3209

TL. 113-52/06

AVRO CANADA

C104/2

ALL—WEATHER

FIGHTER

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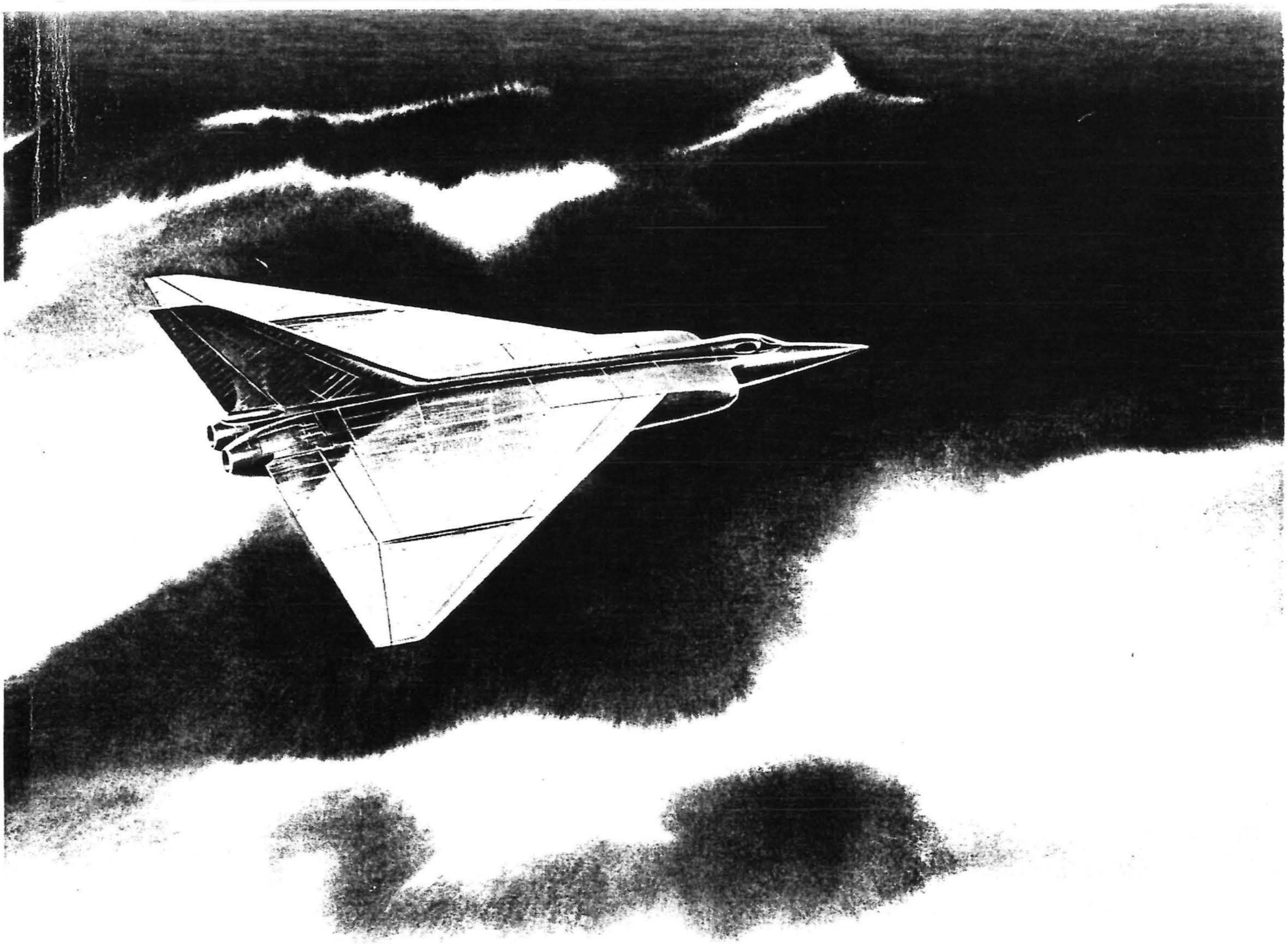

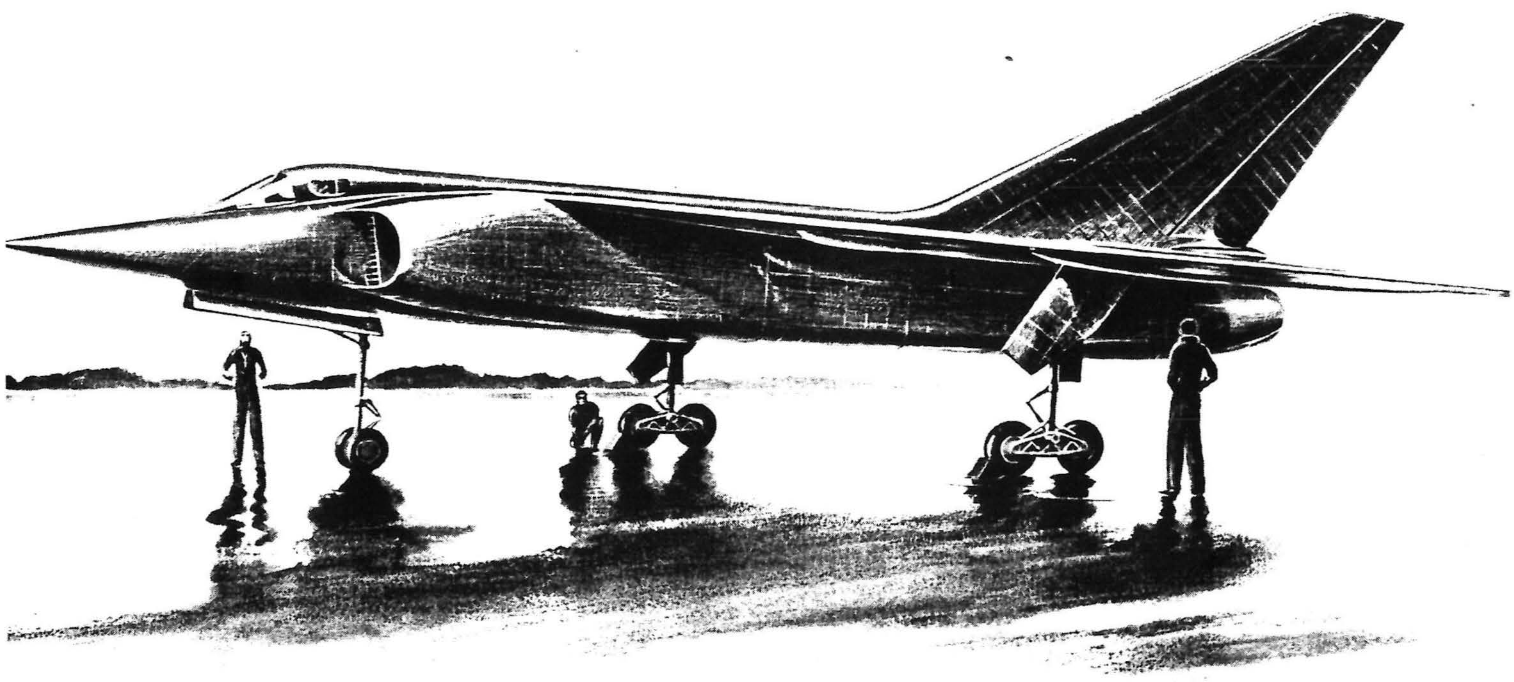

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C 104/2

SUPERSONIC ALL-WEATHER FIGHTER

PROPOSAL NO.2

TWIN ENGINE AIRCRAFT

JUNE 1952

SECRET

AVRO

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

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APPENDICES

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ILLUSTRATIONS

1 SCOPE AND CLASSIFICATION

1.1 Details of the following airplane are covered in this brochure:-

Service Model Designation	Supersonic, all-weather fighter
Designer's Name and Model Designation	A.V.Roe Canada Ltd., - C-104/2
Number and Places for Crew	One (1) Pilot
Number and Kind of Engines........	Two A.V.Roe Canada turbo-jet engines - TR9 or Two Bristol Engine Co. turbo-jet engines - OL3 (fighter version) or Two Curtiss-Wright turbo-jet engines - J67

NOTE: Each type of engine will be fitted with an afterburner.

1.1.1 The mission of this airplane is to intercept and destroy any long-range bombers of the highest performance which are likely to be available to an enemy during the next five to ten years. Guided missiles and air-to-air rockets are used as the main offensive armament, the target tracking, aiming and fire control being automatically computed by airborne electronic equipment working in conjunction with ground signals.

1.1.2 The adaptability of this airplane to perform other missions, in addition to the above basic mission, is discussed in Appendix IV of this brochure and summarized in Appendix V.

2 APPLICABLE SPECIFICATIONS AND OTHER PUBLICATIONS

2.1 Specifications and publications used in the preparation of this brochure are as follows:-

(a) Handbook of Instructions for Aircraft designers, AMC 80-1 Edition, including revisions up to and including April, 1951.

(b) Air Force (USAF) Model Specification MIL-I-6252 dated 18 October, 1950.

3 REQUIREMENTS

3.1 Characteristics:

3.1.1 Refer to Figure 1 on the following page for information on the configuration of the airplane.

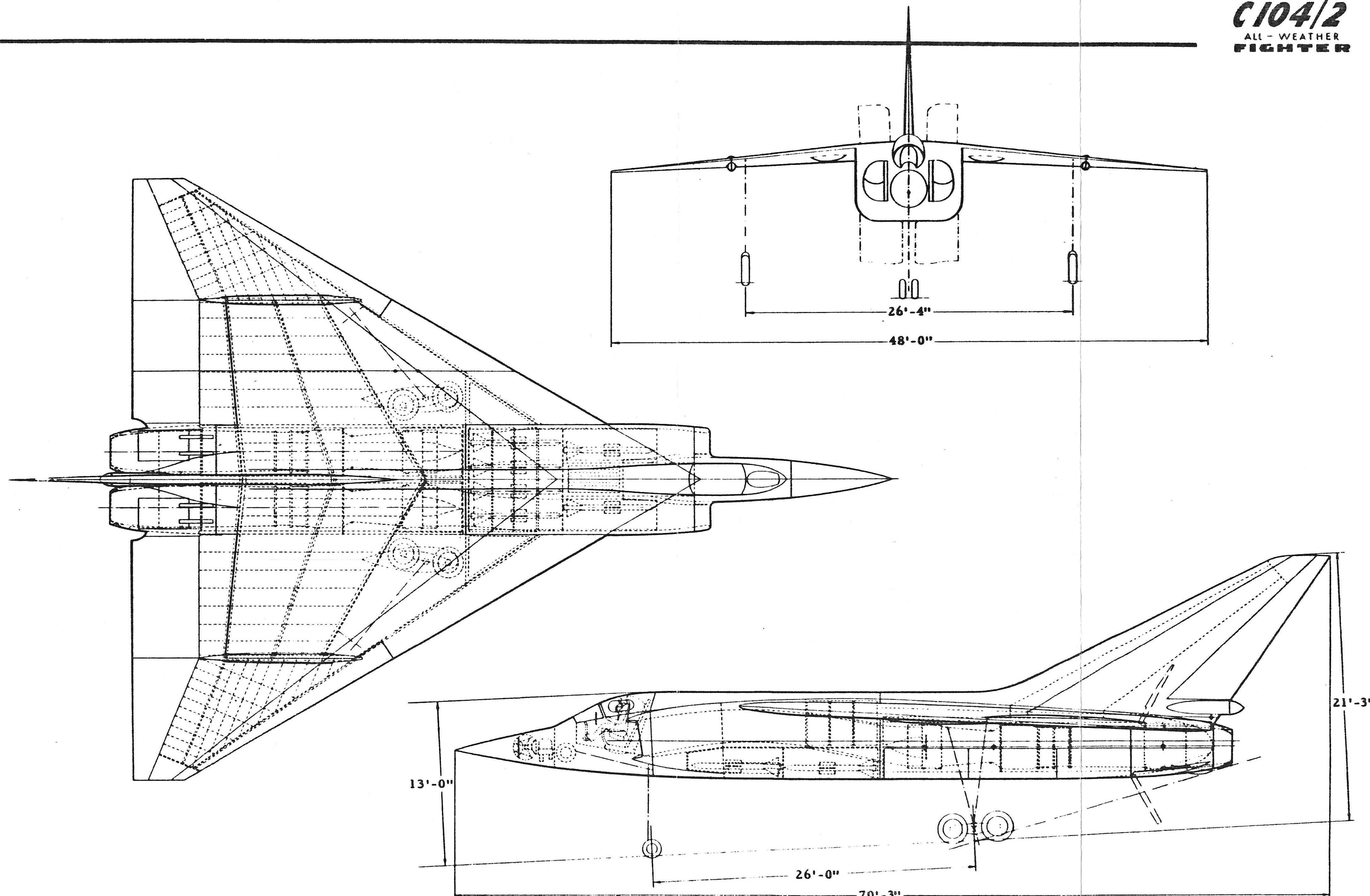


FIG. 1 3 VIEW G.A. OF AIRPLANE

AVRO CANADA
C104/2
ALL-WEATHER
FIGHTER

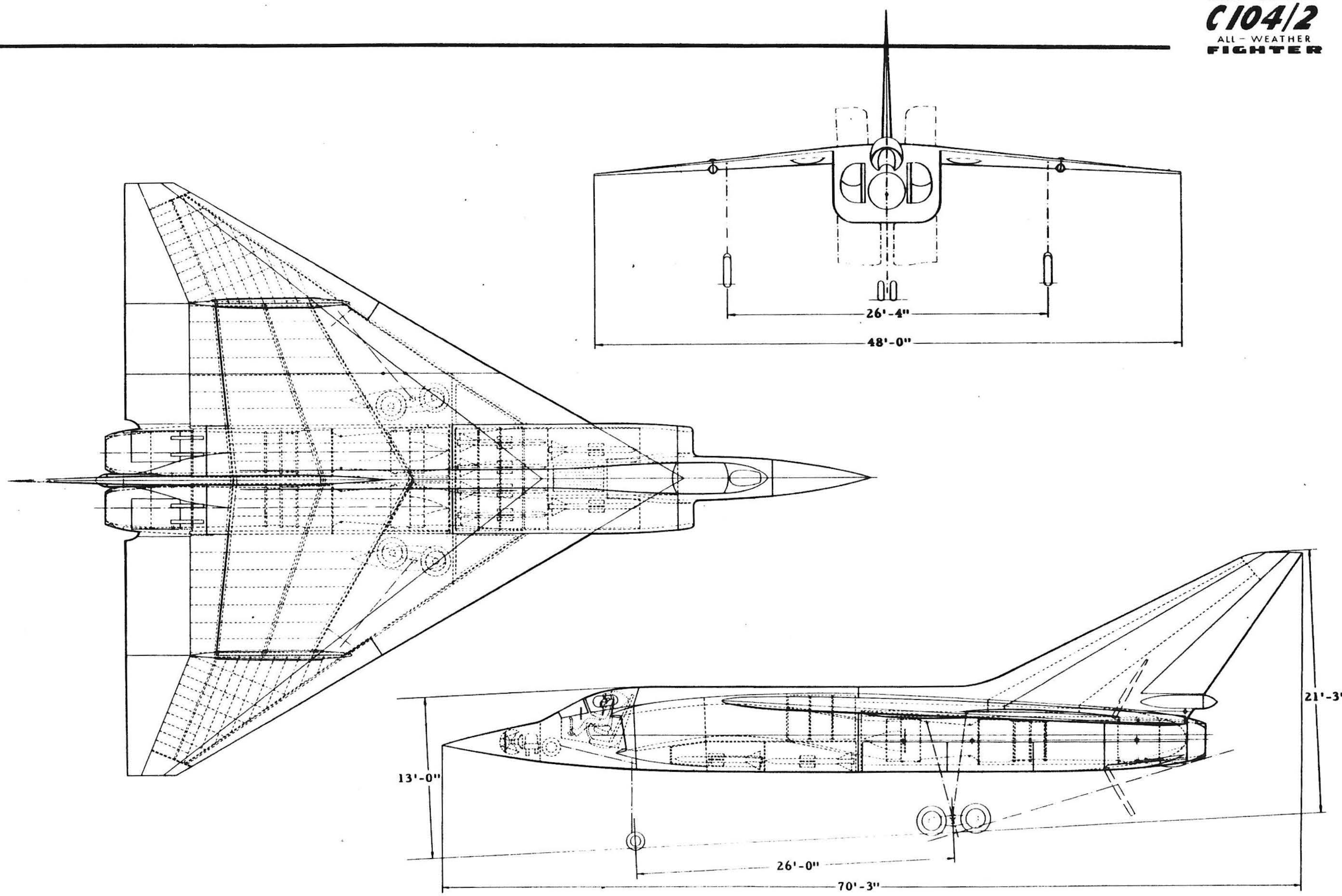


FIG. 1 3 VIEW G.A. OF AIRPLANE

3.1.2 Performance:

3.1.2.1 Tabulated Performance:

TABLE 1
PERFORMANCE UNDER I.C.A.N. STANDARD ATMOSPHERIC CONDITIONS

True Air Speed in Level Flight at Sea Level at Combat Weight (44,000 lb.):	
Maximum Thrust, Afterburners Lit	1,100 knots
Maximum Thrust................................	669 knots
Maximum Continuous Thrust, Afterburners Lit	1,012 knots
Maximum Continuous Thrust	654 knots
True Air Speed in Level Flight at 50,000 ft. Altitude at Combat Weight (44,000 lb.):	
Maximum Thrust, Afterburners Lit	1,140 knots
Maximum Continuous Thrust, Afterburners Lit	1,006 knots
Operational Ceiling (rate of climb = 500 f.p.m.) at Combat Weight (44,000 lb.):	
Maximum Thrust, Afterburners Lit	55,000 ft.
Climb Thrust, Afterburners Lit	53,800 ft.
Maximum Continuous Thrust, Afterburners Lit	51,600 ft.
Maximum Rate of Climb at Sea Level at Maximum Take-off Gross Weight (52,000 lb.):	
Maximum Thrust, Afterburner Lit	42,000 f.p.m.
Climb Thrust, Afterburners Lit	38,300 f.p.m.
Climb Thrust	13,300 f.p.m.
Time to 50,000 ft. Altitude from a Standing Start at Maximum Take-off Gross Weight (52,000 lb.):	
Maximum Thrust, Afterburners Lit	5.2 min.
Climb Thrust, Afterburners Lit	6.8 min.
Take-off Distance over 50 ft. Obstacle with Maximum Thrust, Afterburners Lit, at Sea Level at Maximum Take-off Gross Weight (52,000 lb.)	2,940 ft.
Landing Distance over 50 ft. Obstacle at Sea Level at Combat Weight (44,000 lb.)	5,920 ft.
True Stalling Speed in Landing Configuration at Sea Level at Combat Weight (44,000 lb.)	104 knots
Combat Radius of Action with Combat at 50,000 ft. Altitude:	
High Speed Mission (Table 3)	242 naut.mi.
Maximum Range Mission (Table 4)	390 naut.mi.

3.1.2.1.1 Engine Performance:

TABLE 2 - STATIC PERFORMANCE OF DEVELOPED TR9 ENGINE WITH AFTERBURNER UNDER I.C.A.N. STANDARD SEA LEVEL CONDITIONS

Rating	Time Limit min.	Engine r.p.m.	Thrust lb.	Sp. Fuel Cons. lb/hr/lb.	J.P.T. °C
Maximum Take-off and Combat - Afterburner Lit	15.0 (combined limit)	5,500	21,450	1.848	669+869
Maximum Take-off and Combat - Afterburner Unlit	15.0 (combined limit)	5,500	14,220	.941	669
Climb	30.0	5,400	13,290	.927	631
Maximum Continuous	no limit	5,250	11,780	.901	580

3.1.2.1.2 Combat Radius - High Speed Mission:

TABLE 3 - COMBAT RADIUS OF ACTION - HIGH SPEED MISSION

	Distance naut. mi.	Time min.	Fuel Consumed lb.	Aircraft Weight lb.
A. Start	-	-	-	52,000
B. Taxi and Warm-up	-	4.0	760	51,240
C. Take-off: Maximum Thrust, Afterburners Lit	-	.3	412	50,828
D. Acceleration to Best Climbing Speed: Maximum Thrust, Afterburners Lit	4	.6	880	49,948
E. Climb to 36,090 ft.: Climb Thrust, Afterburners Lit	12	1.4	1,360	48,588
F. Acceleration to Mach No. = 1.5: Maximum Thrust, Afterburners Lit	15	1.3	890	47,698
G. Climb to 50,000 ft.: Maximum Thrust, Afterburners Lit	22	1.5	950	46,748
H. Cruise-out at 50,000 ft. at Mach No.=1.5	189	13.2	4,780	41,968
I. Combat at 50,000 ft. at Mach No.=1.5	-	5.0	1,740	38,978*
J. Descent to 40,000 ft.	17	2.2	90	38,888
K. Cruise-back at 40,000 ft., Economical Cruising Speed	149	16.4	1,520	37,368
L. Descent to 30,000 ft.	24	3.1	200	37,168
M. Stack at 30,000 ft. Maximum Endurance Speed	-	15.0	1,100	36,068
N. Descent to Sea Level	52	6.3	710	35,358
O. Approach, Maximum Endurance Speed	-	5.0	608	34,750
TOTAL:	484	75.3	16,000	

Combat Radius of Action = 242 naut. mi.

*1,250 lb. of ammunition fired

3.1.2.1.3 Combat Radius - Maximum Range Mission:

TABLE 4 - COMBAT RADIUS OF ACTION - MAXIMUM RANGE MISSION

	Distance naut. mi.	Time min.	Fuel Consumed lb.	Aircraft Weight lb.
A. Start	-	-	-	52,000
B. Taxi and Warm-up	-	4.0	760	51,240
C. Take-off: Maximum Thrust	-	.7	318	50,922
D. Acceleration to Best Climbing Speed: Maximum Thrust	6	.9	446	50,476
E. Climb to 36,090 ft.: Climb Thrust	39	4.4	1,325	49,151
F. Cruise-out at 36,090 ft.: Economical Cruising Speed	312	34.4	3,915	45,236
G. Acceleration to Mach No. = 1.5 Maximum Thrust, Afterburners Lit	13	1.1	830	44,406
H. Climb to 50,000 ft.: Maximum Thrust, Afterburners Lit	20	1.4	825	43,581
I. Combat at 50,000 ft. at Mach No. =1.5	-	5.0	1,790	40,541*
J. Descent to 40,000 ft.	17	2.2	90	40,451
K. Cruise-back at 40,000 ft.: Economical Cruising Speed	297	32.7	3,083	37,368
L. Descent to 30,000 ft.	24	3.1	200	37,168
M. Stack at 30,000 ft.: Maximum Endurance Speed	-	15.0	1,100	36,068
N. Descent to Sea Level	52	6.3	710	35,358
O. Approach, Maximum Endurance Speed	-	5.0	608	34,750
TOTAL:	780	116.2	16,000	

Combat Radius of Action = 390 naut. mi. *1,250 lb. of ammunition fired

3.1.2.1.4 Combat Fuel Allowances: The following table sets forth the combat fuel allowance based on 5 minutes at a Mach number of 1.5, together with the equivalent range at a Mach number of 1.5 and at maximum range speed. Comparative allowances for climbing to 50,000 ft. and turning through 180 degrees are also given.

TABLE 5 - COMBAT FUEL ALLOWANCES AT COMBAT WEIGHT (44,000 lb.)

Altitude ft.	Combat Fuel Allowance lb.	Range at M = 1.5 naut. mi.	Max. Range naut. mi.	Fuel to Climb to 50,000 ft. at M =.95 lb.	Fuel for 180° Turn at M=1.5 lb.
50,000	1,800	72	-	-	1,142
45,000	2,040	72	-	612	874
40,000	2,403	72	198	908	811
36,090	2,758	72	232	1,065	774
20,000	5,150	77	347	1,322	892

3.1.2.1.5 The performance specified herein is based on estimated specific fuel consumption of the two TR9 engines, each one fitted with an afterburner.

3.1.2.1.6 Drag Estimate: Performance estimations specified herein are based on the drag estimate detailed in sub-paragraph 3.3

3.1.2.1.7 Combat Radius - Sea Level Mission:

TABLE 6

COMBAT RADIUS OF ACTION - SEA LEVEL MISSION

	Distance naut. mi.	Time min.	Fuel Consumed lb.	Aircraft Weight lb.
A. Start	-	-	-	52,000
B. Taxi and Warm-up	-	4.0	760	51,240
C. Take-off: Maximum Thrust, Afterburners Lit	-	.3	412	50,828
D. Acceleration to Mach No.=.95: Maximum Thrust, Afterburners Lit	4	.6	880	49,948
E. Cruise-out at Mach No.=.95	98	9.4	3,050	46,898
F. Acceleration to Mach No.=1.09: Maximum Thrust, Afterburners Lit	2	.2	326	46,572
G. Combat at Mach No.=1.09	-	5.0	7,020	38,302*
H. Climb to 30,000 ft.: Climb Thrust	21	2.3	780	37,522
I. Cruise-back at 30,000 ft.: Economical Cruising Speed	31	3.3	354	37,168
J. Stack at 30,000 ft.: Maximum Endurance Speed	-	15.0	1,100	36,068
K. Descent to Sea Level	52	6.3	710	35,358
L. Approach: Maximum Endurance Speed	-	5.0	608	34,750
TOTAL:	208	51.4	16,000	

Combat Radius of Action
=
104 naut. mi.

