an airspeed of 240 knots. The thrust from the four operating engines is 24,000 lbs. and the drag load on the two inoperative engines (propellers set at minimum drag) is 3000 lbs. A side gust of 50 f.p.s. is encountered producing a vertical tail load of 14,900 lbs. (Item 3 of C).

Distance c.g. to outboard engines is 42.3 feet.

$$\ell_{VT} = 34.8 \text{ feet}$$

Vertical tail load for trim

$$L_{VT_{trim}} = 15,000 \ \frac{42.3}{34.8} = 18,200 \text{ Lbs.}$$
$$L_{VT_G} = 14,900 \text{ Lbs.}$$
$$\text{Total } L_{VT} = 33,100 \text{ Lbs.}$$

2. Yawing Maneuver

The airplane is initially in straight and level flight at an airspeed of 240 knots and at a zero yaw angle. The rudder is abruptly deflected to its maximum displacement (30 degrees).

$$\ell_{VT} = 34.8 \text{ feet}; \quad b = 84.6 \text{ feet}$$

$$(C_{n_{\delta r}}) = .00155 \quad \text{(Ref. Aero. Section)}$$

$$\begin{aligned} \text{Moment} &= (C_{n_{\delta r}})\,(\delta_r)\,(b)\,(q)\,(S) \\ &= (.00155)(30)(84.6)(.363)^2(1481)(1240) \\ &= 952{,}000 \text{ ft. lbs.} \end{aligned}$$

$$F_{VT} = 952{,}000/34.8 = \underline{27{,}400} \text{ Lbs.}$$

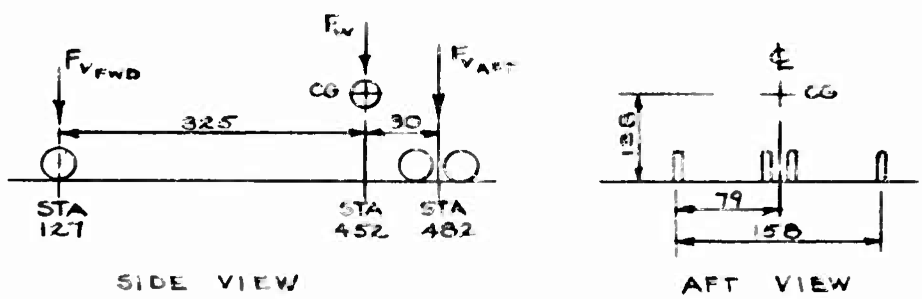
Angular Acceleration

$$\ddot{\psi} = 952{,}000/1{,}100{,}000 = .87 \text{ rad/sec}^2$$

E. Ground Conditions

1. Taxiing at Take-Off

The engines are in the horizontal attitude delivering idling or maximum take-off power. The vertical load factor, n_z, is 1.67 and is applied to the maximum take-off gross weight of 68,000 lbs.



$$F_{V_{FWD}} = (1.67)(68{,}000)\ 30/355 = 9{,}600 \text{ Lbs.}$$

$$F_{V_{AFT}} = (1.67)(68{,}000)\ 325/355 = 104{,}000 \text{ Lbs.}$$

2. Symmetrical Three-Point Vertical Landing

The airplane at the basic landing gross weight of 58,000 lbs. is investigated for a 3 point vertical landing. The sinking speed is 8 f.p.s. and the design ground reaction factor is 2.0. Since the engines are providing a vertical lift of 2/3 W, the airplane landing load factor is 2.67. With a ground reaction factor of 2.0 the total ground load is

$$(2.0)\ (58,000) = 116,000 \text{ Lbs.}$$

Because the oleo travel is 10 inches for each gear (designed for an efficiency of 75%) the ground load is apportioned according to the static loads (as in taxiing at take-off).

$$F_{V_{FWD}} = 116,000 \ \frac{9,600}{113,600} = 9,800 \text{ Lbs.}$$

$$F_{V_{AFT}} = 116,000 \ \frac{104,000}{113,600} = 106,200 \text{ Lbs.}$$

3. Side Drift Landing

The airplane is in the level attitude with only the main gear contacting the ground. A sinking speed of 8.5 fps is used to obtain ultimate gear loads for this condition. At one gear the side component of the ground reaction acts inboard and is 80 percent of the gear vertical load. The other gear ground reaction side component is 60 percent of the gear vertical load and acts outboard.

