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C 104/1

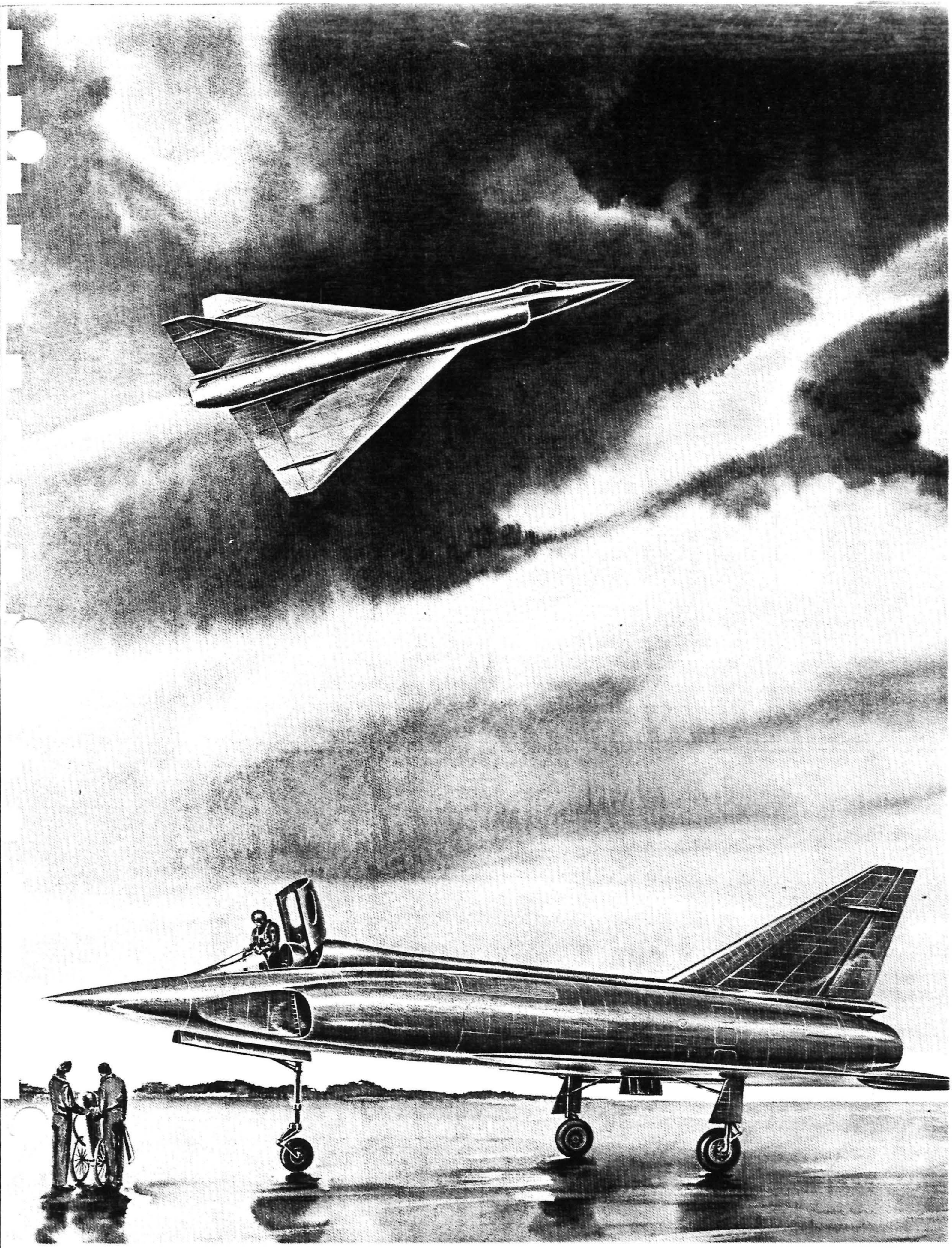
SUPERSONIC ALL-WEATHER FIGHTER

PROPOSAL NO. 1 SINGLE ENGINE AIRCRAFT

JUNE 1952

SECRET UNCLASSIFIED

A. V. ROE CANADA LIMITED
MALTON - ONTARIO



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Issued to: Mr. J. C. Floyd.

Date: 24-7-52

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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-104/1
Number and Places for Crew	One (1) Pilot
Number and Kind of Engines..........	One A. V. Roe Canada turbo-jet engine - TR9 or One Bristol Engine Co. turbo-jet engine - OL3 (fighter version) or One Curtiss-Wright turbo-jet engine - 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.

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.