*1,250 lb. of ammunition fired

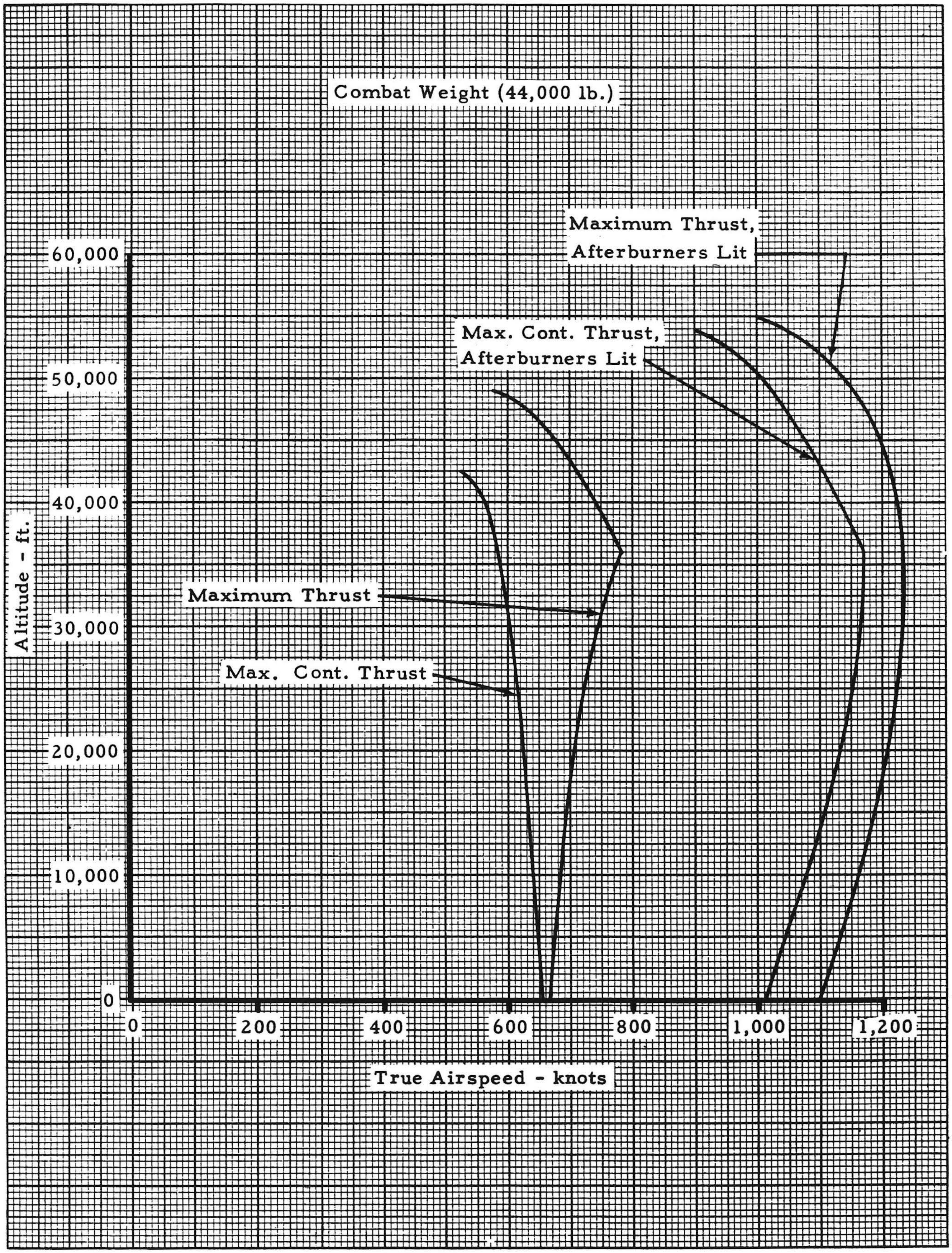


Fig. 2 Level Flight True Airspeed

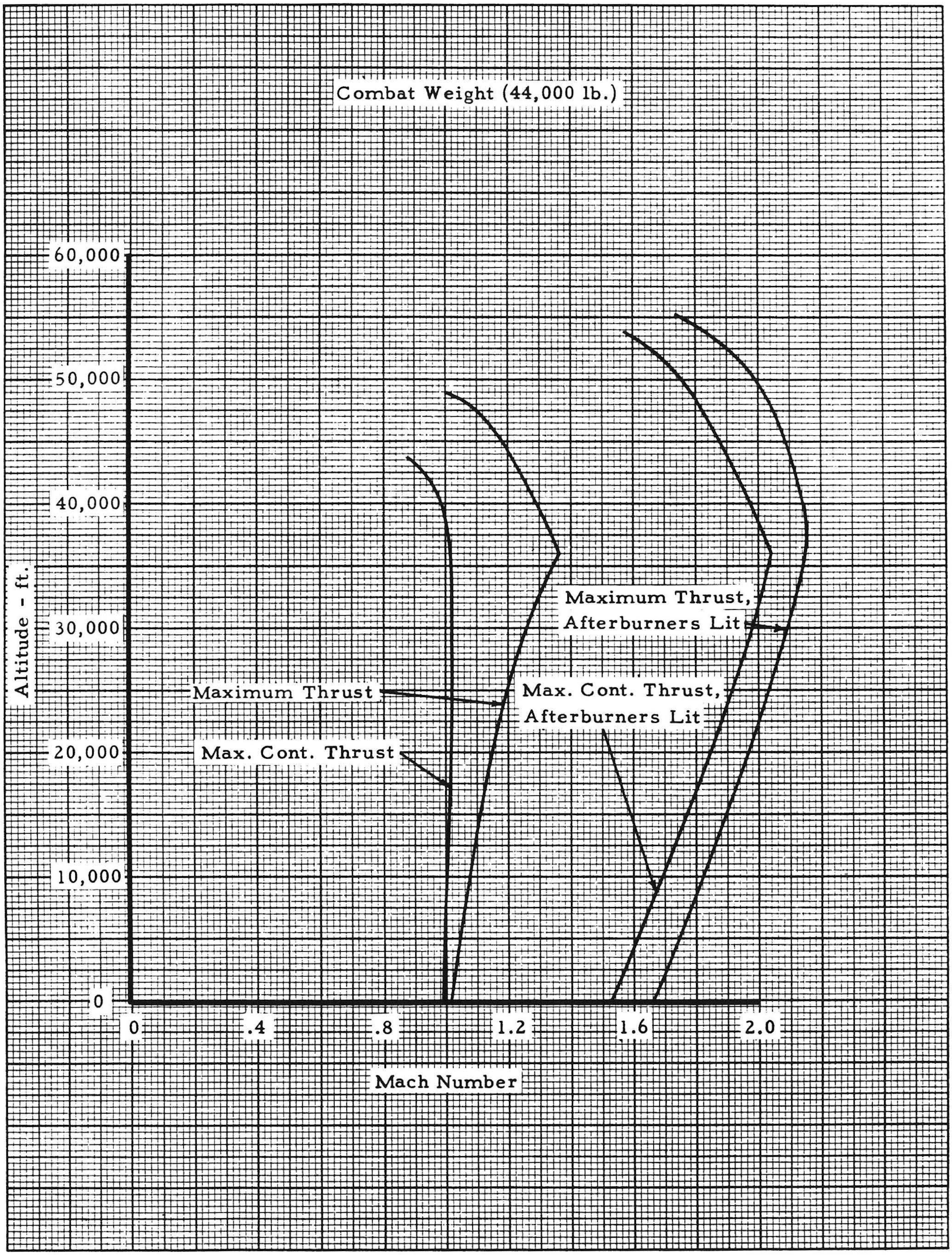


Fig. 3 Level Flight Mach Number

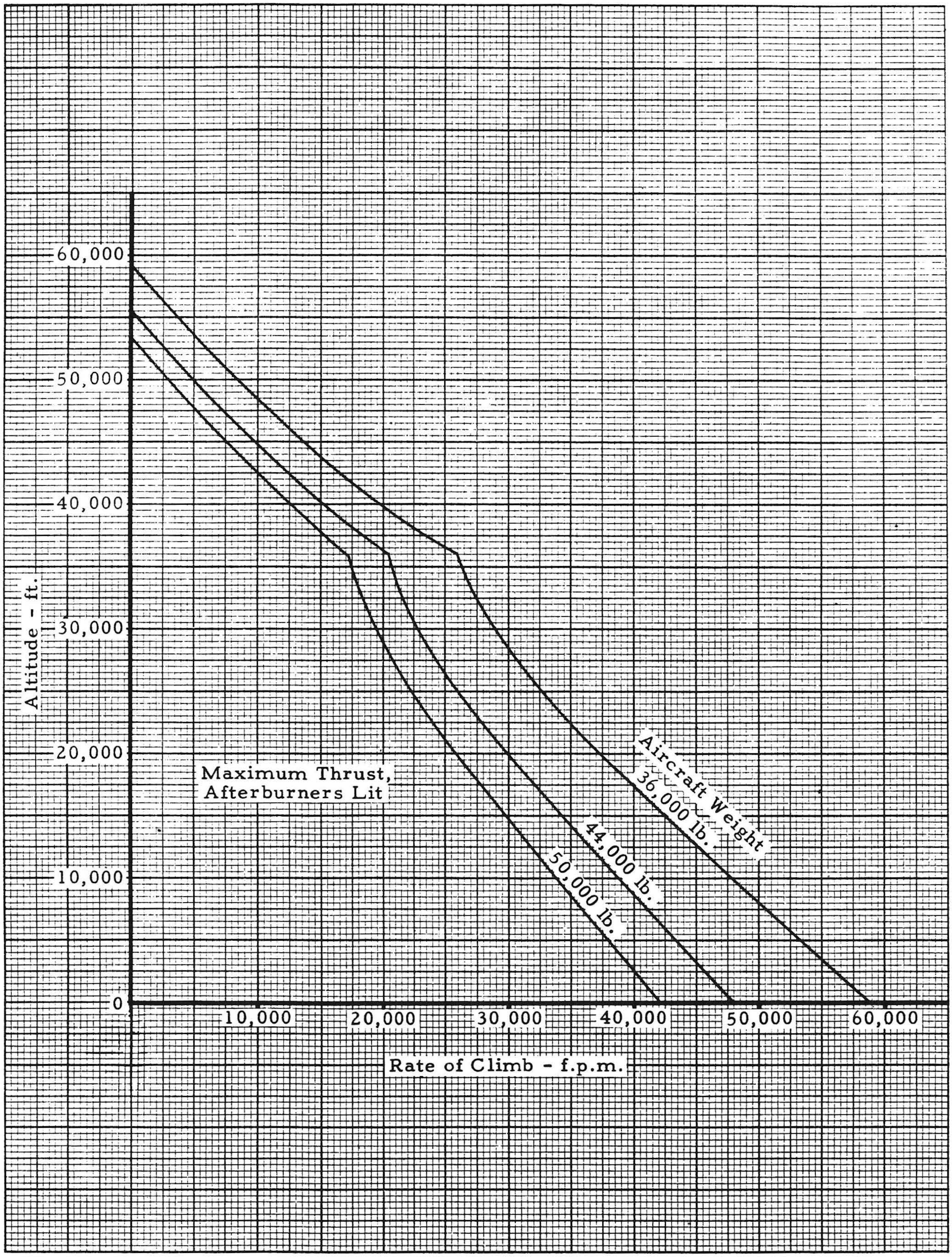


Fig. 4 Maximum Steady Rate of Climb - Maximum Thrust

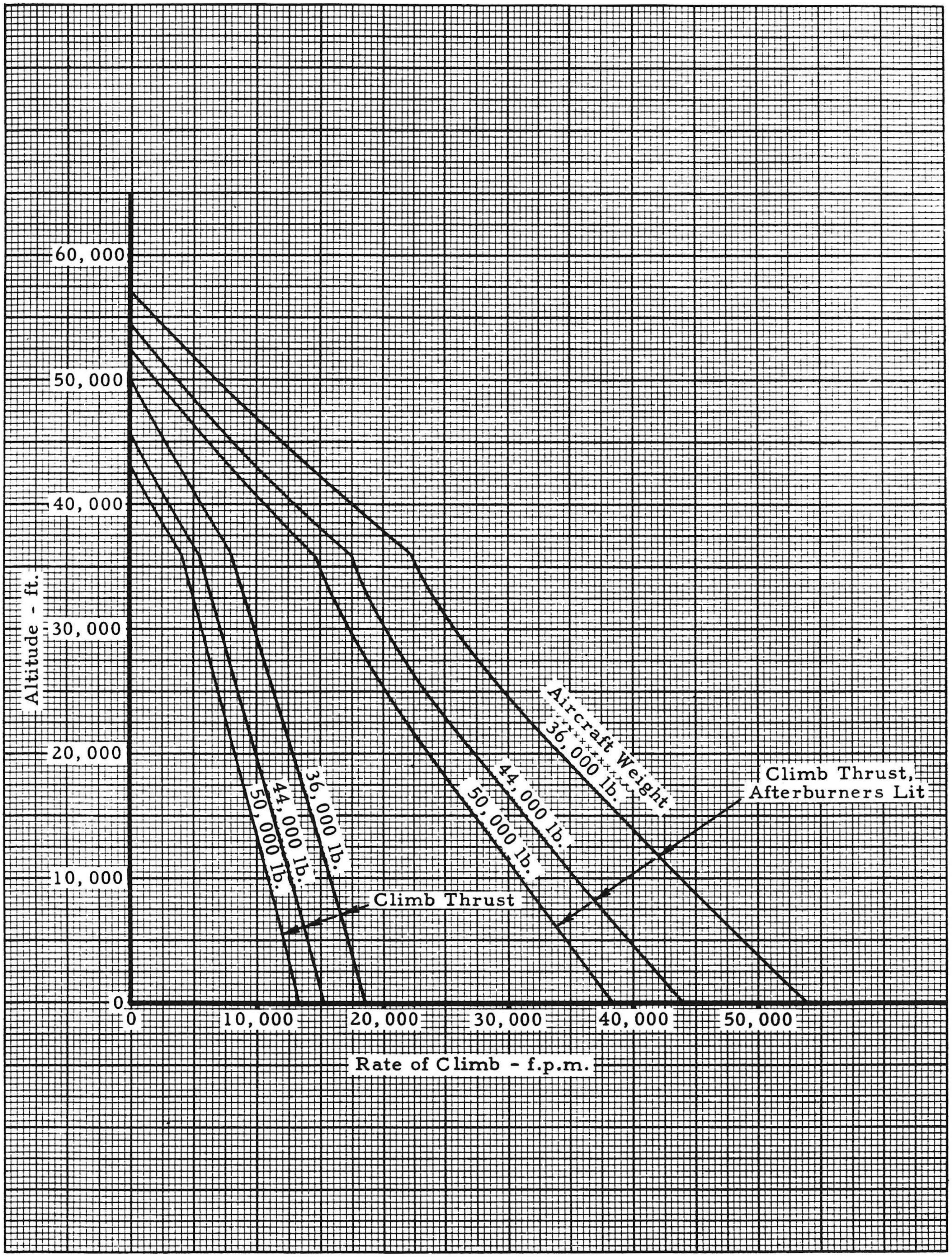


Fig. 5 Maximum Steady Rate of Climb - Climb Thrust

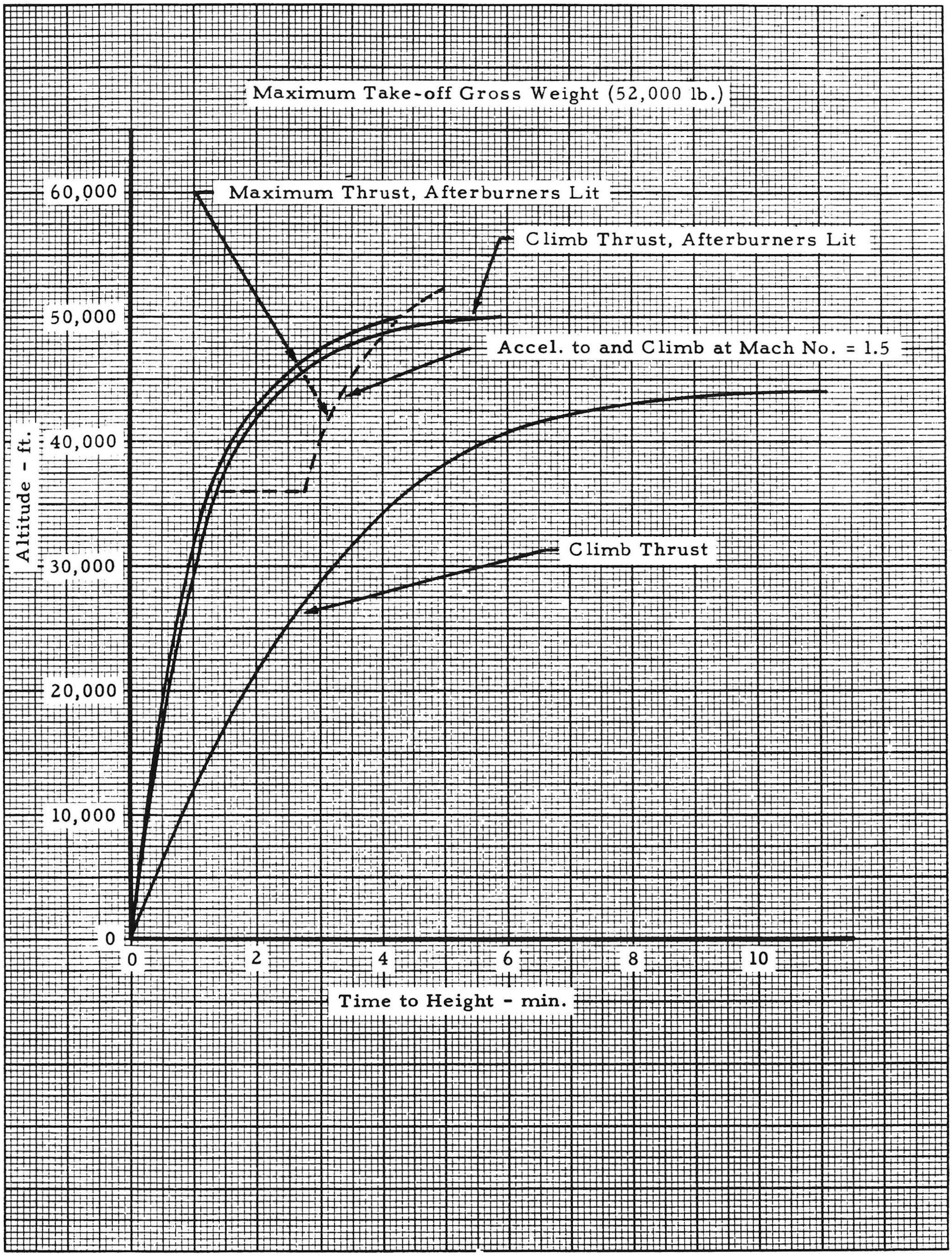


Fig. 6 Time to Height

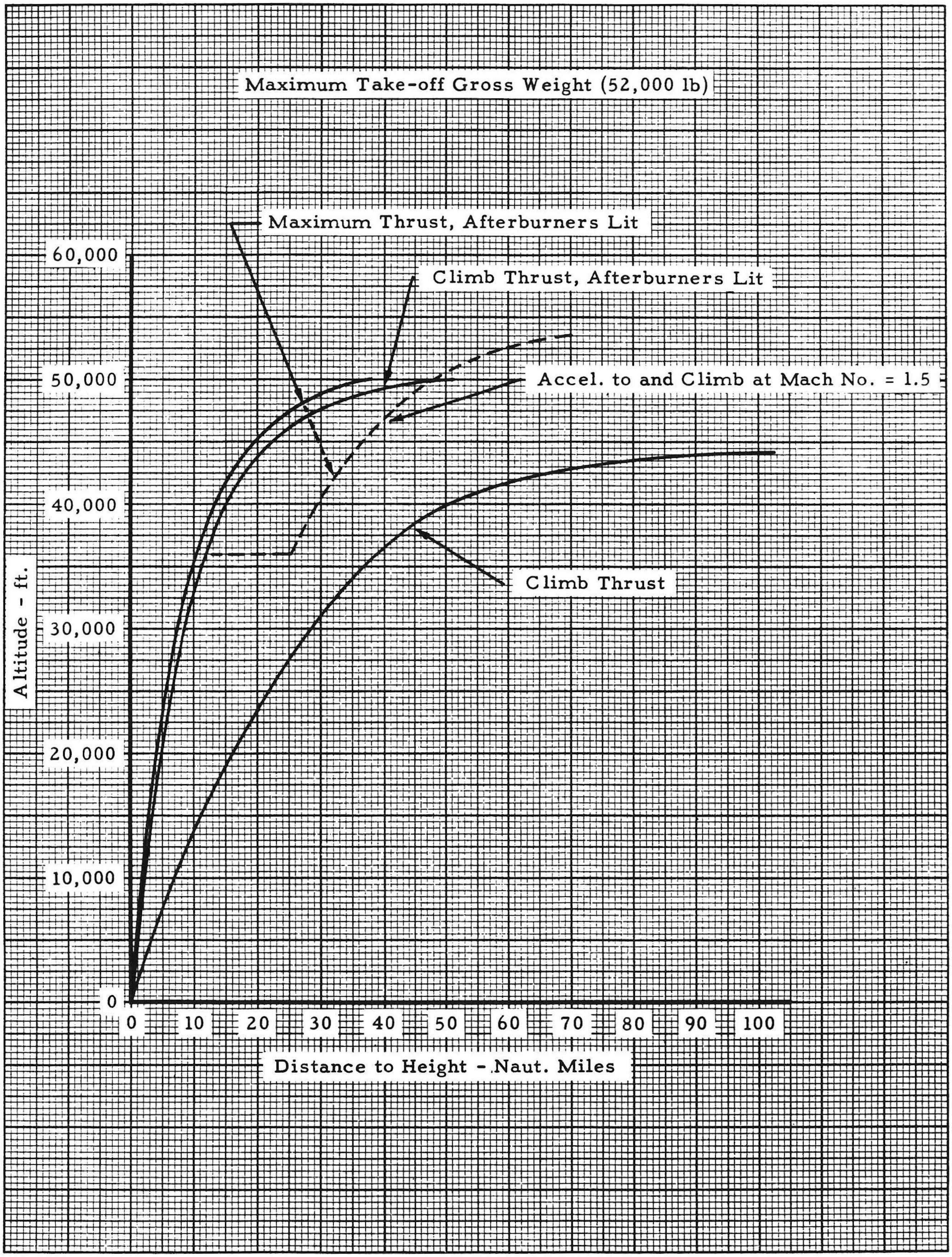


Fig. 7 Distance to Height

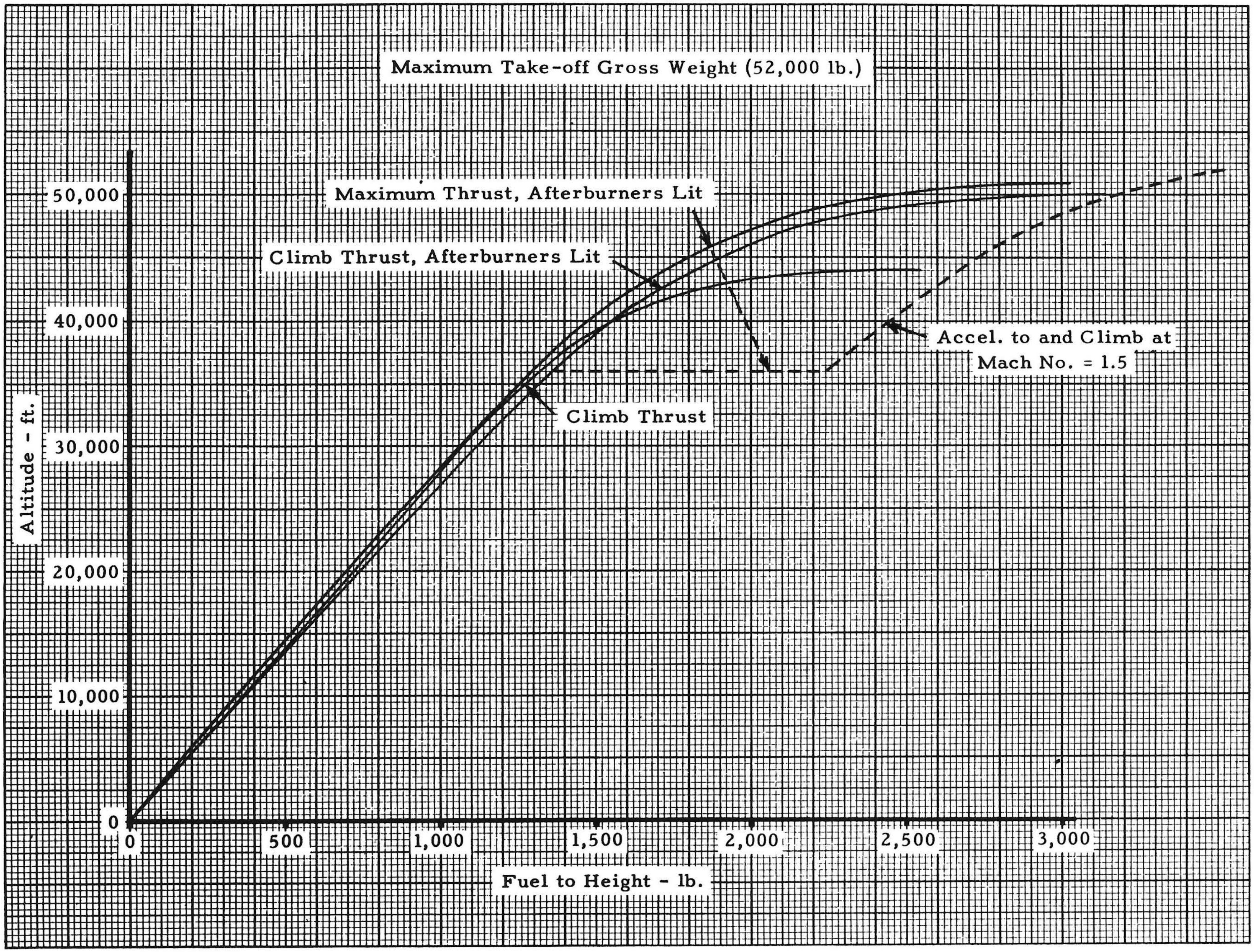


Fig. 8 Fuel to Height

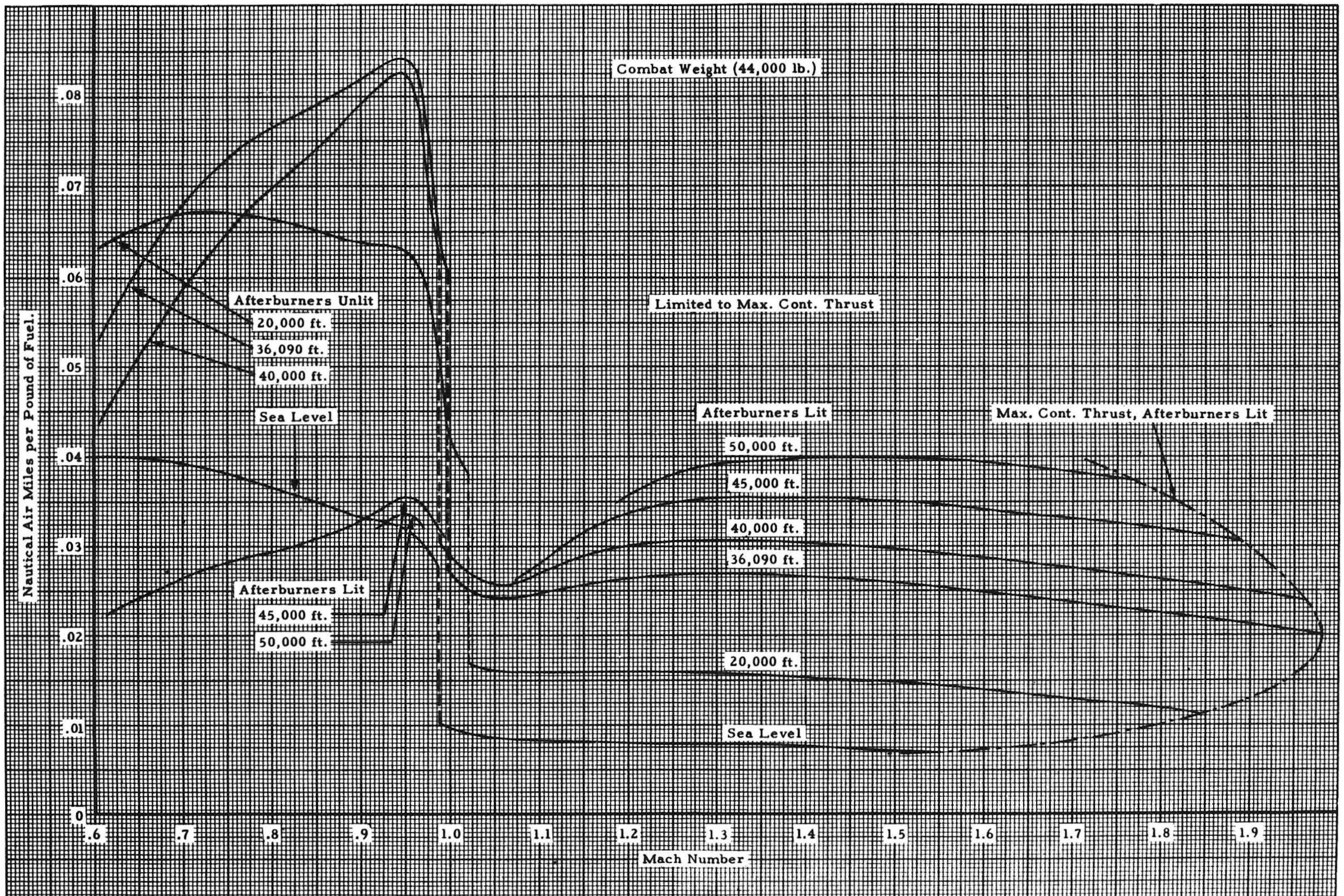


Fig. 9 Air Miles per Pound of Fuel

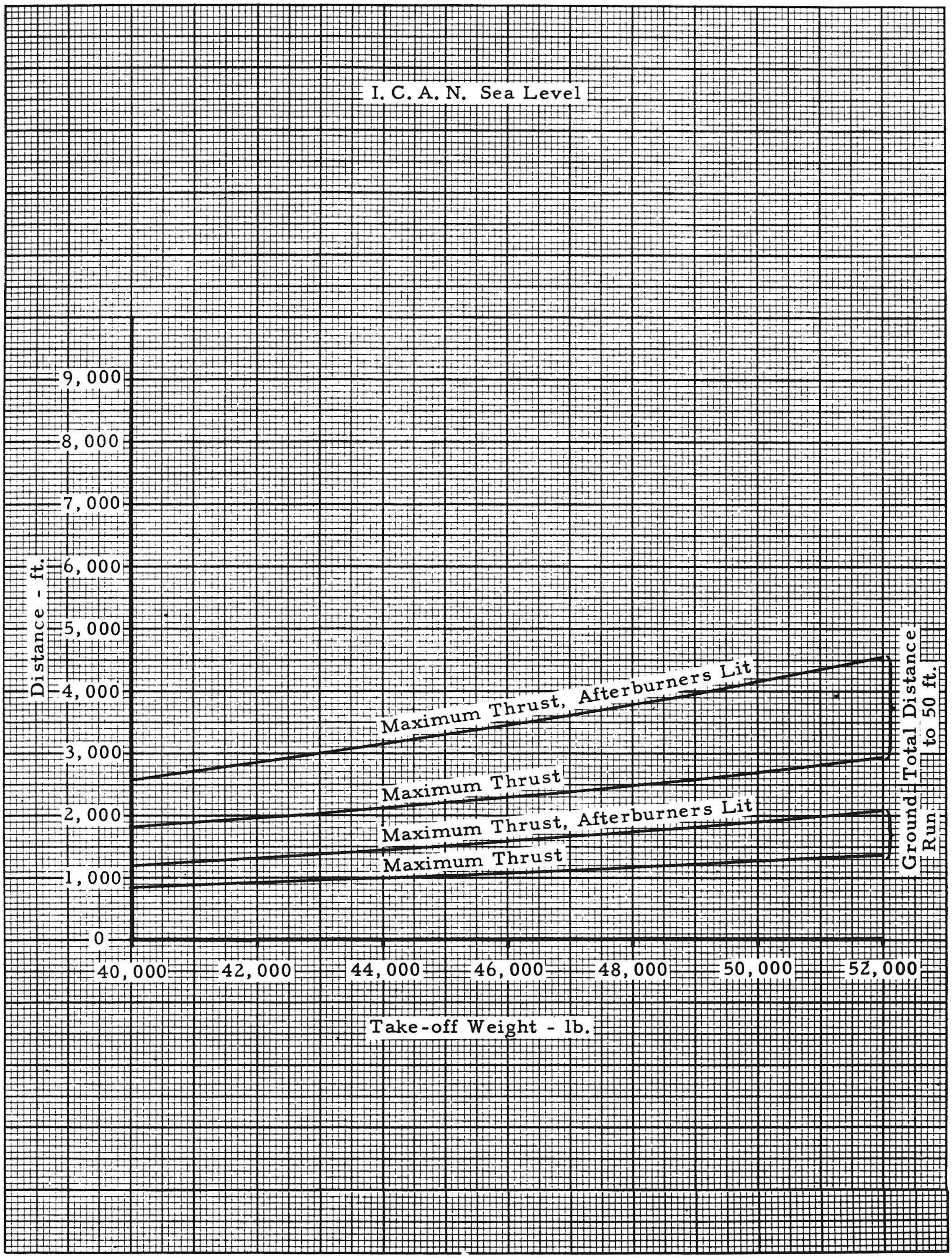


Fig. 10 Take-off Distance

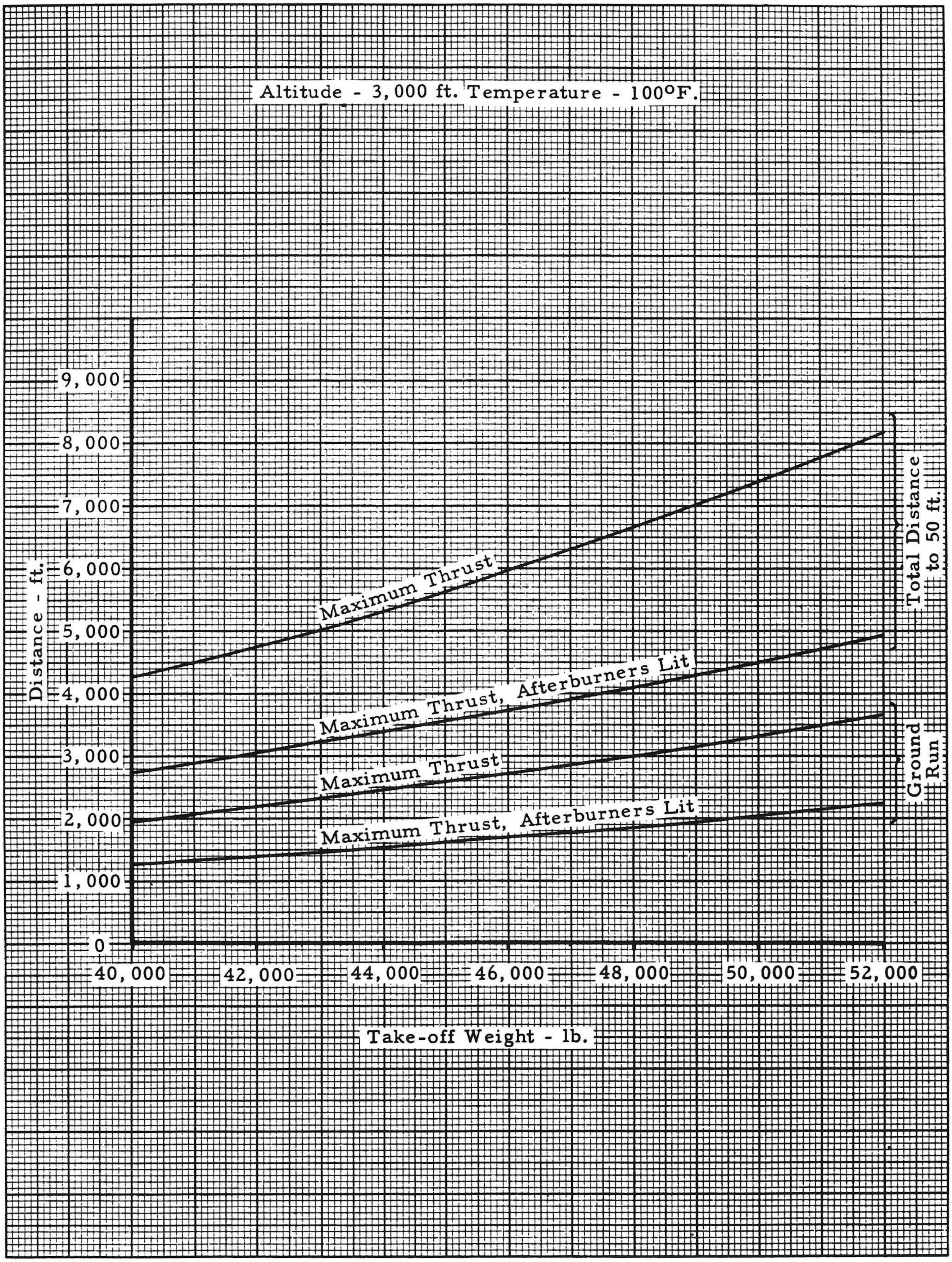


Fig. 11 Take-off Distance

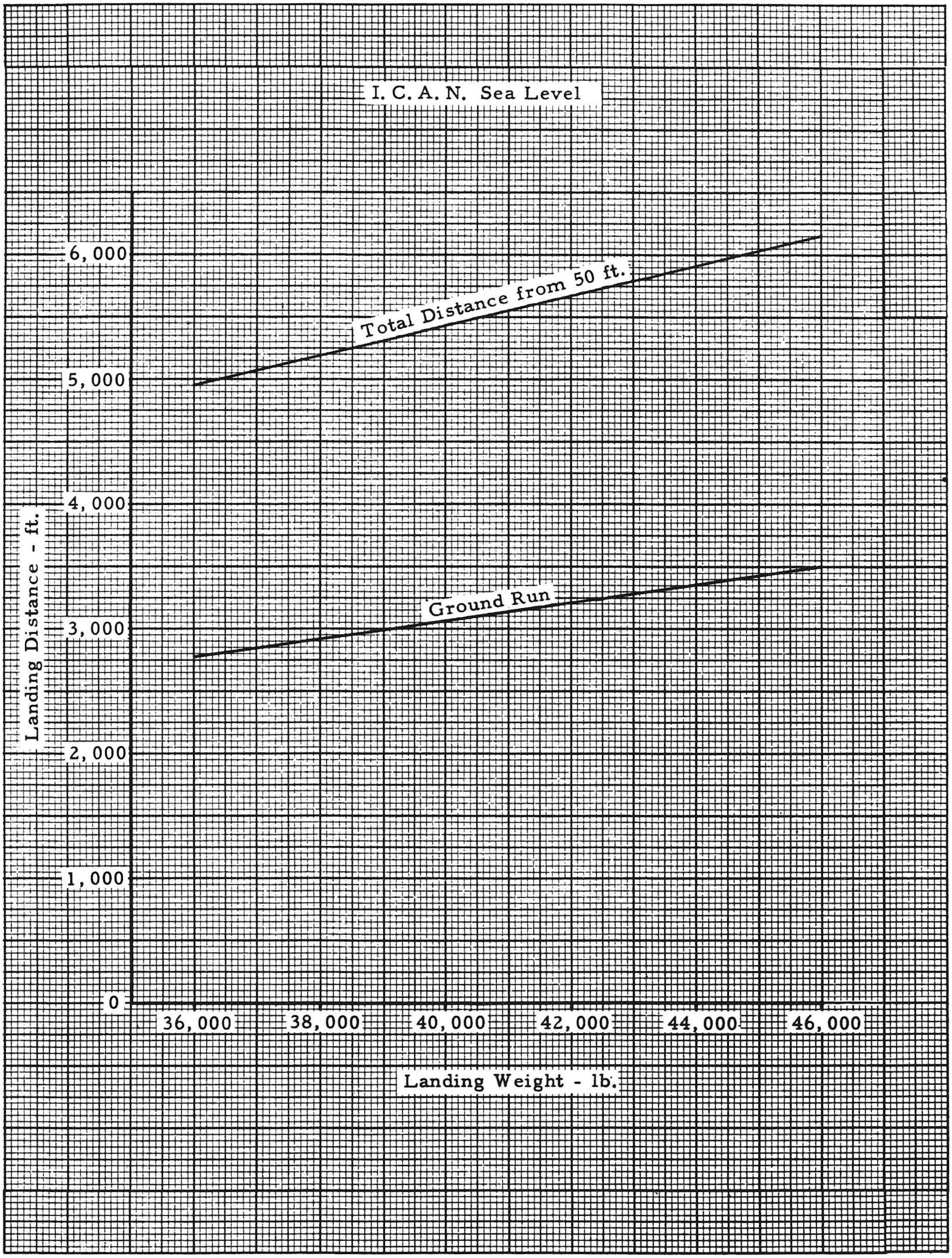


Fig. 12 Landing Distance

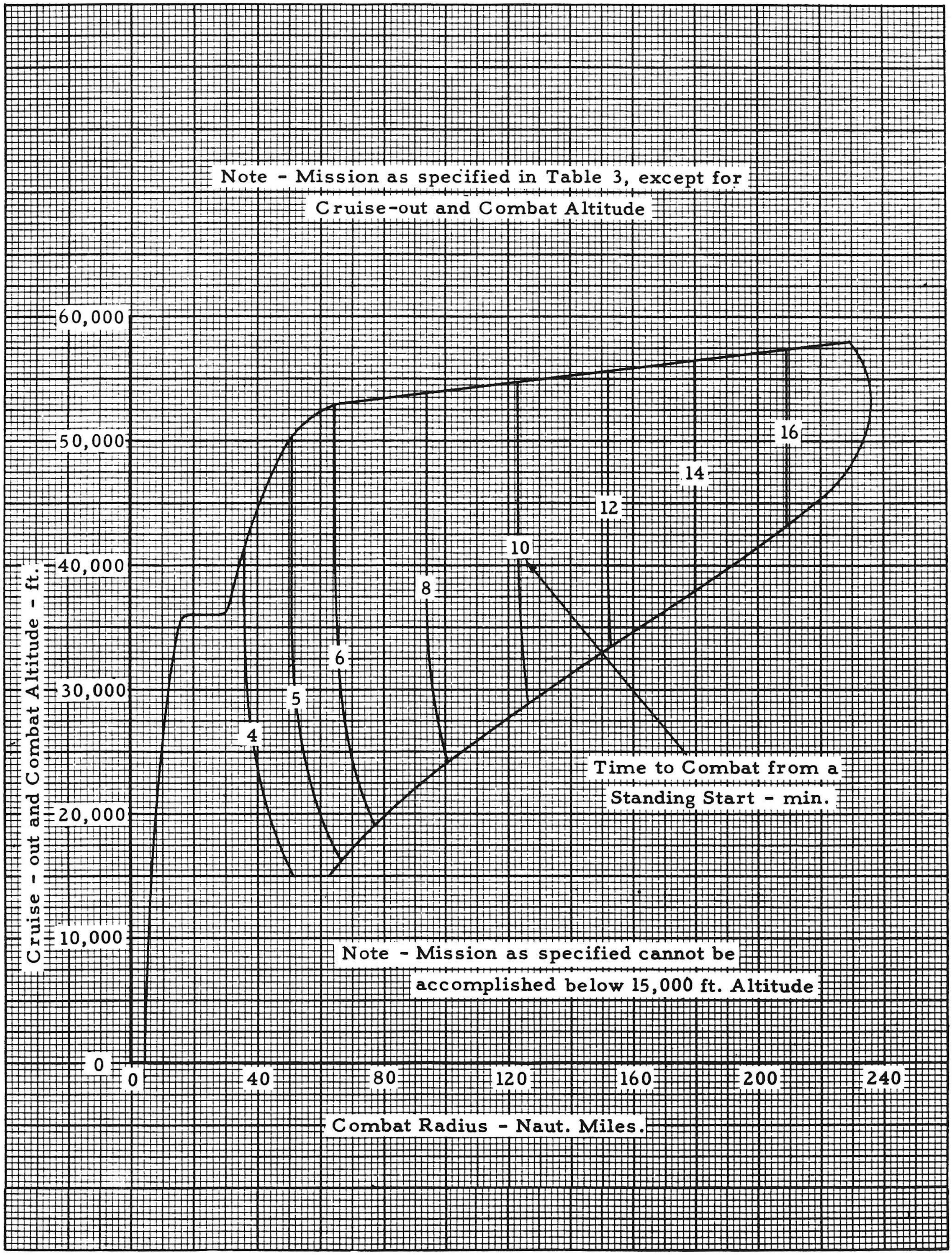


Fig. 13 Combat Radius of Action - High Speed Mission

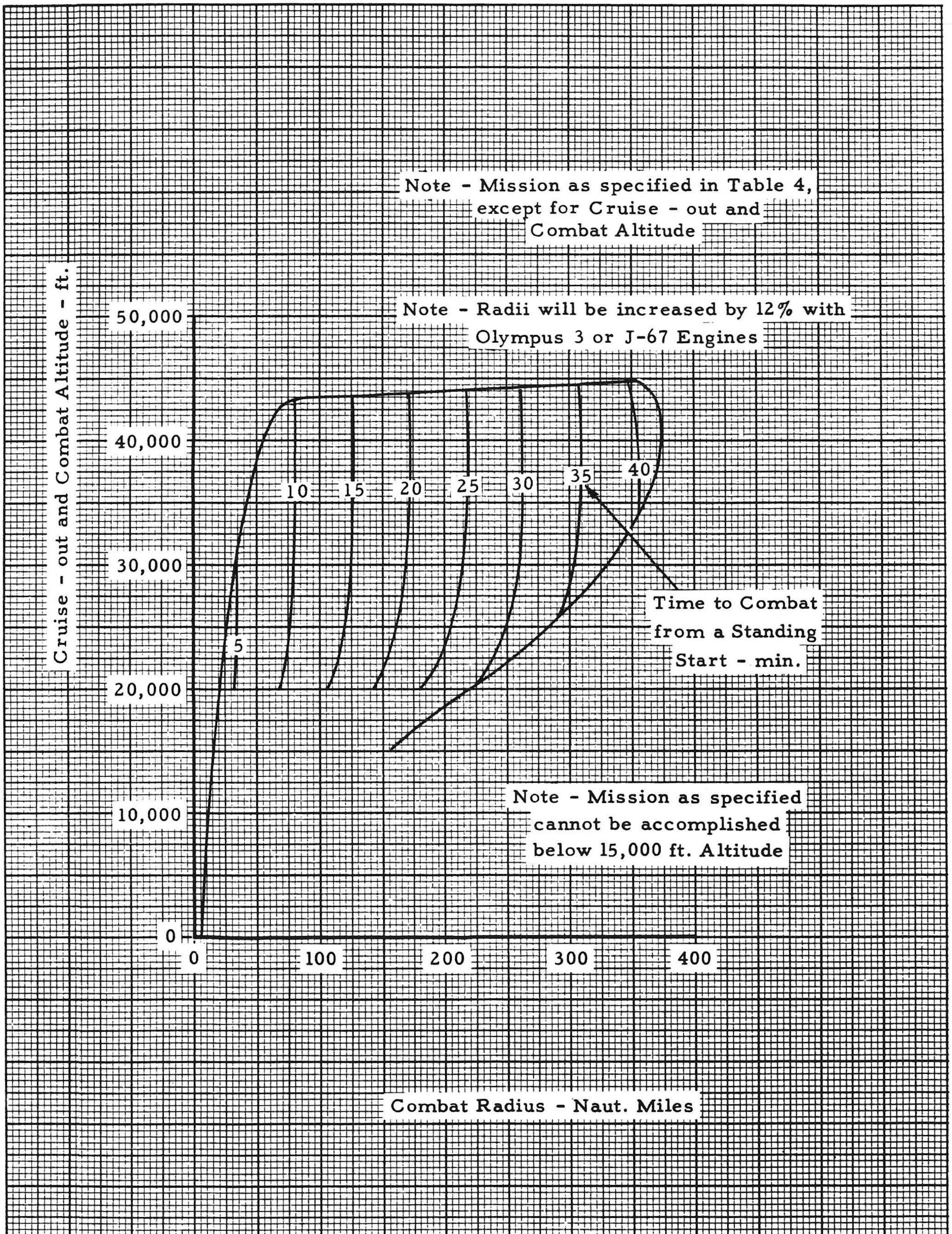


 Fig. 14 Combat Radius of Action - Maximum Range Mission

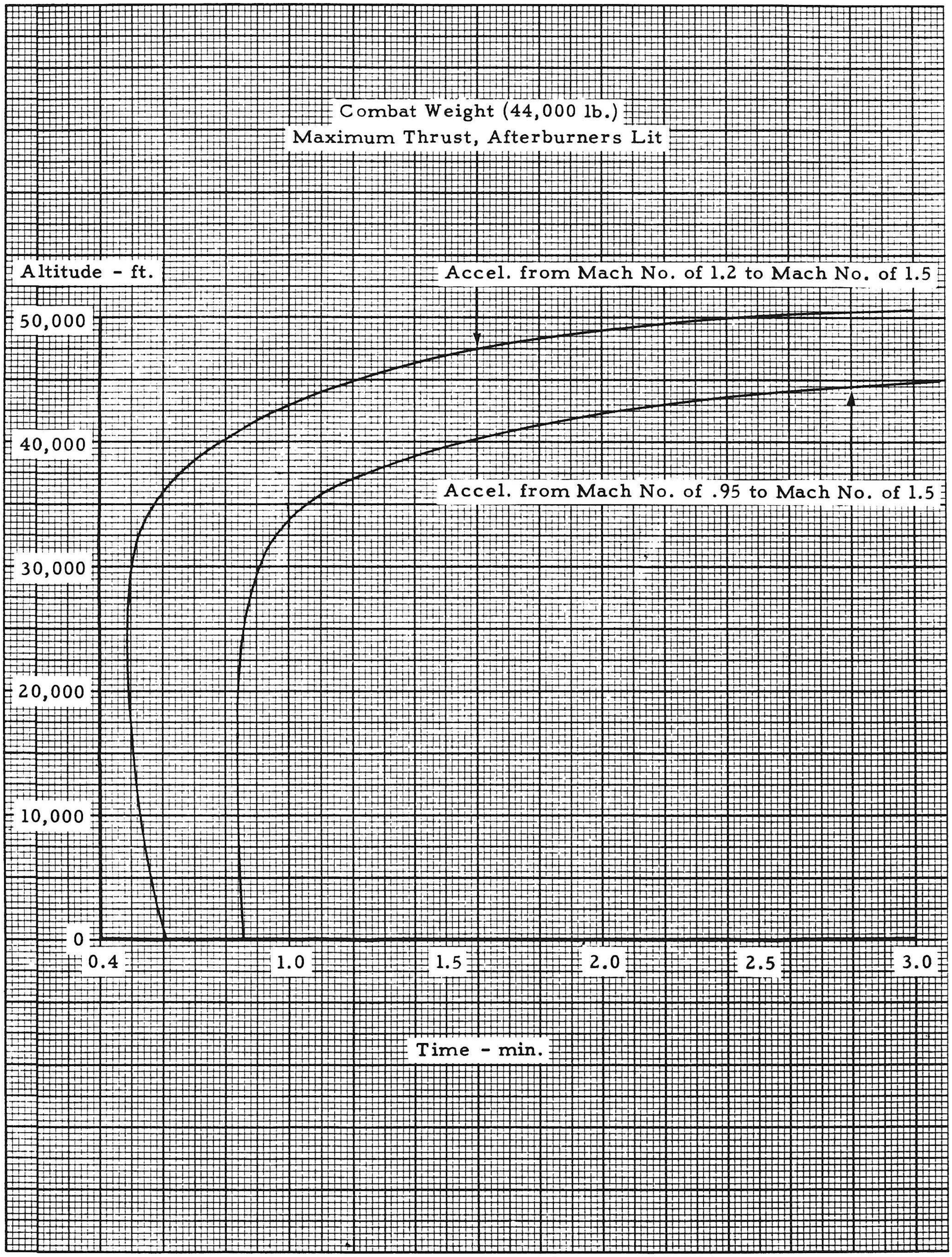


Fig. 15 Acceleration Time in Level Flight

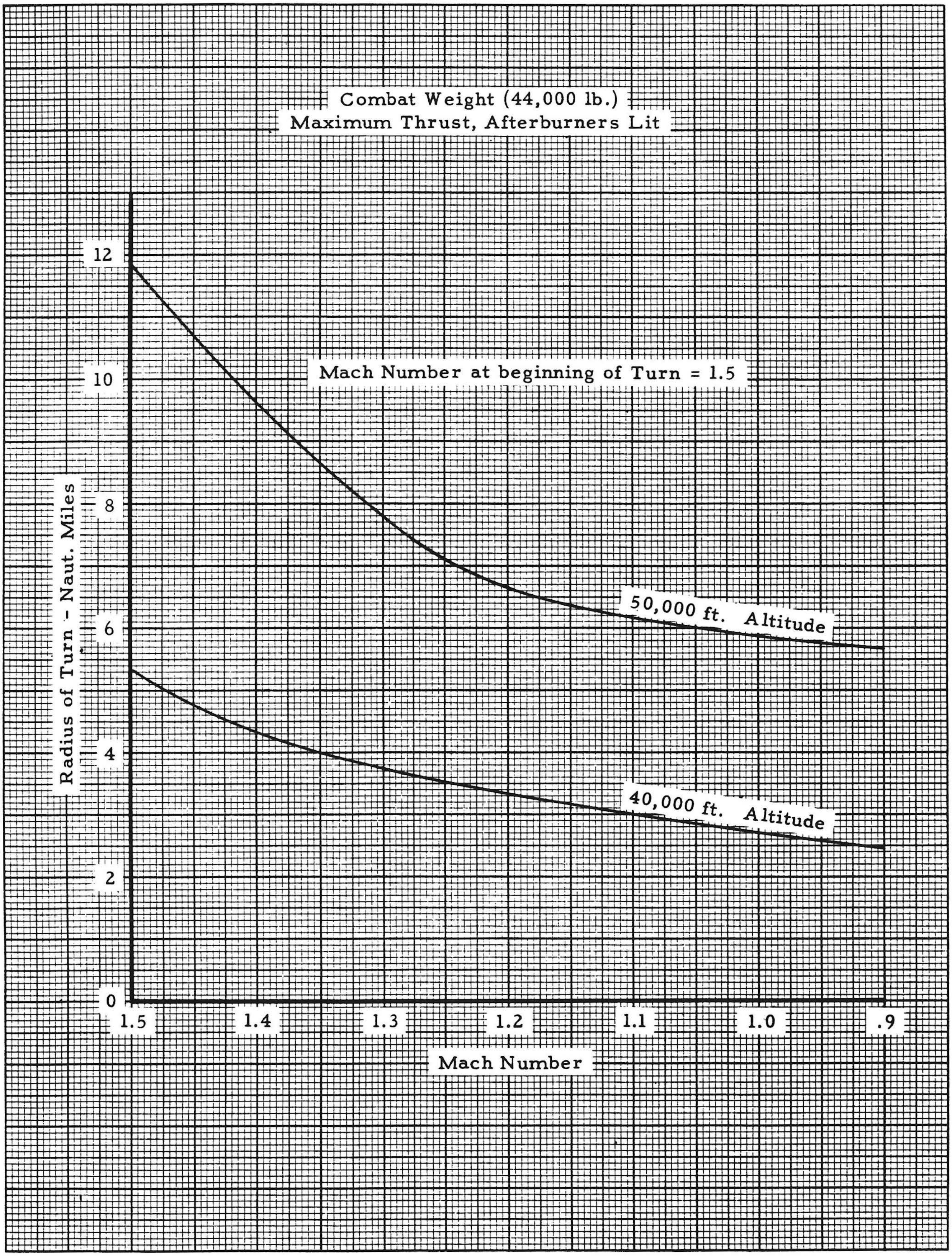


Fig. 16 Decelerating Turns in Level Flight

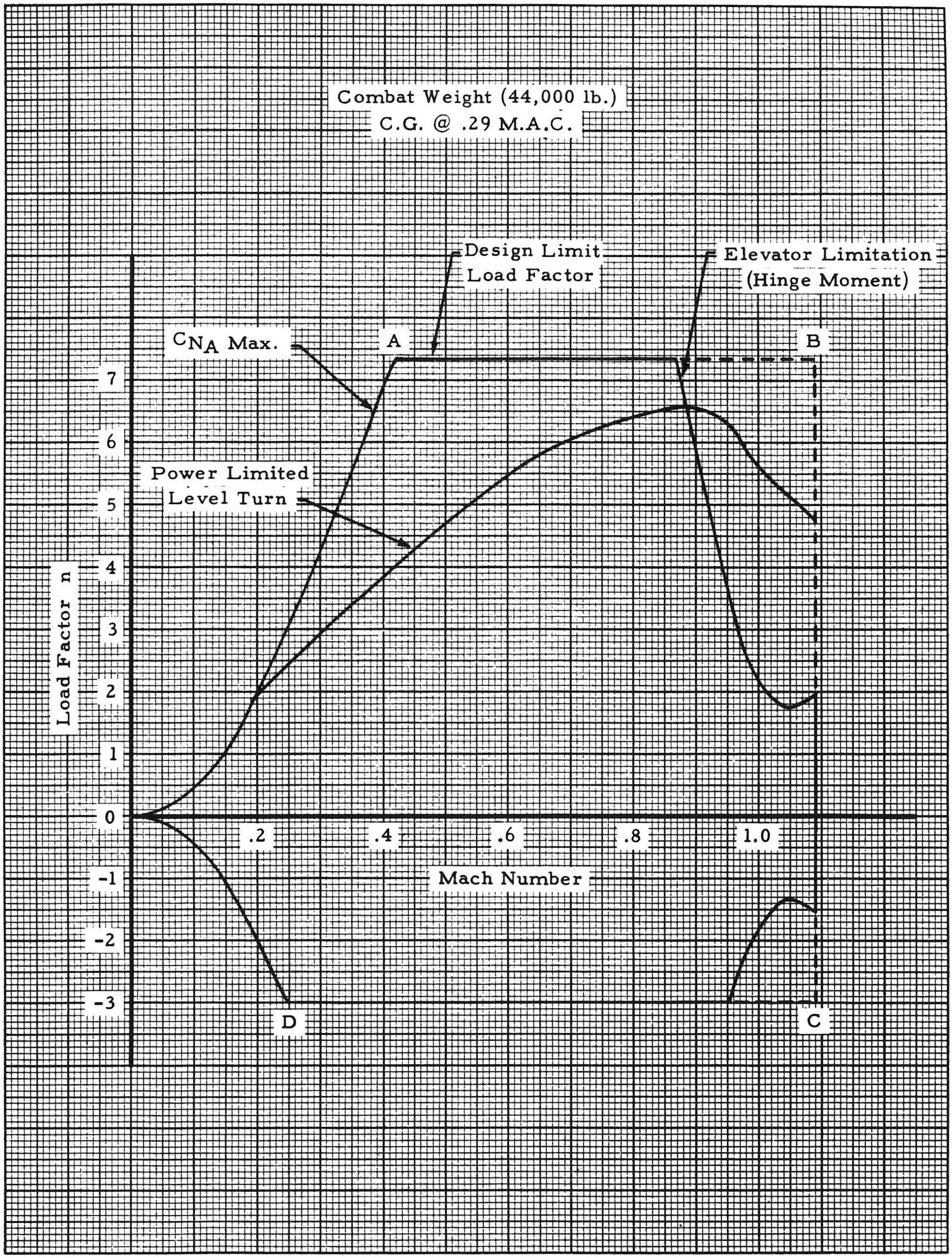


Fig. 17 Performance Flight Envelope at Sea Level

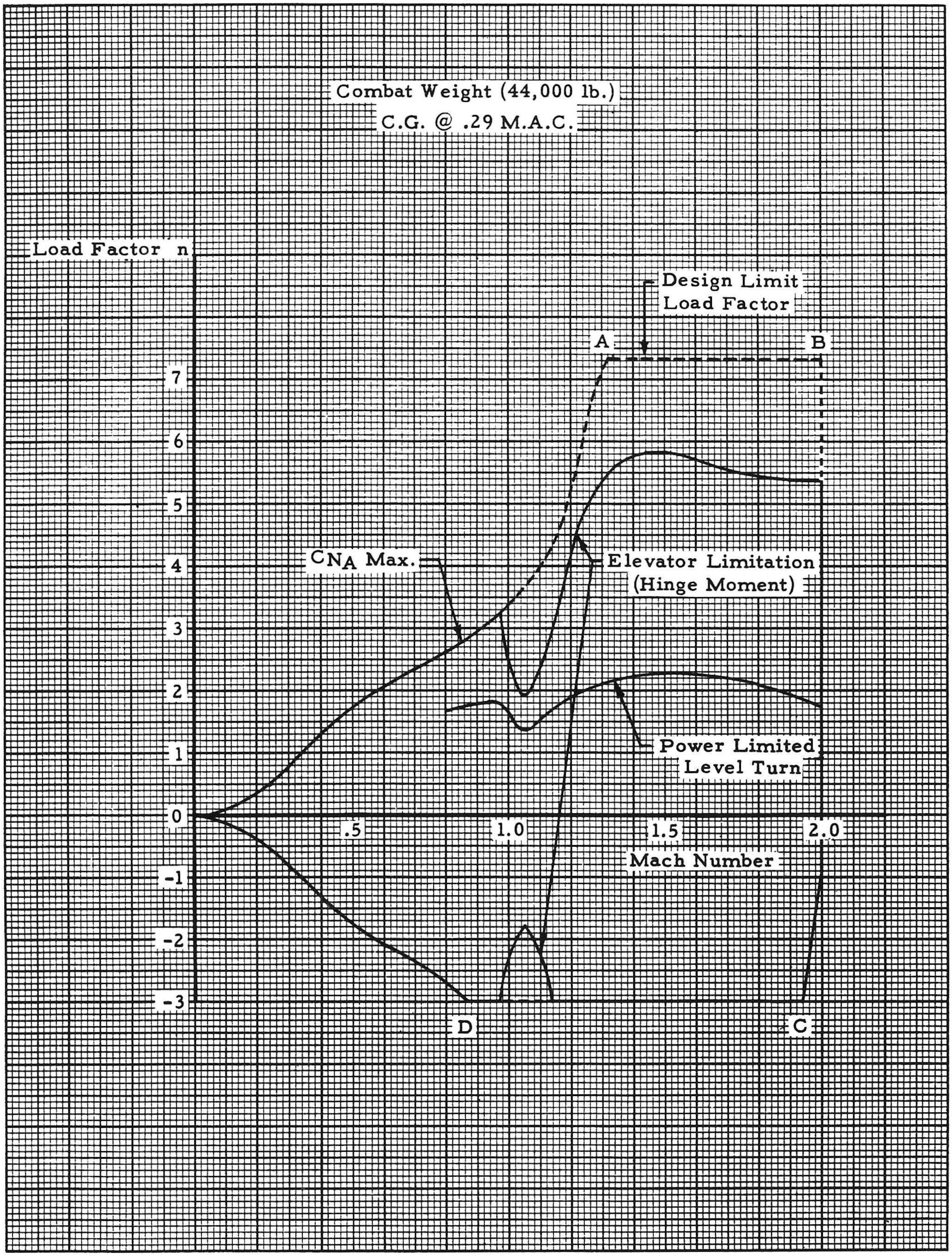


Fig. 18 Performance Flight Envelope at 40,000 ft. Altitude

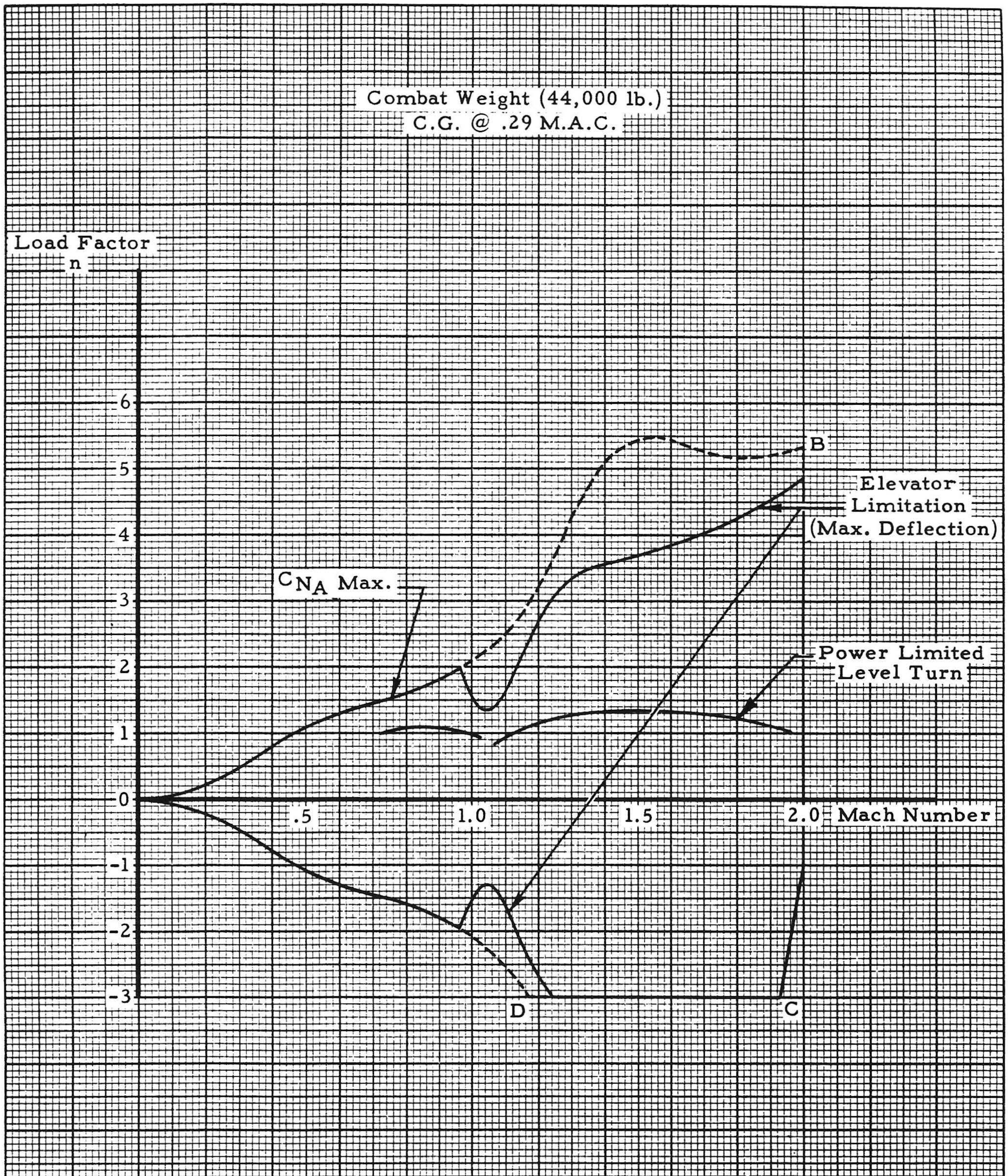


Fig. 19 Performance Flight Envelope at 50,000 ft. Altitude

3.1.3 Weights: Following is a list of the component weights of this airplane. Details of the c.g. balance calculations may be found by referring to Appendix VI at the back of this brochure.

TABLE 7 - STRUCTURE AND POWERPLANT

Item	Weight (lb.)
WING GROUP:	
Wing	9,200
Ailerons	215
Elevators	325
TAIL GROUP:	
Fin and rudder	1,000
BODY GROUP:	
Front fuselage	2,940
Rear nacelle	335
Speed brakes	163
Engine doors	567
Underwing structure	620
LANDING GEAR:	
Main undercarriage - including jacks	2,162
Nose undercarriage - including jacks	427
Tail skid	30
ENGINE SECTION:	
Shrouds, firewall, drainplate	430
Engine mounting	40
POWER PLANT GROUP:	
Engines - dry weight (as per RCAF-Eng-62-1 issue 4, Appendix II)	7,544
Engine controls	20
FUEL SYSTEM:	
Tanks	490
Piping, pumps etc.	440
FIRE EXTINGUISHER SYSTEM	65
ENGINE DE-ICING	185
ACCESSORIES GEAR AND DRIVES	15
AFTERBURNERS - COMPLETE	1,872
TOTAL:	29,085

TABLE 8 - FIXED EQUIPMENT GROUP

Item	Weight (lb.)
INSTRUMENTS	53
SURFACE CONTROLS:	700
Elevators, ailerons, rudder speed brakes - including jacks and artificial feel	
HYDRAULIC SYSTEM	678
ELECTRICAL SYSTEM	790
HUGHES ELECTRONIC EQUIPMENT:	
Nose radar	210
Cockpit equipment	190
Main equipment bay	1,400
ARMAMENT PROVISIONS	200
FURNISHINGS:	
Ejector Seat	132
Emergency accommodations	15
Oxygen	20
ANTI-ICING SYSTEM	270
AIR CONDITIONING AND PNEUMATIC SYSTEMS	355
BRAKE PARACHUTE	75
EXTERIOR FINISH	70
TOTAL:	5,158
TOTAL BROUGHT FWD.:	29,085
WEIGHT EMPTY:	34,243

TABLE 9 - UNIT WEIGHTS

Component	Unit Weight
WING GROUP (Gross Area 1189.4 sq. ft.) lbs. per sq. ft.	8.18
TAIL GROUP (Gross Area 155 sq. ft.) lbs. per sq. ft.	6.45
WEIGHT OF FUEL SYSTEM PER GAL. CAP. (1,958 Imperial gal. of fuel)	.475

TABLE 10 - USEFUL LOAD

Item	Weight (lb.)
CREW - ONE (1) PILOT	207
FUEL:	
Usable	16,000
Total residual	260
OIL	40
ARMAMENT:	
Guided missiles (6)	660
Rockets (24)	590
TOTAL:	17,757
WEIGHT EMPTY:	34,243
GROSS WEIGHT:	52,000

TABLE 11 - DESIGN INFORMATION

Length - max.	70 ft. 3 in.
Height - max.	21 ft. 3 in.
Span	48 ft. 0 in.
Thickness - root chord	3%
Thickness - tip chord	3%
Wing area - net	822 sq. ft.
Taper ratio (Root chord/tip chord)	11.49:1
Length - root chord	45 ft. 7 in.
Length - tip chord	4 ft. 0 in.
Maximum fuselage depth	6 ft. 4 in.
Maximum fuselage width	8 ft. 8 in.

Design gross weight	52,000 lb.
Stressing weight and load factor:	
At combat weight	44,000 lb.
Ultimate load factor	11.00 g.
Limit load factor	7.33 g.
Factor of safety	1.50
Airplane weight immediately after take-off	50,000 lb.
Ultimate load factor	9.67 g.
Limit load factor	7.33 g.
Factor of safety	1.33

3.1.3.1 Alternate Loading: Refer to Appendix IV at the back of this brochure for alternate loading information.

3.1.3.2 Gross weight estimations are as follows:-

Design gross weight .... 52,000 lb.

3.1.4 Center of Gravity Locations:

Take-off gross weight, c.g. location, wheels up:

Aft l.e. of m.a.c. .... 27.65% m.a.c

Below l.e. of m.a.c. .... 1.02 ft.

Design gross weight, c.g. location, wheels down:	
Aft l.e. of m.a.c.	28.10% m.a.c.
Below l.e. of m.a.c.	1.29 ft.
Extreme forward position, c.g. possible in flight regardless of loading at take-off - wheels up:	
Aft l.e. of m.a.c.	27.65% m.a.c.
Below l.e. of m.a.c.	1.02 ft.
Gross weight for this condition	52,000 lb.
Extreme rearward position, c.g. possible in flight regardless of loading at take-off - wheels down:	
Aft l.e. of m.a.c.	30.95% m.a.c.
Below l.e. of m.a.c.	1.74 ft.
Gross weight for this condition	34,750 lb.

3.1.5 Areas:

Wing area, total including ailerons and elevators	1189.4 sq. ft.
Elevator area aft of hinge line (each)	51.55 sq. ft.
Aileron area aft of hinge line (each)	30.25 sq. ft.
Vertical tail area, total	155 sq. ft.
Fin - to rudder hinge	122 sq. ft.
Rudder, aft of hinge	33 sq. ft.
Speed brakes:	
Upper (each) - projected	7.75 sq. ft.
Lower (each) - projected	12.25 sq. ft.

NOTE: The control surfaces are power-operated and do not incorporate aerodynamic balance aids.

3.1.6 Dimensions and General Data:

Wings:	
Span, maximum	48 ft. 0 in.
Chord:	
At root	45 ft. 7 in.
At construction tip (theoretical extended section at tip)	4 ft. 0 in.
Mean aerodynamic	30 ft. 7.2 in.
Airfoil section designation and thickness (percent chord)	NACA (Modified) 0003-63
At root	3%
At construction tip (theoretical extended section at tip)	3%
Average (frontal area divided by wing area) ...	3%

Incidence:	
At root	2 deg.
At construction tip	2 deg.
Sweepback at 25% chord	52 deg. 0 min.
Anhedral	3 deg. 0 min.
Aspect ratio	1.93
Ailerons:	
Span (each)	9 ft. 8 in.
Chord (average percent wing chord)	25%
Elevators:	
Span (each)	9 ft. 10 in.
Chord (average percent wing chord)	17.85%
Speed brakes (fuselage):	
Span - upper (each)	2 ft. 6 in.
Span - lower (each)	3 ft. 6 in.
Chord - upper	4 ft. 0 in.
Chord - lower	4 ft. 4 in.
Location:	
Upper	Upper surface of fuselage, one each side of the fin structure
Lower	Below the engine afterburners on the fuselage lower surface
Tail - vertical:	
Airfoil section designation and thickness	NACA (Modified) 0003-63
Sweep of leading edge	64 deg.
Aspect ratio	.926
Height over highest fixed part of airplane - fin ...	21 ft. 3 in.
Height - wing tip	8 ft. 0 in.
Height to top of cockpit	13 ft. 0 in.

NOTE: The above heights are taken with the airplane in its normal ground attitude - shock-absorber struts static.

Height in hoisting attitude from top of hoisting sling to lowest part of airplane - wheels down ...	Dimensions not yet available
Length, maximum:	
Reference line level	70 ft. 3 in.
Three point attitude	71 ft. 0 in.
Length from hoisting sling to farthest aft point of tail, reference line level, rudder neutral	Dimensions not yet available

Distance from wing m.a.c. quarter chord point to vertical tail m.a.c. quarter chord point... ...	16 ft. 1 in.
Angle between reference line and wing zero lift line	2 deg.
Ground angle	3 deg.
Wheel rim size:	
Main wheels...	29 x 7.7
Nose wheels...	18 x 5.5
Tire size:	
Main wheels...	29 x 7.7
Nose wheels...	18 x 5.5
Tread of main wheels	26 ft. 4 in.
Wheel base	26 ft. 0 in.
Vertical travel of axle from extended to fully compressed position:	
Main wheels...	11 in.
Nose wheel	11 in.
Angle between lines joining center of gravity with points of ground contact of main wheel tires, static deflection of 1 W (front elevation) - take-off condition	112 deg.
Angle of line through center of gravity and ground contact point of main wheel tire to vertical line, reference line level, static deflection of 1 W (side elevation) - take-off condition	20 deg. 30 min.

3.1.7 Control Surface and Corresponding Control Movements: Following is a table of control surface and control movements on each side of neutral position for full movement as limited by stops:-

TABLE 12

Surfaces	Control	Movement
Rudder	Surface	25 deg. RIGHT, 25 deg. LEFT
-	Pedals	According to USAF Spec. AMC80-1
Ailerons	Surface	19 deg. UP 19 deg. DOWN
-	Stick	According to USAF Spec. AMC80-1
Elevators	Surface	30 deg. UP 30 deg. DOWN
-	Stick	According to USAF Spec. AMC80-1
Speed brakes:		
Upper	Surface	60 deg. maximum UP
Lower	Surface	60 deg. maximum DOWN

3.2 General:

3.2.1 General Interior Arrangement: Refer to Fig. 20. This illustration shows the disposition of crew, armament, powerplant and the electrical, electronic, hydraulic, pneumatic and air conditioning systems superimposed on a profile drawing of the fuselage. Additional illustrations are included in the text to show the main items of equipment in detail together with the access to same for purposes of maintenance. These comprise:-

Fig. 32	Armament and Electronics
Fig. 28	Hydraulic, Electrical, Pneumatic and Air Conditioning Systems
Fig. 27	Fuel System
Fig. 26	Engine Installation

Special attention has been given to ease of access for those installations which require frequent servicing; in particular, this applies to the electronics which are grouped together in one large crate which can be quickly lowered out of the airplane by means of an integral electrically operated winch for servicing by personnel standing on the ground. Accessibility is enlarged upon in the paragraphs dealing with equipment in detail.

3.2.2 Materials: These will conform to specifications approved by the RCAF.

3.2.3 Workmanship: This will conform to the usual high grade airplane practice.