Weight = 58,000 Lbs.

$$\text{Total Airplane K.E.} = 58,000 \times (8.5)^2/2\ (32.2)$$

$$= 65,100 \text{ ft-lb.}$$

Engine lift is 2/3 W, therefore potential energy

$$P.E. = W/3 \times d$$

Total Energy = 65,100 + 580,000/36 = 81,200 ft. lb.

Total Main Gear Load = 81,200/d (0.75)

= (81,200) (12)/7.5 = 130,000 Lbs.

Ult. $F_{V_{AFT}}$ = 65,000 Lbs. per Gear

Ult. $F_{S_{AFT}}$ = 52,000 Lbs., inboard for one gear

Ult. $F_{S_{AFT}}$ = 39,000 Lbs., outboard for other gear

F. Duct Air Loads and Moments

The airloads and moments developed on the ducts at the landing approach speed of 175 knots (Mach .265) are presented in Table III. Only three different angles of attack are considered; maximum lift (30 degrees), maximum drag (60 degrees), and 90 degrees. Duct power-off lift, drag, and pitching moment coefficients vs. angle of attack are presented in Figures 3, 4, and 5.

Table III: AIRLOADS AND MOMENTS AT 1/4 CHORD OF DUCTS

DUCT		C_L	C_D	C_m	L	D	M
	degrees				lbs.	lbs.	ft. lbs.
Outboard	30°	.16	.09	-.0225	20,600	11,600	-41,500
	60°	.076	.18	-.025	9,800	23,200	-46,100
	90°	0	.122	-.031	0	15,700	-57,100
Inboard	30°	.086	.044	-.012	11,100	5,700	-22,300
	60°	.046	.108	-.015	6,000	13,900	-27,600
	90°	0	.074	-.0185	0	9,500	-34,100

The reference area for the coefficients is 1240 sq. ft.

The reference chord length for C_m is 14.3 feet.

C_{OD} = 12.5 feet

C_{ID} = 9.6 feet

All loads and moments are values per duct.

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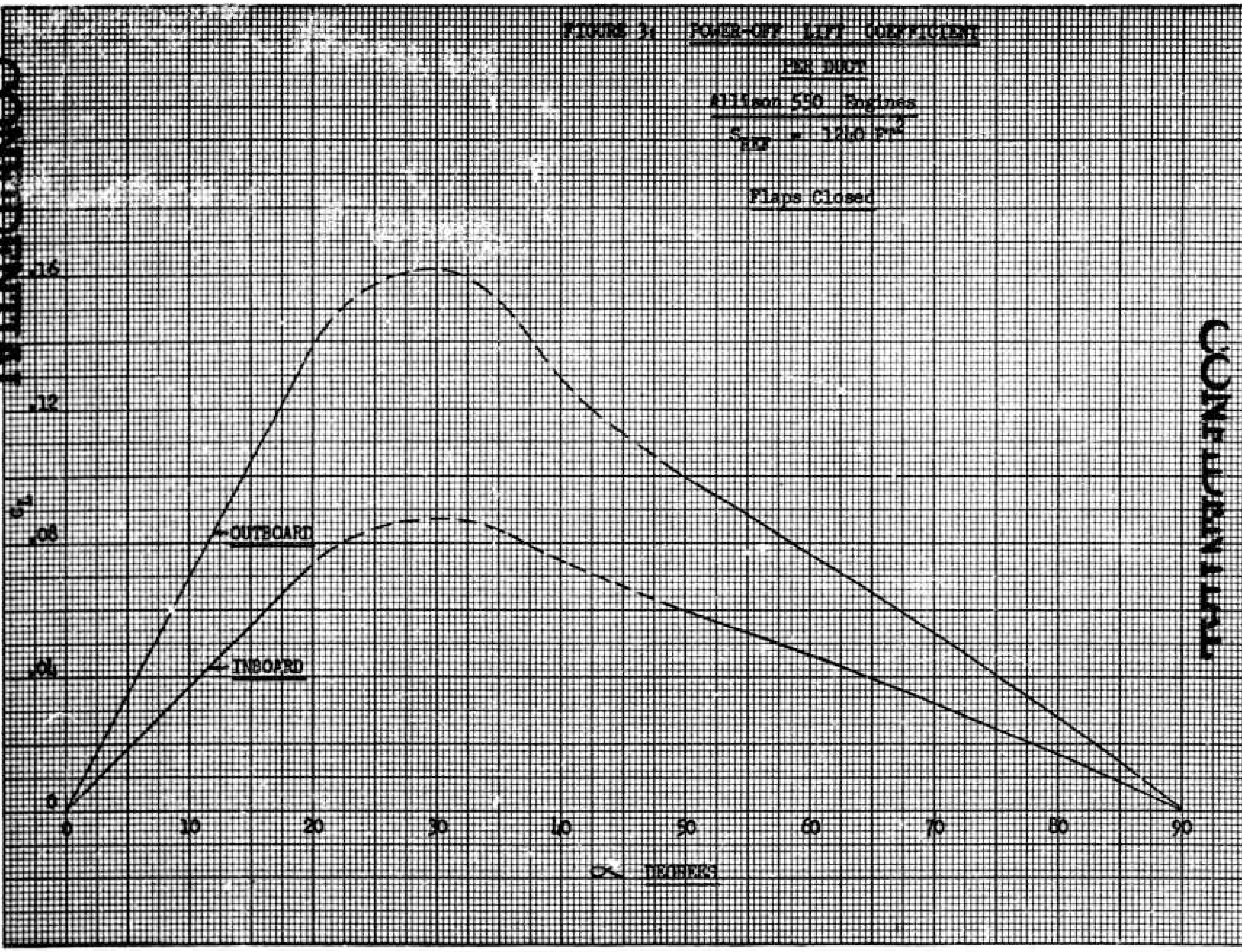