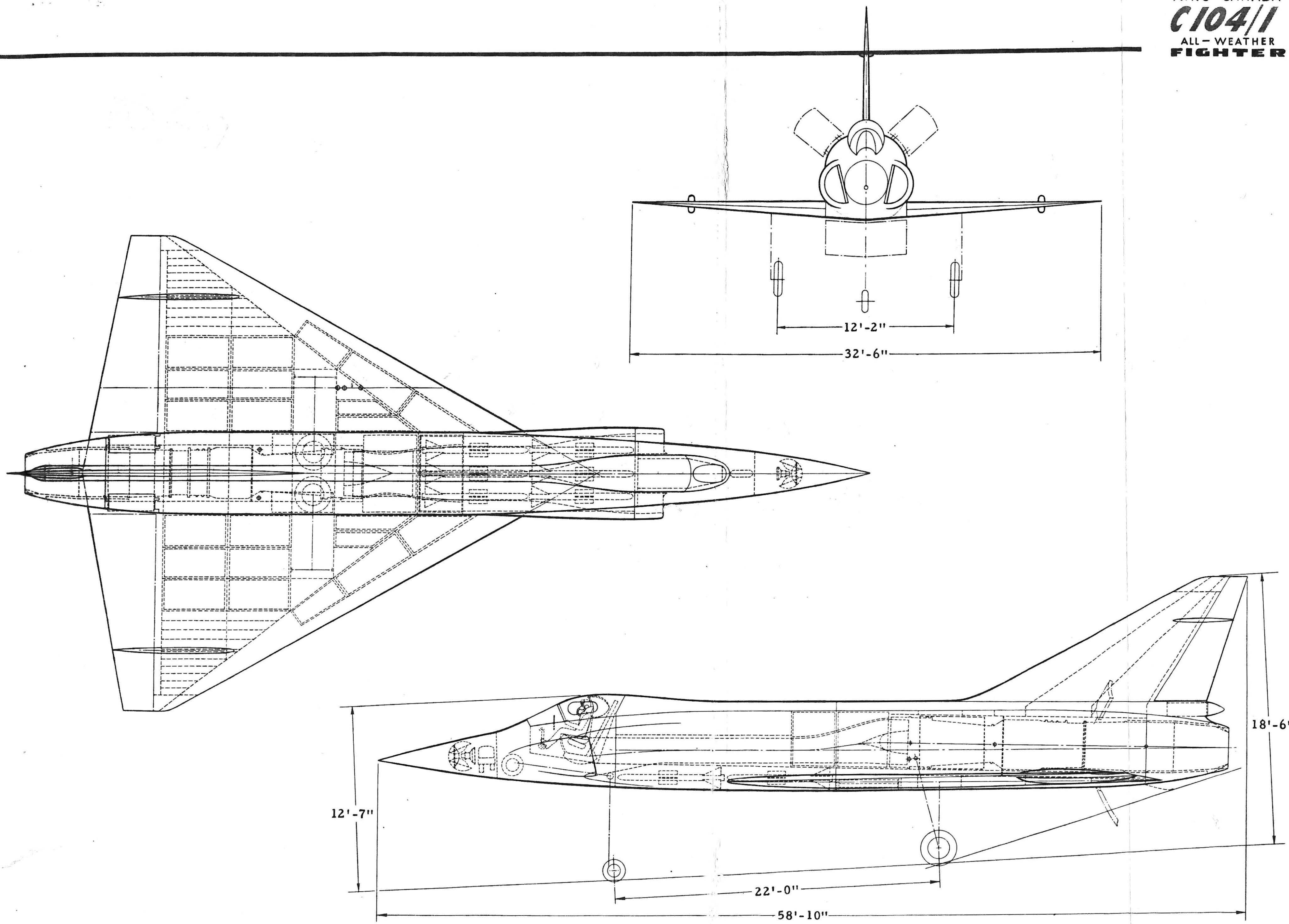


FIG. 1 3-VIEW G.A. OF AIRPLANE

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 (23,700 lb.):	
Maximum Thrust, Afterburner Lit ...	926 knots
Maximum Thrust ...	652 knots
Maximum Continuous Thrust, Afterburner Lit ...	791 knots
Maximum Continuous Thrust ...	641 knots
True Air Speed in Level Flight at 50,000 ft. Altitude at Combat Weight (23,700 lb.):	
Maximum Thrust, Afterburner Lit ...	959 knots
Climb Thrust, Afterburner Lit ...	906 knots
Operational Ceiling (rate of climb = 500 f.p.m.) at Combat Weight (23,700 lb.):	
Maximum Thrust, Afterburner Lit ...	52,800 ft.
Climb Thrust, Afterburner Lit ...	51,000 ft.
Maximum Continuous Thrust, Afterburner Lit ...	49,400 ft.
Maximum Rate of Climb at Sea Level at Maximum Take-off Gross Weight (28,200 lb.):	
Maximum Thrust, Afterburner Lit ...	36,000 f.p.m.
Climb Thrust, Afterburner Lit ...	32,900 f.p.m.
Climb Thrust ...	10,750 f.p.m.
Time to 50,000 ft. Altitude from a Standing Start at Sea Level at Maximum Take-off Gross Weight (28,200 lb.):	
Maximum Thrust, Afterburner Lit ...	5.5 min.
Climb Thrust, Afterburner Lit ...	6.3 min.
Take-off Distance over 50 ft. Obstacle with Maximum Thrust, Afterburner Lit, at Sea Level at Maximum Take-off Gross Weight (28,200 lb.) ...	3,300 ft.
Landing Distance over 50 ft. Obstacle at Sea Level at Combat Weight (23,700 lb.) ...	5,800 ft.
True Stalling Speed in Landing Configuration at Sea Level at Combat Weight (23,700 lb.) ...	106 knots
Combat Radius of Action with Combat at 50,000 ft. Altitude:	
High Speed Mission (Table 3) ...	200 naut. mi.
Maximum Range Mission (Table 4) ...	300 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	-	-	-	28,200
B. Taxi and Warm-up	-	4.0	380	27,820
C. Take-off: Maximum Thrust, Afterburner Lit	-	.3	215	27,605
D. Acceleration to Best Climbing Speed: Maximum Thrust, Afterburner Lit	5	.8	570	27,035
E. Climb to 36,090 ft.: Climb Thrust, Afterburner Lit	15	1.6	750	26,285
F. Acceleration to Mach No. = 1.5: Maximum Thrust, Afterburner Lit	23	1.8	629	25,656
G. Climb to 50,000 ft.: Maximum Thrust, Afterburner Lit	41	2.8	782	24,874
H. Cruise-out at 50,000 ft. at Mach No. =1.5	116	8.2	1,650	23,224
I. Combat at 50,000 ft. at Mach No. =1.5	-	5.0	980	20,994*
J. Descent to 40,000 ft.	17	2.2	45	20,949
K. Cruise-back at 40,000 ft.: Economical Cruising Speed	107	11.8	629	20,320
L. Descent to 30,000 ft.	24	3.1	100	20,220
M. Stack at 30,000 ft.: Maximum Endurance Speed	-	15.0	615	19,605
N. Descent to Sea Level	52	6.3	355	19,250
O. Approach: Maximum Endurance Speed	-	5.0	300	18,950
TOTAL:	400	67.9	8,000	

Combat Radius of Action = 200 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	-	-	-	28,200
B. Taxi and Warm-up	-	4.0	380	27,820
C. Take-off: Maximum Thrust	-	.8	168	27,652
D. Acceleration to Best Climbing Speed: Maximum Thrust	7	1.2	292	27,360
E. Climb to 36,090 ft.: Climb Thrust	46	5.0	750	26,610
F. Cruise-out at 36,090 ft.: Economical Cruising Speed	189	20.8	1,399	25,211
G. Accelerate to Mach No. = 1.5 Maximum Thrust, Afterburner Lit	19	1.7	597	24,614
H. Climb to 50,000 ft.: Maximum Thrust, Afterburner Lit	39	2.7	763	23,851
I. Combat at 50,000 ft. at Mach No. =1.5	-	5.0	1,015	21,586*
J. Descent to 40,000 ft.	17	2.2	45	21,541
K. Cruise-back at 40,000 ft.: Economical Cruising Speed	207	22.8	1,221	20,320
L. Descent to 30,000 ft.	24	3.1	100	20,220
M. Stack at 30,000 ft.: Maximum Endurance Speed	-	15.0	615	19,605
N. Descent to Sea Level	52	6.3	355	19,250
O. Approach: Maximum Endurance Speed	-	5.0	300	18,950
TOTAL:	600	95.6	8,000	

Combat Radius of Action = 300 naut. mi. *1,250 lb. of ammunition fired

3.1.2.1.4 Combat Fuel Allowances: The following table sets forth the combat fuel allowances based on 5 minutes at a Mach number of 1.5, together with 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 (23,700 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	1,020	72	-	-	805
45,000	1,160	72	-	220	571
40,000	1,380	72	197	360	488
36,090	1,605	72	238	450	454
20,000	3,090	77	362	800	635

3.1.2.1.5 The performance specified herein is based on estimated specific fuel consumption of the TR9 engine 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.