3.2.4 Production, Maintenance and Repair: The design of the aircraft shall be such as will ensure ease of production, simple and rapid installation of the engines and equipment, and ease of general maintenance. Special attention has been given to the ease with which component parts of the structure can be inspected, maintained and repaired. The fuselage and wings are designed to facilitate the removal and replacement of damaged sections. To meet these requirements, the fuselage is constructed in seven main sections. These comprise:-

(a) The radome

(b) The cockpit section

(c) The armament and equipment section

(d) The powerplant section superstructure

(e) The powerplant doors

(f) The rear nacelle superstructure

(g) The lower rear nacelles

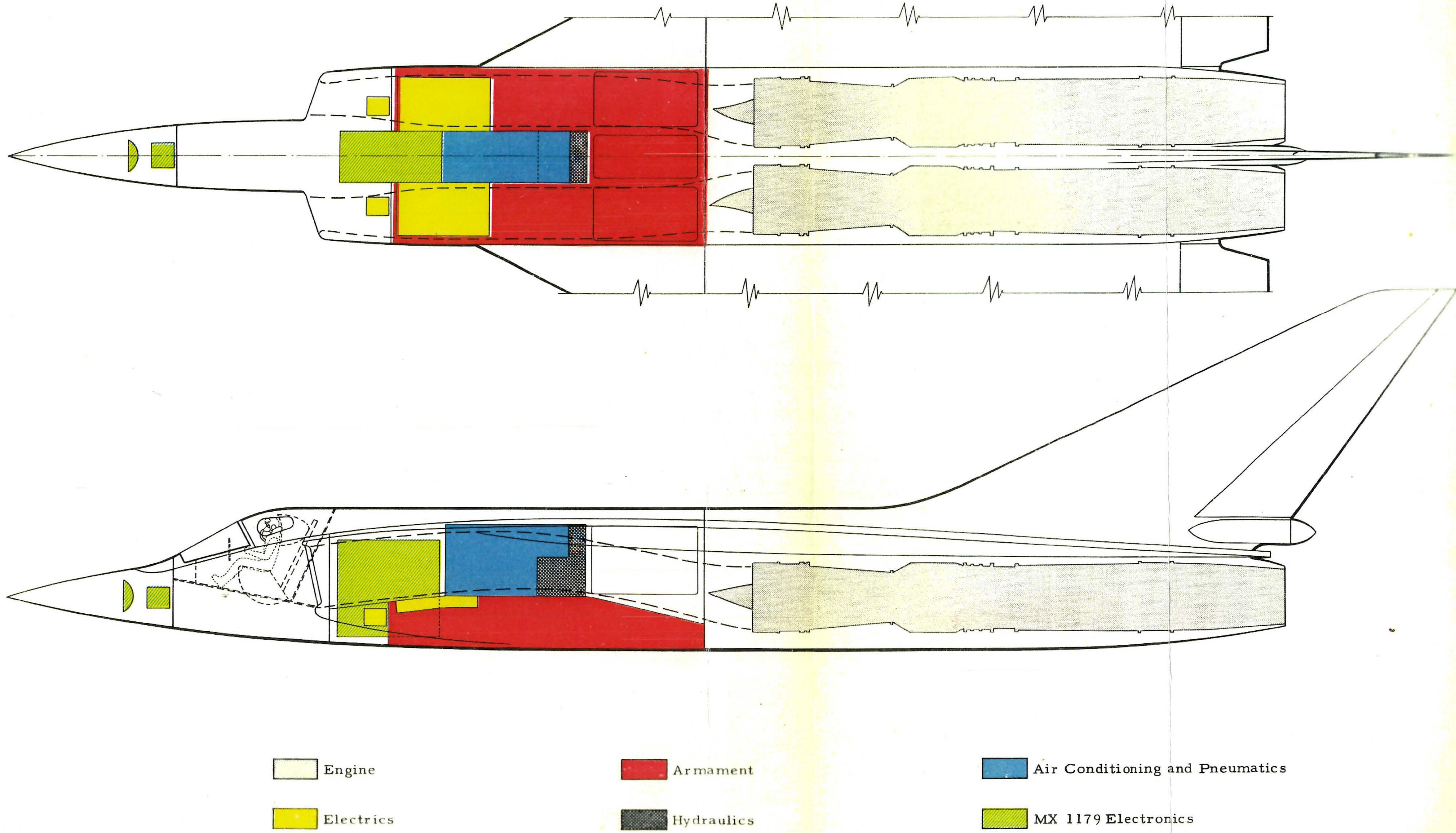


FIG. 20 EQUIPMENT ARRANGEMENT

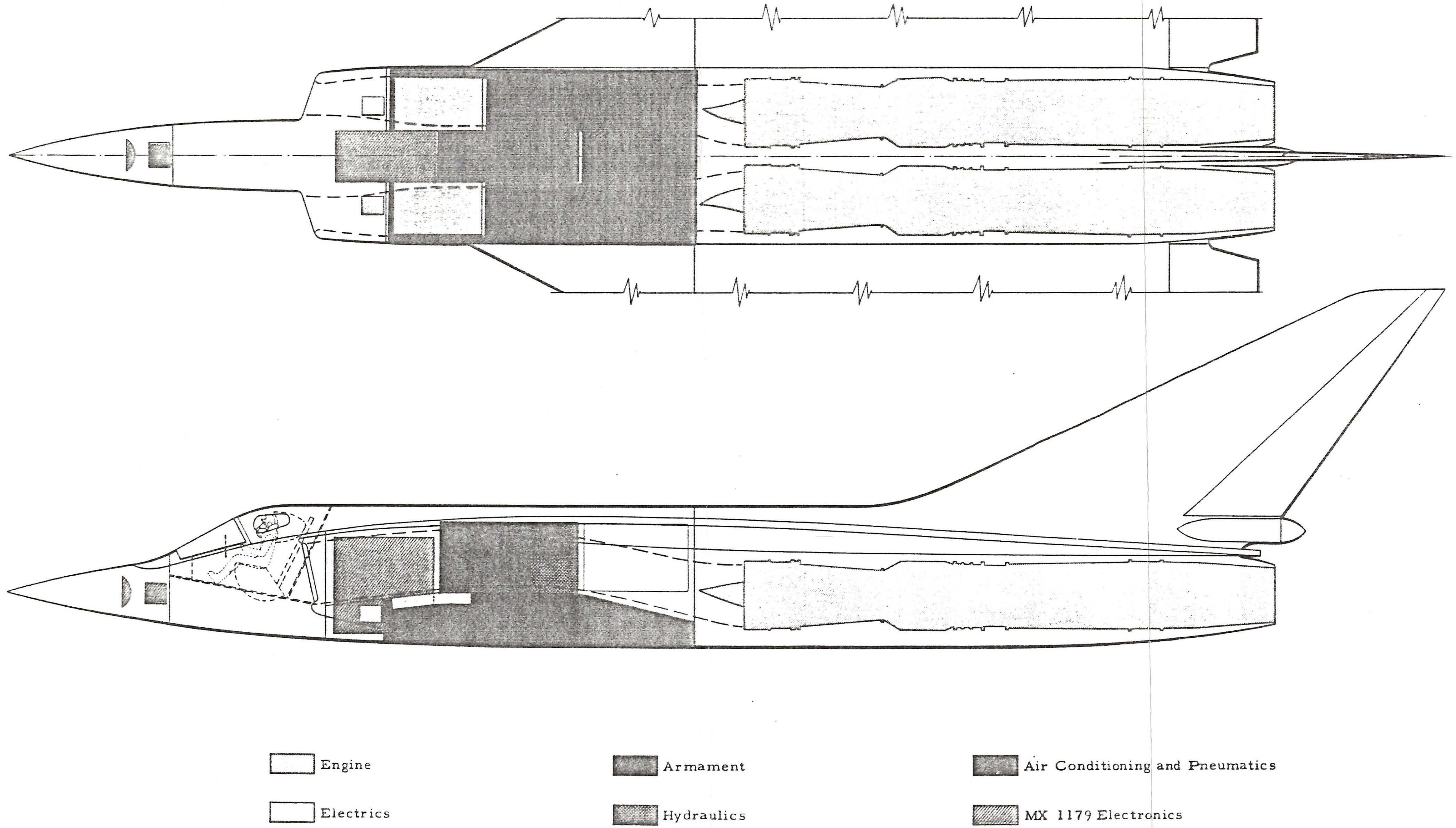
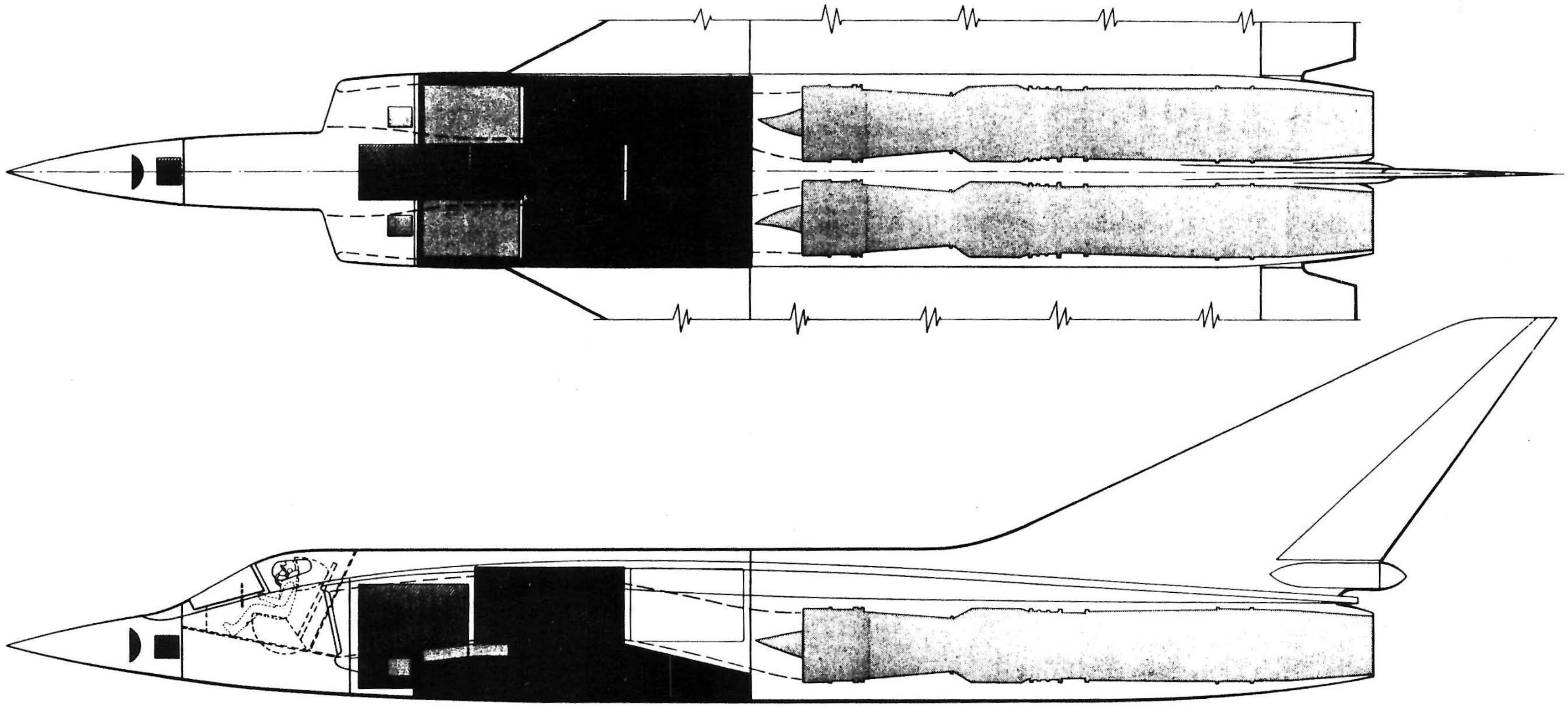


FIG. 20 EQUIPMENT ARRANGEMENT



Engine

Electrics

Armament

Hydraulics

Air Conditioning and Pneumatics

MX 1179 Electronics

FIG. 20 EQUIPMENT ARRANGEMENT

Item (d) forms the largest section and is approximately 20 feet long by 8.67 feet wide. The wing structure, which is continuous across the fuselage, is divided into ten separate parts, i.e. left and right inner wings joined at the aircraft center line, left and right inner wing trailing edges, left and right outer wings, left and right outer wing trailing edges, and left and right wing tips. The elevators and ailerons are separately hinged to the trailing edge assemblies of the inner wings and outer wings respectively. The dimensions of the largest wing panel are approximately 14.3 feet by 31.5 feet.

NOTE: These dimensions will allow transport by road or rail.

Bolted joints connect the fin to the wing structure. This application also complies for the attachment of the main and nose undercarriage units.

3.2.5 Interchangeability and Replaceability: The component parts of all airplanes of the same model, exclusive of experimental and service test airplanes, shall be interchangeable or replaceable in accordance with and to the extent required by Spec. AN-I-21, and shall be manufactured in conformity with the provisions of such specification.

3.2.6 Finish: The finish of the airplane and parts shall be in accordance with RCAF specification Proc. 31-3 for parts, and in accordance with the C-100 process for the exterior surfaces, i.e. polished metal plus spar varnish or its equivalent.

3.2.7 Identification and Marking: The airplane and its components shall be identified and otherwise marked in accordance with RCAF requirements.

3.2.8 Extreme Temperature Operation: The airplane as a whole, including equipment, shall be so constructed that it will function satisfactorily in any or alltemperature conditions that will be encountered. A ground temperature range of:-

-65 degrees Fahrenheit
to
+160 degrees Fahrenheit

has been established as the range in which the airplane shall operate.

3.2.9 Climatic Requirements: The airplane and its equipment shall not be adversely affected by other climatic conditions incident to the temperature range stated in the preceding sub-paragraph, and shall be capable of transfer from one climate to another without penalty of extensive modification and adjustment.

3.2.10 Lubrication: Lubrication of the airplane shall conform to the requirements of AN-L-32.

3.2.11 Standard Parts: Specifications and standards for all materials, parts and

Government certification and approval of processes and equipment, which are not specially designated herein and which are necessary for the execution of the specification, shall be selected in accordance with ANA Bulletin 143.

3.2.12 Crew: The crew shall consist of one (1) pilot.

3.2.13 Equipment Installations: Equipment to be carried in the airplane is listed below.

3.2.13.1 Armament:

Guided missiles (Hughes 'Falcon')	6 off
Folding fin air-to-air rockets, 2.75 in. diameter	24 off

Further information on this equipment is contained in paragraph 3.18.

3.2.13.2 Communications and Navigation Equipment: This consists of the Hughes MX 1179 integrated navigation - communication - interception system, operating details of which are described in the following sub-paragraphs.

3.2.13.2.1 Operation: Due to the high speed of the airplane, it is considered essential to fit automatic interception and navigation equipment to the type specified in the preceding sub-paragraph. An additional function of the equipment is armament fire control. Details of this latter procedure are outlined in paragraph 3.18.

3.2.13.2.2 Interception navigation is effected by referring the G.C.I. broadcast data to the airplane position as determined by an A.P.I., corrected for wind by the use of DME omni range. All of the foregoing data are fed, as pulses, through a single, high speed digital computer which, through magnetically stored instructions processes all the relevant information, and sends the appropriate instructions to the auto-pilot which controls the flight path. A gyro-stabilized platform, which gives a vertical and north reference, is used as a datum for the steering instructions.

3.2.13.2.3 After the interception has been completed automatically, as discussed in sub-paragraph 3.18.3.4, the computer gives the necessary instructions to reach a marshalling area defined by its co-ordinates in the memory of the computer. From this area, the airplane is directed to the landing slot by a signal sent by the ground controller. This can be relayed either through the pilot or by automatic means. Landing will be accomplished by AILS or AGCA, the choice being dependent on which system is first developed to a high state of perfection.

> NOTE: It is assumed that the AILS will meet this criterion but will be displaced, ultimately, by a superior AGCA system.

3.2.13.2.4 Several redundancies occur in the navigation system. These will permit it to function with reasonable efficiency in the event of a number of contingencies. If the ground data link is severed, the wind vector computed from the last DME omni range information, is retained in the memory of the computer and applied, from the severage point onwards, to the API without change. The error involved by this method is often quite small. The path of the target may also be computed from CGI data stored magnetically in the event of failure of this link; reasonable accuracy is ensured providing that no evasive measures are attempted by the target. ADF is also used as an auxiliary source of information and may be compared with the data secured from the other sources to check their veracity. Alternatively, it may be used in the event of their failure.

3.2.13.2.5 All navigational information is displayed to the pilot in a convenient form so that he may take over any part of the system which becomes unserviceable. Enemy interference with the ground data link can be avoided if the pilot changes the wave length. A range of approximately twenty frequencies is provided for this purpose. Any other remedial action, which could be taken by the pilot of a conventional airplane, can also be applied by the pilot of this airplane during an emergency. For this purpose a manual over-ride of all controls is provided, guidance being obtained from the information displayed to the pilot from various sources.

3.2.13.2.6 Under normal circumstances, the only flight functions expected from the pilot are:-

(a) Taxying

(b) Take-off until terrain clearance has been secured, and

(c) Stopping the engines after taxying to the ramp.

Landing is accomplished automatically.

3.2.14 Equipment and Furnishings: Following is a list of equipment and furnishings provided in the airplane:-

Item	Reference Para.
Two turbo-jet engines and afterburners ...	3.12
Auxiliary gas turbine - compressor	3.13
Instruments	3.14
Hydraulic and pneumatic equipment	3.15
Electrical equipment	3.16
Electronic equipment	3.17
Armament	3.18
Furnishings	3.19
Air conditioning and anti-icing equipment	3.20
Auxiliary gear	3.22

3.3 Aerodynamics:

3.3.1 General: As the mission of this airplane requires the use of supersonic speeds, the basic configuration is so designed as to achieve the best possible aerodynamic characteristics in transonic and supersonic flight. This condition can be obtained by using the maximum wing sweep and the minimum t/c. Detailed studies have shown that, with conventional planforms, it is practically impossible to find accommodation in a wing of less than 6% t/c for the necessary equipment. In addition to this, the weight of the wing structure becomes excessive. Utilizing a delta configuration, however, a 3% t/c wing is perfectly practical with a 60 degrees sweep at the leading edge. Furthermore, due to the favourable disposition of material, the weight per square foot does not exceed that of a conventional unswept wing. These considerations made the selection of the delta planform a mandatory requirement. Turning to the tail unit, it has been found that, with the delta configuration, a horizontal tail becomes not only a superfluity but also an embarrassment. The reasons are as follows:-

(a) If a tail is raised substantially above the wing chord line it becomes destabilizing at moderate angles of incidence, thus giving rise to longitudinal instability. This, apart from being dangerous, restricts the airplane's manoeuverability seriously.

(b) If the tail is near the chord plane, these difficulties are overcome but it becomes impossible to secure the necessary ground angle for the low aspect ratio wing. The wake from the wing is also a serious problem in this case.

Accordingly then, the supersonic airplane is made tail-less. Troubles with longitudinal damping which have been associated with this configuration are avoided by using at least 60 degrees sweep. This has been proved by theory and demonstrated in practice on the Convair XF-92.