FIGURE 31: POWER-OFF LIFT COEFFICIENT
PER DUCT
Allison 550 Engines
S_{REF} = 1260 FT^2
Flaps Closed

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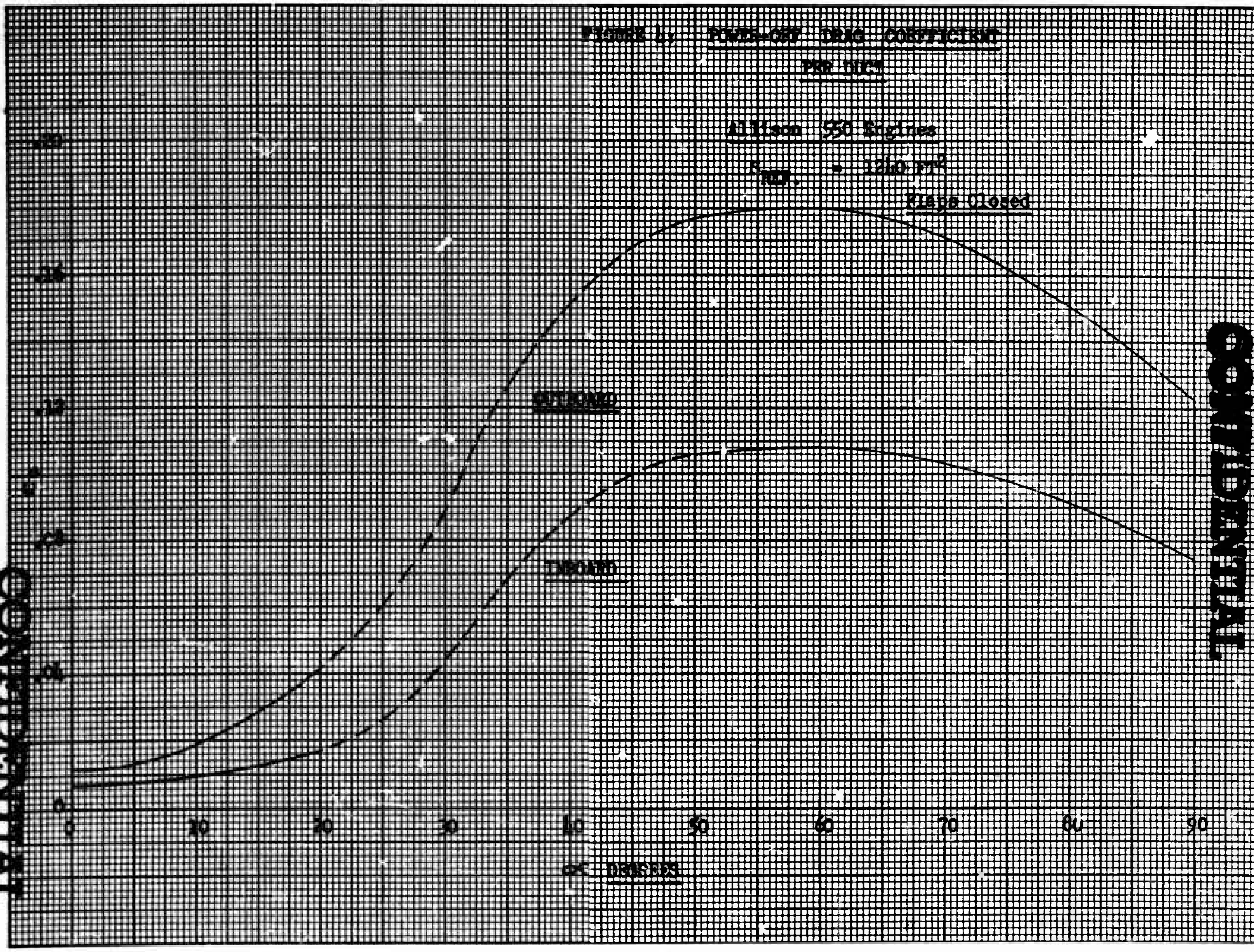