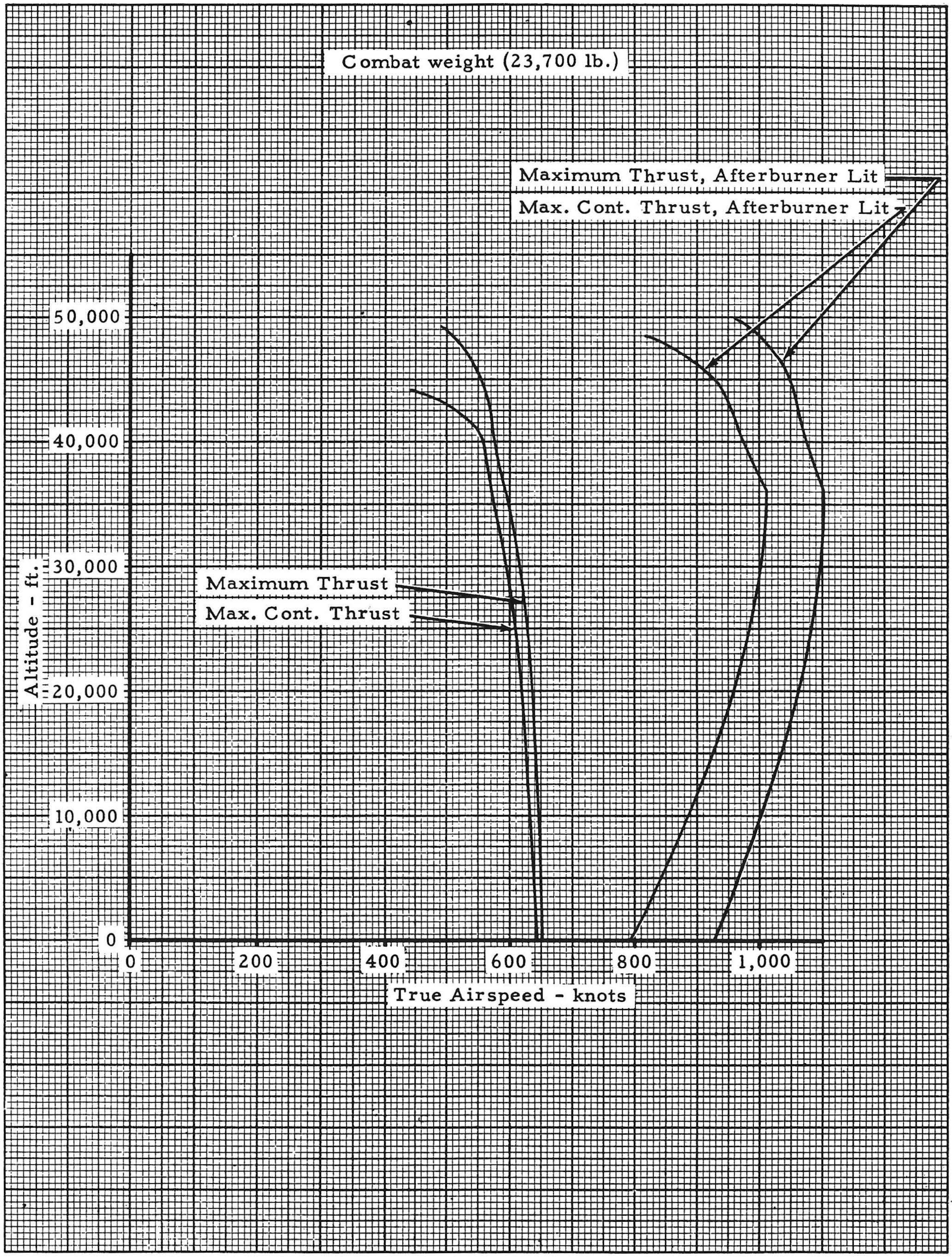


Fig. 2 Level Flight True Airspeed

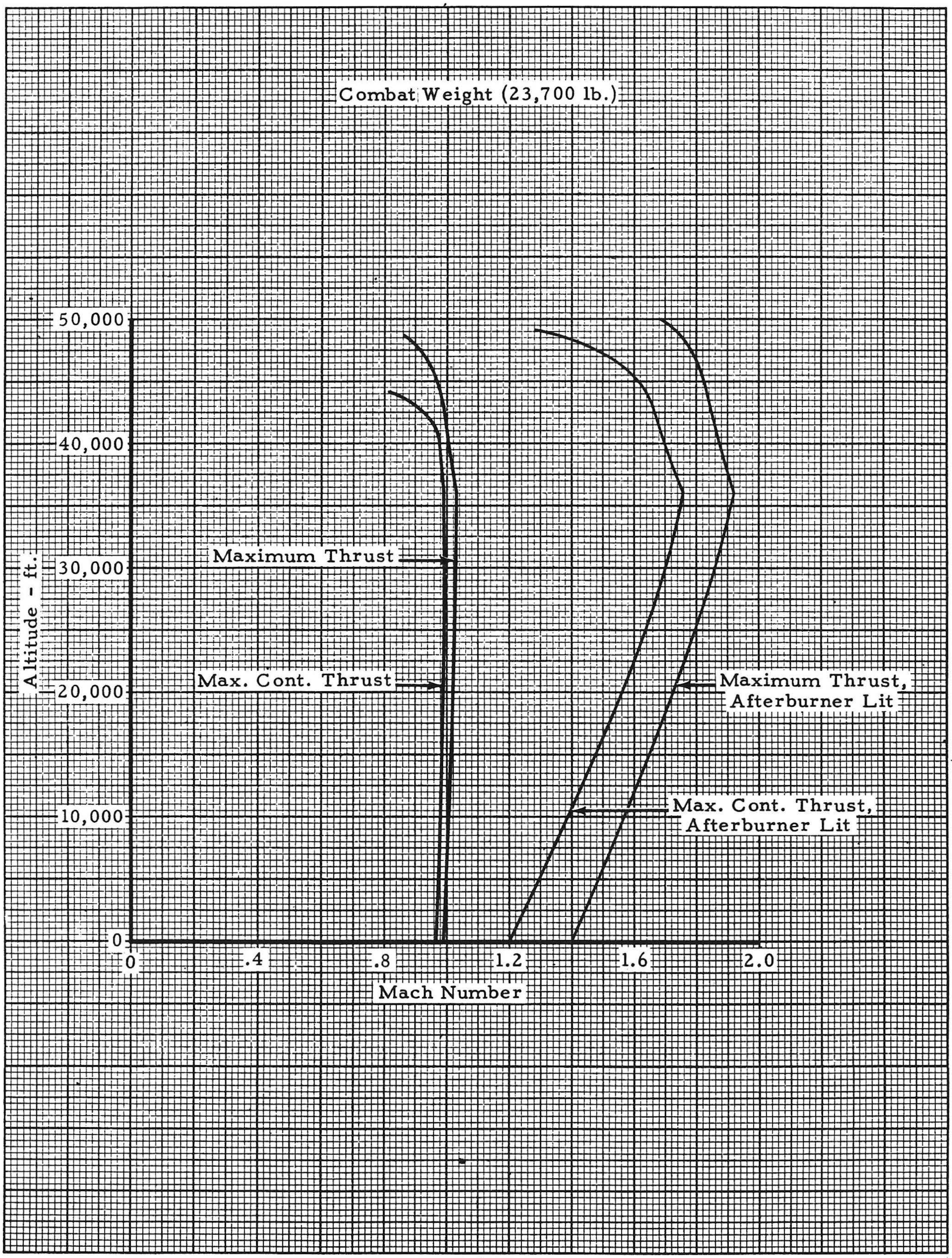


Fig. 3 Level Flight Mach Number

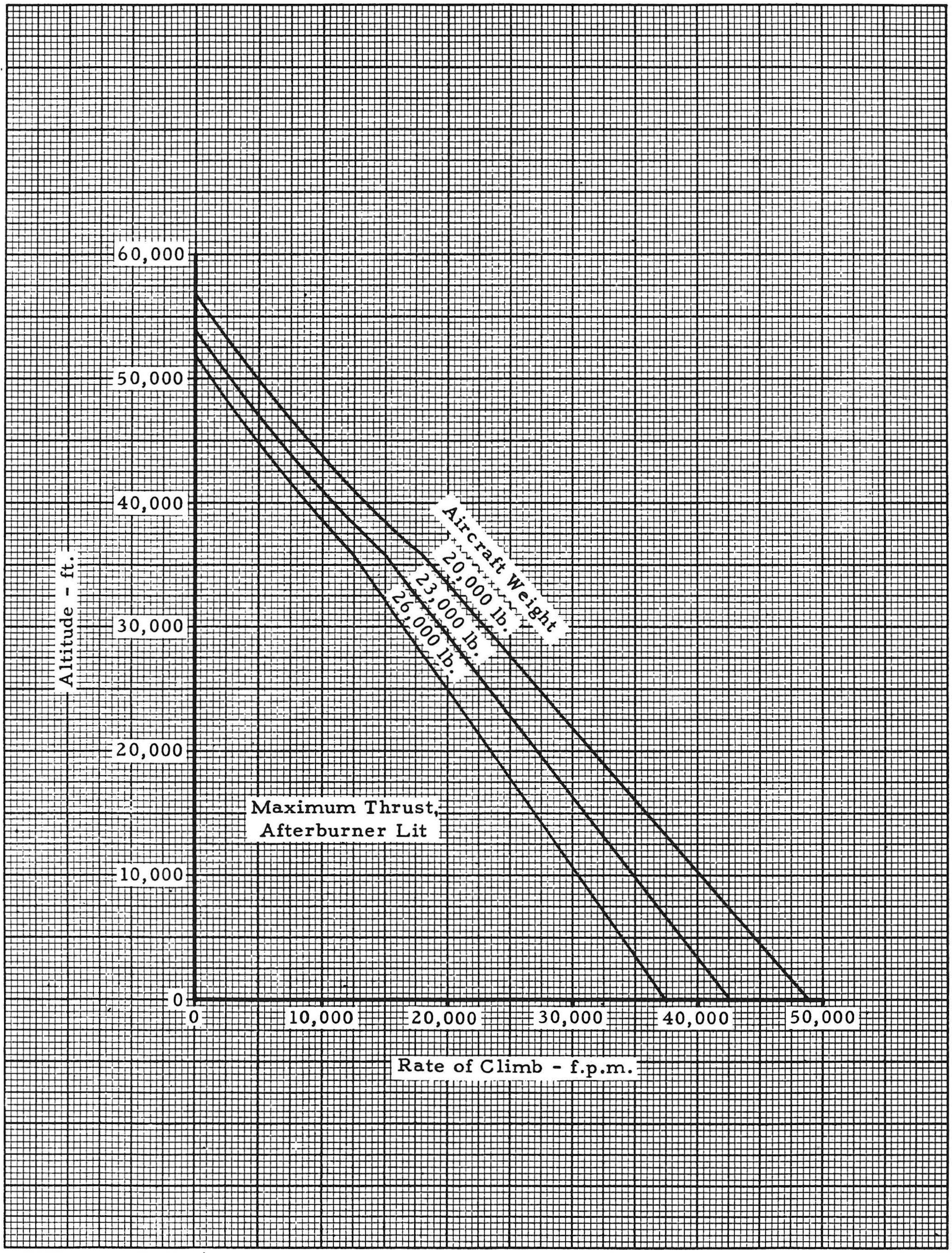


Fig. 4 . Maximum Steady Rate of Climb - Maximum Thrust