3.3.1.1 Subsonic Drag Synthesis: The drag used in the subsonic performance estimates was synthesized by adding up the drags for the various parts including appropriate allowances for roughness and interference. From the result, the drag coefficient of the wetted area (or, to use the technical appellation 'aerodynamic cleanness') was computed. As a further criterion, the ratio of the total drag to the flat plate friction drag was also calculated. This is known as the 'cleanness ratio'. These two quantities were compared with the values achieved by several airplanes in service; the comparison forms a check on the vailidity of the original estimate. Following are details of the synthesis:-

<u>Body - including canopy and side inlets</u>: Due to the jet location in the tail, the diameter-length ratio of the fuselage (d/l) is effectively reduced by approximately 40%. Then, based on the wetted area, the drag for a stream-line body, with transition at the nose for cruise flight conditions, is found. An additional drag

increment of 25% is added for roughness and leakage. The drag is then referred to the gross wing area.

Wing: It has been shown that subsonic drag is independent of the angle of sweep-back providing that the drag is based on the streamwise thickness chord (t/c) and thickness distribution. On this basis, the exposed wing drag is determined. An additional drag increment, due to flush rivets, is taken as .001. The wing drag is then referred to the gross wing area.

Vertical Tail: Data for the vertical tail drag is found in the same way as that for the wing drag.

Interference: The drag of wing + interference has been found from rocket model tests on models similar in configuration, differences occurring primarily in body cross-section area to wing gross area ratio. These, however, bracket the ratio required so that a reasonable estimate of the interference can be obtained.

Miscellaneous: The 25% drag increment already added to the body is only for normal conditions. It is thought that the armament section doors and the inter-cooler system will give rise to more drag.

The component drags stated are enumerated below together with their percentage of the total zero lift drag.

Part	CDo	% Total
Wing	.00389	44.8
Body - including canopy and side air intakes	.00306	35.1
Vertical tail	.00081	9.3
Armament	.00015	1.7
Interference	.00079	9.1
TOTAL:	.0087	100.0

The aerodynamic cleanness and cleanness ratio of several jet airplanes - including the C104/2 - are given in the following list as a check on the low speed profile drag estimate. It should be noted that the smooth flat plate friction drag used to obtain the cleanness ratio is varied in accordance with the averaged Reynolds number of each airplane.

Airplane	Aerodynamic Cleanness	Cleanness Ratio
C100	.00367	1.50
F80	.00299	1.20
F86	.00326	1.30
Meteor 4	.00380	1.52
C104/2	.00311	1.41

The aerodynamic cleanness required for the C104/2 is relatively quite good. This is not as difficult to attain as for the other airplanes due to the much higher Reynolds numbers of the components. The true situation is more clearly illustrated by the cleanness ratio, where this is allowed for; here the C104/2 lies approximately half way between the F86 and the C100. While the F80 sets a very high standard of surface finish, the F86 is not, by comparison, remarkable in this respect. Accordingly, it is felt that a small improvement over the C100 is quite practical and does not represent any undue optimism as to what may be achieved by a certain amount of attention to surface finish. Drag reference data were compiled from:-

Wing and Tail	Rivets and Roughness
RAS data sheets	ACR R&M No. 2258
NACA RMLCJ01a	Journal of the Aeronautical Sciences,
MAP VOLK R&T361	April 1946 and August 1940
RAE AERO 2295	NACA TR667
Fuselage	**Interference**
RAS data sheets	NACA RML50D26
MAP/VG R&T857	NACA RML50I22

3.3.1.2 Supersonic Drag Synthesis: Refer to Fig. 21. The supersonic drag was, in most cases, expressed as a ratio of the subsonic drag. A detailed synthesis was carried out in much the same manner as for the subsonic case. In the following details of the synthesis it will be noted that the body is treated as a basic component with the canopy and spillage details covered separately.

NOTE: The figures in parenthesis in the following text refer to the List of References following sub-paragraph 3.3.1.3.

Body: This component embraces the streamlined body only, without the canopy and side air intakes, and results in a fineness ratio of approximately 10 with the maximum cross-section area occurring about half way down the body length. From rocket model tests (1, 2, 3) the drag rise is given as approximately 150%, tapering off to 100% at 1.5 MN. Base drag has been subtracted since the base area will almost be eliminated by the jet stream. The roughness and leakage drag increment is held constant for all Mach numbers (7).

Canopy: The body wave drag will be increased by the addition of the canopy. Available data (5, 6) so far, are not entirely satisfactory but give a suitable impression of the expected drag increase.

Spillage: At full mass flow (2) and the sharp intake lips at zero angle of attack, the nacelle drag would be zero. Mass flow, however, varies with r.p.m., altitude and speed, and choking must also be prevented. Since the spillage drag

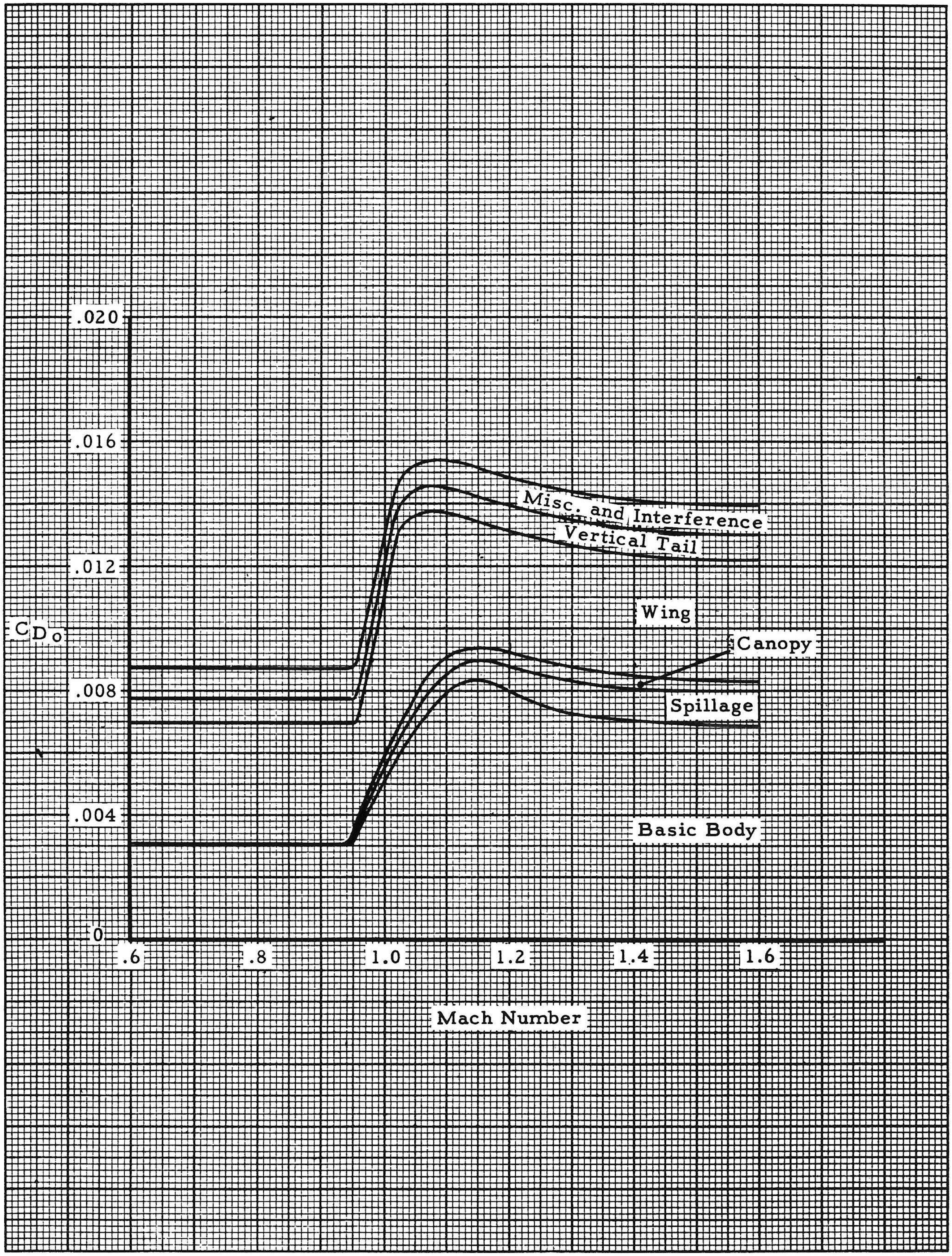


Fig. 21 Profile Drag Coefficient

appears relatively small, it is added only as a variable with Mach number, for a mean of design flight conditions.

Wing: The maximum body cross-section area to gross wing area ratio is almost evenly bracketed by information (3, 4) and shows a slight increase in wing + interference drag for the larger ratio and decrease for the smaller ratio. Thus, a constant wing + interference drag is taken. Since the interference drag has already been separated for low speeds, it is carried through into the supersonic range due to the difficulty of separating the interference drag at supersonic speeds. The wing drag, therefore, is considered constant. While this, of course, is not academically true, it does not affect the final drag answer.

Vertical Tail: Since the subsonic drag of the tail is small and the drag rise is indeterminate because of interference effects, the vertical tail drag is considered constant over the entire Mach number range. This is considered to be a negligible error as the drag rise of a 3% section is small (2).

Interference: Interference drag is covered under the wing section.

Miscellaneous: Miscellaneous drag is above the normal allowance for roughness, rivets and leakage. This extra drag provides for armament doors and intercooler system effects.

3.3.1.3 Drag Efficiency and Elevator Drag: An analysis of all available data (8, 9) on the drag efficiency factor indicates that for a delta configuration with less than 10% t/c, the drag efficiency lies between .4 and .5 for both subsonic and supersonic speeds, see Fig. 22. A major factor in the drag of a delta is that due to the deflected elevators, which may be considerable at high altitudes. Trim curves were calculated for a mean c.g. position and the drags estimated by using reference 10. These were added to the other items to form the total.

3.3.1.3.1 List of References: Following is a list of applicable references used in the compilation of the preceding data:

(1) NACA RM L9130: Flight investigations at high subsonic, transonic, and supersonic speeds to determine zero-lift drag of fin-stabilized bodies of revolution having fineness ratios of 12.5, 8.91 and 6.04 and varying positions of maximum diameter.

(2) RAE TM AERO 132: The estimation of the drag of an aircraft at supersonic speeds.

(3) NACA RM L50D26: Large-scale flight measurements of zero-lift at Mach numbers from 0.86 to 1.5 of a wing-body combination having a 60 deg. triangular wing with NACA 65A003 sections.

(4) NACA RM L50I22: Comparison of large-scale flight measurements of zero-lift drag at Mach numbers from 0.9 to 1.7 of two wing-body combinations having similar 60 deg. triangular wings with NACA 65A003 sections.

(5) NACA RM L7L22: Effect of a pilot's canopy on the drag of an NACA RM-2 drag research model in flight at transonic and supersonic speeds.

(6) NACA RM L8E04: Effect of windshield shape of a pilot's canopy on the drag of an NACA RM-2 drag research model in flight at transonic speeds.

(7) RAE TM AERO 130: Skin friction at supersonic speeds.

(8) NACA RM A50K24a: Lift, drag, and pitching moment of low aspect ratio wings at subsonic and supersonic speeds; plane triangular wings of aspect ratio 2 with NACA 0003-63 section.

(9) NAE: Lab memo AE-6b. The induced drag factor of highly swept delta wings at subsonic speeds by Procter/Westell.

(10) NACA RM L51I04: Flight determination of the drag and longitudinal stability and control characteristics of a rocket-powered model of a 60 deg. delta-wing airplane from Mach numbers of 0.75 to 1.70.

3.3.1.4 Ducts: The main engine air intake duct entries are of D-section and are located immediately aft of the cockpit, one on each side. The intake area has been designed for minimum combined spillage and internal diffusion loss. Generous boundary layer bleeds are fitted to ensure the maximum pressure recovery under all conditions. The Mach number in the ducting is kept below 0.6 under all important conditions.

3.3.1.5 Radome: A sharp-pointed radome has been assumed, the design of which serves as a compromise between the radiation and high-speed aerodynamic requirements. The aerodynamics are not perfect but the penalties involved should not be appreciable. On the other hand, the radiation problem is regarded as very severe but by no means impossible of solution in time. In consequence, this nose is regarded as a target design to be achieved some time in the future, the date of which can not yet be ascertained. As an interim measure, it may be necessary to fit a nose with a more elliptical contour. The penalty for this latter measure is very small up to a Mach number of 1.4, but increases rapidly above that speed. It is assumed that the radome will be formed of a sandwich construction and provision will be made for anti-icing, using either infra-red radiation or hot air circulation.

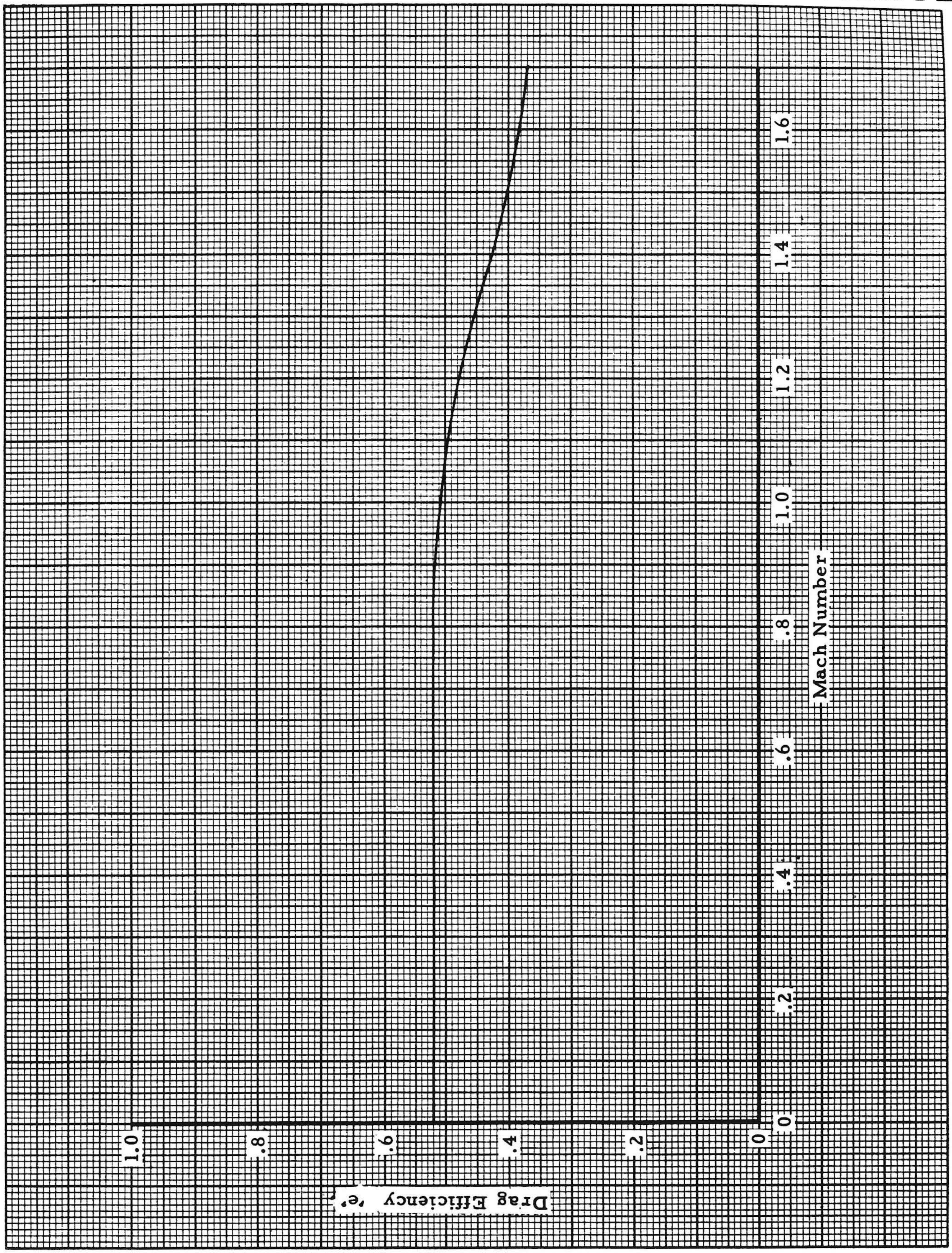


Fig. 22 Drag Efficiency

3.3.2 Stability and Control:

3.3.2.1 Longitudinal Stability and Control: The airplane will have positive static longitudinal stability for all speeds, loadings and power conditions except when the lift coefficient is greater than 0.7. A push will be required to increase, and a pull to decrease the speed for all conditions except when the lift coefficient exceeds 0.7 or when the Mach number is between .95 and 1.05. The static margin (measure of static stability) increases steadily with Mach number until it is approximately six times the low speed value at supersonic speeds.

3.3.2.1.1 Dynamic Longitudinal Stability: At all speeds and altitudes short period pitching oscillations will be damped. Long period (phugoid) oscillations will be damped by an electronic damper which is included as part of the auto-pilot.

3.3.2.1.2 Longitudinal control is achieved by plain flap-type elevators which are power operated and fully irreversible. Aerodynamic balancing aids are not fitted to the surfaces. The effectiveness of the controls at:-

(a) Sea level is shown in Fig. 17

(b) 40,000 feet is shown in Fig. 18

(c) 50,000 feet is shown in Fig. 19

It will be seen from the graphs that nearly all potential manoeuverability - as limited by the maximum lift coefficient obtainable at a given Mach number - is realized by the elevators at a cost of 62,000 ft. lb. hinge moment required at each side.

3.3.2.1.3 Take-off: It has been established that the main parameters involved in take-off characteristics from the control point of view are:-

(a) The main undercarriage position in relation to the c.g., and

(b) The angle between the wing chord and the ground line (airplane attitude - nose wheel on the ground).

Suitable choice of these parameters has allowed the speed required to raise the nose to equal 90% of the take-off speed and has reduced to a minimum the stick movement required between the nose-up and take-off positions to trim in the air near ground level (equivalent to approximately 5 degrees elevator movement). In consequence, the undesirable aspects of 'stick pumping' at take-off and 'jumping off' the runway have been avoided. These are noticeable characteristics of some contemporary delta wing airplanes. For the C104, however, a take-off technique similar to orthodox high-speed fighters is envisaged differing only in the rather large ground angle involved. Take-off speed is approximately 160 knots at a forward c.g. of 27% m.a.c..

3.3.2.1.4 Landing: Landing will be accomplished at small incidences using the flying-in technique. A tail parachute, streamed either automatically or by manual control, will be used to achieve braking in order to keep the landing run within reasonable limits.

3.3.2.1.5 Balked Landing: Stability difficulties will not arise as long as the speed is not allowed to fall below 140 knots. The destabilizing effect of power required at this speed will be counteracted by the built-in downwards inclination of the jet pipes. At speeds below 140 knots, however, with full power on, the airplane will have negative static longitudinal stability similar to that occurring on most conventional airplanes.

3.3.2.2 Centre of Gravity Limits: Considerable difficulty arises, for an airplane operating through a large Mach number range, in fixing c.g. limits from a longitudinal stability point of view. This difficulty is created by the following:-

(a) The extensive aft travel of the aerodynamic center with Mach numbers, and

(b) A forward travel with an increasing lift coefficient for moderate speeds (up to M = .7).

Aggravation of the situation is caused by the destabilizing effects of power at high lift coefficients. In normal practice, it is customary - with orthodox subsonic airplanes - to provide for a static margin of approximately 5% m.a.c. at low speeds and high lift coefficient. To achieve a similar provision with supersonic airplanes it would mean, in the case of the C104, an increase of its supersonic static margin by 40%. At high altitudes and speeds (because of the low value of aerodynamic damping), elevator effectiveness in executing pull-outs and turns is almost directly proportionate to the static margin. In effect, this means that the penalties at high speeds and altitudes - which are the operational requirements for this airplane - would be either a 40% reduction in pull-out and turning performance or an even higher percentage increase in hinge moments required to maintain the performance specified in the flight envelope graphs (Fig. 17-19). Additionally, it would mean a higher overall drag due to higher elevator deflection and consequent greater loss of height in high 'g' turns. The whole performance, therefore, as an efficient interceptor airplane, would be affected. It would appear, then, that to aim at the traditional low speed static margin is an unacceptable solution when dealing with supersonic airplanes. Consequently the aft c.g. limit on this airplane is fixed at 31% m.a.c. which gives:-

(a) A static margin of 3% at low speeds for C_L range of 0 to .3, and

(b) A static margin of 1% at low speeds for a C_L range of .3 to .7

Both limits are with power off. The destabilizing effect of power will be counteracted by the downward deflection (approximately 6 degrees) of the jet pipes. In the following table data are given on static margins at low speeds of other delta airplanes to indicate that other designers have reached similar conclusions.

TABLE 13 - COMPARISON OF NEUTRAL POINT, AFT C.G. AND STATIC MARGIN FOR DELTA WING AIRPLANES

Aircraft	Neutral Point	Aft C.G. Limit	Static Margin $0 < C_L < .3$	Static Margin $.3 < C_L < .7$ (estimated)
Avro 707B	32.5 (flight test)	30.0	2.5	+ .5
Convair XF92	35.0 (flight test)	31.0	4.0	2.0
C104	34.1 (estimated)	31.0	3.1	1.1

NOTE: All figures in the above table are represented in % m.a.c.

The forward c.g. limit is fixed at 27% m.a.c. with a corresponding take-off speed of approximately 160 knots.

3.3.2.2.1 Summary of C.G. Limits in % M.A.C.: C.G. limits, represented as a percentage of the main aerodynamic chord, are summarized as follows:-

Forward	Aft
.27%	.31%

3.3.2.3 Lateral Stability Control: The airplane will posses a small degree of positive static directional stability. This is characteristic of most modern high speed fighters. A further characteristic, which is common to all swept back wings, is the satisfactory dihedral effect which will become somewhat excessive at high lift coefficients. Dynamic lateral stability (Dutch roll) will be maintained at a satisfactory level by an electronic damping device incorporated in the auto-pilot.

3.3.2.3.1 Lateral control is achieved by plain flap-type ailerons and the rudder. All the surfaces are fully irreversible and are power operated. The ailerons are capable of giving a maximum rolling performance of 205 deg. per second at all altitudes. (Reference: U.S.A.F. Spec. 1815-B, para. 6.3.4). In the evaluation of these estimates account has been taken of aeroelastic distortions.

3.3.2.3.2 Low Speed Response: It is recognized that delta wings exhibit, in general, a rather sluggish initial response to aileron command at low speeds. This condition is covered by U.S.A.F. Spec. 1815-B, para. 6.3.4 requirement of pb = 10 ft. per sec. for conventional designs. It should be noted that the ailerons on the C104 are nearly four times more powerful than the above requirement.

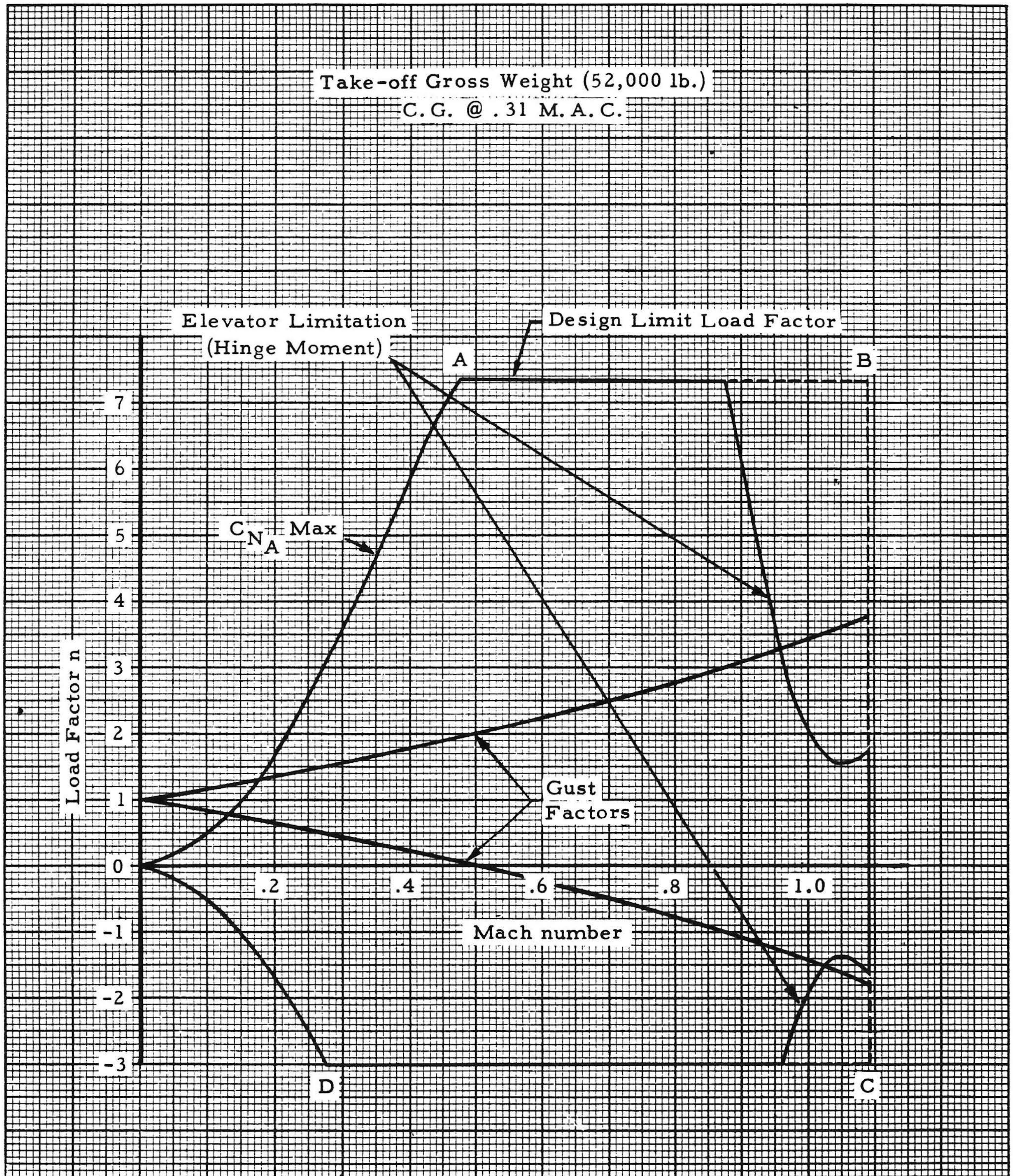


Fig. 23 Structural Flight Envelope at Sea Level

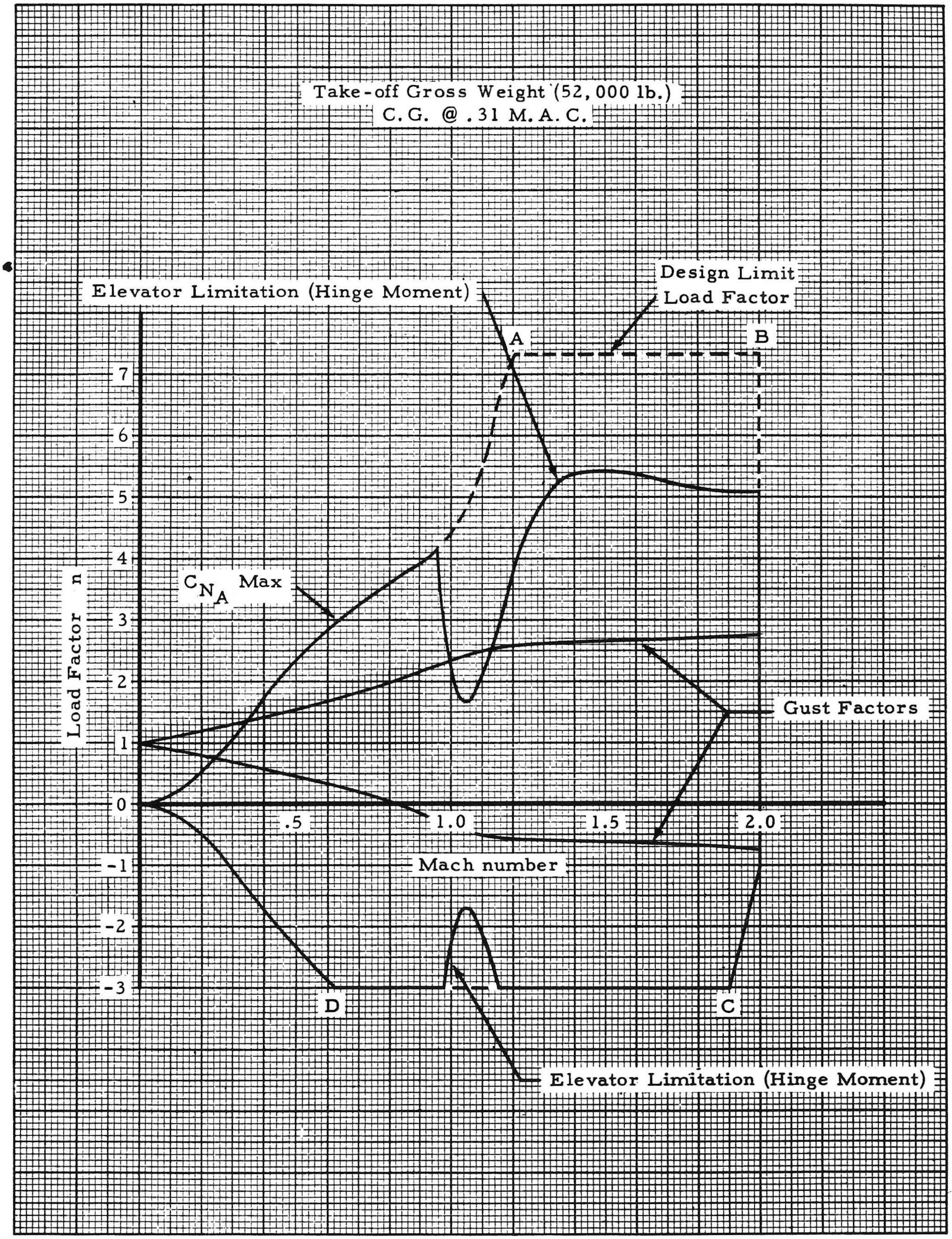


 Fig. 24 Structural Flight Envelope at 30,000 ft. Altitude

3.3.2.3.3 Wing Dropping: Flight reports on high speed aircraft (e.g. Bell X-1, Douglas Sky Rocket) indicate a very serious and troublesome aspect of transonic flight manifested by a sudden loss of lateral trim. This condition leads to large amplitude lateral oscillations which are extremely difficult to control. It is believed that this phenomenon is, primarily, a function of wing thickness. By using a 3% wing section it is assumed that this transonic flight disadvantage will, in all probability, not be encountered on the C104. This belief is confirmed by the experience gained on the Convair XF-92.

3.3.2.3.4 The rudder is powerful enough to meet manoeuvering requirements as specified by U.S.A.F. Spec. 1815-B, para. 5.3.

3.4 Structure Design Criteria:

3.4.1 Limit Flight Load Factors: The critical flight envelopes occur at sea level and 30,000 ft. and are shown on Fig. 23 and 24. At sea level, the transonic phenomena are associated with a higher equivalent air speed than at 30,000 ft., thus giving critical conditions for this range. At 30,000 ft. the Mach range reaches the maximum design value, and accordingly must be investigated in its entirety. The elevator limitations have been investigated for a c.g. position extended 1% m.a.c. aft of the aft limit to ensure that the most critical loads are obtained.

3.4.2 Limit Ground Load Factors:

Ground Take-off, n = 2.5 at Design Gross Weight
Ground Landing, n = 2.5 at Design Gross Weight
Ground Landing, n = 3 at Landing Weight = 44,000 lbs.

3.4.3 Limit Diving Speed: The limit diving speed is either 720 knots EAS or a Mach number of 2.0, whichever is the lesser speed.

3.5 Wing Group:

3.5.1 Description and Components: Refer to Fig. 25. The wing, which is of triangular shape in plan (so-called delta shape), forms a continuous structure through the top of the fuselage. The reason for adopting the high wing configuration is that this arrangement makes it possible to design an ideal installation for the side-by-side engines and afterburners which are located below the main wing structure. The size of this airplane is just large enough to obtain a satisfactory main landing gear installation with a high wing layout. This latter consideration necessitates the anhedral angle of three degrees to the wing, which is acceptable from aerodynamic considerations. Maximum wing thickness is three percent of the chord at all stations along the span. The wing is manufactured from a number of sub-assemblies which are bolted together at transport joints. These assemblies are as follows:-

(a) Two inner wing assemblies joined at the airplane center line

(b) Two inner wing trailing edge assemblies

(c) Two outer wing assemblies

(d) Two outer wing trailing edge assemblies, and

(e) Two wing tips.

An aileron and an elevator are fitted to each wing trailing edge, the aileron being outboard.

3.5.2 Construction:

3.5.2.1 Inner Wing Assembly: This component consists of a large conventionally constructed main sparbox, located aft of the landing gear well, and a leading-edge structure positioned forward of the landing gear well. The main sparbox consists of three spanwise spars, a large number of chordwise ribs intercostal between the spars, outer surface skin panels reinforced by extruded Z-section stringers, and inner skins which attach to the flanges of the stringers and to the ribs. The inner skins cater for bladder-type fuel cells which are housed in the compartments bounded by the spar and rib webs. Fuel cell access doors are provided in the bottom skins similar to those used on the CF-100. This type of structure is very efficient from a weight standpoint and also gives an excellent aerodynamic surface, since no skin buckling will occur below the limit load factor. The skin and stringers will carry the end-loads due to bending and also the shears due to torque. The vertical shear loads are taken by the spar webs while the ribs, apart from stabilizing the skin-stringer panels, cater to the high chordwise bending moments which are characteristic in a delta wing. The main spar box also absorbs the greater part of the main landing gear loads which are transmitted into it by a very rugged end rib located at the outer wing transport joint. Two robust ribs are located in line with the fuselage sides and another one at the center transport joint of the left and right inner wings. Bolted to these three ribs is the powerplant bay fuselage structure which consists essentially of:-

(a) A center keel beam and

(b) Two fuselage-side beams.

These three beams form a structural unit with the wing ribs to which they are connected on assembly of the complete airplane. The design of the wing transport joints consists of robust external angles of extruded light alloy material which are bolted externally to the skin of the sparbox. These attach to similar angles on the mating component by means of tension-bolts. A streamlined shroud covers the angles on final assembly; at the center transport joint such shrouds are not required. The fin structure attachment to the left and righ main sparboxes is by means of transport joint bolts. The fin bending moment is, therefore, transferred to the wing main spar boxes. The upper speed brakes are hinged from a local structure cantilevered aft from the rear spar of the sparboxes. The inner wing leading edge structure, forward of the landing gear well, consists, in the first place, of a heavy leading edge spar which attaches to the wing transport joint rib at the outboard end and is pin-jointed to the fuselage side at the inboard end. Further, another transverse spar attaches to the leading edge spar (outboard end) and continues across the body to the center line rib to which it attaches. The rib in line with the fuselage side attaches to this transverse spar. A bladder-type fuel cell is housed in the triangular compartment between the leading edge spar and the transverse spar. The leading edge proper is riveted to the leading edge spar, and consists of closely spaced riblets normal to this spar and a thick skin covering. Provision is made in the leading edge for hot air anti-icing. The main landing gear well between the main spar box and the leading edge structure is a conventional structure of diaphragms, arched ribs and stringer-stiffened top skin. The landing gear pivot axle is attached in fittings which are bolted, respectively, to:-

(a) The front spar of the main sparbox, and

(b) The leading edge spar.

In order to detach the main gear from the wing it is proposed to extract this pivot axle forward through a detachable panel in the leading edge structure. The main gear side stay is attached to a robust built-up structure cantilevered forward from the front spar of the main spar box. Large wheel doors will be hinged to the inboard edge of the wheel well to cover the undercarriage bogie and wheels when these are retracted. Since the bogie does not fit completely inside the lower airfoil surface, these doors are slightly bulged, forming a streamlined shallow fairing.

3.5.2.2 Inner Wing Trailing Edge Assembly: This assembly is made from a series of magnesium alloy castings which are bolted to the rear spar and outer skin of the inner wing assembly. The castings consist essentially of a grid of webs with flanges. They are bolted together where the inboard web of one casting joins the outboard web of the casting next to it. Special extruded piano-type hinges are bolted to the rear webs of the castings to form the elevator attachment to the wing. Inside this assembly is housed the elevator control actuating mechanism which consists of a long push-pull rod driving seven bellcranks, each of which operates the elevator through a short push-pull rod. Excellent access to this mechanism is provided by a large number of quickly-detachable doors which are flush with the castings when closed, thus maintaining a smooth wing surface.