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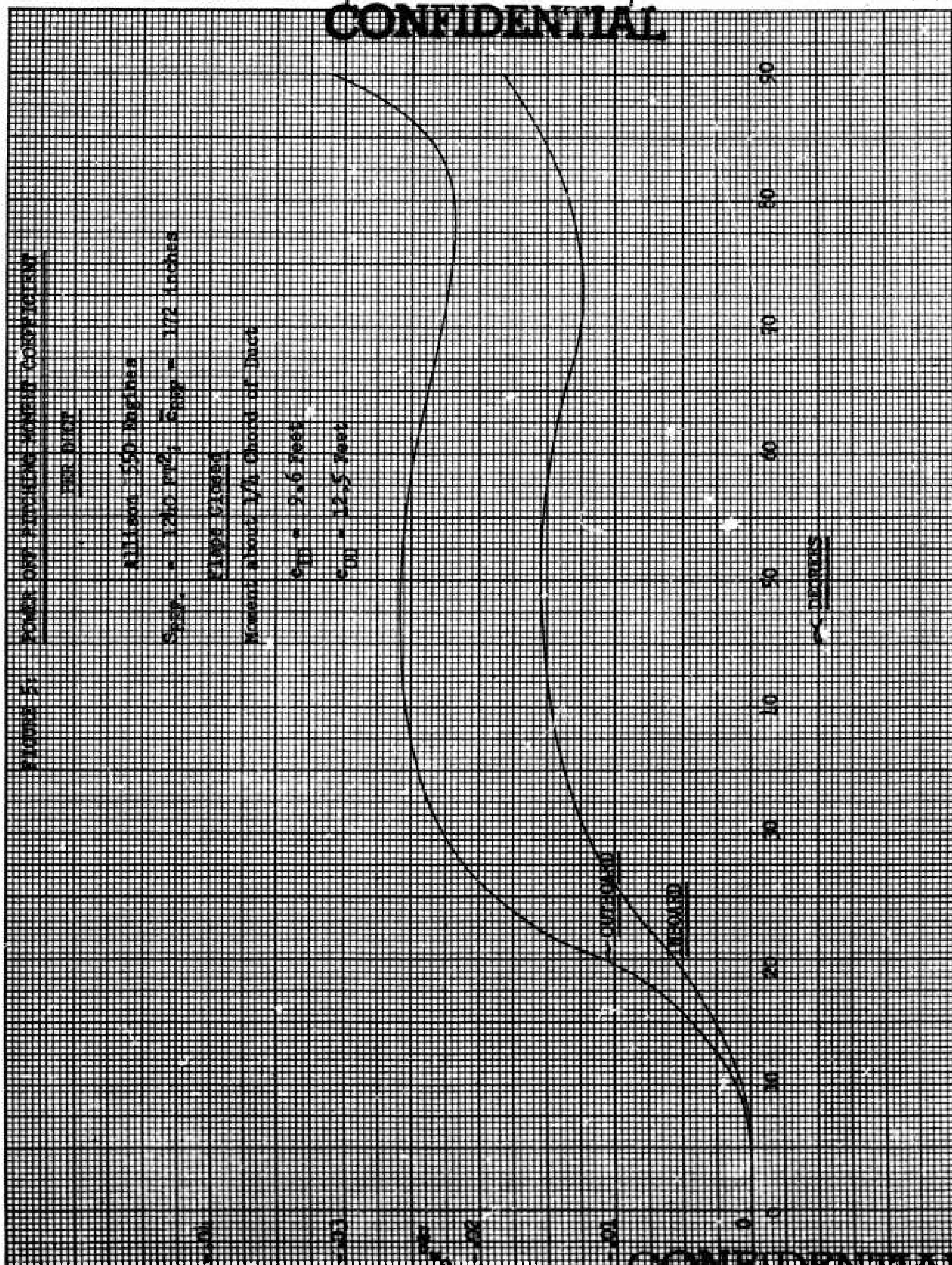