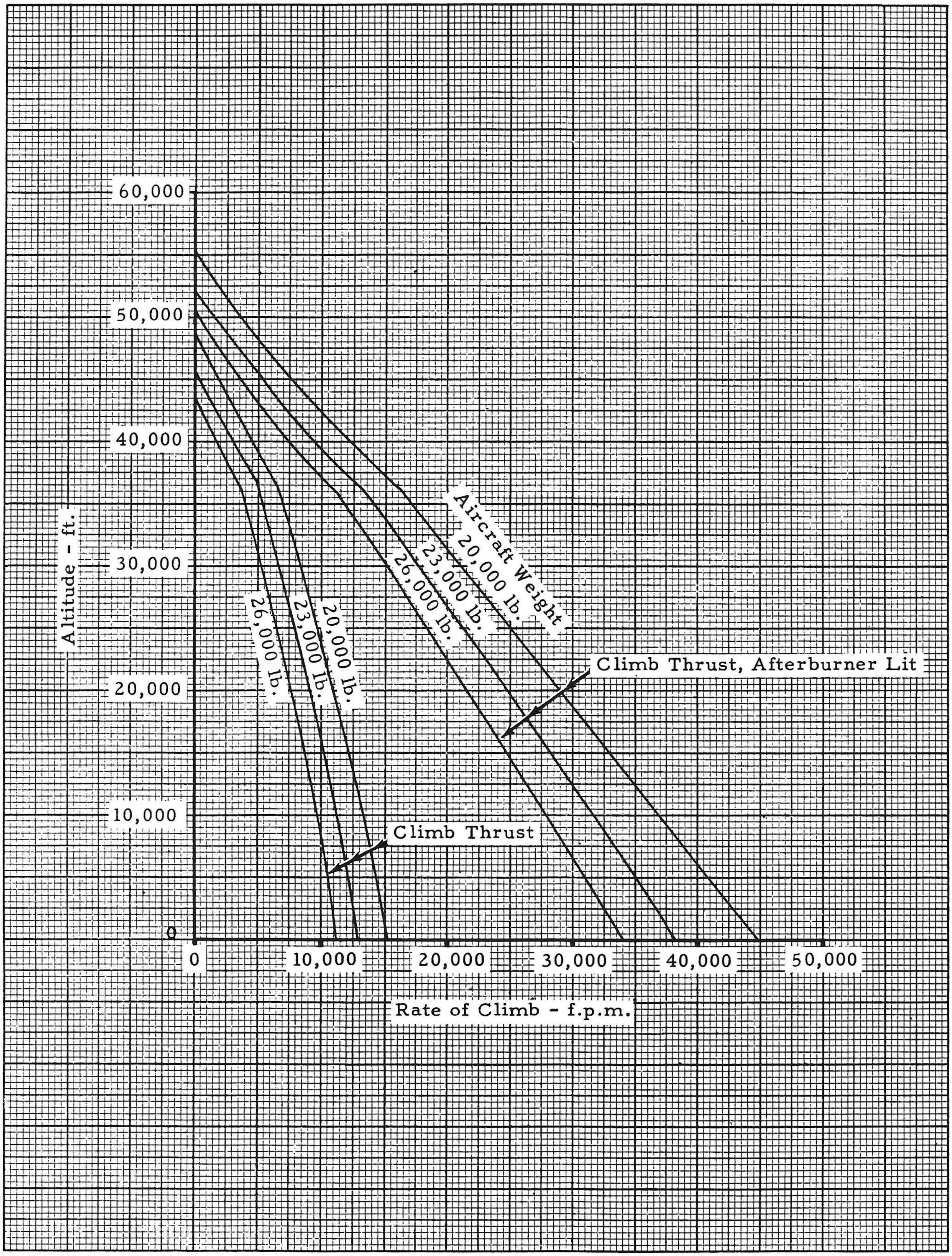


Fig. 5 Maximum Steady Rate of Climb - Climb Thrust

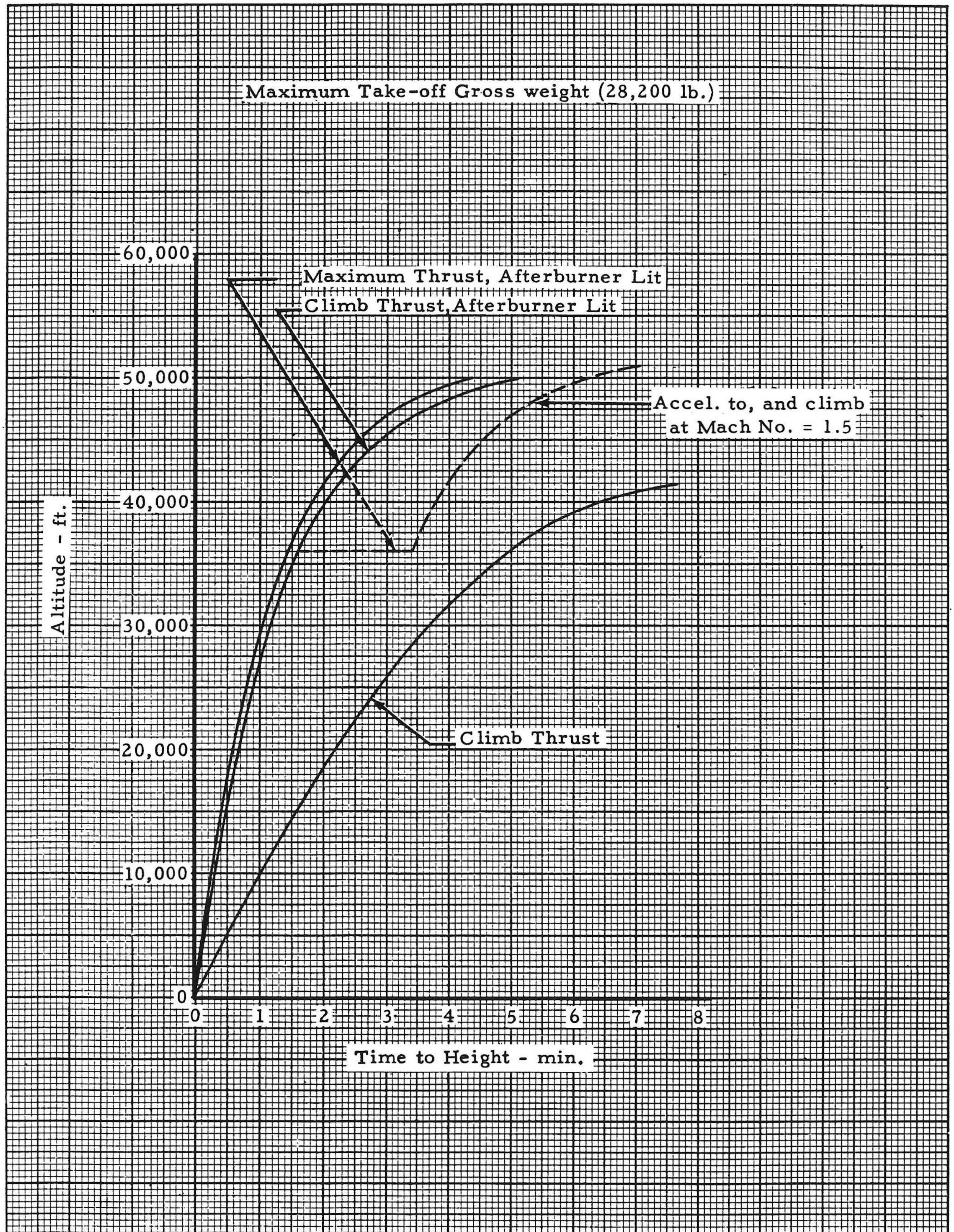


Fig. 6 Time to Height

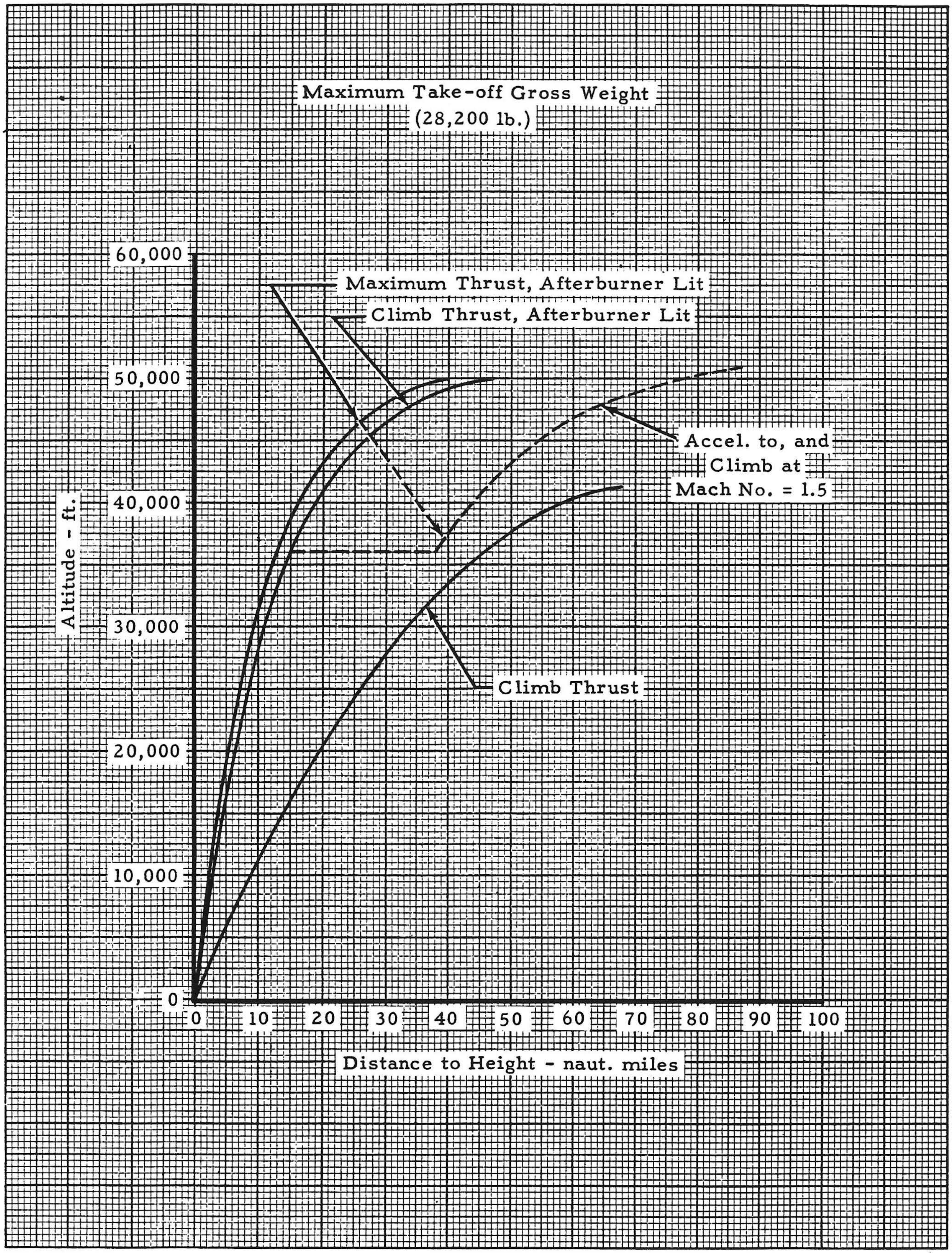


Fig. 7 Distance to Height