3.5.2.3 Outer Wing Assembly: This component consists of a multi-cell arrangement of spars and ribs bounded by a leading edge spar and a rear spar, the whole assembly being covered with relatively thick aluminum alloy skin. The spars are of formed channel sections. The ribs are intercostal with shear-attachments through the spars. An excellent aerodynamic surface is obtained, since skin buckling will not occur below the limit load factor. The leading edge portion of this assembly will be manufactured

LEGEND

(1) Radome
(2) Radome/Cockpit Section Transport Joint
(3) Cockpit Section
(4) RH Engine Air Intake
(5) Electronic Equipment Compartment
(6) Air Intake Structure
(7) Hydraulic & Pneumatic Equipment Compartment
(8) Longitudinal Diaphragms
(9) Access Aperture to Main Fuel Cells
(10) Access Panels - Main Fuel Cells
(11) Fuel Cell Compartments
(12) Inner Wing - Leading Edge Spar
(13) Control Trough
(14) Main Undercarriage Retracting Bay
(15) Main Undercarriage Pivot Axle
(16) Intermediate Wing Spar - Forward
Outer Wing - Leading Edge Spar
(18) Fuel Cell Compartments
(19) Intermediate Wing Spar - Aft
(20) Detachable Wing Tip
(21) Aileron
(22) Outer Wing - Rear Spar
(23) Inner Wing - Rear Spar
(24) Elevator
(25) Multi-cell Fin Structure
(26) Fin Casting - Trailing Edge
(27) Detachable Fin Tip
(28) Rudder
(29) Brake Parachute Housing
(30) Fuselage Speed Brake - Upper
(31) Variable Afterburner Nozzle
(32) Rear Nacelle Section of Fuselage
(33) Elevator Hinge - Piano Type
(34) Elevator Control Linkage
(35) Inner Wing - Trailing Edge Casting
(36) Outer Wing - Trailing Edge Casting
(37) Transport Joint - Power Plant Section/Rear Nacelle Section
(38) Transport Joint - Inner Wing/Outer Wing
(39) Transport Joint Shroud
(40) Fin Structure Attachment to Wing Transport
(41) Wing Transport Joint Center Line of Aircraft
(42) Hinged Access Doors - Engine Compartment, Aft
(43) Side Walls - Engine Compartment
(44) Main Undercarriage Fairing Door
(45) Formers - Engine Compartment
(46) Shroud - Engine Compartment
(47) Hinged Access Doors - Engine Compartment, Center
(48) Longitudinal Keel Structure
(49) Front Spar - Lateral Wing Section Attached to Fuselage Transport Joint
(50) Transport Joint - Equipment Section/Power Plant Section
(51) Air Intake Shroud
(52) Air Intake Duct
(53) Lateral Bulkhead - Equipment/Fuel Cell Compartments
(54) Lateral Bulkhead - Armament Compartment
(55) Floor Structure - Equipment Compartment
(56) Armament & Electrical Equipment Compartment
(57) Lateral Bulkhead Forming Aft Wall of Electronics Compartment
(58) Transport Joint - Cockpit Section/ Equipment Section
(59) Boundary Layer Bleed - Air Intake
(60) Nose Undercarriage Pivot Axle
(61) Cockpit Floor Structure
(62) Nose Wheel Door

(63) Bulkhead - Cockpit/Radome Section

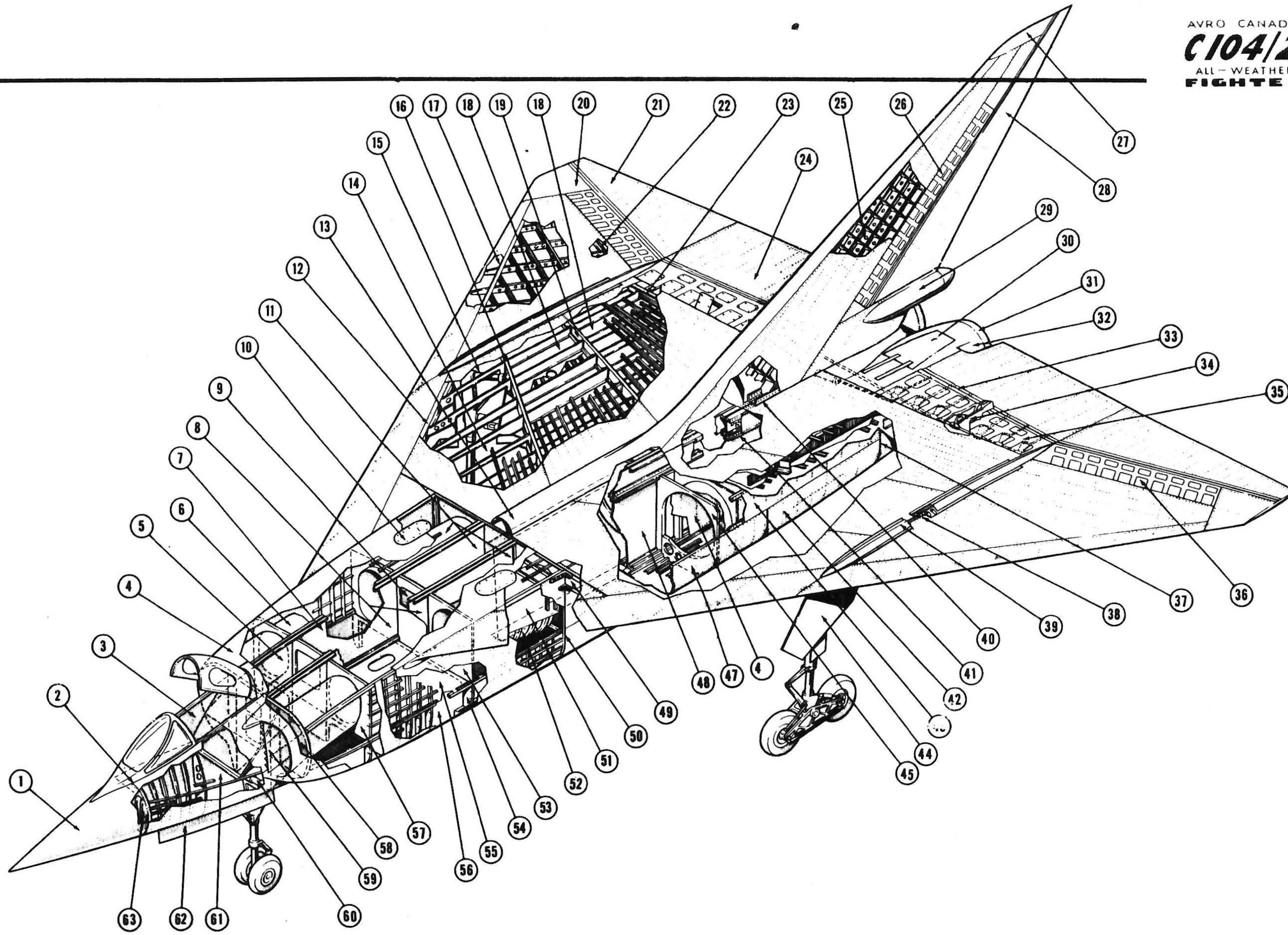


FIG. 25 STRUCTURE

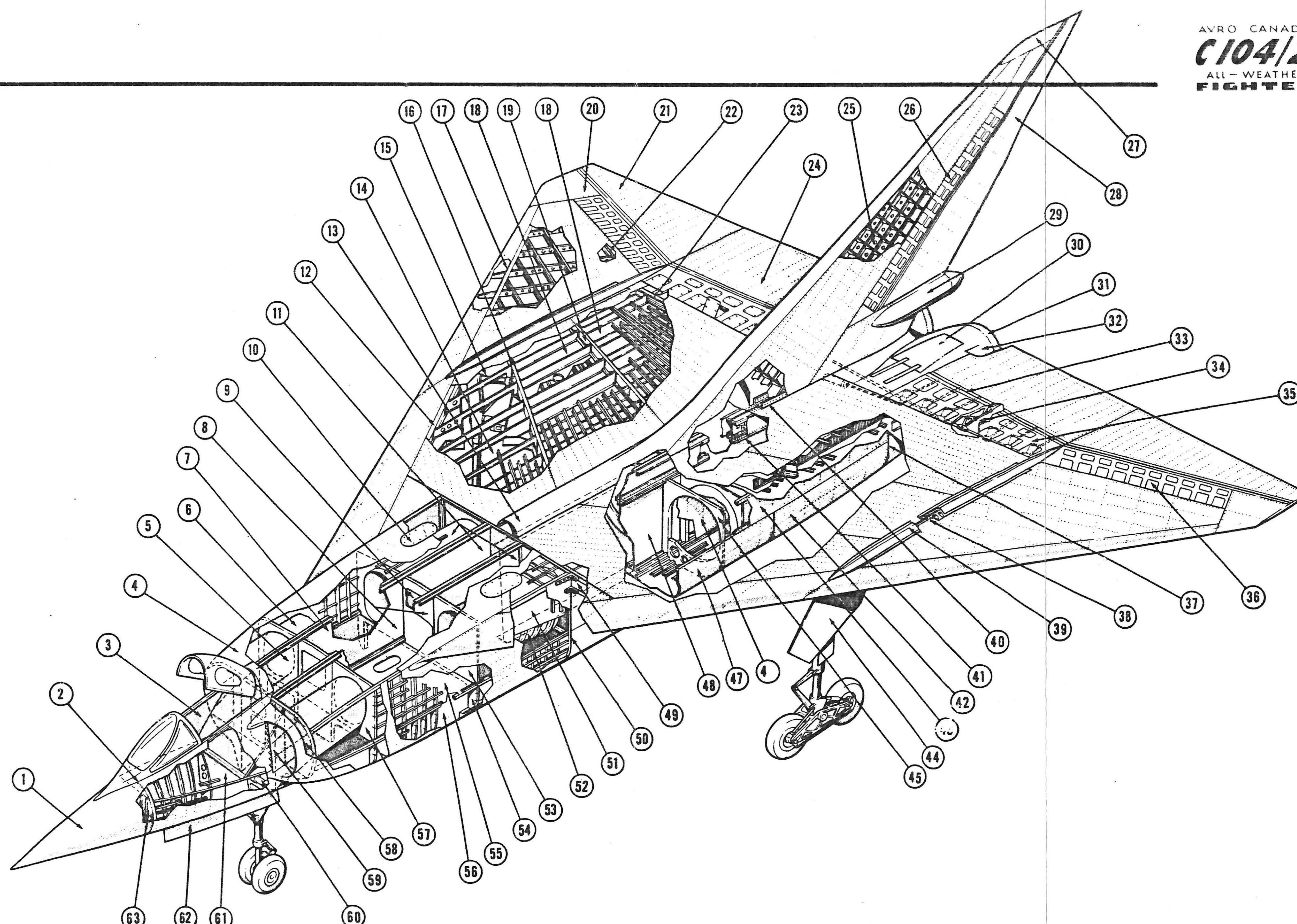


FIG. 25 STRUCTURE

as a separate sub-assembly but it is not intended to make this a detachable component. Hot air anti-icing will be provided for in the construction of the leading edge. The outer wing assembly attaches to the inner wing by means of an external type of transport joint with tension bolts (refer to sub-paragraph 3.5.2.1).

3.5.2.4 Outer Wing Trailing Edge Assembly: This assembly, which is made from a series of magnesium alloy castings similar to the inner wing trailing edge assembly, houses the aileron control actuating mechanism which consists of ten bell cranks and push-pull rods. Access to every bell crank is obtained in the same manner as described in sub-paragraph 3.5.2.2.

3.5.2.5 Wing Tip Assembly: This assembly is conventionally constructed and attaches to the outer wing by means of screws so that it may be quickly replaced in case of damage.

3.5.2.6 Material: In stressing the wing structure, account will be taken of the effect of elevated temperature due to air friction at the design speeds of this airplane. The effect of this temperature, which may reach a value of 160° under tropical summer conditions at a Mach number of 2, is to decrease somewhat the strength and stiffness of the light alloy material. The materials to be used in the wing structure are as follows:-

All skin panels and ribs and outer wing spars	24ST clad aluminum alloy
Sparboom extrusions	75ST aluminum alloy
Spar webs of inner wings	75ST clad aluminum alloy
Trailing edge castings	DNDC-21-97 magnesium alloy

3.5.3 Elevator: This component is hinged to the trailing edge assembly of the inner wing by means of a special extruded piano hinge which is bolted to the top skin and the leading edge spar of the elevator. The elevator is actuated by seven push-pull rods equally spaced along the span. These rods attach to lugs on the elevator. As the lugs protrude slightly below the lower surface of the wing they are shrouded by small fairings. In order to keep these lugs as small as possible, uni-ball type self-aligning bearings are used. The structure of the elevator consists of a leading edge spar and closely spaced ribs covered with thick aluminum alloy skins. A blunt trailing edge is used which will consist of a light alloy extrusion; the reason for this is that this type of trailing edge improves the torsional stiffness considerably and yet causes no additional drag at supersonic design speed. Mass balance or aerodynamic balance devices are not incorporated in the design of the elevator and no tabs are fitted.

3.5.3.1 Elevator Motion: The elevators are fully power-operated by double piston hydraulic jacks; details of the control system are described elsewhere in this brochure. Elevator motion is as follows:-

UP 30 deg.
DOWN 30 deg.

3.5.4 Lift and Drag Increasing Devices: These are not fitted on the wing of this airplane.

3.5.5 Speed Brakes: These are not fitted on the wing of this airplane (refer to sub-paragraph 3.10.2.2.

3.5.6 Aileron: This component is similar in design and construction to the elevator and is hinged to the outer wing trailing edge assembly by means of a special extruded piano hinge. The ailerons, which are fully power operated, are actuated by ten push-pull rods equally spaced along the inner seventy five percent of the span. Most of the lugs to which these rods attach protrude slightly below the lower wing surface and are therefore shrouded by very shallow fairings. As on the elevator, uni-ball type self-aligning bearings are used. The outboard twenty five percent of the aileron will be covered by a somewhat thicker skin than the inboard portion; the reason for this is that, because of the extremely thin airfoil section, it is impossible to fit any actuating rods to this outboard portion. No mass balance or aerodynamic balance is incorporated and no tabs will be fitted.

3.5.6.1 Aileron Motion: Details of control system of the aileron are discussed elsewhere in this brochure. The motion of the aileron is as follows:-

19 degrees UP, and
19 degrees DOWN.

3.6 Tail Group:

3.6.1 Description and Components: Refer to Fig. 25. The tail group consists of a fin and rudder which are situated on top of the wing. The fin is sharply swept back and the maximum airfoil thickness is three percent of the chord.

3.6.2 Stabilizer: A horizontal stabilizer is not fitted to this airplane.

3.6.3 Elevator: The elevators are fitted to the trailing edge of the wing. Refer to sub-paragraph 3.5.3 for details.

3.6.4 Fin: The fin mounts directly on to the top surface of the wing and its root-shear, bending moment and torque are distributed directly into the wing structure. It is obvious then that the wing structure dictates to a great extent the type of structure that has to be employed for the fin. The structure of the fin consists of a multi-cell arrangement of spars and ribs bounded by a leading edge spar and a rear spar, the whole assembly being covered with relatively thick aluminum alloy skin. The spars are formed channels sections. The ribs which are at right angles to the rear spar, are intercostal with shear attachments through the spars. An excellent aerodynamic surface is obtained since skin buckling will not occur below the limit load factor. The leading edge portion of the assembly will be manufactured as a separate sub-assemblv but it is not intended to make it detachable. Hot air anti-icing will be provided for in the construction of the leading edge. The attachment of the fin to the wing is via a rugged fin root rib. This rib is bolted, by means of transport bolts, to the top surface of the wing and through it to two wing ribs disposed on either side of the wing center transport joint rib. The detachable wing tip is of similar conventional construction as the wing tips. The lower part of the fin extends aft to the trailing edge of the airfoil section and is manufactured as a separate sub-assembly. It incorporates the brake-parachute container which fairs elegantly into the airfoil lines of the fin. The material for the main fin structure will be 24ST clad aluminum alloy. As for the wing, account in stressing will be taken of the effects of elevated temperature due to skin friction at the supersonic design speed. The trailing edge of the fin in front of the rudder is similar in construction to the wing trailing edge and consists of a series of magnesium alloy castings (Specification DNDC-21-97) which are bolted together and house the rudder control actuating mechanism. Excellent access to this mechanism is available by the large number of quickly opening doors which are recessed in the casting when closed, thus maintaining a smooth aerodynamic surface.

3.6.5 Rudder: This component is hinged to the trailing edge castings of the fin by means of a special extruded piano hinge which is bolted to the right hand skin surface and the leading edge spar of the rudder. It is actuated by a large number of push-pull rods equally spaced along the lower seventy percent of the span, in a manner similar to the ailerons. The construction of the rudder is similar to that of the ailerons; mass balance and aerodynamic balance devices are not incorporated and no tabs will be fitted.

3.6.5.1 Rudder Motion: The rudder is fully power operated by a double piston hydraulic jack and has a motion of 25 degrees either way. Details of the control system are described elsewhere in this brochure.

3.7 Body Group:

3.7.1 Fuselage:

3.7.1.1 Description: Refer to Fig. 25. The fuselage extends the full length of the airplane, the cross-sectional shape being mainly rectangular with rounded corners at the bottom and sides. Accommodated in the fuselage are the pilot, equipment (including armament), nose landing gear, engines intake ducts, the engines and afterburners, flying and powerplant controls, speed brakes, a brake parachute, tail skid, and fuselage fuel tanks. The fuselage is to be manufactured in seven sections which are made as separate, replaceable assemblies which are bolted together at transport joints. These sections are as follows:-

(a) The radome section

(b) The cockpit section (incorporating the pilot's canopy)

(c) The equipment and armament section

(d) The powerplant section superstructure

(e) The powerplant doors

(f) The rear nacelle superstructure, and

(g) The lower rear nacelles.

A bullet-shaped radome is proposed. This contour gives the best entry for supersonic speed; it is known that this type of radome is under development by manufacturers of electronic equipment, but a more conventionally shaped radome could be fitted if necessary. The engine intake ducts are curved upwards slightly in order to provide maximum possible depth to the armament bay. Otherwise the ducts form a direct path from the cheek-type air intakes located just aft of the pilot's station, one on each side. These intakes are elegantly faired into the general fuselage lines. Running over the top of the fuselage and wing is a detachable dome-shaped fairing which fairs into the canopy at the forward end into the fin at the rear; controls, etc., are housed in this portion of the structure. The speed brakes are mounted as far aft as possible and consist of four separate flaps, one on each side of the rudder, hinged to the wing structure and housed flush in the rear nacelle superstructure, and two are hinged to and housed flush in the lower rear nacelle. Special attention has been given to the methods of housing equipment from the point of access for servicing and maintenance; further details of this are contained in the following sub-paragraphs and in the paragraphs dealing with the various equipment items. The whole configuration of the airplane has been dictated by the accessibility requirements of the side-by-side engines and afterburners and, in this respect, ideal installation has been achieved. This aspect of the design is enlarged upon in the following sub-paragraphs and in the sub-paragraph dealing with the engine installation. The wide fuselage necessitated by the engine arrangement plus the upward curve in the air intake ducts has been utilized to the fullest extent by providing this airplane with an extraordinary large armament bay. This important feature makes the airplane very adaptable to the carrying of different types of armament. This is enlarged upon in Appendix IV at the end of this brochure.

3.7.1.2 Construction:

3.7.1.2.1 Radome: This component, which houses the radar scanner and some electronic equipment, will be made of suitable dielectric material probably of sandwich construction. Anti-icing will be provided.

3.7.1.2.2 Cockpit Section: This component comprises the pilot's compartment, the windscreen, the canopy, the nosewheel compartment and the air intakes for the engines. The general structure follows conventional practice with skin, stringers, formers and longerons. The pilot's compartment is pressurized and is bounded by the cockpit sides, the pilot's floor, the radome bulkhead and a sloping seat bulkhead. Immediately aft of the seat bulkhead is a transport joint which attaches the cockpit section to the equipment and armament section. The material to be used will be 24ST clad aluminum alloy. The canopy will hinge upwards to provide access to the cockpit and will be power operated; jettison gear will be fitted. Canopy structure will be of conventional light alloy construction with glass windows set flush with the surrounding metal. This method of construction is necessary to cater to the air friction temperatures which are encountered at the limit design speed of this airplane. Suitable transparent material is not available at present to give the necessary strength required at these elevated temperatures. In consequence, a fully transparent canopy can not be fitted. It is the designer's intention that, when such a material becomes available, a fully transparent canopy will be substituted for the one at present proposed. The glass windscreen, which is bullet resistant, is of conventional construction and is optically flat; provision will be made for anti-icing and de-misting. The air intakes are of conventional light alloy construction and incorporate a boundary layer scavenge duct.

3.7.1.2.3 Equipment and Armament Section: This component houses the air intake ducts, the electronic, electrical, pneumatic and air conditioning equipment, the hydraulic equipment (except the main pumps), the armament and the fuselage fuel tanks. The structure of this component has been designed in such a way as to provide maximum accessibility for servicing of the various equipment and armament items. The basic structural scantlings consist of the following:-

(a) Two parallel central-beams, in line with the cockpit sides, which attach to the top of the fuselage surface, to the armament bay roof, and to the front and rear transport joint bulkheads.

(b) The fuselage sides.

(c) The fuselage top surface which lines up with the top surface of the wing at the rear transport joint of this component.

(d) The armament bay roof which extends right across the fuselage below the intake ducts. This item forms the main bottom structural boundary of this fuselage section since the armament bay structure below it is designed as a subsidiary structure. It attaches, like the fuselage top surface, to the front and rear transport joint bulkheads of this component. The forward part of this roof between the central beams is cut away to accommodate a crate which contains the electronic equipment.

(e) The front transport joint bulkhead, to which are attached the nose landing gear retracting axle, the two vertical central beams, the fuselage sides, the fuselage top surface, the armament bay roof and the bottom surface of the fuselage (subsidiary structure). The sides and floor of the cockpit assembly are connected to it by means of transport joint bolts.

(f) A bulkhead separating the electronic equipment compartment from the pneumatic and hydraulic equipment compartment.

(g) A bulkhead separating the rear equipment compartment from the fuel cell compartments.

(h) The rear transport joint bulkhead, to which are attached the two vertical central beams, the fuselage sides, the fuselage top surface, the armament bay roof and the bottom surface of the fuselage (subsidiary structure). The powerplant superstructure assembly and the transverse spar of the inner wing are connected to it by means of transport joint bolts.

It will be seen that these scantlings form a multi-cellular structural box, consisting of two cells forward of the bulkhead mentioned under (f) above, and of three cells aft of it. Below the box is the armament bay which consists, as previously mentioned, of subsidiary structure. The air intake ducts pass through the outer cells of the box. In the middle space are located:-

(a) The electronic equipment. This is all housed together in a separate crate which is attached to the structure by means of rubber mountings. For ease of maintenance the complete crate may be lowered through a large hole in the armament bay roof and through the armament bay and bottom skin to the ground by means of a self-contained electrically operated winch (Fig. 32).

(b) The pneumatic and hydraulic equipment: This may be reached through a manhole located in the bulkhead separating this compartment from the electronic crate compartment and another door located in the bulkhead separating this compartment from the center fuel cell compartment. There is, therefore, plenty of access to install and service this equipment.

(c) The center fuel bladder tank. This tank is installed and serviced through a manhole in the armament bay roof.

The outer fuel bladder tanks are located in the aft part of the two outer cells of the structural box and are draped over the air intake ducts. These are installed and serviced through manholes in the top surface of the fuselage which are normally covered by stress carrying, screwed on panels. The greater part of the vertical bending moment and shear in this section of the fuselage is taken by the two central beams, although some is taken by the fuselage sides. The horizontal bending moment and shear is taken by the armament bay roof and by the fuselage top skin. The torque on the fuselage is taken by the walls of the multi-cellular box in shear. The general construction follows conventional practice with skins, stringers, diaphragms, formers and longerons. The material will be 24ST light alloy with 75ST light alloy for extruded longerons. Skin formers are notched to allow the stringers to pass through, except at the front and rear transport bulkheads where the stringers terminate in fittings to pick up corresponding fittings on the other side of the bulkheads with transport joint bolts.

The fuel cell compartments cater to bladder-type cells, an inner skin being fitted over the stringers to provide a smooth interior. The majority of electrical equipment is housed in the armament bay where it is suspended from the armament bay roof and is accessible through the fuselage bottom skin.

3.7.1.2.4 Powerplant Section Superstructure: This assembly supports the engines and afterburners. To it are attached also, by means of transport joints, the powerplant doors, the rear nacelle superstructure and the lower rear nacelle. This assembly is suspended, by means of transport bolts, from the inner wing with which it forms an integral structural unit when bolted tight. The assembly also attaches to the transport joint bulkhead of the armament and equipment section of the fuselage by means of transport bolts. The main structural scantlings consist of a very rugged center keel beam extending from the bottom surface of the wing to the bottom of the fuselage and two robust fuselage side beams. The latter extend from the wing bottom surface to approximately half-way down the fuselage sides and incorporate the hinges from which the powerplant doors are suspended. These three beams are stabilized by a large number of closely spaced transverse formers. Attached to the inside flanges of these formers are the engine and afterburner shrouds which are constructed of thin stainless steel sheet aft of the engine compressor outlet and of aluminum alloy sheet forward of this station. Between the top of this steel shroud and the top of the formers (wing bottom surface) there is another stainless steel shroud which is intercostal between formers and riveted to them. These latter shroud panels are covered with 'Alfol'. The reason for this double shrouding is to isolate the hot sections of the powerplant from the wing fuel cells. The structural function of the three beams of this assembly is to transmit vertical shears, bending moments and torques from the front fuselage into the wing. The greater part of the shear and bending is transmitted by the center keel beam, whereas the side beams transmit the torque by virtue of differential shear and bending. Horizontal shears and bending moments from the front fuselage are transmitted into the wing via the wing front transverse spar and the wing top skin panel between the wing root ribs.

3.7.1.2.5 Powerplant Doors: Six doors are provided, three to each powerplant. They are hinged from the fuselage side beams of the superstructure, probably by means of continuous piano-hinges. The doors, when closed, register with the bottom longeron of the center keel beam by means of two dowels per door and are locked by a number of robust cowling fasteners of the quick disconnecting toggle type. The door structure is conventional, incorporating formers, stringers and boundary angles. These doors do not form a part of the main structural scantlings of the airplane.

3.7.1.2.6 Rear Nacelle Superstructure: This component forms a continuation of the powerplant section superstructure to which it is attached by means of transport bolts. It is also bolted to the trailing edge portion of the wing. Wells, used to house the upper speed brakes when these are retracted, are also incorporated in the structure. The main structural function of the nacelle is to transmit the tail skid loads, in a tail down landing, to the center keel beam.

3.7.1.2.7 Lower Rear Nacelles: These two components each contain a lower speed brake complete with operating mechanism and jacks. The nacelles are normally attached to the rear nacelle superstructure by means of a number of registering dowels and robust cowling fasteners of the quick disconnecting toggle type. Four links also attach each of these components to the afterburner which it encloses. For purposes of routine servicing, a complete engine-afterburner installation may be lowered approximately two feet on hydraulic jacks. To accomplish this, the relevant lower rear nacelle will be quickly disconnected from the rear nacelle superstructure so that it is suspended only by the links from its afterburner. The powerplant may then be lowered the requisite amount through the open engine doors with the lower rear nacelle attached to it. For complete removal of an afterburner, its lower rear nacelle may be disconnected from it by detaching the links which connect these two components. The structure of these components is conventional light alloy construction suitably stressed to cater to the speed brake loads.

3.7.1.3 Crew Station: The crew consists of one pilot, seated in an ejector-type seat, accommodated in the cockpit section of the fuselage. If required by the RCAF, an ejection capsule, of the type now being developed by the U.S. Navy, could, in all probability, be fitted at a sacrifice of weight. In the three view drawing in this brochure (Fig. 1) such a capsule has been indicated but in the accompanying weight estimate the ordinary type of ejection seat has been used for the calculations. The pilot will be provided with the normal flying and engine controls, a radar scope, some flight instruments, the usual engine instruments, and switches, instruments etc. for the integrated avionic MX 1179 equipment. The cockpit will be pressurized to a maximum differential of 3.5 lb. per sq. in. and will be temperature controlled. The pilot's position and attitude is in accordance with the requirements laid down in the Handbook for Aircraft Designers - AMC-80. Special attention has been given to the pilot's forward view over the nose; the angle between the pilot's line of sight over the nose and the wing chord is 17 degrees. Furthermore, in order to provide the pilot with the maximum possible view sideways and downwards, the air intakes are positioned just aft of the pilot's eye station. The jettisonable pilot's canopy is power operated and hinges upwards and rearwards to provide access to the cockpit.

3.7.1.4 Cargo Compartment: Cargo compartments are not fitted in this airplane.

3.7.1.5 Equipment Compartments: These have already been described in the preceding paragraphs dealing with the fuselage. Reference may further by made to the equipment illustrations in this brochure and the paragraphs dealing with equipment.

3.7.1.6 Speed Brakes: Four separate speed brakes are fitted and comprise two flaps mounted on top of the rear fuselage (termed upper speed brakes) and two flaps mounted on the bottom of the rear fuselage (the lower speed brakes).

3.7.1.6.1 Upper Speed Brakes: These components are mounted flush with the fuselage contour and each are actuated by two hydraulic jacks, which rotate the flaps into the air flow about two hinges attached to a trailing edge structure cantilevered aft from the main inner wing sparbox. The structure of the flaps consists of a conventional reinforced skin panel made of light alloy material.

3.7.1.6.2 Lower Speed Brakes: These components are mounted flush with the fuselage contour and are actuated by hydraulic jacks which rotate the flaps into the airflow about hinges attached to the lower rear nacelle structure. These brakes and their actuating mechanisms are completely self-contained and form part of the lower rear nacelles.

3.8 Alighting Gear:

3.8.1 General Description and Components: Refer to Fig. 25. The alighting gear is the conventional type of tricycle undercarriage with a retractable tail-skid to cater to the high angle of incidence of the airplane when landing. The nose undercarriage retracts forward into a compartment below the pilot's floor. The main undercarriage folds sideways and forwards into a compartment inboard of its pivot-axis inside the wing. The tail skid pivots from the center keel beam of the rear fuselage and, when retracted, lies flush in the rear fairing between the afterburners. The main gear consists of a two-wheel bogie, which, when retracted, can not be completely enclosed within the airfoil contour of the wing. This fact necessitates a shallow blister on the bottom surface of the wing which, however, is tolerable from aerodynamic considerations. The main wheels are positioned relative to the center of gravity of the airplane so that a line drawn from the aft c.g. limit of the airplane (31% m.a.c.), and normal to the tail down static ground line, passes through the center of the bogie chassis pivot axle; this line makes an angle of 18 degrees with a line drawn normal to the wing chord. The angle between the wing chord and the static tail up ground line is 5 degrees. By this arrangement, instability of the airplane in pitch, during take-off, is obviated.

3.8.2 Main Landing Gear:

3.8.2.1 Description: This gear consists essentially of a two-wheel bogie. Retraction is effected about an inclined pivot axle, the motion of the gear being inboard and forward, so that the wheels in their retracted position are considerably ahead of their extended position. Due to the inclination of the pivot axle in plan view, it is necessary to rotate the bogie chassis about the leg center line during retraction, by about forty-five degrees. In addition, it is necessary to tilt the bogie chassis about its pivot axle, in order to accommodate the gear in the smallest possible compartment adjacent to the fuselage side. The main leg casing will be a large aluminum alloy forging. The front main pivot bearing is forged integral with the leg. All torque on the leg will be transmitted by this bearing into the wing structure. The rear main pivot bearing is separate and joined to the main leg by a forged link. These bearings are pinned to the pivot shaft, which rotates in forged bearing fittings bolted to the wing structure. The whole gear will be made quickly detachable from the airplane. For this reason, the main pivot shaft is designed so that it can be extracted through a detachable portion of the wing leading edge. The bogie chassis will be manufactured from a light alloy casting and some of the major fittings on the gear will be made from high grade steel.

It is proposed to use needle bearings wherever possible; this is in accordance with best contemporary practice and results in lower friction losses during retraction and hence smaller hydraulic jacks.

3.8.2.2 Wheel Brakes and Brake Control System: The main wheels will be fitted with brakes which will be operated hydraulically by means of links connected to the rudder pedals. These pedal links are connected to hydraulic valves which meter the hydraulic fluid to the brakes. A hand operated parking brake will be installed in the cockpit. The brake drums of the front and rear wheel of the bogie are interconnected by a link, so that the braking torque from the rear wheel is transmitted to the front drum. The total braking torque is then transmitted from the front drum to the under-carriage leg, via another link and the bogie chassis.

3.8.2.3 Tires and Tubes: The tire size will be 29 x 7.7, in accordance with USAF drawing SF 51 F 601 type VII - military aircraft. The air pressure will be 220 lb. per sq. in.

3.8.2.4 Shock Absorbers: The main shock absorber, which will be of the liquid spring type, is housed inside the leg casing. Total travel is of the order of eleven inches. In addition to the main shock absorber, there is a small oleo-pneumatic bogie chassis damper strut. This damper strut connects the bogie chassis to the main leg and serves the following purpose:-

(a) To damp out oscillations of the bogie due to sudden load transference between the front and rear wheels during wheel spin-up and braking while landing the airplane.

(b) To act as a spring to position the bogie in its correct touch-down attitude prior to landing.

(c) To act as a subsidiary to the main shock absorber during the early part of a touch-down.

The damper strut will be designed so as to completely eliminate the usual high drag and anti-drag design cases. This may be accomplished as follows. The damper strut spring will position the bogie - chassis so that the rear wheel will hit the ground first. The characteristics of the damper will be made such that spin-up of this rear wheel is accomplished completely before the front wheel hits the ground, with its subsequent spin-up drag load. This characteristic along with the fact that the wheel inertias, resisting spin-up, are small due to the relatively small wheels, results in a very low drag load on the main gear. In fact, the bogie arrangement acts so as to spread out, over a longer period of time, the initial shock at landing and wheel spin-up. Further-more, the usually critical spring back condition, causing high anti-drag forces on a landing gear, is completely eliminated by this mechanism.

3.8.2.5 Retracting, Extending and Locking Systems: It is proposed to use a hydraulic jack operating on a folding sidestay and also on the upper end of the undercarriage leg, with the jack extending during the retraction cycle. This gives an efficient system with the smallest jack diameter. The folding sidestay supports the undercarriage leg in its extended position against side and drag loads. The downlock will be incorporated in the knuckle of the folding sidestay and will be operated through a 'lost motion' device from the jack. In the retracted position, it is proposed that the landing gear be supported by the wheel doors, which will themselves be positively locked in the UP position to the wing structure. Closely associated with the retraction mechanism is the means for rotating the bogie chassis about the vertical shock absorber axis and tipping it about its pivot axle. Rotation of the bogie chassis is accomplished by a torque sleeve situated around the lower portion of the main leg. This torque sleeve is connected by a connecting rod, with universal end-bearings, to a fixed point on the wing structure, slightly offset from the main gear pivot axle. The torque sleeve is provided with a profiled cam slot, similar to those in a propellor pitch change barrel. The slot engages with a roller which is fixed to the leg proper. To the lower end of the torque sleeve is rigidly attached a torque-locking collar, which in turn connects to the bogie chassis via conventional torque scissor links. This collar engages, during the DOWN position of the gear, with the lower end of the leg proper by means of large dogs. These dogs locate and lock the bogie chassis in the correct fore-and-aft position for landing and take off. The mechansim of retraction is therefore as follows:-

(a) As the gear is folding sideways, the connecting rod causes the torque sleeve to move down the leg casing, thus disengaging the torque locking collar dogs from the leg proper.

(b) Due to the profiled cam slot in the torque sleeve, further downward movement of this sleeve is accompanied by a rotary movement about the leg which in turn, is transmitted through the torque scissor links to the bogie chassis.

(c) The rotary movement of the bogie chassis is accompanied by a tilting of the bogie about its pivot axle. This is achieved by a cable system attached to the folding sidestay. The movement of the sidestay pulls the bogie chassis about its pivot axle against the resistance of the air-spring which is incorporated in the bogie damper strut. The cable load has been calculated to be only of the order of 350 lb.

It should be noted that the rotating feature of the bogie chassis requires the whole shock absorbing element inside the leg proper to be capable of rotation. Emergency lowering in case of hydraulic failure will be effected by compressed air stored at 3000 lb. per sq. in. The cable is attached to the bogie chassis near to the rear wheel axle.

3.8.2.6 Doors and Fairings: The bogie chassis and wheels are covered by a large hydraulically operated door which hinges from the bottom surface of the wing close to the fuselage side. The hinge line will be parallel to the airflow. The outboard portion of the main gear leg and retracting mechanism is covered by a door which is linked to the main leg but which hinges about a line parallel to the airflow. A narrow fairing between these two doors is mounted rigidly on the main gear leg. Only this latter small door will be at an angle to the airflow when the landing gear is extended. It is proposed that, in the retracted position, the main gear be supported by the large door covering the bogie chassis. This door will be hooked to the wing structure and positively locked in the up position. The hydraulic jacks operating the door will be sequenced from the main gear retracting jack; this makes it possible to close this large door after the main gear is extended.

3.8.2.7 Inspection and Maintenance: Provision for this will be similar to that on the CF-100.

3.8.3 Auxiliary Landing Gear - (tail wheel): A tail wheel unit will not be fitted on this airplane. A retractable tail skid will be provided as described in paragraph 3.8.1.

3.8.4 Auxiliary Landing Gear - (nose wheel):

3.8.4.1 Description: This component will be of conventional design in every respect. The gear retracts forward and a side-by-side wheel arrangement is used. Axle travel for shock absorption will be approximately eleven inches and a liquid spring shock absorber will be used. A drag stay consisting of two links will be used; retraction will be by means of a hydraulic jack. The retraction pivot axle is supported by fittings attached to the transport bulkhead behind the cockpit. The design of the attachments of the gear to the airplane structure will be such as to permit rapid replacement of the complete unit. Emergency lowering of the gear in case of hydraulic failure will be effected by compressed air stored in a 3000 lb. per sq. in. air bottle.

3.8.4.2 Wheel Brakes: These will not be fitted on the nose wheel.

3.8.4.3 Tires and Tubes: The tire size will be 18 x 5.5, type VII - military aircraft. The air pressure will be 170 lb. per sq. in.

3.8.4.4 Shock Absorber: This will be of the liquid spring type, manufactured by Dowty Equipment Limited.

3.8.4.5 Retracting, Extending and Locking Systems: These will be similar to the CF-100 systems.

3.8.4.6 Doors and Fairings: Two doors will cover the well into which the nose gear retracts.

3.8.4.7 Steering Control: The nose wheel will be steerable by hydraulic means.

3.8.4.8 Inspection and Maintenance: Provision for this will be similar to that on the CF-100.

3.8.5 Auxiliary Landing Gear: A retractable tail skid will be fitted as described in sub-paragraph 3.8.1.

3.9 Alighting Gear (water type): Not applicable.

3.10 Surface Control System:

3.10.1 Primary Flight Control Systems: The primary flight control surfaces comprise:-

(a) An elevator, mounted on the inboard trailing edge of each wing

(b) An aileron, located outboard of the elevator on each wing, and

(c) A rudder.

These installations are illustrated in Fig. 25. All three surfaces are actuated by irreversible hydraulic jacks which are connected to the related surfaces at multiple points through a combined push rod/bellcrank linkage (refer to Fig. 25). By using piano-type hinges on the surfaces, the whole mechanism can be housed internally. This method of driving the surfaces reduces very greatly the stiffness required from them and, in consequence, serves to lighten them. Mass balance is not required and the amount of twist under load is negligible. This internal system is not applicable to airplanes incorporating elevon control, due to the high tip loads and can not be successfully used on very small airplanes owing to space restrictions. Two pistons are fitted to each hydraulic jack. These are actuated by the separate hydraulic systems as outlined in sub-paragraph 3.15.1.2.1. The necessary control valves are accommodated in the fuselage and are normally operated by electric actuators controlled by the auto-pilot. A straight-run push rod, however, leads from the valves to the cockpit, to link up with the pilot's conventional controls. Artificial 'feel' for the pilot is obtained by a suitable spring system which can be biased to give trim conditions (refer to sub-paragraph 3.10.3). A bob weight is provided in the longitudinal control circuit in order to sense 'g' applications. In general, the system will be similar to that used on the F86E airplane.

3.10.2 Secondary Flight Control Systems:

3.10.2.1 Lift and Drag Increasing Devices: Lift increasing flaps are ineffective on this type of wing planform and, in consequence, are not fitted. Speed brakes (refer to the following sub-paragraph) will not be used for landing purposes since the lower one will foul the ground in the tail down attitude. Drag at landing speed is greatly increased by the release and subsequent streaming of a tail parachute. This item is stored in a housing (covered by a quick-release end cap) immediately below the rudder

(Fig. 25). The method used for parachute ejection is similar to that required for anti-spin purposes. Successful development of this equipment has been reached on the AVRO 707 series of airplanes. Initially, the ejection will be manually controlled from the cockpit; later it is intended that the parachute release is tied in with the automatic landing system. The parachute diameter will be 16 feet.

3.10.2.2 Speed Brakes: Upper and lower speed brakes are fitted at the rear end of the fuselage as illustrated in Fig. 25, and are disposed about the vertical center of gravity in order to give no change of lift or pitching moment when opened. Additional adjustment can be obtained by differential settings. Adequate ventilation between the brake flaps and the adjacent surface is provided to ensure freedom from buffetting at all speeds. The location of the brakes is such that a minimum of interference with the primary flight control surfaces is obtained. Hydraulic jacks, limited in capacity so that the flaps will blow down when a fore-and-aft acceleration of 1'g' is exceeded, are used to operate the brakes. The maximum time from brakes closed to opened will not exceed three seconds.

3.10.3 Trim Control System: Trim is effected by adjusting the centering position of the spring system which gives artificial feel to the primary flight controls. Tabs are not fitted since these would be ineffective with irreversible controls.

3.10.4 Automatic Pilot: This airplane is designed to be operated almost entirely under automatic control. For this reason the auto-pilot must be integrated with the automatic navigation, fire control and landing systems, while still exercising those functions normally associated with an auto-pilot. Elaborating on this requirement, provision will be made for:-

(a) Damping of at least 60% critical, about all three axes for both short and long period oscillations.

(b) Turns to be co-ordinated so as to eliminate sideslip irrespective of drag asymmetry.

(c) The response of the system will be such as to give a smoothing time with respect to a steering signal of 0.2 seconds for purposes of fire control. Steering signals from all sources are computed relative to a gyro-stabilized platform which uses the earth's gravitational and magnetic fields as primary references in the normal manner. The method of deriving the automatic steering signals is referred to in sub-paragraph 3.18.

Suitably amplified voltages are then fed to the servo motors which are located in the proximity of the hydraulic valves serving the surface controls. For the location of the valves refer to sub-paragraph 3.10.1. The servo motors are just powerful enough to operate the hydraulic control valves and, accordingly, can be over-ridden easily by the pilot at any time. The force necessary to accomplish this operation will not exceed 5 lb. It will also be possible to disconnect any or all of the steering channels from the auto-pilot without affecting automatic stabilization and damping. It should be noted that although it will be possible to fly the airplane with the auto-pilot completely disconnected, the synthetic damping will normally be retained even if all the other functions are rendered inoperative. As far as possible, auto-pilot failure will be arranged to exclude possible flight hazards and a system to cater for checks on the circuits will be incorporated in the basic design.

NOTE: As the electronic part of the auto-pilot is integral with the MX 1179 system, the equipment is stored in the crate provided in the fuselage for electronic apparatus.

3.11 Engine Section:

3.11.1 Description and Components: Refer to Fig. 26. The engines and afterburners are housed in the powerplant section of the fuselage as has been described in sub-paragraphs 3.7.1.2.4, 3.7.1.2.5 and 3.7.1.2.7. Refer also to sub-paragraph 3.12.3.

3.11.2 Construction: Refer to the references mentioned in the preceding sub-paragraphs.

3.11.3 Engine Mounts: The engines are supported on two trunnions located near their center of gravity on either side of the engine center line and by front steady supports. The afterburners are supported on two trunnions located near their center of gravity on either side of the center line and further by the engine exhaust casings (to which the afterburners are attached so that shear loads only can be transmitted). The design of these mountings is such as to allow the engines and afterburners to expand laterally and longitudinally.

3.11.4 Vibration Isolators: These shall not be fitted.

3.11.5 Firewalls: Each complete engine and afterburner is housed in a tunnel shaped shroud, of which the upper half is attached to the fuselage superstructure and the lower half to the powerplant access doors. These shrouds shall be of aluminum alloy sheet over the engine compressors and of stainless steel sheet over the engine combustion chambers and afterburners. A firewall shall isolate the compressor compartment from the combustion chamber compartment; this firewall shall be of stainless steel sheet. An additional shroud is fitted below that part of the wing which contains wing fuel bladder tanks (refer to sub-paragraph 3.7.1.2.4).