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VI. WEIGHT AND BALANCE

The estimated weight and balance of the Assault Transport, reference BAC Drawing D181-960-009, is presented in Table IV. Conventional methods of weight estimation are used in determining the structural weights. Duct weights are based on data available from previous BAC duct designs.

Table IV: WEIGHT AND BALANCE ESTIMATE

	WEIGHT	ARM	MOMENT	ARM	MOMENT
WING	5200	458	2381600	166	863200
TAIL					
Horizontal	685	900	616500	160	109600
Vertical	486	870	422820	260	126360
BODY	7423	440	3266120	110	816530
LANDING GEAR					
Nose	300	135	40500	30	9000
Main	1930	483	932190	28	54040
SURFACE CONTROLS					
Flight Controls	500	400	200000	135	67500
Reaction Controls (Pitch)	400	920	368000	180	72000
ENGINE SECT. (Duct Around Prop)					
Inboard (Vertical Position)	2780	340	945200	135	375300
Outboard (Vertical Position)	3920	475	1862000	175	686000
PROPULSION					
Engines (2) Inb'd.- Allison 550-B1	3150	340	1071000	107	337050
Engines (4) Outb'd.	6300	475	2992500	140	882000
Inboard Gear Boxes (2)	980	340	333200	165	161700
Outboard Gear Boxes (2)	2180	475	1035500	215	468700
Engine Mounts (Inboard)	125	340	42500	107	13375
(Outboard)	245	475	116375	140	34300

Table IV: WEIGHT AND BALANCE ESTIMATE (Cont'd)

	WEIGHT	ARM	MOMENT	ARM	MOMENT
PROPULSION (Cont'd)					
Duct Supports (Inboard)	300	370	111000	135	40500
(Outboard)	500	475	237500	170	85000
Rotating Mech. (Inboard)	60	370	22200	135	8100
(Outboard)	100	475	47500	170	17000
Lubricating Sys. 6.5#/Gal.	195	400	78000	130	25350
Fuel System 2310 x 0.20#/Gal.	460	430	197800	160	73600
Water Injection System	200	430	86000	135	27000
Engine Controls	50	300	15000	130	6500
Starting System	150	400	60000	120	18000
Propeller Installation					
Inboard (Vertical Position)	1094	340	371960	155	169570
Outboard (Vertical Position)	1522	475	722950	200	304400
Auxiliary Power Plant	80	200	16000	100	8000
Instruments	160	60	9600	140	22400
Hydraulics (Brakes & Nose Ster.)	50	300	15000	60	3000
Electrical	800	155	124000	125	100000
Electronics	500	150	75000	110	55000
Furnishings (No Paratroop Seats)	465	100	46500	120	55800
Air Conditioning & Anti-Ice	500	200	100000	140	70000
Auxiliary Gear (Jacking, Towing)	25	500	12500	150	3750
TOTAL WEIGHT EMPTY	43815		18974515		6169625
USEFUL LOAD					
Crew (3)	645	90	58050	130	83850
Oil-Engines 25 Gal.	188	430	80840	145	27260
Gear Boxes (6% Box Wt.)	140	400	56000	140	19600
Fuel Gals. at 6.5#	13295	430	5716850	150	1994250
Water	1297	430	557710	140	181580
Payload	8000	425	3400000	90	720000
TOTAL USEFUL LOAD	23565		9869450		3026540
GROSS WEIGHT (VTOL Position)					
TOTAL WEIGHT EMPTY	43815		18974515		6169625
USEFUL LOAD	23565		9869450		3026540
GROSS WEIGHT	67380	428.1	28843965	136.5	9196165

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