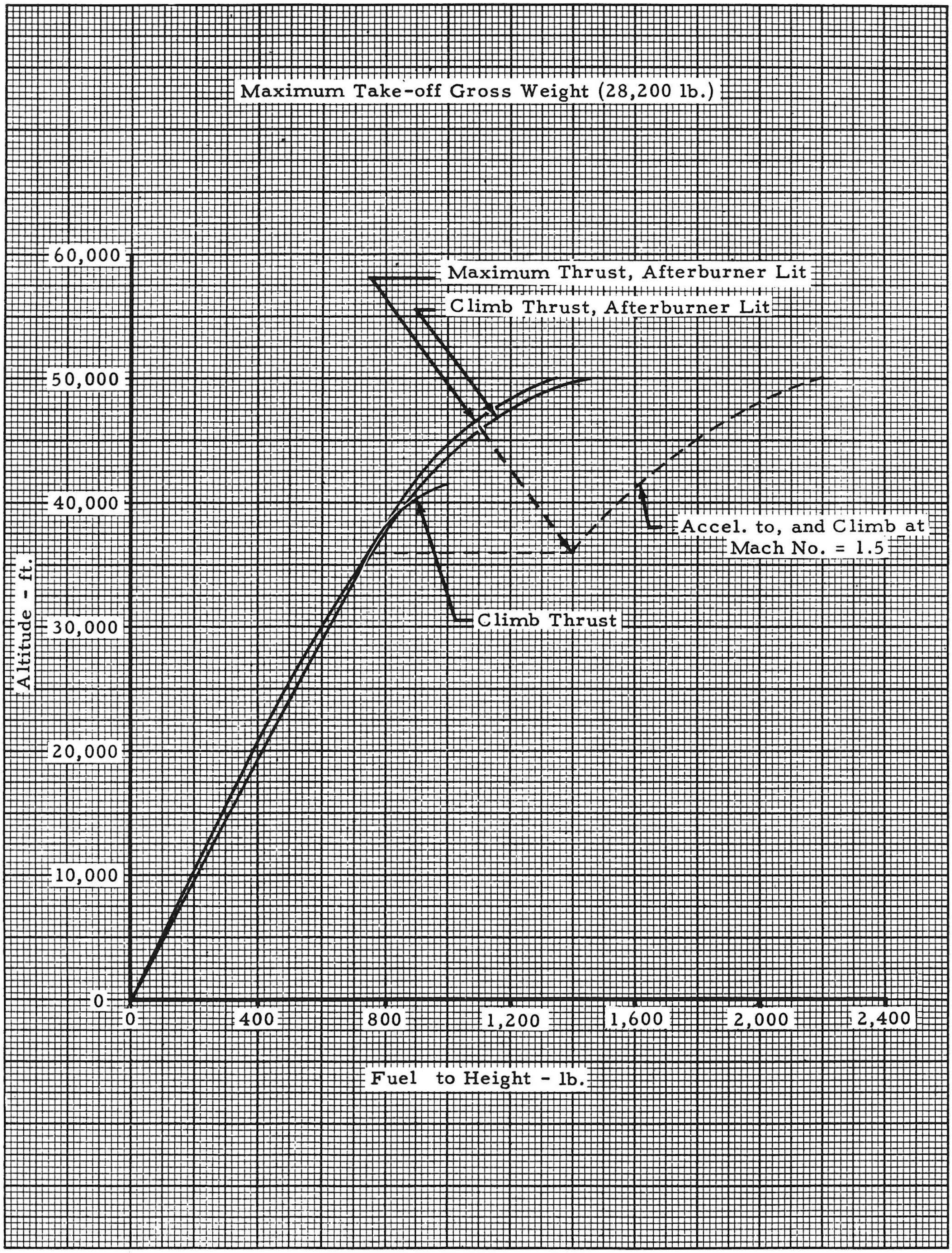


Fig. 8 Fuel to Height

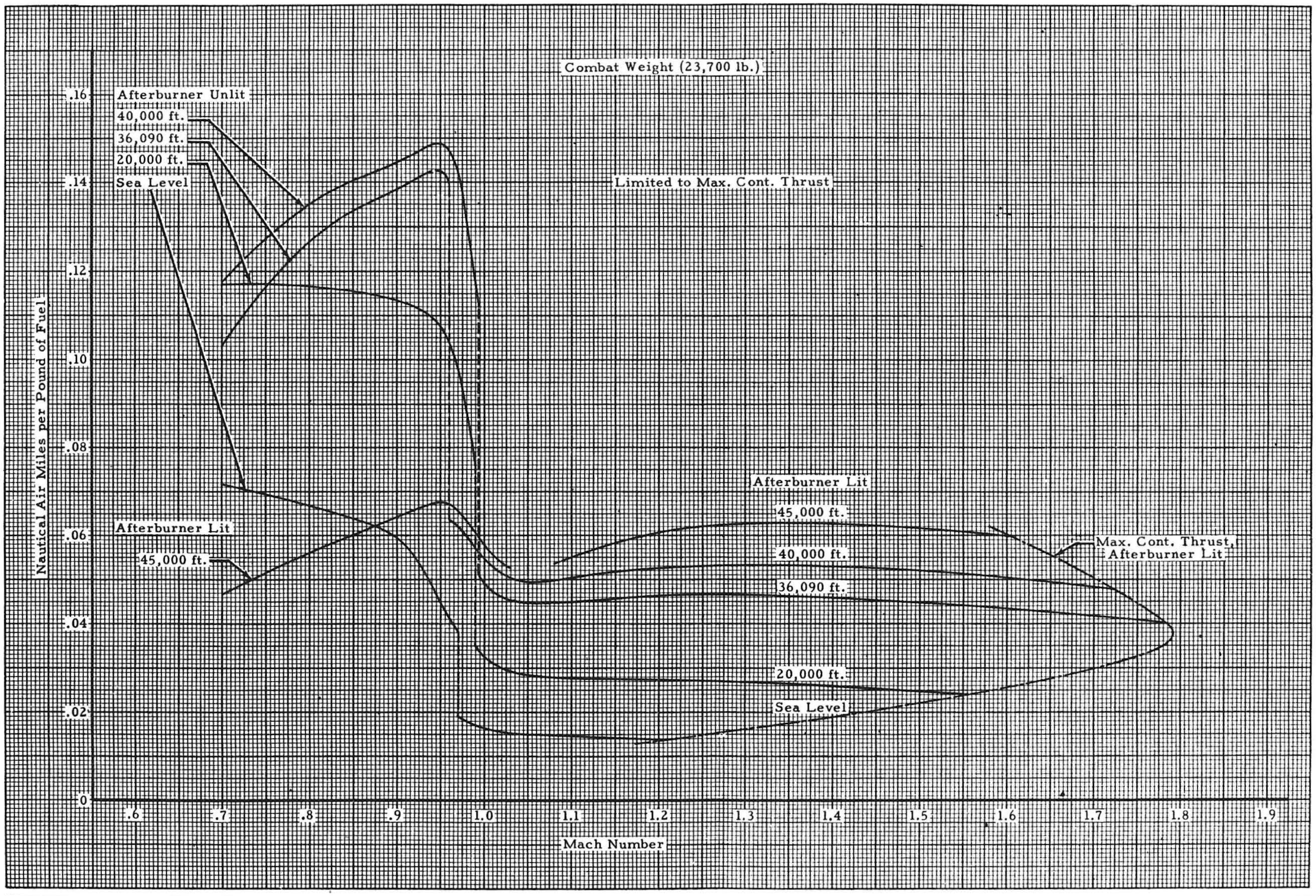


Fig. 9 Nautical Air Miles per Pound of Fuel

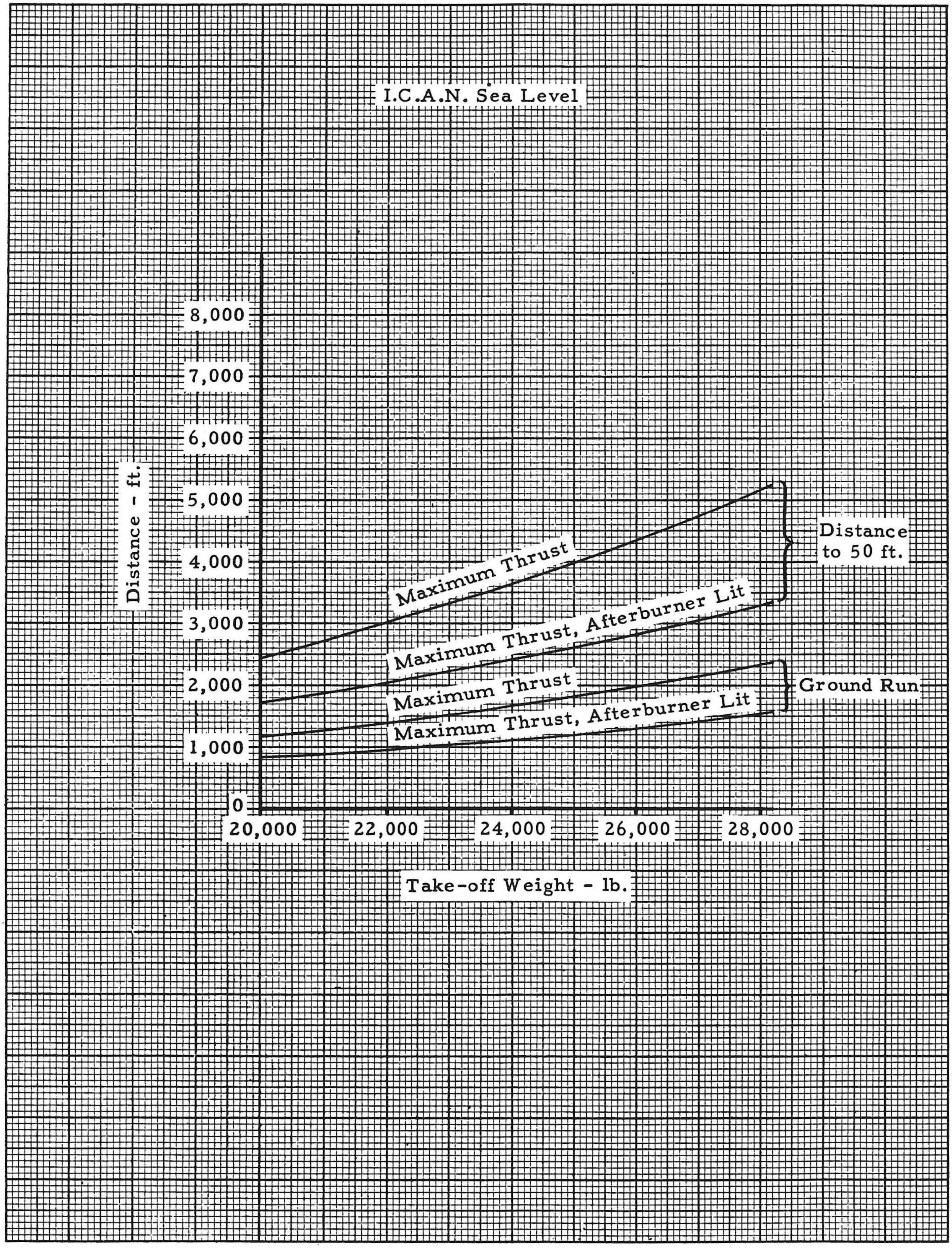


Fig. 10 Take-off Distance