3.11.6 Cowling and Cowling Flaps: These are not applicable to this airplane.

3.11.7 Access for Inspection and Maintenance: Refer to Fig. 26 which shows the engine and afterburner installation and to the description under sub-paragraph 3.12.3.2.

3.12 Propulsion:

3.12.1 General Description and Components: The airplane is propelled by two turbo-jet engines, each with an afterburner. The complete installation is mounted inside the rear of the fuselage as shown in Fig. 26.

3.12.2 Main Propulsion Unit: Any one of the following turbo-jet engines may be installed:-

(a) The AVRO Canada TR9

(b) The Curtiss Wright J67

(c) The Bristol Olympus OL3 (fighter version)

Afterburners will be fitted to whichever engines are used. All these engines will give approximately the same performance. As the engine sizes and weights are very similar, the installation details will not differ greatly. Maximum sea level static thrust of each engine will be 15,000 lb. in their fully developed state, although the rating of initial engines may only be from 13,000 - 14,000 lb. Of the engines proposed, the Olympus is already in an advanced stage of development, a slightly reduced version now giving 10,600 lb. thrust with a specific fuel consumption of 0.78 lb/lb/(thrust)/hour. A larger version of this engine (OL3 - bomber version) is scheduled to run in the middle of 1953. Accordingly, it is reasonable to expect that engines of this type and specification will be available by 1955. The Olympus - and its Curtiss Wright variant - have two compressors and turbines mounted on separate concentric shafts but working in series. This permits a very low fuel consumption, obtained at the expense of some mechanical complication. The AVRO TR9, on the other hand, is a conventional engine having only one shaft. This gives a slightly simpler and cheaper engine but with a higher fuel consumption.

3.12.2.1 The missions for which this airplane is primarily designed are accomplished using supersonic speeds. These are obtained with the afterburner lit for the greater part of the time. Under these conditions, a reduction in the fuel consumption of the main engine does not result in any appreciable change in the overall picture. The range, however, would be increased somewhat for subsonic missions if an Olympus type engine were used. All engines will be suitable for use at supersonic speeds and are stressed for fighter load factors.

3.12.2.2 Afterburners: The afterburner will be designed to use a temperature of 1,800 deg. K. It will be fitted with an adjustable nozzle which is infinitely variable in cross section between the required limits. The control system will be so devised that the afterburner can be operated with a maximum of efficiency down to not more than 80% of the maximum permissible engine r.p.m. The nozzle operating mechanism will be constructed so that the external lines are sufficiently well faired, for all nozzle openings, in order to achieve a negligible base drag.

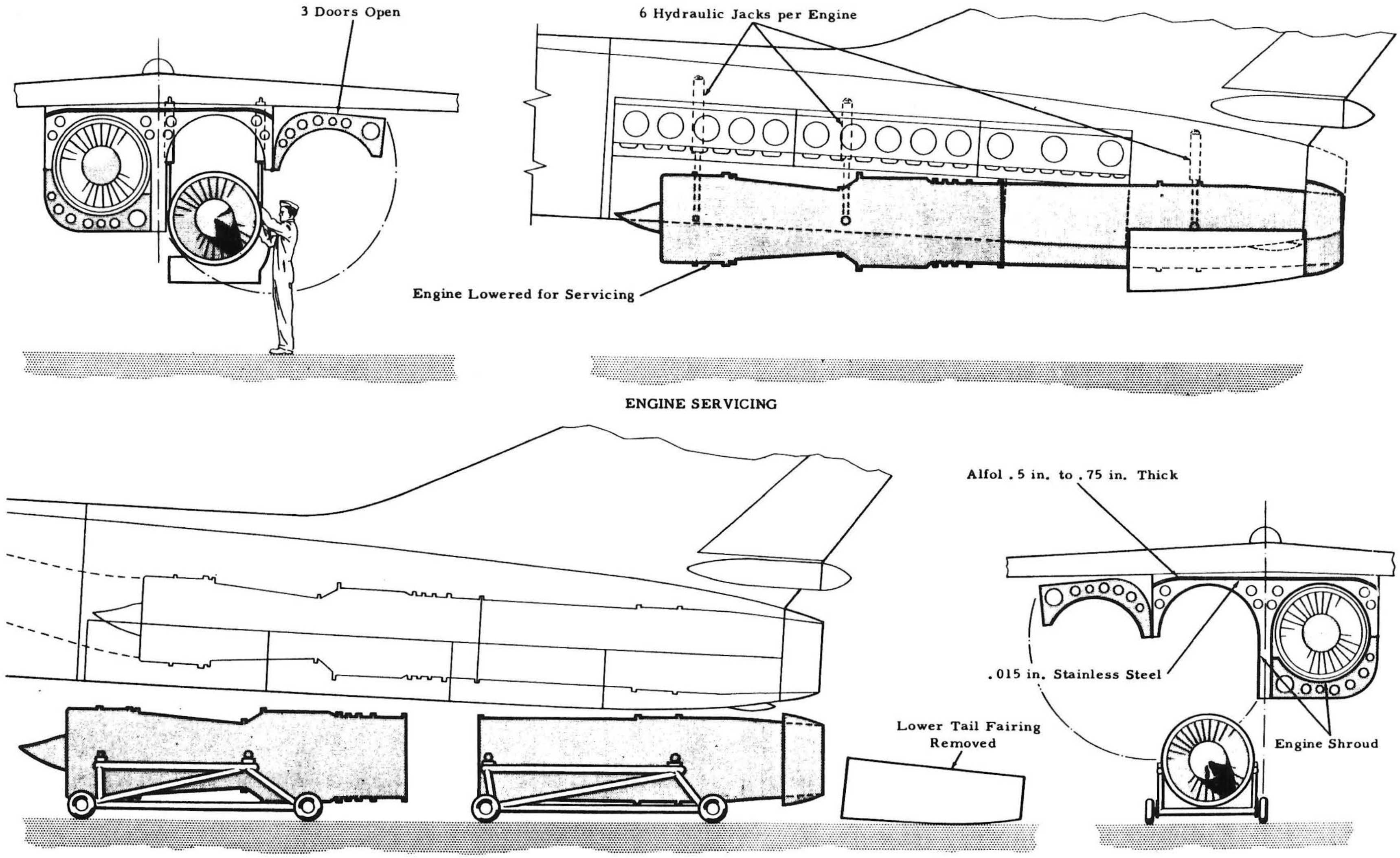
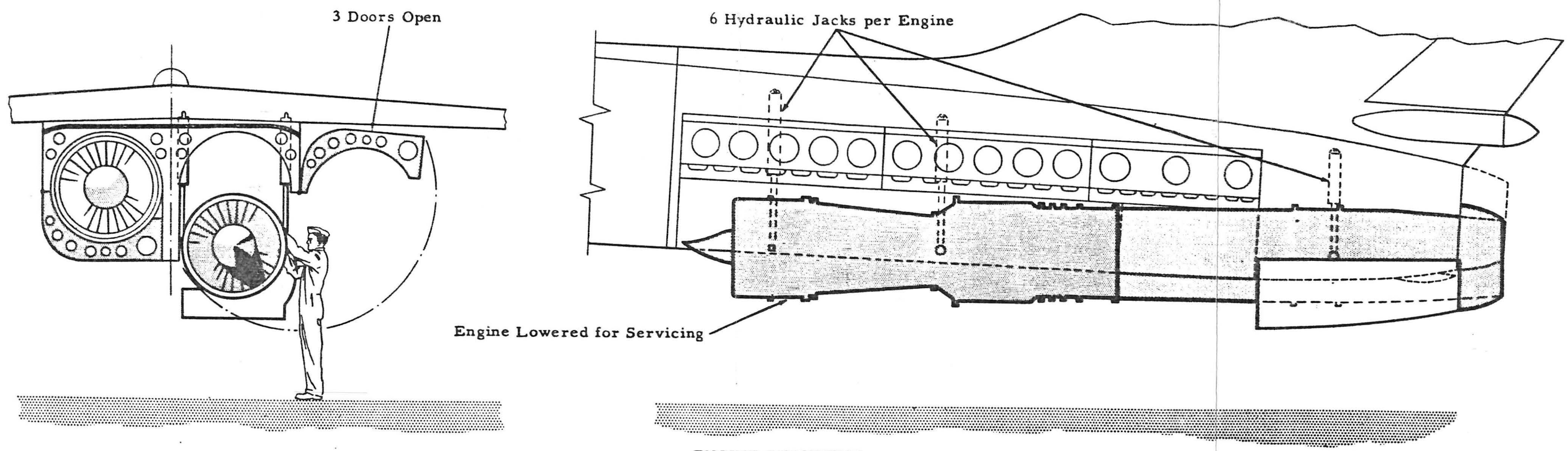
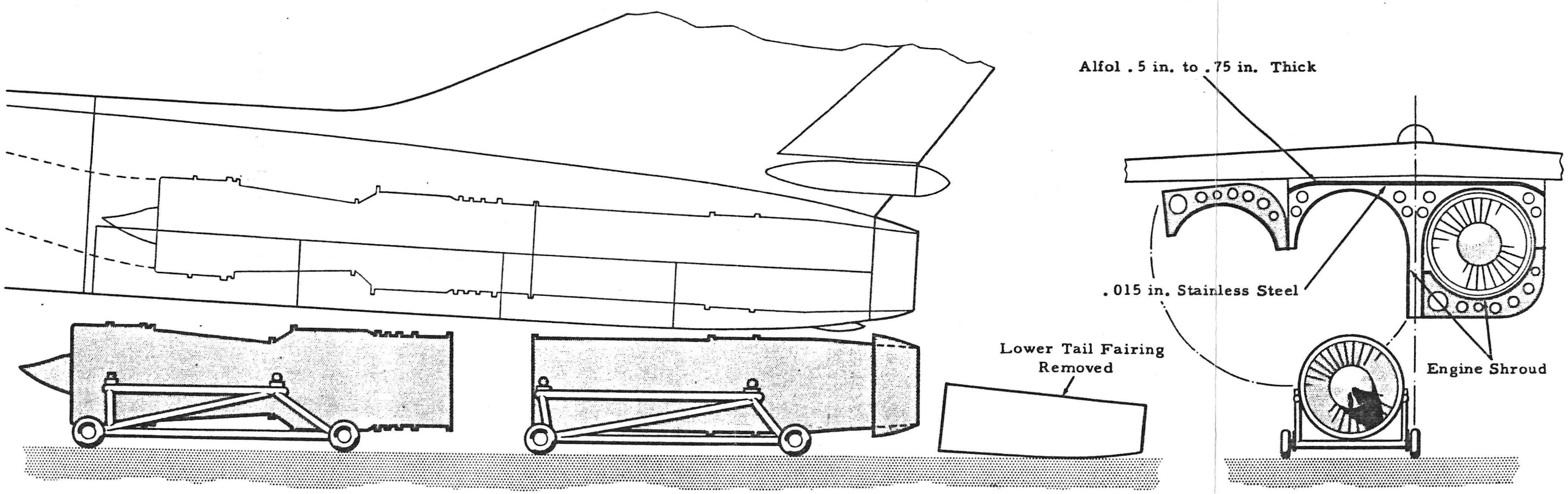


FIG. 26 ENGINE AND AFTERBURNER INSTALLATION



ENGINE SERVICING



ENGINE REMOVAL

FIG. 26 ENGINE AND AFTERBURNER INSTALLATION

3.12.3 Mounting, Access and Removal:

3.12.3.1 Mounting: Refer to Fig. 26. The engines and afterburners are slung from a central and two outer keel members which run underneath the wing. The afterburner is supported by the engine and by supports permitting expansion. All the mountings are designed so that they may be unclamped and the load then transmitted to six small hydraulic jacks which are used to lower the power unit a distance of approximately two feet.

3.12.3.2 Access Provisions: Access to an engine is obtained by opening the three large doors underneath the engine as illustrated in Fig. 26. If further access is required it will be necessary to:-

(a) Remove the engine starter access panel

(b) Uncouple the lower rear nacelle from the fuselage, and

(c) Release the main trunnions, the front steady support and the after-burner support.

This will allow the engine to be supported by the hydraulic jacks. The complete power-plant may then be lowered for a distance of approximately two feet, as described in the preceding sub-paragraph, without disconnecting any of the engine connections except the throttle. When this has been accomplished, almost perfect access is secured for all routine maintenance.

3.12.3.3 Removal: When removing an engine, it will be necessary to uncouple all fuel, air and electrical connections to both the engine and afterburner. After the weight of the engine has been taken by a suitable hoist, it is then possible to uncouple the hydraulic jacks. The complete installation may then be lowered onto a trolley as shown in Fig. 26.

3.12.4 Engine Driven Accessories: A hydraulic pump is mounted on an accessory pad on each engine. All other accessories which are not integral with the engine are driven by air bled from the compressor as described in sub-paragraph 3.15.2. (pneumatic system).

3.12.5 Air Induction System:

3.12.5.1 Description and Components: The air induction system consists of the air intake ducts described below.

3.12.5.2 Air Intakes: The air enters through D-shaped ducts located one on each side of the fuselage just aft of the cockpit. From these points it is led to the engines by separate ducts which become circular in section as they near the engines. The aero-dynamic features of this system are discussed in sub-paragraph 3.3.1.4.

3.12.5.3 Ice Protection System: Compressor entry de-icing equipment will be provided by the engine manufacturer as an integral part of the engine. The lips of the air intake ducts will be de-iced by passing hot air, bled from the engine compressor, through the annular space between the inside and outside skins.

3.12.5.4 Dust Protection System: Normal type debris screens will be fitted in front of the compressor entry. These screens will be made retractable when the airplane has obtained sufficient altitude or when icing conditions prevail.

3.12.6 Exhaust System: This system is not applicable in this airplane.

3.12.7 Cooling System: An annulus around the propelling nozzle will be so arranged that it acts as an ejector and sucks a satisfactory volume of cooling air through a shroud enclosing the afterburner. This component is designed as an integral part of the afterburner. The air entering this shroud will be secured from the engine compartment and will be used for cooling the engine.

3.12.8 Lubricating System: A self-contained lubricating system will be provided as an integral part of the engine.

3.12.9 Fuel System:

3.12.9.1 Description and Components: Refer to Fig. 27. The total internal fuel capacity is 16,000 pounds of usable kerosene which is contained in a number of tanks situated in the wing and in the fuselage. All tanks are of the 'bladder-cell' type. The fuel system will supply fuel to the engines and the afterburners under all flight conditions including inverted flight. Four immersed-type booster pumps, which are located in the fuselage tanks forward of the engines, supply the engines and afterburners. Normally, the pair of pumps located in the front tank supplies the right engine and afterburner, while the pair of pumps located in the rear tank supplies the left engine and afterburner. These front and rear fuselage tanks in which the booster pumps are located are not interconnected. The wing fuel is fed into these fuselage tanks by transfer pumps, the right wing pumps normally supplying the front fuselage tank and the left wing pumps supplying the rear fuselage tank. In the event of failure of any or all of the booster pumps it is possible for the pilot to:-

(a) Direct fuel from the left wing transfer pumps to the left engine or to both engines, and

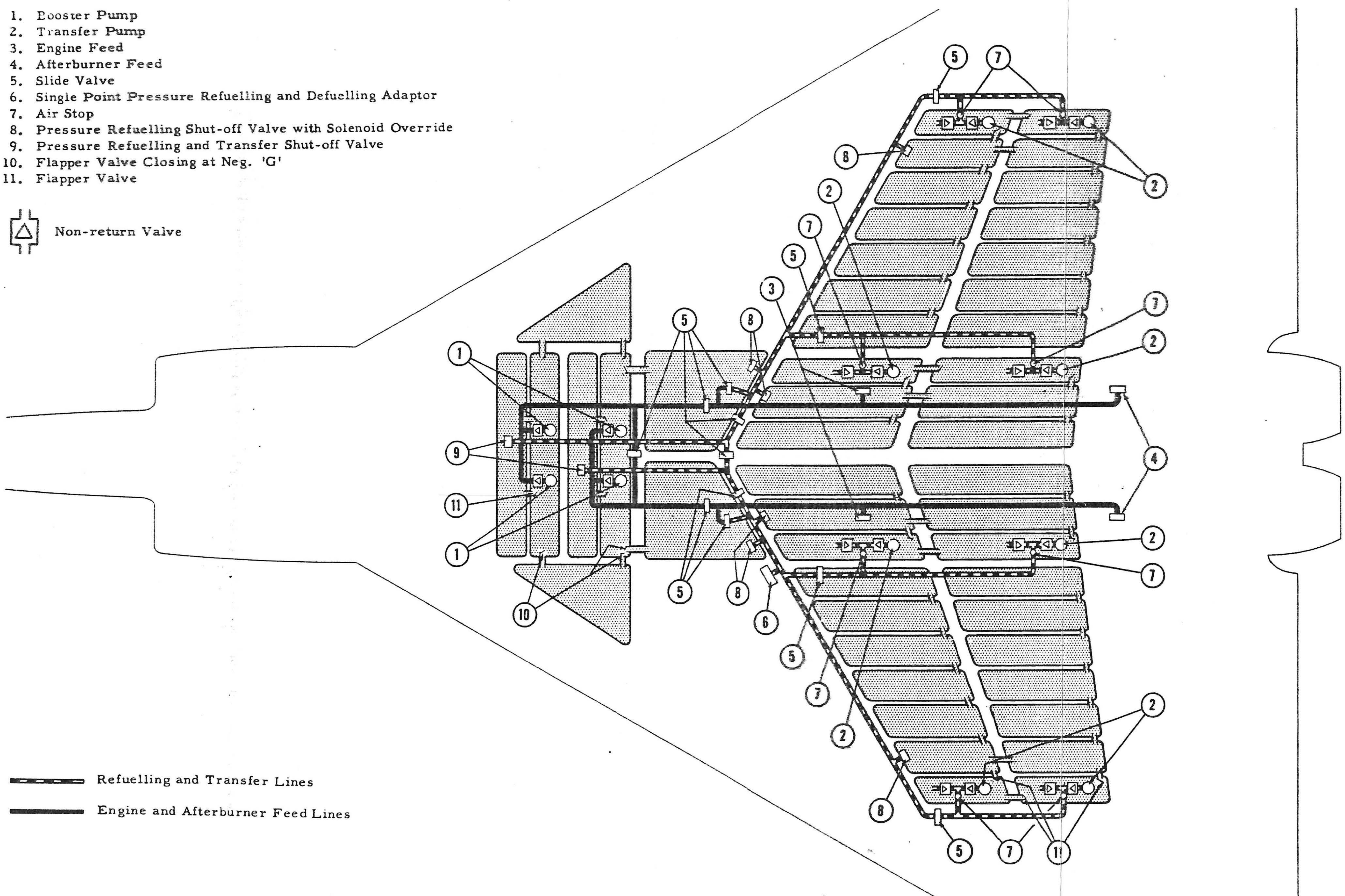
(b) Alternatively to direct fuel from the right wing transfer pumps to the right engine or to both engines.

In the case of failure of an engine, it is possible for the pilot to direct the whole of the fuel supply for the defective engine to the serviceable engine.

3.12.9.2 Pumps: It is proposed to drive the four booster pumps, which are located in the fuselage, by air turbines fed from the pneumatic system. The reason for adop-

1. Booster Pump
2. Transfer Pump
3. Engine Feed
4. Afterburner Feed
5. Slide Valve
6. Single Point Pressure Refuelling and Defuelling Adaptor
7. Air Stop
8. Pressure Refuelling Shut-off Valve with Solenoid Override
9. Pressure Refuelling and Transfer Shut-off Valve
10. Flapper Valve Closing at Neg. 'G'
11. Flapper Valve

Non-return Valve



Refuelling and Transfer Lines

Engine and Afterburner Feed Lines

FIG. 27 FUEL SYSTEM

ting pneumatic power to drive these large pumps instead of electric power is that the electric power source (two alternators) is energized by the pneumatic system which uses air fed from the main engine compressor. Hence, it is obviously more efficient, and much lighter, to drive the pumps directly from the primary source of power. There are eight transfer pumps which transfer the fuel in the wing tanks to the fuselage tanks in normal circumstances, or directly to the engines in an emergency. These transfer pumps are driven by immersed a.c. electric motors. The reason for using four transfer pumps in each wing is discussed in the following sub-paragraph. Because of the anhedral angle of the wing, the pumps are placed well outboard in each battery of fuel cells.

3.12.9.3 Tanks (fixed): The number of cells and their respective capacities are as follows:-

Forty-four bladder cell wing tanks	11,800 lb.
Four bladder cell fuselage tanks	4,200 lb.
TOTAL:	16,000 lb.

Of the twenty-two cells in each wing, two empty directly by means of gravity feed into the fuselage tanks. The other twenty are split up into an inboard battery of six cells and an outboard battery of fourteen cells, each with its own set of two transfer pumps. This arrangement prevents excessive fuel pressures in the outboard cells when the airplane rolls. The bladder cells will be of conventional construction similar to the ones installed in the CF-100.

3.12.9.4 Tanks (droppable): These will not be carried on this airplane.

3.12.9.5 Vent System: An adequate fuel venting system will be incorporated.

3.12.9.6 Piping and Fittings: Refer to Fig. 27 which shows the proposed piping layout.

3.12.9.7 Valves: Refer to Fig. 27 which shows the main valves required in the fuel system.

3.12.9.8 Strainers and Filters: An adequate number of these will be incorporated in the system.

3.12.9.9 Quantity Gages and Flowmeters: An adequate number of these will be incorporated in the system.

3.12.9.10 Drainage Provisions: These will be dealt with adequately in the design of the system

3.12.9.11 Fuel Vapor Inertion: Whether an inertion system will be incorporated depends on RCAF requirements in this respect. Refer also to sub-paragraph 3.18.6 on Passive Defence.

3.12.9.12 Fuel Evaporation Control: It is proposed to pressurize the fuel tanks to 1 lb. per sq. in. differential pressure, assuming JP-4 fuel. This pressure differential is also required to prevent the bladder cells from collapsing when empty.

3.12.9.13 Refuelling Provisions: Ground refuelling will be carried out from a single point pressure refuelling adaptor. Float valves will be provided in the tanks to automatically shut off the fuel when a tank is filled to capacity. Provision for in-flight refuelling is not incorporated in this airplane.

3.12.9.14 Defuelling Provisions: Ground defuelling may be carried out from the same single point adaptor as used for refuelling.

3.12.10 Water Injection System: This is not fitted on this airplane.

3.12.11 Propulsion System Controls: This is dependent on the design of the engines and afterburners on which no final information is available at present.

3.12.12 Starting System:

3.12.12.1 Description and Components: It is proposed to use air turbine starter motors located in the nose-bullets of the engines, which will be energized by either the auxiliary powerplant (gas turbine compressor) carried in the airplane or alternatively, by an external ground compressor unit which plugs into the airplane pneumatic system.

3.12.13 Propellor: Not applicable.

3.12.14 Rocket Propulsion System: This is not required on this airplane.

3.13 Auxiliary Powerplant:

3.13.1 Description and Components: An auxiliary gas turbine compressor, which also supplies shaft power, will be carried in this airplane. The proposed powerplant is manufactured by Airesearch Ltd. Details of this powerplant are as follows:-

Model	GTCP65-1
Consumes	Air and Fuel
Delivers	Compressed air and shaft power
Size	7 cubic feet
Weight	180 pounds (dry)
Starter	Electrical 24-28 volt d.c.
Fire Detection and Suppression Equipment	Self-contained
Continuous Output	70 hp.
Maximum Output	84 hp.
Shaft Speed...............	6,000 r.p.m. (nominal)
Complete Metal Enclosure of Self-contained Unit	

This auxiliary powerplant is used to supply compressed air for the operation of pneumatic services plus an emergency hydraulic pump connected to its output shaft in the event of both main engines failing in flight. It is designed to provide adequate power to land the airplane at normal speeds. On the ground, this powerplant will supply pneumatic power and air to operate the following:

(a) The electrical system (in order to warm up the electronic equipment)

(b) The air conditioning system (in order to air condition the cockpit and the equipment and armament compartments)

(c) The hydraulic system (in order to operate and test the flying controls and other hydraulic services), and

(d) The main engine starter motors.

Starting of the auxiliary powerplant may be accomplished by using the airplane battery as power source. The air required for this powerplant is tapped from one of the main engine intake ducts.

3.13.2 Installation: This self-contained gas turbine compressor unit, together with its emergency hydraulic pump, is installed in the central compartment of the equipment and armament section of the fuselage. Access to the unit can be obtained through a manhole door in the armament bay roof.

3.14 Instruments and Navigational Equipment:

3.14.1 Instruments: The instruments which will be fitted in the cockpit comprise the following:-

(a) The normal engine and afterburner instruments, and

(b) Such flight instruments as are required to satisfactorily fly and land the airplane in case of failure of the electronic MX 1179 equipment.

3.14.2 Navigational Equipment: This will comprise the MX 1179 integrated electronic system as described elsewhere in this brochure.

3.14.3 Installation: The installation of instruments and equipment will comply with RCAF requirements and specifications. The electronic equipment is housed compactly in the equipment section of the fuselage inside a self-contained crate. This crate may be lowered out of the fuselage onto a ground-testing trolley, as has been described elsewhere in this brochure (see Fig. 32).

3.15 Hydraulic, Emergency Air and Pneumatic Systems:

3.15.1 Hydraulic and Emergency Air Systems:

3.15.1.1 Descriptions and Components: Refer to Fig. 28 and 29. A dual hydraulic system is used to actuate the power flying controls and aircraft services, each system being hydraulically independent of the other, with separate pump, accumulator, reservoir, etc. Each main engine drives one hydraulic pump through a power take-off shaft. The pumps are of the variable displacement type. A hydraulic fluid cooler is incorporated in each circuit to control the system temperature. One system, known as the utility circuit, actuates the airplane services and supplies half of the total requirements of the flying controls. The other system, identified as the flying control circuit, supplies the remaining half of the requirements of the flying controls. This latter system incorporates a third pump, normally inoperative, driven by the auxiliary powerplant and operated in the event of failure of one or both engines. Normal operating pressure of each system is 3000 lb. per sq. in. using fluid to Specification MIL-O-5606. Pressure lines are of annealed stainless steel and suction and return lines of aluminum alloy. Standard AN type tube fittings are used. On the ground, it is possible to operate each system by a hydraulic rig through conventional ground connections or, the flying control circuit only, by running the auxiliary powerplant.

3.15.1.2 Summary of Actuated Items:

3.15.1.2.1 Flying Control Circuits: The surfaces to be actuated are the elevators, the ailerons and the rudder. Each elevator is actuated by one jack and the left and right elevators are mechanically interconnected. The utility circuit supplies one elevator jack and the flying control circuit supplies the other elevator jack. Each aileron is actuated by one double piston jack, due to the fact that mechanical interconnection between the left and right ailerons is not possible. The utility circuit supplies one piston of the left aileron jack and one piston of the right aileron jack. The flying control circuit supplies the other pistons. The rudder is actuated by one double piston jack, each circuit supplying one piston. Fluid supply to the jacks is controlled by one dual sliding valve for each pair of elevators and ailerons and for the rudder. Each valve in the dual sliding valve body serves one of the two independent circuits. The design of the dual sliding valve body is such that in the event of failure of one of the two circuits, the fluid between each side of the jack pistons in the faulty circuit can freely by-pass the corresponding valve in this circuit. In this manner the other circuit can operate the remaining piston or jack without restriction. The actuation of the sliding valves is described in sub-paragraph 3.10.1.

3.15.1.2.2 Landing Gear Circuit: The landing gear circuit derives its power for normal operation from the utility system and from a high pressure air system for emergency use. On a DOWN signal from the cockpit, fluid is directed by a solenoid operated control valve through lines to the nose gear jack and the main undercarriage door jacks, via a shuttle valve in each case. A line branching off from the down-line to the doors connects to a dual sequence valve, one line out of this valve connecting

LEGEND

(1) Conditioned Air Inlet to Electronic Equipment Compartment (Cockpit Exhaust Air)
(2) Conditioned Air Inlet to Electronic Equipment Compartment (Alternator Turbine Exhaust Air)
(3) Air Intake of Auxiliary Power Unit - Gas Turbine
(4) Auxiliary Power Unit - Gas Turbine Compressor
(5) Air Turbine Motors Driving Alternators
(6) Heat Exchanger Cooling Air Inlet
(7) Air Delivery Duct from Heat Exchanger
(8) Air-to-Air Heat Exchanger
(9) Air Turbine Motor Driving Heat Exchanger Fan
(10) Heat Exchanger Cooling Air Outlet
(11) Hydraulic Reservoirs
(12) Hot Air Supply to Right Wing Leading Edge - Anti-icing
(13) Hot Air Supply to Fin Leading Edge - Anti-icing
(14) Hot Air Supply to Left Wing Leading Edge - Anti-icing
(15) Pressure Air Collector Ring - one on each Engine Compressor
(16) Main Hydraulic Pumps - one on each Engine Gearbox
(17) Air Turbine Starter Motors of Main Engines
(18) Air Outlet - Armament Compartment
(19) Air Duct to Air Turbine Starter Motors - from Auxiliary Power Unit
(20) Main Pressure Air Supply Duct - from Engine Compressors
(21) Hydraulic Accumulators
(22) Conditioned Air Duct to Armament Compartment
(23) Ground Charging and Test Unit for Hydraulic and Pneumatic Systems (in Armament Compartment)
(24) Emergency Air Bottles (3000 psi)
(25) Rectifiers (28 volt a.c. to 28 volt d.c.) - 2 off
(26) Alternator Circuit Breakers - 2 off
(27) Emergency Hydraulic Pump - Shaft Driven from Auxiliary Power Unit
(28) Emergency Air Supply Duct - from Auxiliary Power Unit
(29) Electrical Fuel Contents Units - 2 off
(30) Alternator Control Panels - 2 off
(31) Main Electrical Junction Box - left hand
(32) Exhaust of Auxiliary Power Unit - Gas Turbine
(33) Synchronizing Unit for Alternators
(34) Battery
(35) Electrical Distributor Box
(36) Electrical Relay Box
(37) Electrical Panel for Engine Starting
(38) Main Electrical Junction Box - right hand
(39) Conditioned Air Inlet to Cockpit

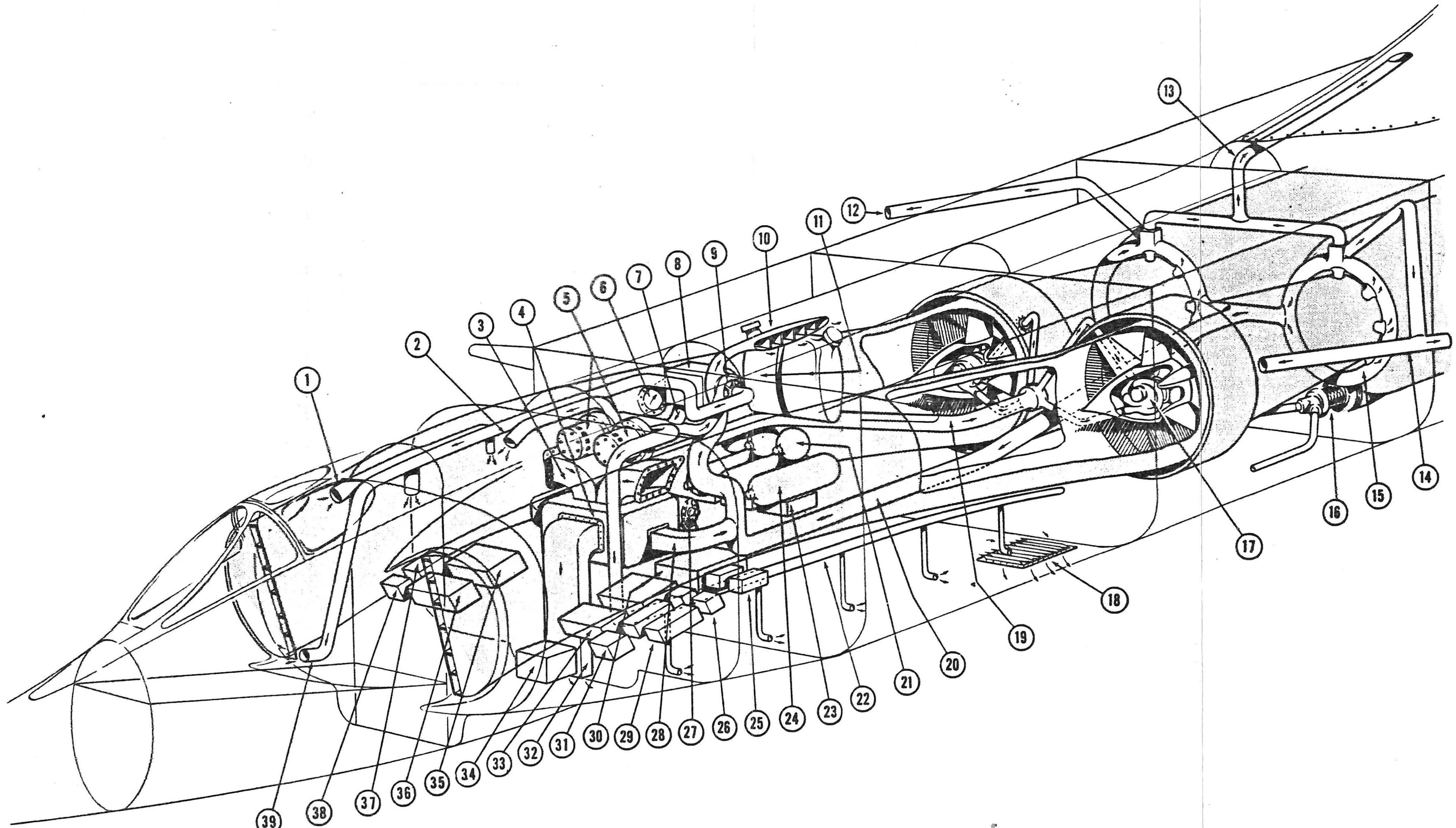


FIG. 28 ELECTRICAL, PNEUMATIC, AIR CONDITIONING AND HYDRAULIC EQUIPMENT

to the uplock jack and the other line to the main undercarriage jack. Thus, when the doors are open, the pressure builds up to actuate the uplock portion of the dual sequence valve, following which, the main jacks are actuated. The rate of extension of the undercarriage is controlled by a variable flow valve in the main up-line without restricting the rate of retraction. For UP operation, fluid is directed straight to the jacks. When the main gear is almost retracted, a mechanically operated valve is tripped, admitting fluid to the door jacks. This valve admits free flow in the opposite direction. For emergency DOWN operation of the landing gear, air is stored at 3000 lb. per sq. in. and is released by an emergency valve which admits this air to the shuttle valves, thereby closing the hydraulic port in these valves and, to the jacks operating doors, uplocks and retraction mechanisms. An airline branches off from the main supply to a jettison valve which permits the return of fluid in the up-side of the jacks to the hydraulic reservoir.

3.15.1.2.3 Speed Brakes Circuit: The speed brakes circuit consists of six hydraulically connected jacks, two for each upper flap and one for each lower flap. Operation is controlled by a solenoid-operated selector valve; a relief valve is incorporated in the system to protect the structure against overloads. The circuit is designed for rapid operation. Refer also to sub-paragraph 3.10.2.2.

3.15.1.2.4 Wheel Brake Circuit: The wheel brake circuit comprises a power circuit and a foot motor circuit. The foot motor circuit operates the power brake valve from the pilot's rudder pedals, each pedal incorporating one foot motor for left and right brakes operation. An independent circuit is provided for each foot motor. These terminate at an actuating cylinder at the power valve. The component has dual supply inlets, one directly from the utility system, the other from the same feed line via a non-return valve, with an accumulator fitted between the latter and the brake valve. Both feeds pass through a common pressure reducing valve to give a pressure of 1500 lb. per sq. in. The right and left foot motor systems each operate two identical units in the power valve, each unit being supplied by the feed line. These, in turn, supply pressure to the dual brake cylinder through separate hydraulic lines. Either unit on the brake valve will give full braking effort and the accumulator will supply enough pressure to actuate the brakes in the event of main supply failure. Parking is also accomplished by utilizing accumulator pressure. Brake pressure gauges are fitted in the main wheel wells.

3.15.1.2.5 Steering and Anti-shimmy Circuit: The steering cylinder is attached to the nose leg and is spring-centered, acting also as a shimmy damper. Steering is effected through a mechanical linkage attached between the rudder pedals, and a control valve and follow-up mechanism on the leg. A solenoid-operated stop valve is energized and supplies pressure to the control valve. This admits fluid to the desired side of the steering cylinder. A relief valve is fitted which admits fluid from one side of the cylinder to the other when a pressure build-up is caused through excessive turning loads.

LEGEND

WHEEL BRAKES

(1) Brake Signal Pumps
(2) Brake Relay Valve
(3) Pressure Gauge
(4) Brake Unit
(5) Air Charging Valve
(6) Accumulator
(7) Non-Return Valve

NOSE WHEEL STEERING

(8) Steering Motor
(9) Pressure Relief Valve
(10) Control Valve
(11) Restrictor
(12) Stop Valve

UNDERCARRIAGE

(13) Variable Flow Valve
(14) Main Undercarriage Jack
(15) Gauge and Charging Connection
(16) Air Bottle
(17) Emergency Air Release Valve
(18) Jettison Valve
(19) Dual Pressure Sequence Valve
(20) Non-Return Valve
(21) Mechanical Valve
(22) Undercarriage Door Jack
(23) Restrictor
(24) Nose Undercarriage Jack
(25) Shuttle Valve
(26) Main Shuttle Valve
(27) Up Locks
(28) Control Valve

UTILITY POWER SYSTEM

(29) Hydraulic Pressure Gauge
(30) Pressure Warning Switch
(31) Non-Return Valve
(32) Ground Test Connection - Suction
(33) Ground Test Connection - Pressure
(34) Pump Relief Valve
(35) Oil Cooler
(36) Reservoir
(37) Filter
(38) Hydraulic Pump
(39) Air Charging Valve
(40) Accumulator

FLYING CONTROL POWER SYSTEM

(41) Emergency Hydraulic Pump
(42) Hydraulic Pump
(43) Oil Cooler
(44) Reservoir
(45) Filter
(46) Non-Return Valve
(47) Pressure Warning Switch
(48) Ground Test Connection - Suction
(49) Ground Test Connection - Pressure
(50) Pump Relief Valve
(51) Hydraulic Pressure Gauge
(52) Air Charging Valve
(53) Accumulator

FLYING CONTROLS

(54) Elevator Follow-up Control Valve
(55) Elevator Jack
(56) Aileron Dual Follow-up Control Valve
(57) Aileron Jack
(58) Rudder Jack
(59) Rudder Dual Follow-up Control Valve

DIVE BRAKES

(60) Upper Dive Brake Jack
(61) Pressure Relief Valve
(62) Control Valve
(63) Lower Dive Brake Jack
(64) Non-Return Valve

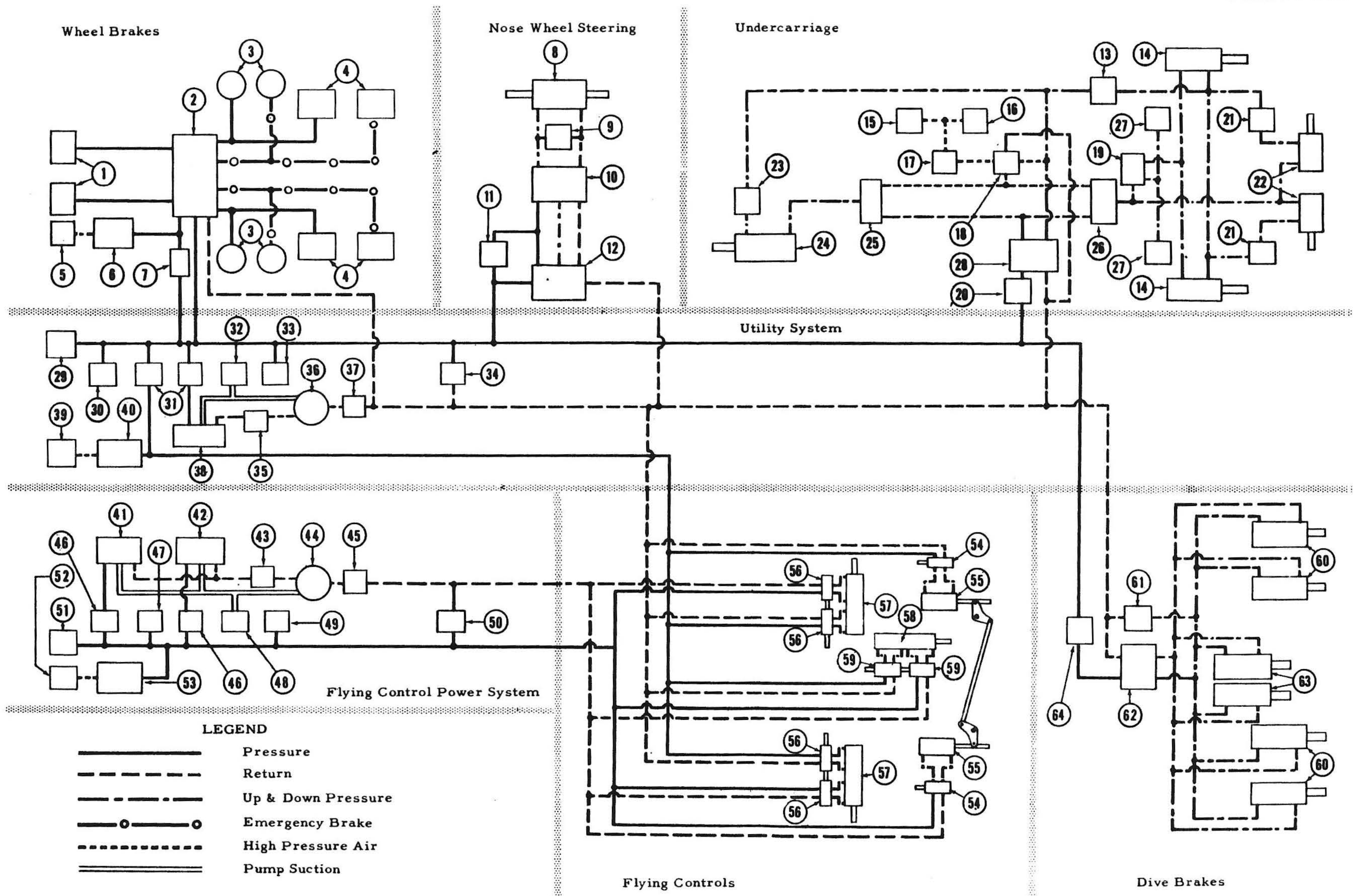


FIG. 29 HYDRAULIC SYSTEM DIAGRAM

3.15.2 Pneumatic System:

3.15.2.1 Description and Components: Refer to Fig. 28 and 30. A low-pressure pneumatic system will be fitted, operating at a pressure of approximately 100 lb. per sq. in. In flight, the system will function on air bled from the main engine compressors; for ground operation, the system will draw compressed air from a portable gas turbine compressor unit or the auxiliary gas turbine compressor unit installed in the airplane. Following is a list of services operated by the system:-

(a) Electrical equipment: Two air turbine motors will drive the two alternators which provide the electrical power supply for the operation of all electrical equipment and services, and the electronic equipment.