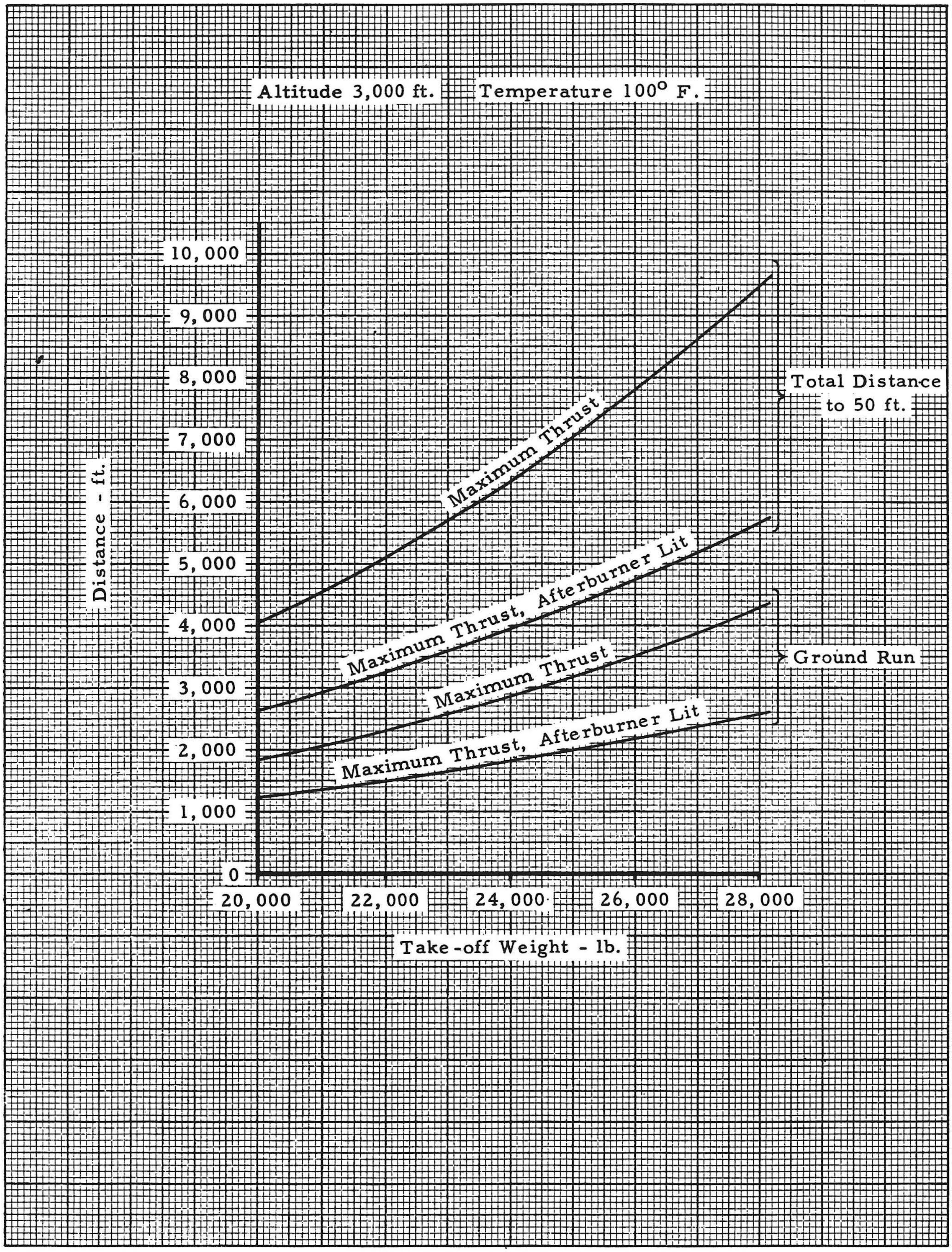


Fig. 11 Take-off Distance

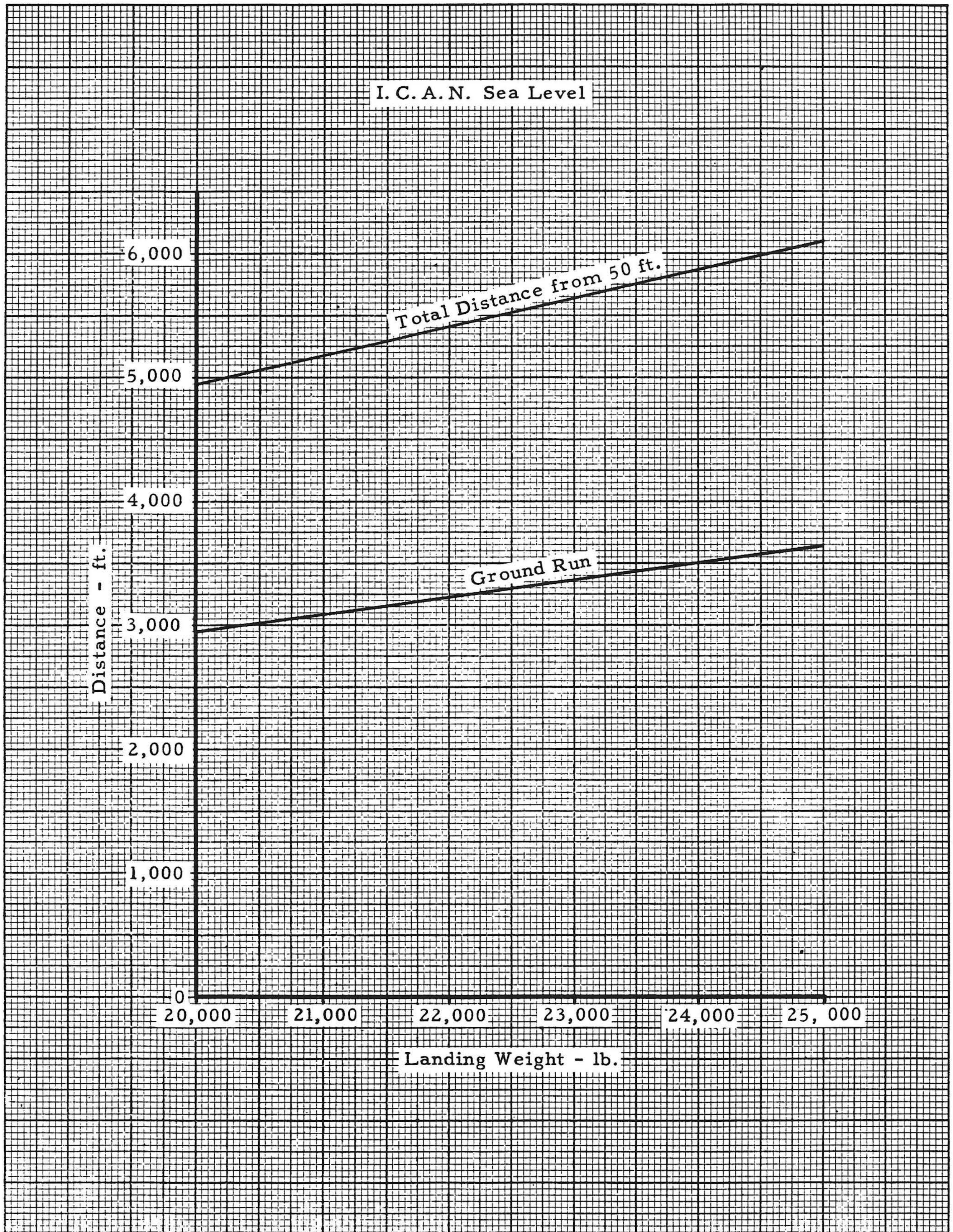


Fig. 12 Landing Distance

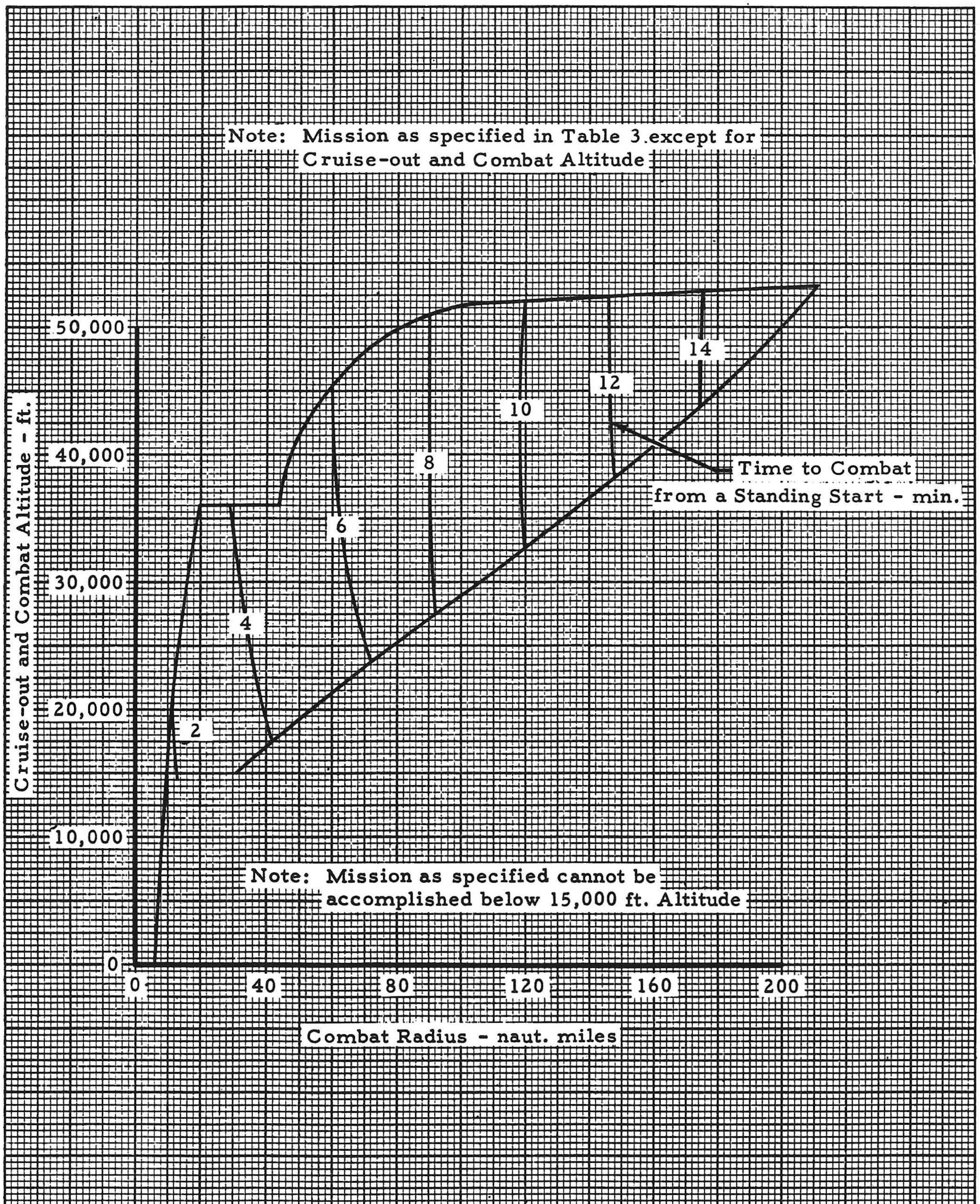


Fig. 13 Combat Radius of Action - High Speed Mission

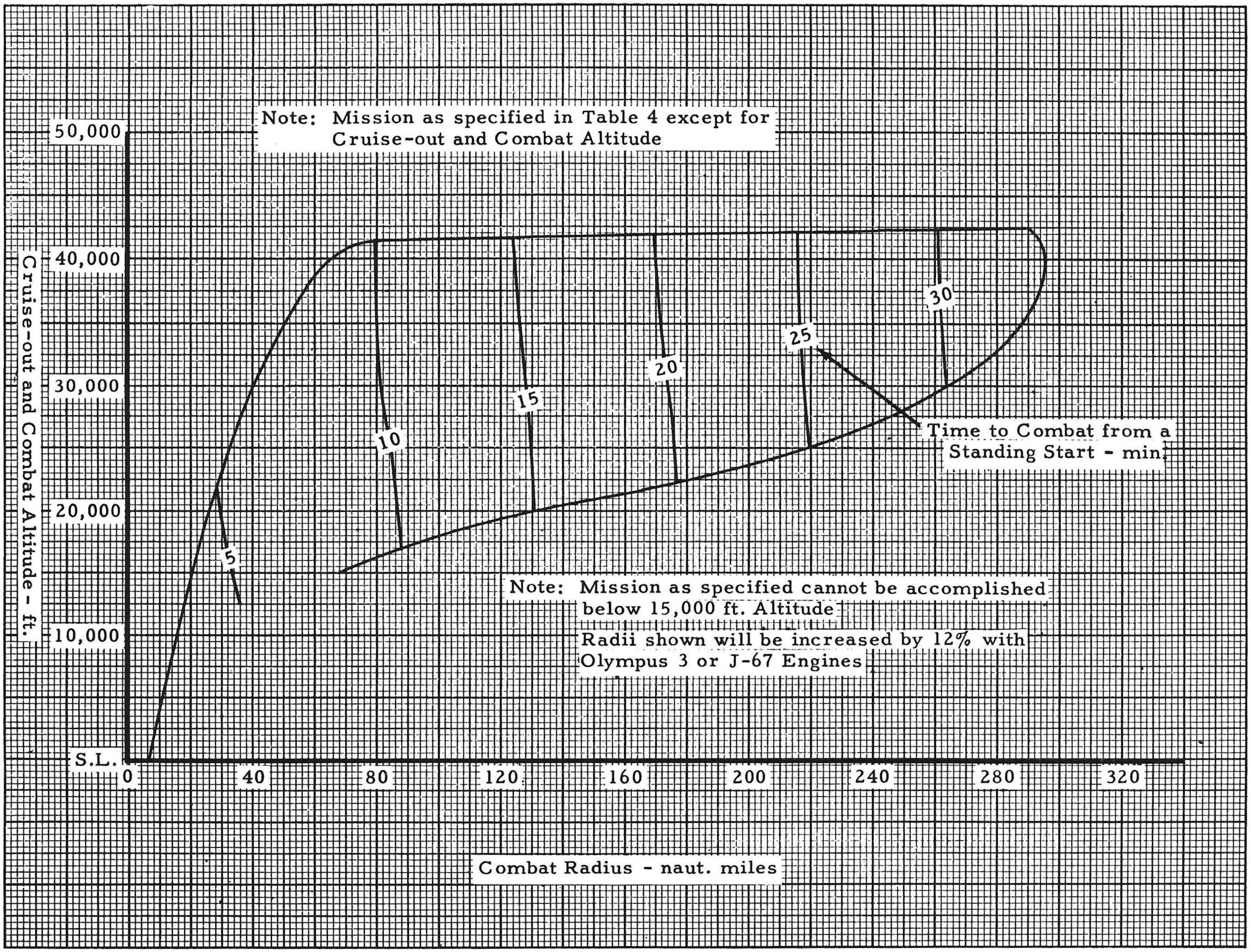


Fig. 14 Combat Radius of Action-Maximum Range Mission

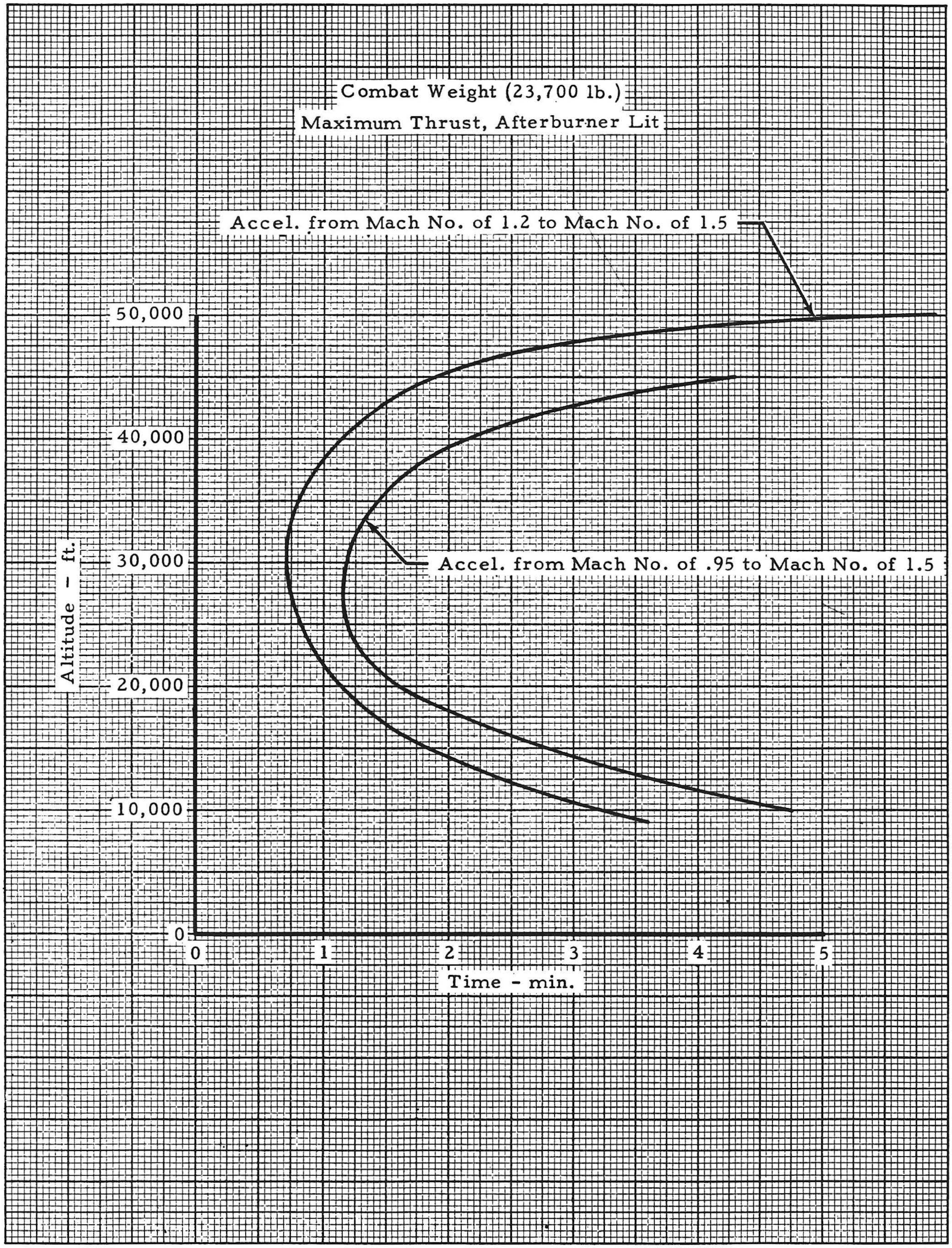


Fig. 15 Acceleration Time in Level Flight

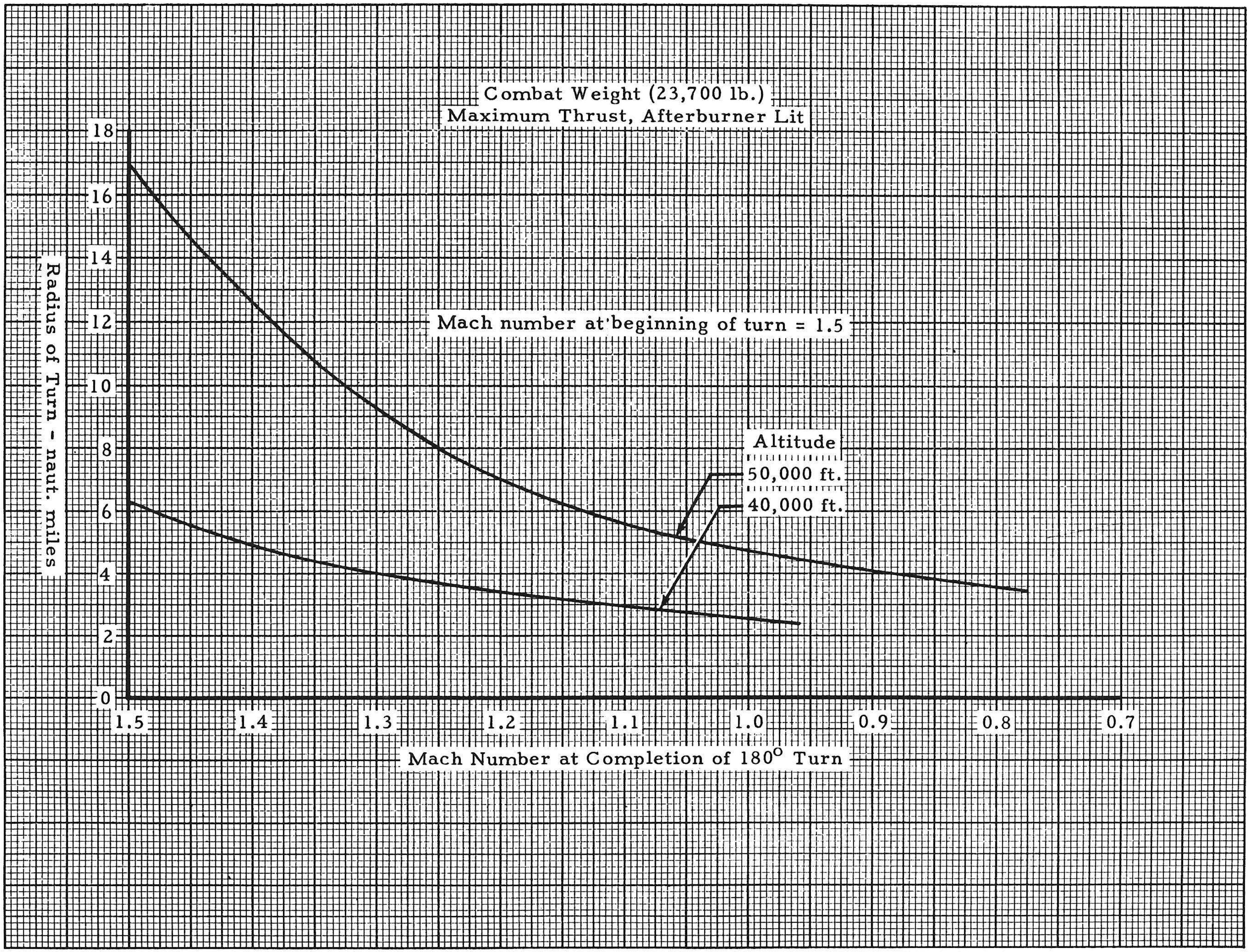