(b) Fuel System: Air turbine motors will operate the two fuel pumps serving the afterburner main fuel supply, the two afterburner fuel booster pumps, and the two engine fuel booster pumps.

(c) Air Conditioning System: This system, which provides heated or cooled air for the conditioning and pressurization of the cockpit, will be dependent on the pneumatic system for operation (refer to sub-paragraph 3.20.1).

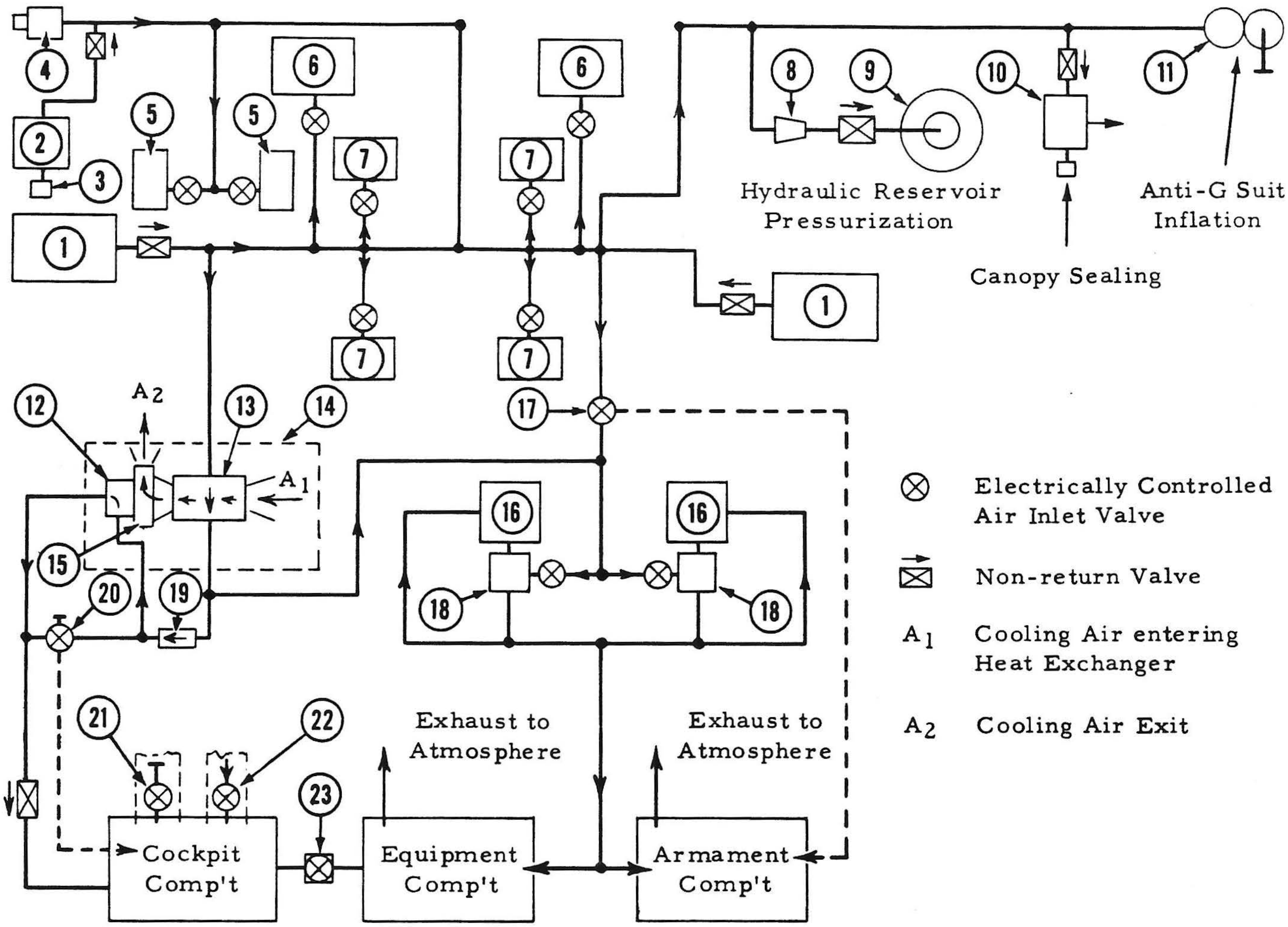
(d) Engine Starting: Two air turbine engine starters operating on compressed air taken from either a portable ground compressor unit or the auxiliary gas turbine compressor, will be used for engine starting purposes.

(e) Miscellaneous services: In addition to the aforementioned, the system will supply air to the pilot's anti-'g' suit, the canopy sealing, hydraulic reservoir pressurization and windscreen demisting.

The design of ducting and associated fittings will, in general, follow the requirements of AMC Manual No. 80-1. A diagram of the system is given in Fig. 30.

3.15.2.2 Summary of Main Units: Following is a summary of units of the pneumatic system:

(a) Constant speed air turbine motors: Two of these motors are used to drive the alternators. For these units an accurate control of shaft output speed is required. This is accomplished usually by limiting the inlet temperatures and pressures and drawing a substantial flow of air for operating purposes. For this reason, the supply is ducted from the source through a heat exchanger from which it is led to the motors. Air discharged from the motors, at outside ambient atmospheric pressure, is then circulated in the equipment section of the fuselage to provide cooling in this compartment.



1 Cockpit Comp't Temp. - Controlled to ±5°F of Selected Temp. within range +20°F to 100°F

2 Armament Comp't Temp. - Controlled at 80°F ±10°F

3 Equipment Comp't Temp. - not to exceed 150°F

LEGEND

1. Engine Compressor
2. Auxilliary Gas-Turbine Compressor
3. Emergency Hydraulic Pump
4. Ground Connection
5. Air Turbine Engine Starter
6. Afterburner Fuel Supply Pump
7. Fuel Booster Pumps
8. Pressure Reducing Valve
9. Hydraulic Reservoir
10. Canopy Seal Inflation Valve
11. Regulating Valve
12. Expansion Turbine
13. Air-to-Air Heat Exchanger
14. Refrigerating Unit
15. Cooling Air Fan
16. Alternators
17. Armament Temperature Control Valve
18. Constant Speed Air Turbine Motor
19. Flow Control Valve
20. Cabin Temperature Control Valve
21. Safety and Inward Relief Valve
22. Emergency Ram Pressure from Engine Intake
23. Cockpit Pressure - Control Valve

 FIG. 30 PNEUMATIC AND AIR CONDITIONING SYSTEMS DIAGRAM

(b) Air turbine driven fuel pumps: Air, at relatively high temperatures and pressures, is ducted directly to the fuel pump air turbines for driving purposes. In this manner the air flow demands are reduced to a minimum. Air discharge from the turbines is exhausted directly to the atmosphere.

(c) Air cycle refrigeration unit (air conditioning system): This installation will consist of a heat exchanger and an expansion air turbine and cooling air fan. Cooling air, taken from one of the engine intake ducts, will be drawn by the fan and directed over the heat exchanger and discharged to atmosphere. Hot compressed air will be ducted directly to the refrigeration unit where it will be passed through the air-to-air heat exchanger and expansion turbine and then delivered to the cockpit. A hot air by-pass, combined with a flow control valve, electronic temperature control unit, temperature sensing elements and other associated controls, regulate the volume and temperature of the air delivered to the cockpit. This arrangement provides the required heating, cooling, ventilation and pressurization to the cockpit. Discharged cockpit air will be led into the equipment compartments and circulated to provide additional cooling.

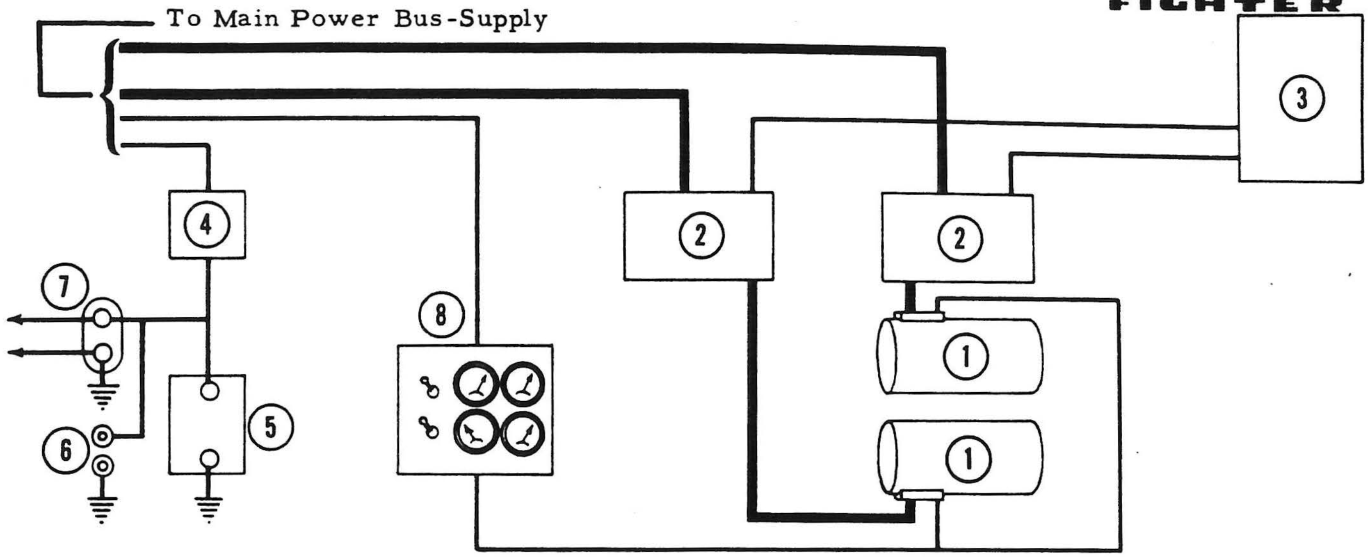
(d) Air turbine engine starters: Two starting motors will be required, one for each engine. Air supply to the motors will be taken from the sources mentioned in sub-paragraph 3.15.2.1 (d).

(e) Auxiliary gas turbine compressor: This unit will provide the air pressure required for the operation of all pneumatic services in the event of main engine failures in flight. Additionally, it can be used for ground operation of the system if an external air supply is not available. An emergency hydraulic pump will be mounted on the output shaft of the unit.

For the location of the various items of equipment refer to Fig. 28.

3.16 Electrical:

3.16.1 Description: All electrical power - except from the batteries - is generated by two 12 kw., 400 cycle alternators connected in parallel and electrically synchronized. Each alternator is driven by a constant-speed air turbine motor. This arrangement ensures that the alternator power output is of a constant frequency. Various voltages (i.e. 28v., 115v., 205v. and higher) may be tapped off the alternators for supply to the items of radar equipment. Direct current is obtained from rectifiers connected to the 28v. a.c. bus. The battery is connected in parallel with the 28v. d.c. bus for initial power requirements. A schematic illustration of the system is shown in Fig. 31.



LEGEND

1 12 KW - 400 Cycle Alternator
2 Alternator Control Panel
3 Electrical Synchronizing Panel
4 Selenium Rectifiers
5 24 Volt 34 A.H. Battery
6 External Supply Socket
7 28 Volt D-C Bus-
8 Alternator Switches and Instrument Panel

ELECTRICAL SERVICES

A LIGHTING
- Landing
- Downward Ident
- Navigation
- Cockpit
- Equipment Bay

B INSTRUMENTS
- Turn and Bank
- Artificial Horizon
- Outside Air Temp
- Ice Warning
- Heated Pressure Heads

C FLYING CONTROLS
- Dive Brake Actuation
- Undercarriage Actuation
- Undercarriage Indication

D ENGINE INSTRUMENTS
- Starting and Ignition
- Tachometer
- Oil Pressure Indication
- Inlet Oil Temp
- Jet Pipe Temp
- Bearing Temp
- Fire Detection
- Fire Extinguisher
- Scavenger Oil Temp

E FUEL SYSTEM
- Wing Transfer Fuel Pumps
- Fuel Contents
- Fuel Pressure Warning
- Fuel Pressure Indication
- Afterburner System
- Fuel Shut-off Valves

F AIR CONDITIONING
- Air Conditioning System
- Cabin Pressure Warning

G ARMAMENT
- 2.75" Rockets
- Falcon Missiles

H ELECTRONICS
- Radio and Radar
- Electronics Rack Hoist

I AUXILIARY POWER UNIT STARTING

FIG. 31 ELECTRICAL CIRCUIT DIAGRAM

3.16.2 Electrical Power Supply: The electrical load to be supplied by the alternators is as follows:-

Supply	Demand
Electronics	10 kw
Fuel pumps	3 kw
Armament (pulse load)	5 kw
Variable load (lighting, flying controls, etc.) ...	2 kw

In the event of failure of an alternator the remaining one will carry the load for any length of time. The normal load will be approximately 13 - 14 kw. A control panel is provided for each alternator in the fuselage armament section. The synchronizing unit and appropriate circuit breakers are located adjacent to the panel. Cockpit controls comprise:-

Alternator starting switches	Wattmeter - varmeter
Voltmeter	Frequency meter, and
Ammeter	Over temperature warning unit

Good voltage regulation, in spite of the numerous voltages required to be controlled, is possible due to the fact that nearly all the electrical load is constant. Provision is also made in the power circuit for over voltage protection, feeder protection, reverse current cut-out, faulty alternator disconnection and load equalization. As previously mentioned, the 24v., 34 ampere-hour battery is connected in parallel with the 28v. a.c. bus. The chief load on the battery is the starting of the auxiliary power unit. An external supply is provided for the battery only. All other power is obtained by supplying air to the pneumatic system which, in turn, operates the turbines which drive the alternators. Danger of alternator overheating is obviated by the circulation of cooling air supplied by the pneumatic system.

3.16.3 Electrical Power Conversion: Some d.c. power is required and is obtained from selenium rectifiers connected to the 28v. a.c. bus. These units are cooled by the air conditioning system.

3.16.4 Equipment - Installation: Refer to Fig. 28. The alternators are installed in the upper central part of the fuselage equipment section. Removal of the alternators can be accomplished through an access door in the bulkhead behind the electronic crate, by lowering the crate. Located above the rocket pack is an access door through which minor adjustments and servicing of the alternators can be made. The alternator control panels and rectifier units are located on the left hand side of the armament bay at the forward end. Access is gained to these latter components when the rocket pack is removed. Forward of the armament bay, on the left hand side, is a compartment used to house the battery. A door, in the fuselage lower surface, gives easy access to the battery for servicing purposes. The alternator switches, instruments and power buses are located in, and are accessible from, the cockpit.

3.16.5 Wiring: Wiring will be fitted in accordance with Spec. MIL-W-5086. Plastic tubing will enclose the cable assemblies.

3.16.6 Bonding: All flying controls, moving surfaces, electrical panels and junction boxes will be bonded to the main structure in accordance with the instructions issued in Spec. MIL-B-5087.

3.16.7 Controls: Trip free circuit breakers will be used on d.c. circuits; fuses will be used on a.c. circuits. Current limiters will be used in the power distribution lines.

3.16.8 Lighting: Lighting will be provided for the following:-

(a) Cockpit and instrumentation
(b) Landing
(c) Navigation, and
(d) Downward identification

3.16.9 Starting and Ignition: Air turbine motors will be used to start the engines (refer to sub-paragraph 3.12.12). When engine speed reaches about 3,500 r.p.m., a starter fuel solenoid valve will release fuel to the torch igniters, which receive high tension sparks from the booster coils.

3.16.10 Receptacles: An external power receptacle supplies the 28v. d.c. supply only. Standard AN connectors will be used on all junction boxes and panels. Provision will be made for fuel line grounding and a cable will be fitted to ground the fuselage structure.

3.16.11 Indicators: For details of the electrical indicators refer to sub-paragraph 3.16.2. Flash warning lamps are provided for:-

(a) Ice warning
(b) Fire detection
(c) Cabin pressure warning
(d) Alternator overtemperature
(e) Fuel pressure warning, and
(f) Turbine failure.

3.16.12 Electrical Drives: The eight wing transfer fuel pumps will be electrically driven as will the auxiliary power unit starting motor and the electronic rack hoist.

3.16.13 Relays: Relays will be used in the following circuits:-

(a) High power
(b) Alternator control
(c) Battery, and
(d) Starting and ignition

3.16.14 Booster Coil: A high tension booster coil will be fitted to provide the initial spark for engine combustion (refer to sub-paragraph 3.16.9).

3.16.15 Radio Filters: These items will be incorporated as required.

3.17 Electronics:

3.17.1 List of Equipment: A completely integrated electronic system, the Hughes MX 1179 - will be installed. This system will cover:-

(a) Communication

(b) Navigation, and

(c) Fire control of missiles and rockets

Due to the large volume of space required for the equipment, coupled with the problem of removal and servicing, special design consideration has been given to location and accessibility of the units. The radar scanner and transceiver are, by necessity, located in the airplane nose. This structure, designated as the radome, is hinged so that it can be easily swung open for pre-flight servicing and adjustment of the equipment. Items of equipment in the cockpit comprise the control units, a radar screen for target display and an instrument which will show all the integrated navigational information. Numerous channel selectors and radar controls will be provided to allow the pilot to overcome detection or interference with the equipment by unfriendly forces. The bulk of the equipment is installed in the electronic rack located just aft of the cockpit in the equipment section of the fuselage (Fig. 32). The entire rack can be raised or lowered by means of an electrically driven hoist. This arrangement gives simplified servicing. Quick-disconnect mountings are provided with all units. In order to achieve efficient packing, the entire rack is shock-mounted. Each unit can be checked rapidly, provision being made for the replacement of a faulty unit without complicated alignment problems. It is intended to pack the equipment in uniform size and shape containers to obtain the maximum efficiency of installation. Air conditioning and temperature control will be provided in the compartment.

3.17.2 Communications: A V.H.F. transceiver system will be used for air-to-air and ground-to-air communication. A number of channels will be provided to prevent jamming and interference. Information from the ground will be recorded by a computer (sub-paragraph 3.18.3.5) for future instructions.

3.17.3 Navigation Equipment: A complete navigation system will be provided incorporating a number of redundant features which, in the event of their failure, will still allow the remaining equipment to function. A radio compass will be installed in the system together with an omni-range unit. This arrangement will increase the versatility of the equipment. Inaccurate information received from a unit will be rejected by the digital computer. Distances from different stations are provided by DME. Ground fix and altitude position can be obtained from the DME and omni-range. An A.P.I. is also provided. Either an ILS glide path receiver working in conjunction with a master beacon receiver or a radar AGCA system will be used. A flight path controller, operating in harmony with the auto-pilot system, will use the information from the various navigational and radar services to guide the airplane over the calculated course.

3.17.4 Radar: Refer to sub-paragraph 3.17.1.

3.17.5 Electronic Countermeasure: Analyzing equipment is incorporated in the computer. This equipment can discriminate between accurate information and absurd or interfering material. Radical changes or a discontinuity in the signals impinging on the computer make the discrimination possible.

3.17.6 Electronic Guidance Equipment: Refer to sub-paragraph 3.17.1.

3.17.7 Static Dischargers: Provision will be made for static dischargers as necessary.

3.18 Armament:

3.18.1 Description: The Hughes MX 1179 integrated electronic system is used and provides for:-

(a) Automatic navigation
(b) Interception
(c) Fire control, and
(d) Return to base.

The basic feature of this system is that data from all the relevant sources are fed into a single high-speed, digital airborne computer which then, successively, supplies the information to the airplane controls required for navigation and fire control. In the following sub-paragraphs only those functions pertaining to the final approach to the target and the firing of the aimed and guided rocket armament is discussed. Navigation aspects are dealt with in sub-paragraph 3.2.13.2.

3.18.2 Fixed Guns: Equipment of this nature is not carried on this airplane.

3.18.3 Rockets: Two types of rocket armament are carried as follows:-

(a) The folding fin air-to-air rocket (2.75 in.), and

(b) The Hughes 'Falcon' guided missile.

These items are discussed below.

3.18.3.1 Air-to-air Rockets: Normally, these projectiles are fired at altitudes below approximately 10,000 feet. At heights up to this figure, ground echoes interfere with the guidance system of the 'Falcon' projectiles. Twenty-four Hughes 2.75 in. diameter rockets are stowed in an extensible pack located in the fuselage belly (refer to Fig. 32). The operation of this pack is similar to that of the F86D and also to the proposal for the CF-100, Mark 4. Extension is by means of hydraulic jacks, the period from fully retracted to fully lowered taking 0.3 seconds. All the rockets are fired in a salvo, after which the pack is retracted. An interlock circuit is provided so that, once the firing signal has been given and any rocket fails to leave its container the whole pack is jettisoned. The stowage compartment is air conditioned to within the required limits of temperature (refer to sub-paragraph 3.20.1.1.2).

3.18.3.2 Optical Sights: At altitudes below 1,000 feet, the airborne radar tracking of the target is ineffective. Any firing of rockets at or below this height will be accomplished by the pilot using optical sighting. A special sight, similar to the G.G.S. (modified for rocket firing), will be designed by Hughes for this purpose. Sighting by this method is suitable only for stern attacks.

3.18.3.3 'Falcon' Guided Missiles: Six 'Falcon' missiles are stowed in the bottom compartment of the fuselage as shown in Fig. 32. Normally, all the missiles are fired during one pass and at least one must hit the target to achieve lethal effects (the missile war head is relatively small). A semi-active guidance system homes the missiles on to the target. The missile compartment is air conditioned to keep it within the required limits described in sub-paragraph 3.20.1.1.2. Electric power is supplied to the missiles while they are stowed to:-

(a) Warm up the electronic elements of the guidance system

(b) Erect the gyros

(c) Provide power for the thermostatically controlled heating system.

Each missile is suspended on two arms which are extended by a hydraulic jack. As the missiles are lowered they open the individual spring-loaded doors which enclose them. Firing is accomplished a few milliseconds after the missiles have been lowered and the radar system has locked on the target.

3.18.3.4 Radar System: In general, the radar system is a development of the APG40. The scan is through ± 70 degrees using a power of 250 kw. This is sufficient to give a search radius of approximately 30 miles. The information on the target is fed through the computer which enables it to be displayed to the pilot on a cathode ray tube in relation to the optimum interception course. In order to distinguish enemy aircraft from friendly aircraft, an airborne IFF is used which removes the response of friendly aircraft from the screen. If it is desired to investigate a particular echo more closely, a sector scan may be chosen. To lock on any particular target, the pilot must set a cursor over the display of the selected target. The radar then supplies the computer with sufficient data to generate the steering and firing signals required to complete the attack and finally to pull out in the optimum manner. The sole function of the pilot in this instance is target discrimination and he must judge whether the echo to be attacked is really an airplane and if he wishes to attack it. If there are several targets he must decide, in conjunction with the pilots of any other fighters in the vicinity, which aircraft he will attack. Although the remainder of the attack is automatic, it can be accomplished manually in the event of failure of the automatic controls. Insofar as the pilot is relieved of the majority of his present duties he is free to exercise target discrimination which cannot as yet be dealt with adequately by mechanical means. By necessity, this function (when flying this airplane at combat speed) must be handled in an extremely short space of time. The addition of other duties would make it impossible for the pilot to give the necessary judgements in a satisfactory manner.

3.18.3.5 Computer: All navigational, flight and fire control problems are handled by a single computer. This instrument is of the digital type incorporating a very large magnetic instruction and memory storage capacity. Basically the unit comprises:-

(a) Approximately 100 tubes

(b) Weighs in the region of 100 lb., and

(c) Occupies a volume of 1 cubic foot (approx.),

The various data for computation are sent through the computer in the form of pulses, each with their own appropriate instructions. In some cases, these are sufficient to make judgements. Thus, if the enemy drops material which reflects radiation, the computer will recognize that the speed of this material is not comparable with that of the airplane being tracked and will not allow the radar to lock on this extraneous echo.

3.18.3.6 Installation: All the electronic packages comprising the MX 1179 system are located in the electronic equipment crate as illustrated in Fig. 32. It will be seen, quite readily, that these are easily accessible for adjustment or complete removal, since they are all fitted with plug connectors designed for ease of removal. Special arrangements are made so that standard functioning tests may be applied to every box as a routine procedure. In this manner, any box which is shown to be defective can be removed in a matter of minutes. Following accepted principles, the radar dish is mounted in the nose radome. This section is hinged at its base so that it may be swung open to give quick and easy access to the antennae components.

3.18.4 Flexible Guns: The installation of a flexible gunnery system is not applicable to this airplane.

3.18.5 Stores: Stores are not normally carried on this airplane.

3.18.6 Passive Defence: As this airplane is designed to operate within a friendly GCI area it is, therefore, not expected to encounter anti-aircraft fire. Furthermore, it is anticipated that the armament of any airplanes which are likely to be engaged will be sufficiently heavy to make any armour carried to be of a prohibitive weight. Some protection will be offered by the heavy gauges of surface skin required.

3.19 Furnishings and Equipment:

3.19.1 Accommodation for Personnel: The pilot will be provided with a type of ejector-seat. If required, an ejection capsule, of the type being developed by the

U.S. Navy, could probably be fitted at a sacrifice of weight. Safety harness, cushions and anti-'g' equipment will be fitted in accordance with RCAF requirements.

3.19.2 Miscellaneous Equipment: The following may be fitted in accordance with RCAF requirements:-

- Rear view mirror
- Airplane flight report holder
- Map case
- Airplane check list holder, and
- Pyrotechnic equipment.

3.19.3 Furnishings: Furnishings will not be provided.

3.19.4 Emergency Equipment: Fire extinguishing equipment will be fitted if required. A fire detection system is installed in the main powerplant bay and in the auxiliary powerplant.

3.19.5 Oxygen Equipment: Oxygen equipment will be fitted for the pilot.

3.19.6 Emergency Rescue Equipment: This may be fitted if required by the RCAF.

3.20 Air Conditioning and Anti-icing Equipment:

3.20.1 Air Conditioning:

3.20.1.1 In-Flight Air Conditioning: The air conditioning system will be designed to meet the requirements of AMC 80-1. The main source of air pressure will be taken from the engine compressors and directed through the pneumatic system components as described in sub-paragraph 3.15.2. From these points, the supply will then lead into compartments which require pressure and temperature regulation. Pressure vs temperature requirements are detailed in the following paragraphs. Refer to Fig. 28 and 30 for schematic and equipment location illustrations.

3.20.1.1.1 Occupied Compartments: As the airplane is a single place interceptor the cockpit will be the only compartment occupied. Provision will be made for the pilot to select any desired cockpit air temperature within a range of +20 deg. F. to +100 deg. F., the required temperature being maintained to within ±5 deg. F. Cockpit pressurization will be governed by the pressure vs altitude schedule given in Fig. 33. In the event of an emergency, cockpit pressurization will be maintained by directing ram air from the engine intake ducts into the compartment. This provides sufficient pressure for survival right up to the ceiling, due to the high speed of the airplane.

3.20.1.1.2 Other Compartments: Other compartments to be air conditioned comprise:-

(a) The armament compartment, and

(b) The equipment compartments.

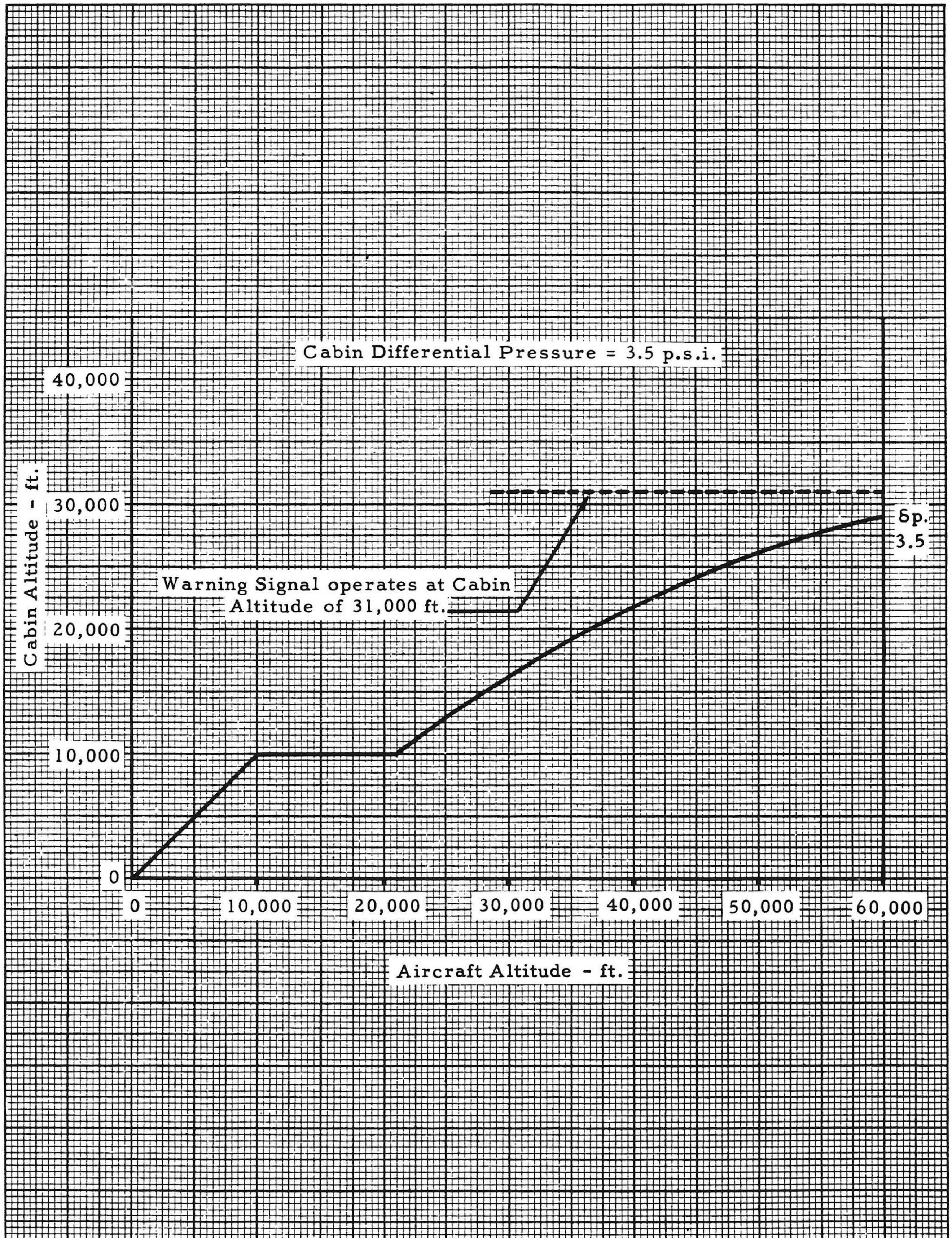


Fig. 33 Pressure vs Altitude Schedule

Dealing with item (a) first, the compartment will contain rockets and guided missiles and will not be pressurized. An air temperature of 80 deg. F. ±10 deg. F. will, however, be maintained within the compartment. In the equipment compartments, which contain the electronic and other heat generating equipment (e.g. electrical, hydraulic and pneumatic units), the majority of the equipment in the compartments will operate efficiently over a temperature range of -50 deg. F. to +150 deg. F. Rigorous temperature control, therefore, will not be necessary. In order, however, to provide for reasonable rates of heat dissipation from the heat generating equipment, ventilating air will be circulated through the compartment to maintain a temperature of approximately 100 deg. F. As the compartment is not pressurized, discharge of the circulating air can be easily accomplished.

3.20.1.2 Ground Air Conditioning: Provision will be made for pre-flight air conditioning of the airplane. A portable ground compressed air source or, alternatively, the installed gas turbine compressor unit will supply the necessary air conditioning for all conditions. Temperature ranges to be attained are given in the following sub-paragraphs.

3.20.1.2.1 Occupied Compartments: The cockpit will be the occupied compartment only which will require air conditioning. Following are the temperature requirements to be achieved before take-off.

Condition	Outside Air Temp.*	Temp. before Take-off
Summer	100 deg. F.	Cooled to 60 deg. F.
Winter	-5 deg. F.	Raised to 80 deg. F.

*Examples only

3.20.1.2.2 Other Compartments: Other compartments to be air conditioned during pre-flight preparations are the equipment and armament compartments. For the former, the electronic equipment will require a pre-flight warm-up period of approximately 20 - 30 minutes. Provision will be made in the air conditioning system for the dissipation of heat generated by the electronic equipment during the warm-up period. Ventilating air will be circulated in the compartment to ensure that the air temperature does not exceed 150 deg. F. This figure will be applicable for both summer and winter conditions. In the armament compartment the temperature will be maintained summer and winter at 80 deg. F. ±10 deg. F.

3.20.2 Anti-icing:

3.20.2.1 Anti-icing of Non-transparent Areas: The radome and the leading edge of the wing, the fin and the air intake ducts will be fitted with anti-icing means. X

3.20.2.1.1 Flight Operation: Since the high-altitude mission of this airplane will require only mild anti-icing, it is proposed to provide this for the wing, fin and air intakes by means of tapping hot air from the engine compressor and ducting this into the leading edges. This structure will be designed in such a manner that the available heat is distributed most efficiently to the outer skin. The radome anti-icing may be accomplished in similar manner or, alternatively, by means of infra-red radiation.

3.20.2.1.2 Ground Operation: This system can be operated on the ground to remove frost by means of running the auxiliary powerplant of the airplane. Refer to sub-paragraph 3.13.

3.20.2.2 Anti-icing, Defrosting and Defogging of Transparent Areas: The areas to be protected are the windscreen and the canopy windows. The exact method which will be used is not known at present since it depends largely on present and future research and development.

3.21 Photographic Equipment: This equipment will not be fitted in this airplane.

3.22 Auxiliary Gear: Provision in the design of this airplane will be made for the following:

(a) Towing
(b) Jacking
(c) Mooring
(d) Hoisting
and
(e) Leveling

4. SAMPLING, INSPECTION AND TEST PROCEDURES

4.1 These will be in accordance with requirements to be specified by the RCAF.

5. PREPARATION FOR DELIVERY

5.1 This will be in accordance with requirements to be specified by the RCAF.

6. NOTES

6.1 Explanatory Information: Since the content of this brochure is as exhaustive as is possible, pending the issue of a comprehensive specification by the RCAF for this type of airplane, no further explanatory information is available at present.

6.2 Additional Information: This may be found in the following appendices included at the end of this brochure.

Appendix III - Applicable Drawings
Appendix IV - Optional Arrangements
Appendix VI - Balance Calculations.

APPENDIX I

The above appendix, which covers Government and contractor furnished equipment lists, is not available at present.

APPENDIX II

DEVIATIONS

Pending the issue of a type specification for this airplane no deviation from standard parts, specifications, designer's handbook requirements and related documents, are available at present.

APPENDIX III

LIST OF APPLICABLE DRAWINGS AS SUBMITTED

Drawing Number	Title
SK 20433	C104/2, General Arrangement Drawing
SK 20426	C104/2, Inboard Profile and Fixed Equipment

APPENDIX IV

OPTIONAL ARRANGEMENTS

The inherent size of this twin engine airplane makes it possible to install different types of fighter armament, thus achieving desirable flexibility in the selection of a suitable installation. This flexibility is considered worthwhile for the following reasons:-

(a) As an insurance against the possibility of the Hughes MX 1179 missile installation not fulfilling expectations by a certain date.

(b) To cater to a change in the tactical situation requiring other types of armament for special purposes.

Accordingly, as an example of what can be done, the following alternative armament installations are discussed in this appendix:-

(a) 4-CARDE missiles installation

(b) 4-30 mm cannon plus 56 F.F.A.A. rockets installation

Furthermore, the basic airplane may be rapidly converted so as to perform a number of other missions other than the primary one of interceptor fighter. The following additional versions of the airplane are discussed in this appendix:-

(c) Long Range Fighter Version

(d) Operational Trainer Version

(e) Unarmed Bomber Version

(f) Photographic Reconnaissance Version

Summarizing:-

THIS AIRPLANE HAS QUALITIES WHICH MAKE IT SUITABLE AS A SUPERSONIC MULTI-PURPOSE AIRPLANE POSSESSING INHERENT FLEXIBILITY REGARDING THE FITTING OF DIFFERENT TYPES OF ARMAMENT OR EQUIPMENT.

APPENDIX IV A

Alternative Armament—Fighter Version

Four CARDE Missiles

Model Designation: C104/2/A

The exceptionally large armament bay of the basic fighter version makes it possible to install four of these missiles without changing the external shape and geometry of the basic airplane. Refer to Fig. 34. This illustration shows the armament installation. Assuming the same weight of electronic equipment as for the basic airplane the gross weight remains practically unchanged. Since the fuel capacity is also the same as for the basic airplane, it follows that the performance of this version is as tabulated in sub-paragraph 3.1.2 of this brochure. Conversion from the basic airplane can be accomplished easily because the armament bay consists of subsidiary structure only and does not contribute in any way to the strength and stiffness of the main fuselage structure; refer to sub-paragraph 3.7.1.2.3 in the first part of this brochure. Ballast is not required in this version. Details of the geometry of these missiles were derived from drawing SD/51082904/D, supplied by the Canadian Armament Research and Development Establishment. The weight of each missile has been quoted as being 305 pounds. The weight of armament provisions has been assumed at 150 pounds instead of 200 pounds for the MX 1179 armament. It should be noted that it is not possible, in this version, to carry additional internally stored F.F.A.A. rockets.

MODEL DESIGNATION C104/2/A

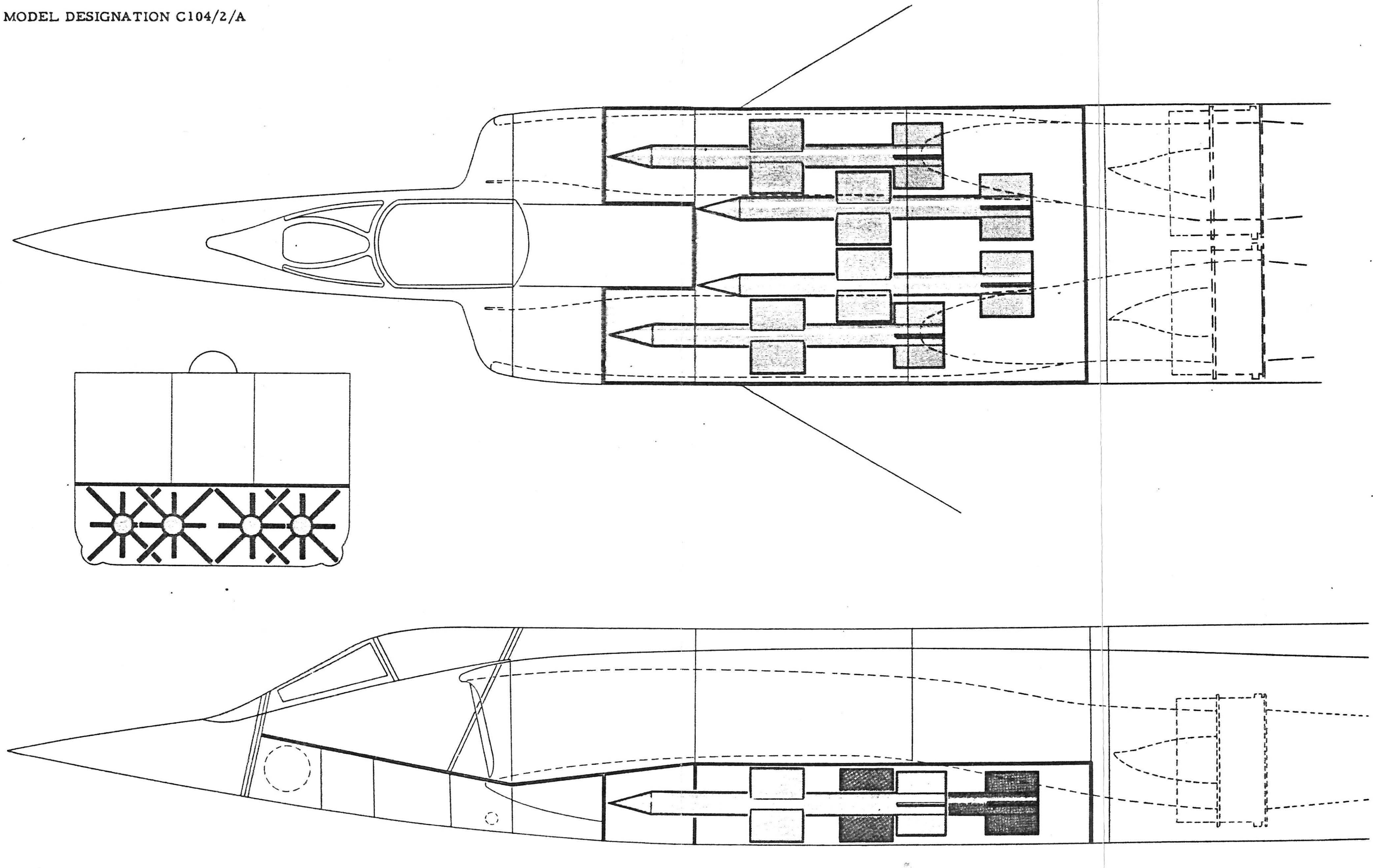


FIG. 34 CARDE MISSILES INSTALLATION

APPENDIX IV B

Alternative Armament Fighter Version

Four-30 mm 'Aden' Cannon with 200 shells each

Fifty-six F.F.A.A. Rockets (2.75 in. dia.)

Model Designation: C104/2/B

This armament installation can be stored easily inside the space allotted to the armament bay of the basic fighter airplane. Refer to Fig. 35 for details of this installation. Assuming the same weight of electronic equipment as for the basic airplane, the gross weight of this version is 2,160 lb. heavier. This includes 446 lb. of ballast, which is required at a point 56 feet from the nose datum in order to keep the c.g. of the airplane within the correct limits. The altitude performance of this version would therefore be somewhat inferior to that of the basic airplane. Conversion is easily accomplished due to the subsidiary nature of the armament bay structure.

MODEL DESIGNATION: C104/2/B

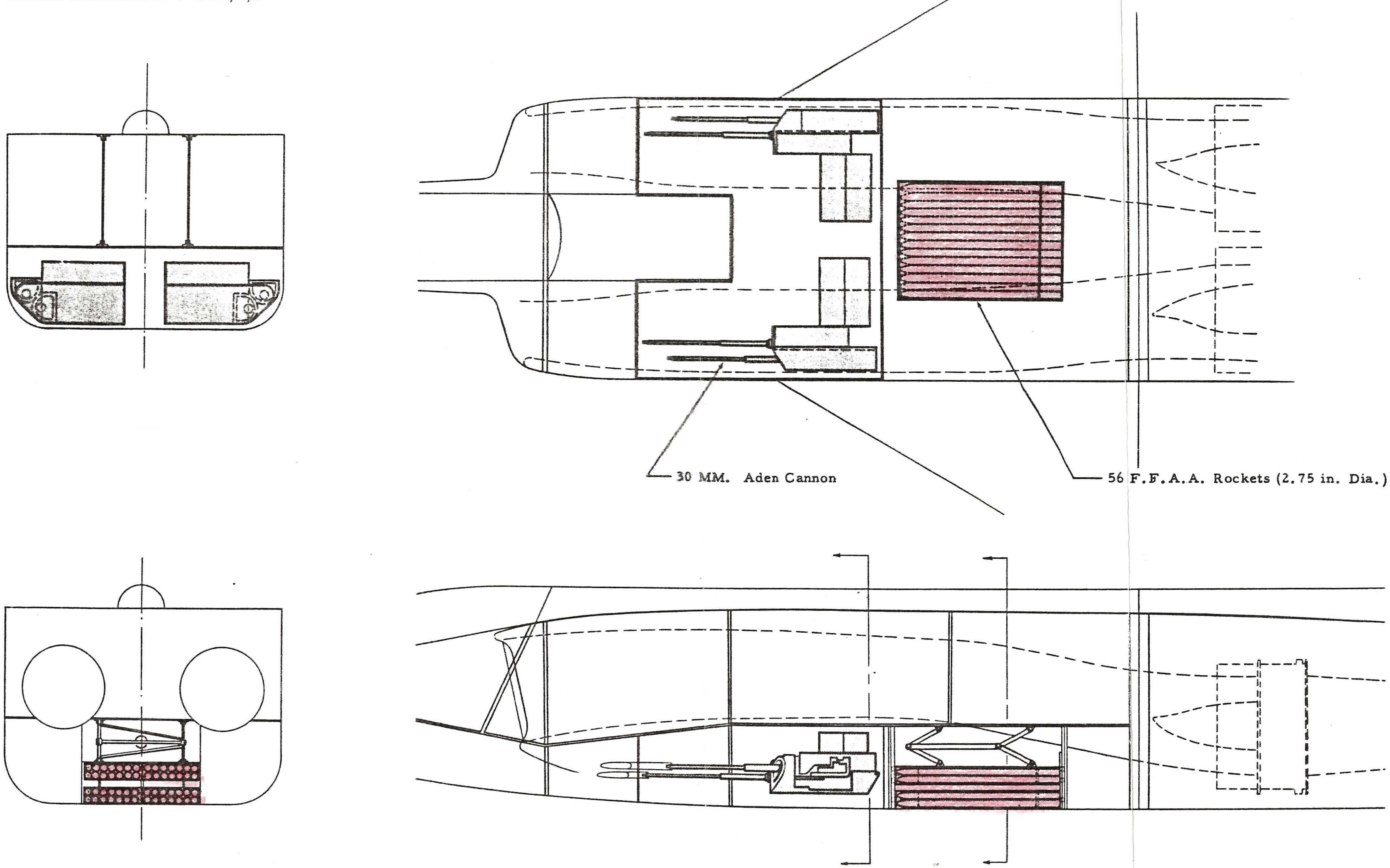


FIG. 35 CANNON AND ROCKETS INSTALLATION

APPENDIX IV C

Long Range Fighter Version

Model Designation: C104/2/C

By means of changing the wing bladder cell fuel tanks to integral tanks, and by inserting a 22 in. bay in the fuselage between the equipment and powerplant sections, it is possible to increase the total fuel capacity of the airplane to 20,000 pounds. The gross weight would then be of the order of 56,000 pounds.

The versatility of the airplane would be improved in this manner, by increasing its range considerably. This may be appreciated by comparing the mission profile tables included in this appendix with those presented on pages 6 to 8 for the basic airplane.

The high altitude operational radius with outbound cruising speed of M=1.5 would be increased from 242 nautical miles to 368 nautical miles. The operation radius with a cruising speed of M=.95 would be increased from 390 nautical miles to 581 nautical miles.

It is at sea level, however, where the improvement in range would be most marked. The sea level mission has been assumed to be as follows:-

> Outbound cruise at M=.95, with 5 minutes combat at the placard speed of 720 knots,
>
> and
>
> Return to base in the most economical manner. The operational radius for this mission would be increased from 111 nautical miles to 198 nautical miles.

While it may be considered that the high altitude operational radii are adequate with the basic airplane, it is felt that the sea level range is very marginal without the extra amount of fuel.

TABLE 1

COMBAT RADIUS OF ACTION, HIGH SPEED MISSION

	Distance naut. mi.	Time min.	Fuel Consumed lb.	Aircraft Weight lb.
A. Start	-	-	-	56,000
B. Taxi and Warm-up	-	4.0	760	55,240
C. Take-off: Maximum Thrust, Afterburners Lit	-	.4	490	54,750
D. Acceleration to Best Climbing Speed: Maximum Thrust, Afterburners Lit	4	.6	949	53,801
E. Climb to 36,090 ft.: Climb Thrust, Afterburners Lit	13	1.5	1420	52,381
F. Acceleration to Mach No. = 1.5: Maximum Thrust, Afterburners Lit	16	1.4	1032	51,349
G. Climb to 50,000 ft.: Maximum Thrust, Afterburners Lit	22	1.6	886	50,463
H. Cruise-out at 50,000 ft. at Mach No. = 1.5	313	21.8	7637	42,826
I. Combat at 50,000 ft. at Mach No. = 1.5	-	5.0	1693	39,883
J. Descent to 40,000 ft.	17	2.2	90	39,793
K. Cruise back at 40,000 ft.: Economical Cruising Speed	275	30.2	2505	37,288
L. Descent to 30,000 ft.	24	3.1	200	37,088
M. Stack at 30,000 ft.: Maximum Endurance Speed	-	15.0	1020	36,068
N. Descent to Sea Level	52	6.3	710	35,358
O. Approach: Maximum Endurance Speed	-	5.0	608	34,750
TOTAL:	736	98.1	20000	

Combat Radius of Action = 368 naut. mi.

*1250 lb. of ammunition fired

TABLE 2

COMBAT RADIUS OF ACTION, MAXIMUM RANGE MISSION

	Distance naut. mi.	Time min.	Fuel Consumed lb.	Aircraft Weight lb.
A. Start	-	-	-	56,000
B. Taxi and Warm-up	-	4.0	760	55,240
C. Take-off: Maximum Thrust	-	.7	340	54,900
D. Acceleration to Best Climbing Speed: Maximum Thrust	7	1.0	482	54,418
E. Climb to 36,090 ft.: Climb Thrust	41	4.5	1390	53,028
F. Cruise-out at 36,090 ft.: Economical Cruising Speed	501	55.2	6300	46,728
G. Acceleration to Mach No. = 1.5 Maximum Thrust, Afterburners Lit	14	1.2	900	45,828
H. Climb to 50,000 ft.: Maximum Thrust, Afterburners Lit	18	1.3	736	45,092
I. Combat at 50,000 ft. at Mach No. = 1.5	-	5.0	1700	42,142*
J. Descent to 40,000 ft.	17	2.2	90	42,052
K. Cruise-back at 40,000 ft.: Economical Cruising Speed	488	53.7	4764	37,288
L. Descent to 30,000 ft.	24	3.1	200	37,088
M. Stack at 30,000 ft.: Maximum Endurance Speed	-	15.0	1020	36,068
N. Descent to Sea Level	52	6.3	710	35,358
O. Approach: Maximum Endurance Speed	-	5.0	608	34,750
TOTAL:	1162	158.2	20000	

Combat Radius of Action = 581 naut. mi.

*1250 lb. of ammunition fired

TABLE 3

COMBAT RADIUS OF ACTION, SEA LEVEL MISSION

	Distance naut. mi.	Time min.	Fuel Consumed lb.	Aircraft Weight lb.
A. Start	-	-	-	56,000
B. Taxi and Warm-up	-	4.0	760	55,240
C. Take-off: Maximum Thrust, Afterburners Lit	-	.4	490	54,750
D. Acceleration to Mach No. = .95 Maximum Thrust, Afterburners Lit	4	.6	949	53,801
E. Cruise-out at Sea Level at Mach No. = .95	192	18.4	6000	47,801
F. Acceleration to Mach No. = 1.09 Maximum Thrust, Afterburners Lit	2	.2	326	47,475
G. Combat at Sea Level at Mach No. = 1.09	-	5.0	7060	39,165*
H. Climb to 40,000 ft.: Climb Thrust	33	3.8	1050	38,115
J. Cruise-back at 40,000 ft.: Economical Cruising Speed	89	9.8	827	37,288
K. Descent to 30,000 ft.	24	3.1	200	37,088
L. Stack at 30,000 ft.: Maximum Endurance Speed	-	15.0	1020	36,068
M. Descent to Sea Level	52	6.3	710	35,358
N. Approach: Maximum Endurance Speed	-	5.0	608	34,750
TOTAL:	396	71.6	20000	

Combat Radius of Action = 198 naut. mi.

*1250 lb. of ammunition fired

APPENDIX IV D

Operational Trainer Version

Model Designation: C104/2/D

A trainer version of the basic airplane is shown in Fig. 36. As will be seen from the illustration, this is essentially the same airplane as the fighter version but with a longer cockpit section. Conversion from the basic airplane is rapidly accomplished by unbolting the cockpit section of the fuselage forward of the transport joint bulkhead to which the nose gear leg is attached and replacing it with the longer trainer cockpit section.

If it is desired to carry the same armament and electronic equipment in the trainer as in the operational fighter, the resulting increase in gross weight together with the longer fuselage will, of course, detract somewhat from the performance as quoted in this brochure for the basic fighter airplane. Ballast is not required in the tail of the airplane since the most forward center of gravity is still within the limits allowed from considerations of stability and control.

MODEL DESIGNATION C104/2/D

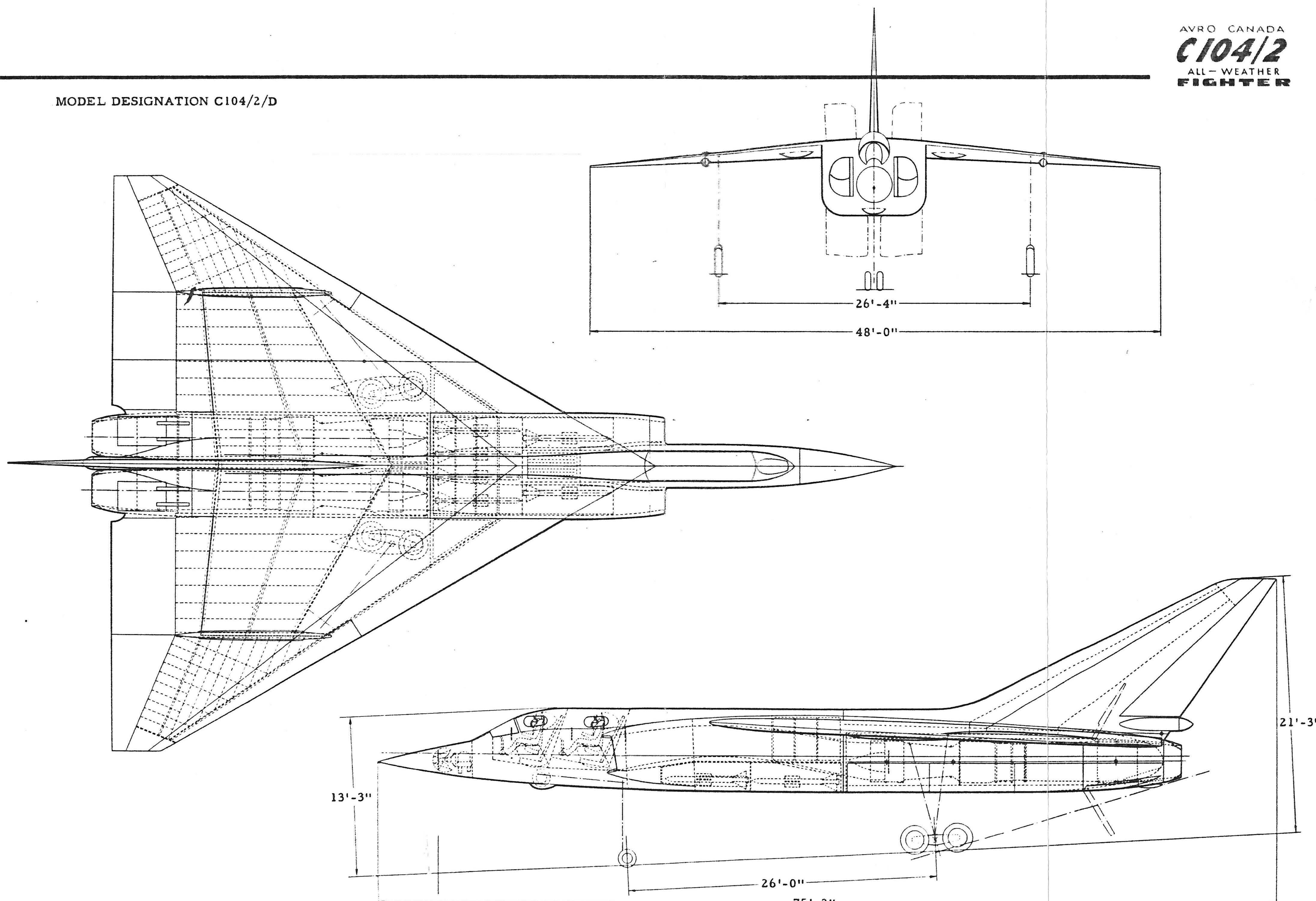


FIG. 36. 3 VIEW G.A. OF AIRPLANE

APPENDIX IV E

Unarmed Bomber Version

Four-1000 pound G.P. Bombs

Model Designation C104/2/E

Refer to Fig. 37. This illustration shows the bombs stored inside the space allotted to the armament bay of the basic airplane. It has been assumed that the tactical role of this airplane would be the destruction of a special target by means of dive bombing attack. For this type of attack, an elaborate bombing sight and a second crew member would not be required. The radar search and fire control equipment could also be deleted, which would save a weight of approximately 1000 lb. On the other hand, however, it is necessary to fit ballast of 700 lb. at a point 6.8 feet from the nose datum in order to keep the c.g. of the airplane within allowable limits. The weight of the bombs plus provisions for supporting and ejecting same will be of the order of 4340 lb. The gross weight of this version would therefore be:-

$$52000 - 1450 - 1000 + 4340 + 700 = 54{,}590 \text{ lb.}$$

The operational radius of this Bomber Version, flying subsonically during the outbound cruise, would be of the order of 420 nautical miles, on the basic airplane's fuel capacity.

MODEL DESIGNATION C104/2/E

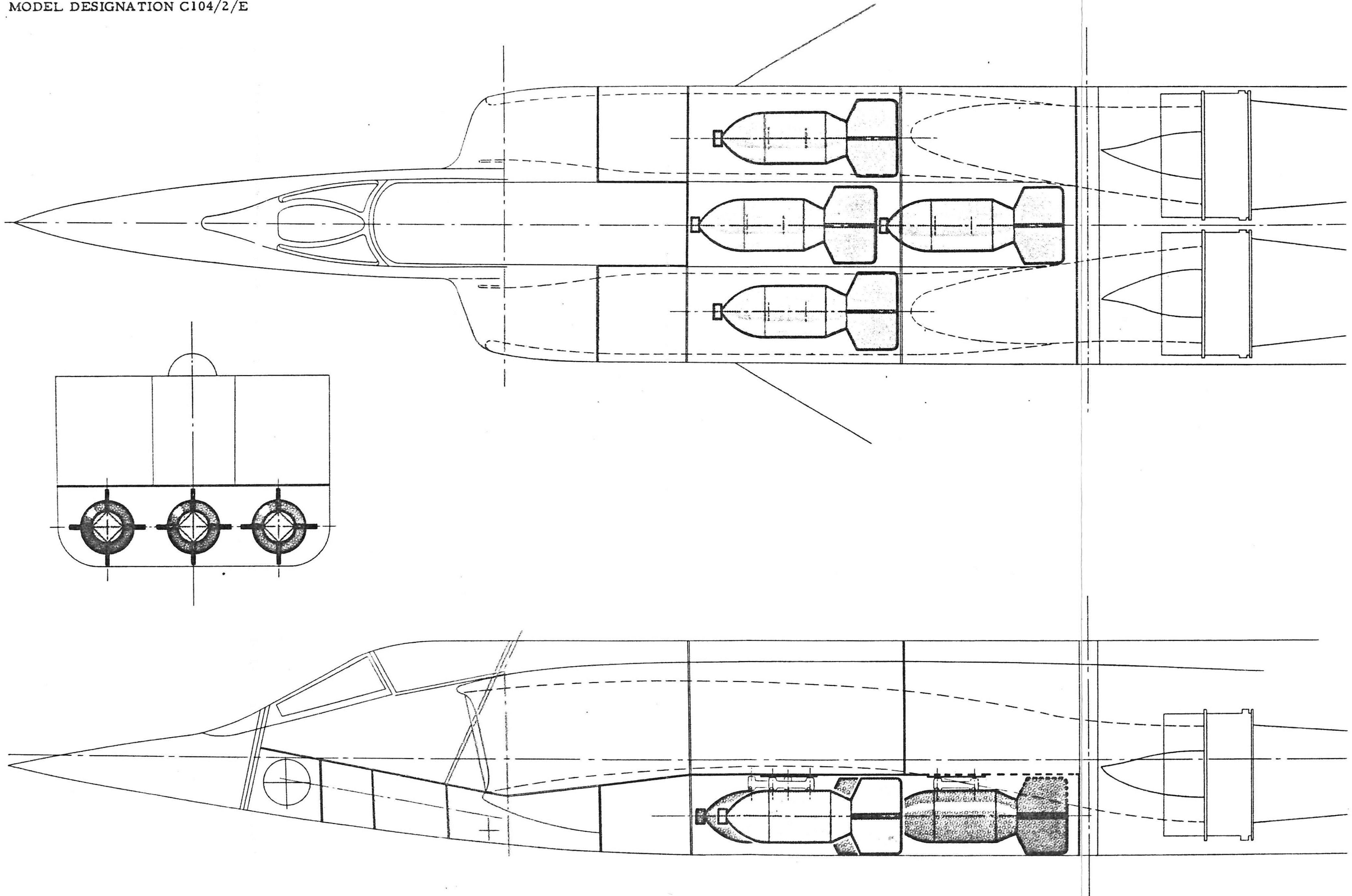


FIG. 37 INSTALLATION OF 4 - 1000 LB. G.P. BOMBS

APPENDIX IV F

Photographic Reconnaissance Version

Model Designation: C104/2/F

The exceptionally large armament bay of the basic airplane makes it possible to accomodate all equipment required for an unarmed photographic reconnaissance version of this airplane and furthermore to provide for an additional 1940 lb. of fuel. Refer to Fig. 38 which shows a proposed installation of the photographic equipment, together with the extra fuel tanks. The photographic equipment which has been assumed conforms to RCAF Operational Requirement No. ORI/1-31 which was issued in draft form for a CF-100 conversion on December 20, 1949. The following equipment has been shown installed:-

For day operation:-

(a) Left and right lateral oblique K/F24 cameras (7/8 in. lense).

(b) A pair of split vertical F52 cameras (20 in. lense).

(c) A vertical Sonne 57A stereo camera with 7 in. lense and ground speed synchronizer.

(d) Left and right forward facing oblique K/F24 cameras with 7/8 in. lenses. These will be mounted in wing tip fairings and are not shown in the illustration.

For night operation:-

(a) One vertical K19B camera with photo electric pick-up unit and amplifier.

(b) Ten multiple chamber flash cartridge dischargers containing sixty cartridges (SIS5343 and AP1661).

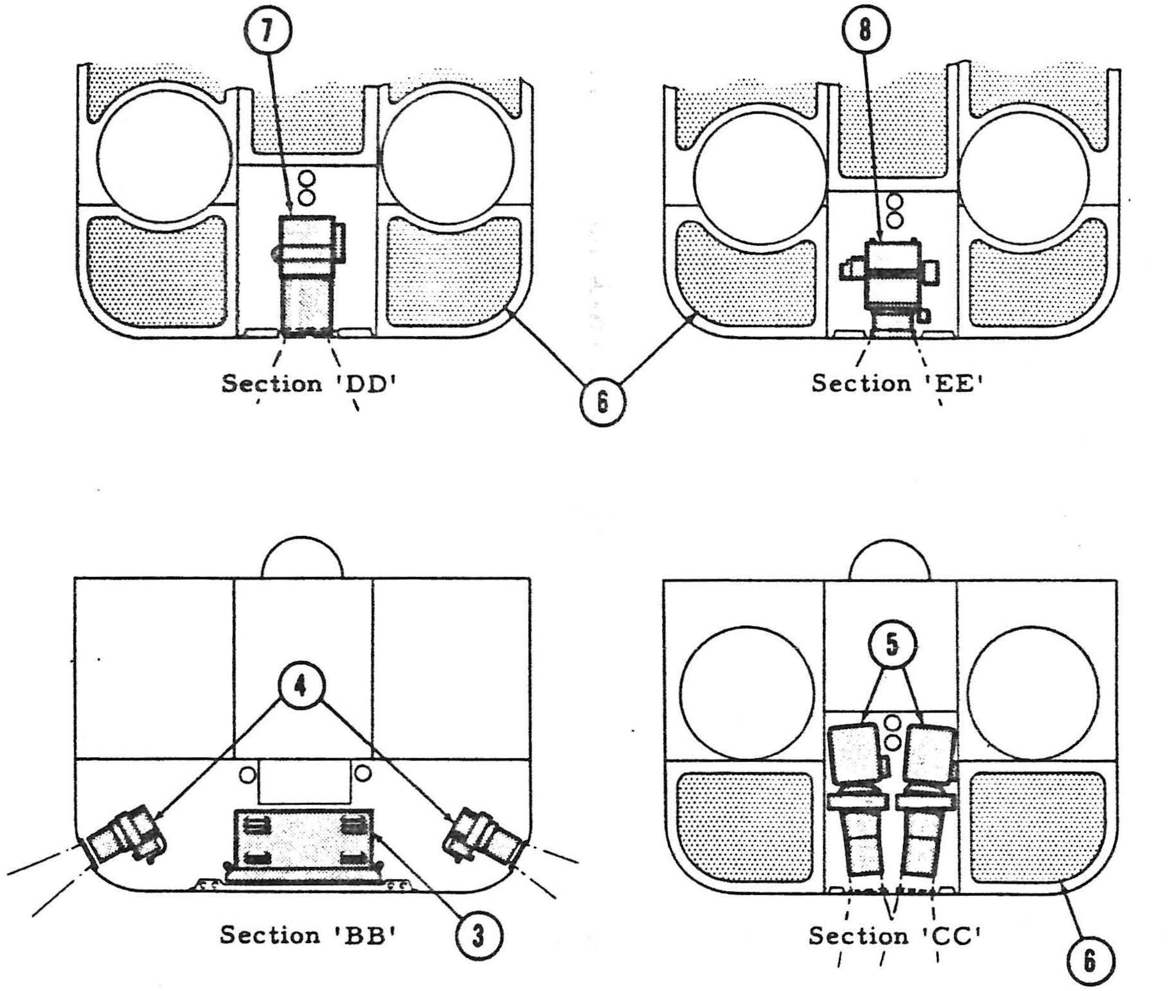
(c) Floodlight (29.75 in. dia. - Raytheon A-123G-X1).

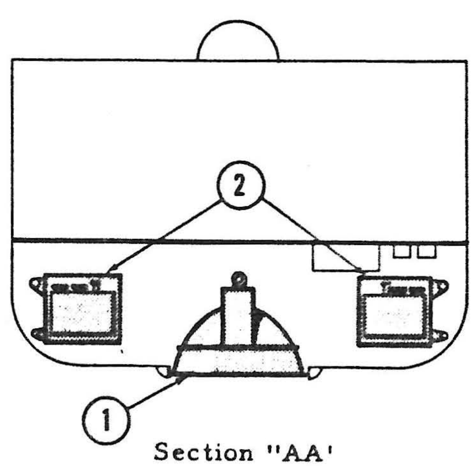
(d) Power unit (Raytheon A-123F-X1).

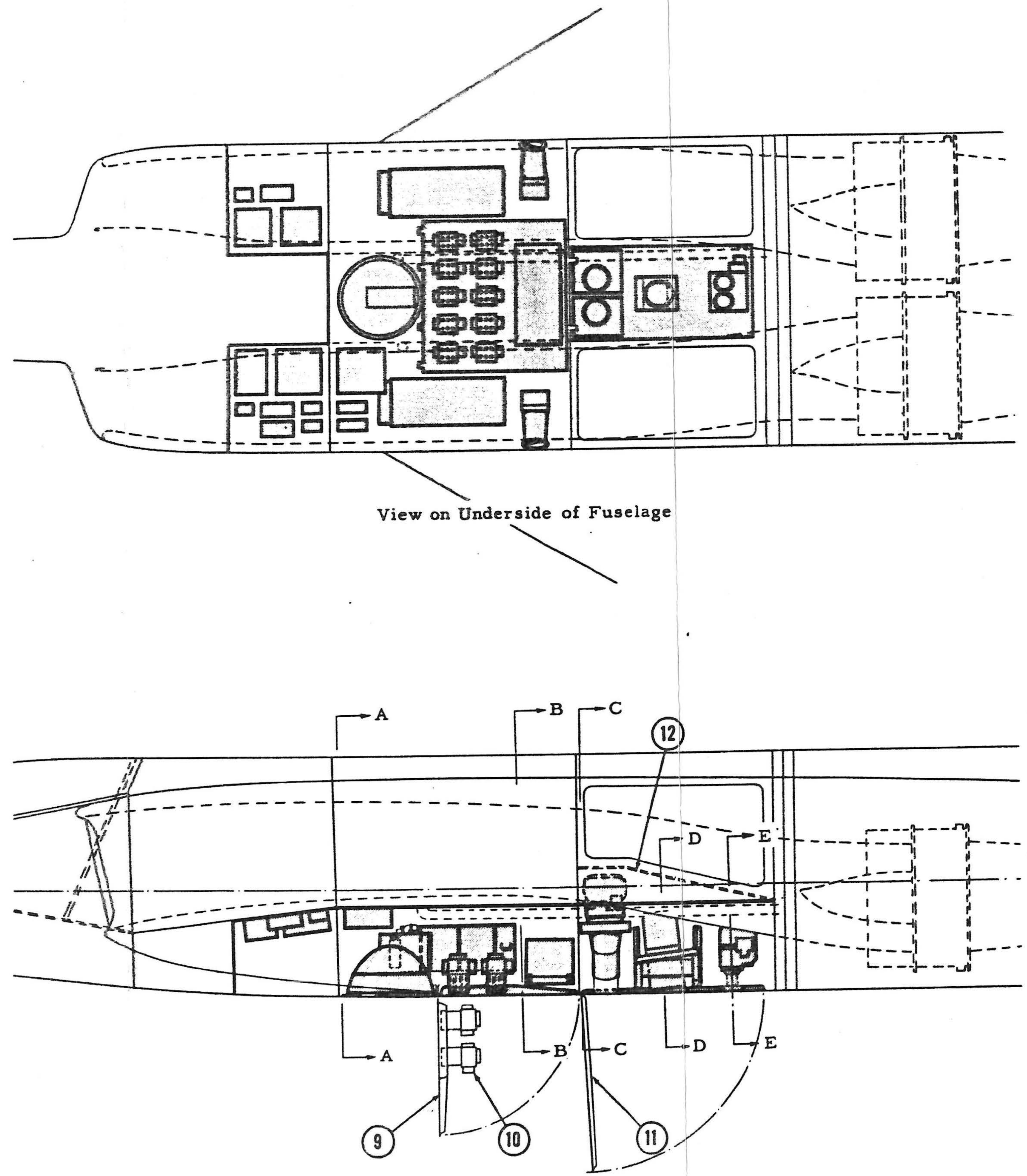
(e) Two condensers (Raytheon A-123E-X1).

The gross weight of this airplane with the extra amount of fuel is 1800 lb. heavier than the basic fighter version. Balast is not required. The range of this P.R. version will be of the order of 550 nautical miles.

MODEL DESIGNATION C104/2/F







LEGEND

1. Floodlight
2. Condenser Packages
3. Power Unit
4. K/F24 Oblique Camera
5. F. 52 Split Vertical Cameras
6. Fuel Tanks
7. K19B Vertical Night Camera
8. Sonne S7A Vertical Stereo Camera
9. Flash Cartridge Mounting Door
10. Flash Cartridge Containers
11. Access Door
12. Centre Tank Floor

FIG. 38 PHOTOGRAPHIC EQUIPMENT INSTALLATION

APPENDIX V

SUMMARY OF OPTIONAL ARRANGEMENTS

Aircraft Model:	Basic Fighter C104/2	Fighter C104/2/A	Fighter C104/2/B	Longer Range Fighter C104/2/C	Operational Trainer C104/2/D	Bomber C104/2/E	P.R. - Airplane C104/2/F
Reference Drawing:	See Fig. 1	See Fig. 34	See Fig. 35	Not available	See Fig. 36	See Fig. 37	See Fig. 38
Airplane Length:	70 ft. 3 in.	70 ft. 3 in.	70 ft. 3 in.	72 ft. 1 in.	75 ft. 3 in.	70 ft. 3 in.	70 ft. 3 in.
Crew:	1 pilot	1 pilot	1 pilot	1 pilot	2 pilots	1 pilot	1 pilot
Armament:	6-Falcon missiles 24-F.F.A.A. rockets	4-CARDE missiles	4-30 mm 'Aden' Cannon 56 F.F.A.A. rockets	6-Falcon missiles 24-F.F.A.A. rockets	6-Falcon missiles 24-F.F.A.A. rockets	4-1000 lb. G.P. bombs	8-Cameras and Illumination Equipment
Fuel Capacity:	16,000 lb.	16,000 lb.	16,000 lb.	20,000 lb.	16,000 lb.	16,000 lb.	17,940 lb.
Ballast:	None	None	446 lb. in rear nacelle	None	None	700 lb. in fuselage nose	None
Empty Weight:	34,243 lb.	34,193 lb.	35,988 lb.	34,243 lb.	34,962 lb.	33,933 lb.	34,243 lb.
Design Gross Weight:	52,000 lb.	51,920 lb.	54,160 lb.	56,000 lb.	52,926 lb.	54,590 lb.	53,800 lb.
Foward C.G. in Flight:	27.65% M.A.C.	27.81% M.A.C.	27.03% M.A.C.	27.65% M.A.C.	25.82% M.A.C.	25.56% M.A.C.	27.88% M.A.C.
Aft C.G. in Flight:	30.95% M.A.C.	30.98% M.A.C.	29.80% M.A.C.	30.95% M.A.C.	28.11% M.A.C.	30.98% M.A.C.	30.88% M.A.C.
Performance:	See para. 3.1.2	See para. 3.1.2	See para. 3.1.2	See appendix IVC	Not available	See appendix IV E	See appendix IVF

APPENDIX VI

BALANCE CALCULATIONS

The following horizontal balance calculations and center of gravity positions are for the basic fighter airplane. Center of gravity positions of the various items are located in feet aft of a vertical datum passing through the extreme nose of the aircraft. The formula which converts these center of gravity positions into percent of the mean aerodynamic chord of the wing is as follows:-

$$\% \text{ M.A.C.} = \frac{A - 30.21}{30.60} \times 100$$

where A is the center of gravity position in feet aft of nose datum. The following table should be read in conjunction with Fig. 39.

TABLE NO. 1

No.	Item	Weight In Pounds	Arm In Feet	Moment In Foot Pounds
	WING GROUP:			
1	Wing	9,200	42.75	393,300
2	Ailerons	215	58.60	12,599
3	Elevators	325	57.85	18,801
	TAIL GROUP:			
4	Fin and rudder	1,000	57.74	57,740
	BODY GROUP:			
5	Front fuselage	2,940	22.50	66,150
6	Rear nacelle	335	57.00	19,095
7	Speed brakes	163	56.55	9,218
8	Engine doors	567	44.30	25,118
9	Underwing structure	620	46.45	28,799
	LANDING GEAR:			
	Retracted:			
10a	Main gear including jacks	2,162	39.80	86,048
11a	Nose gear including jacks	427	11.26	4,808
12	Tail skid including jack	30	59.85	1,796
	Extended:			
10b	Main gear		42.25	91,345
11b	Nose gear		15.50	6,619

CONTINUATION OF TABLE 1.

No.	Item	Weight In Pounds	Arm In Feet	Moment In Foot Pounds
	ENGINE SECTION:			
13	Shrouds, drain plate	430	47.40	20,382
14	Engine and afterburner mounting	40	42.85	1,714
	POWER PLANT GROUP:			
15	Engine complete	7544	43.20	325,901
16	Engine controls	20	15.80	316
17	Fuel tanks	490	41.20	20,188
18	Fuel system	440	38.10	16,764
19	Fire extinguishing system	65	35.85	2,330
20	Engine anti-icing	185	30.88	5,713
21	Accessory gears and drives	15	43.57	654
22	Afterburner complete	1872	56.70	106,142
	FIXED EQUIPMENT GROUP:			
23	Instruments	53	11.00	583
24	Surface controls (including hydraulic jacks and artifical feel)	700	44.00	30,800
25	Hydraulic system	678	35.50	24,069
26	Electrical system	790	21.00	16,590
	HUGHES ELECTRONIC EQUIPMENT:			
27	in radome	210	6.80	1,428
28	in cockpit	190	11.30	2,147
29	in equipment bay	1400	17.75	24,850
30	Armament provisions	200	24.09	4,818
	FURNISHINGS:			
31	Ejector seat	132	13.50	1,782
32	Emergency provisions	15	12.90	194
33	Oxygen	20	10.10	202
34	Pneumatic and air condition systems	355	23.90	8,485
35	Anti-icing system	270	40.25	10,868
36	Brake parachute	75	59.80	4,485
37	Exterior finish	70	37.80	2,646
	WEIGHT EMPTY landing gear up	34243	39.64	1,357,523
	landing gear down		39.85	1,364,631

CONTINUATION OF TABLE 1

No.	Item	Weight In Pounds	Arm In Feet	Moment In Foot Pounds
	NON-EXPENDABLE USEFUL LOAD:			
38	Crew (one pilot)	207	13.00	2,691
40	Residual fuel	260	38.08	9,901
41	Oil	40	37.55	1,502
	GROSS WEIGHT LESS FUEL AND ARMAMENT:	34750		
	Landing gear up		39.47	1,371,617
	Landing gear down		39.68	1,378,725
	EXPENDABLE USEFUL LOAD:			
39	Fuel	16000	38.08	609,280
42	ARMAMENT:		24.09	30,113
	6 Falcon missiles	660		
	24 F.F.A.A. Rockets (including jettisonable container)	590		
	GROSS WEIGHT:	52000		
	Landing gear up		38.67	2,011,010
	Landing gear down		38.81	2,018,118

CENTER OF GRAVITY POSITIONS

(1) **Design gross weight condition, landing gear down:**
The C.G. in percent of mean aerodynamic chord is:

$$\frac{38.81 - 30.21}{30.60} \times 100 = \frac{860}{30.60} = 28.10\% \text{ M.A.C.}$$

(2) **Weight Empty condition, landing gear down:**
The C.G. in percent of mean aerodynamic chord is:

$$\frac{39.85 - 30.21}{30.60} \times 100 = \frac{964}{30.60} = 31.50\% \text{ M.A.C.}$$

This is the furthest aft C.G. but is not a flight condition.

CONTINUATION OF CENTER OF GRAVITY POSITIONS

(3) Design gross weight less expendable useful load, landing gear down:
The C.G. in percent of mean aerodynamic chord is:

$$\frac{39.68 - 30.21}{30.60} \times 100 = \frac{947}{30.60} = 30.959\% \text{ M.A.C.}$$

This is the furthest aft C.G. in flight.

(4) Design gross weight, landing gear up:

$$\frac{38.67 - 30.21}{30.60} \times 100 = \frac{846}{30.60} = 27.65\% \text{ M.A.C.}$$

This is the furthest forward C.G. in flight.

The estimated limits of Center of Gravity travel as determined by aerodynamic requirements of stability and control are:-

From 27% to 31% for the fighter versions

and

From 25% to 31% for the unarmed bomber and the operational trainer version.

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