Fig. 16 Decelerating Turns in Level Flight

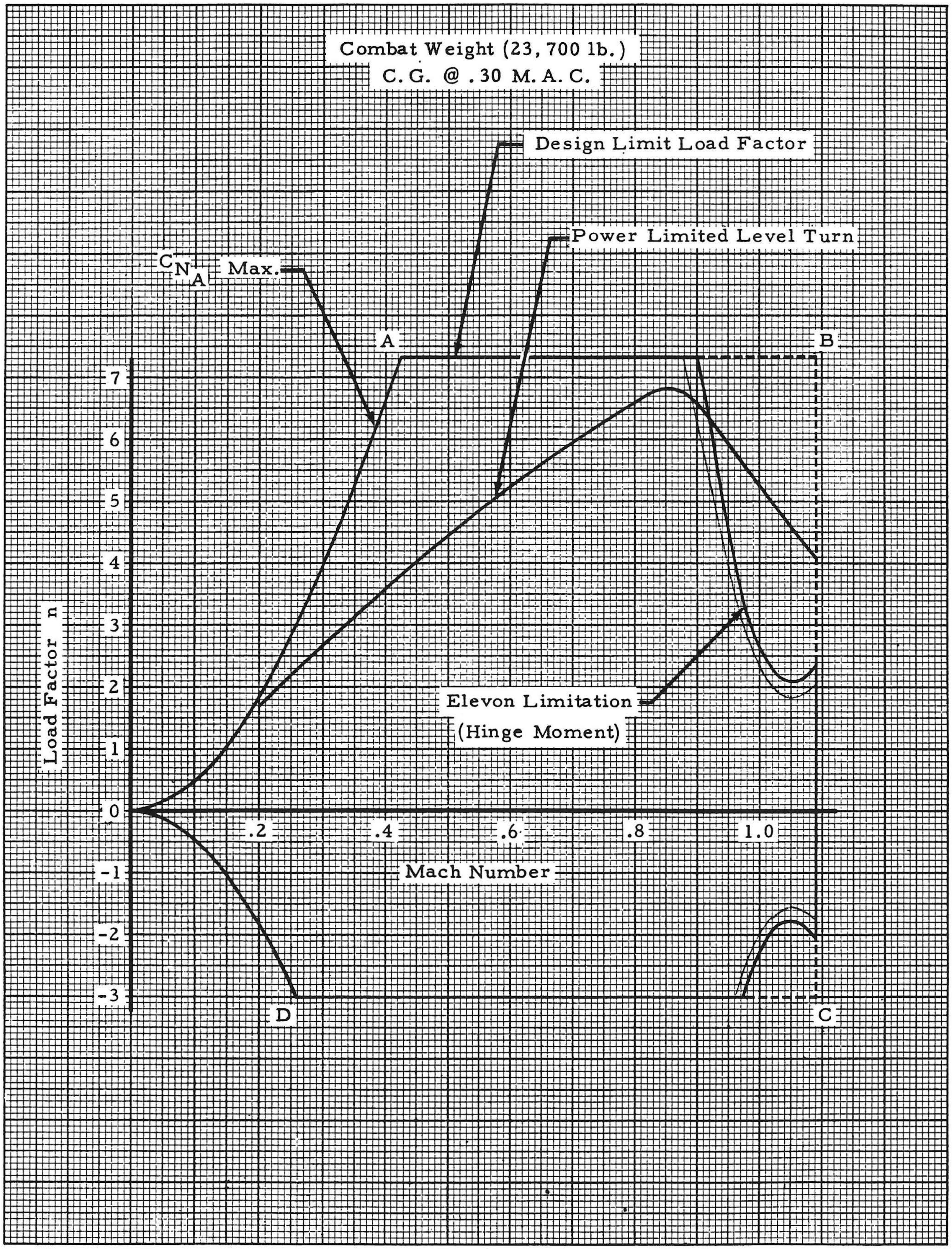


Fig. 17 Performance Flight Envelope at Sea Level

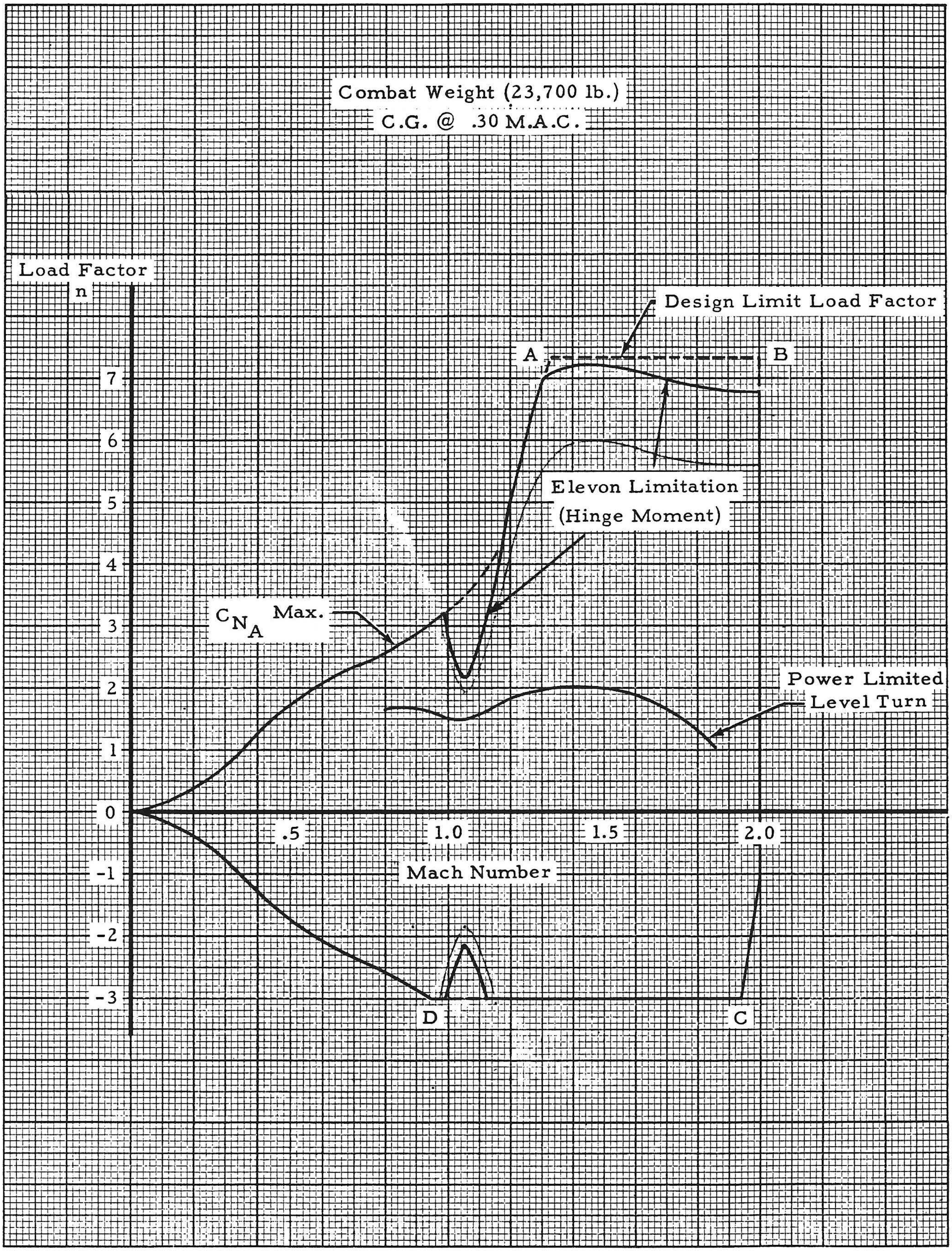


 Fig. 18 Performance Flight Envelope at 40,000 ft. Altitude

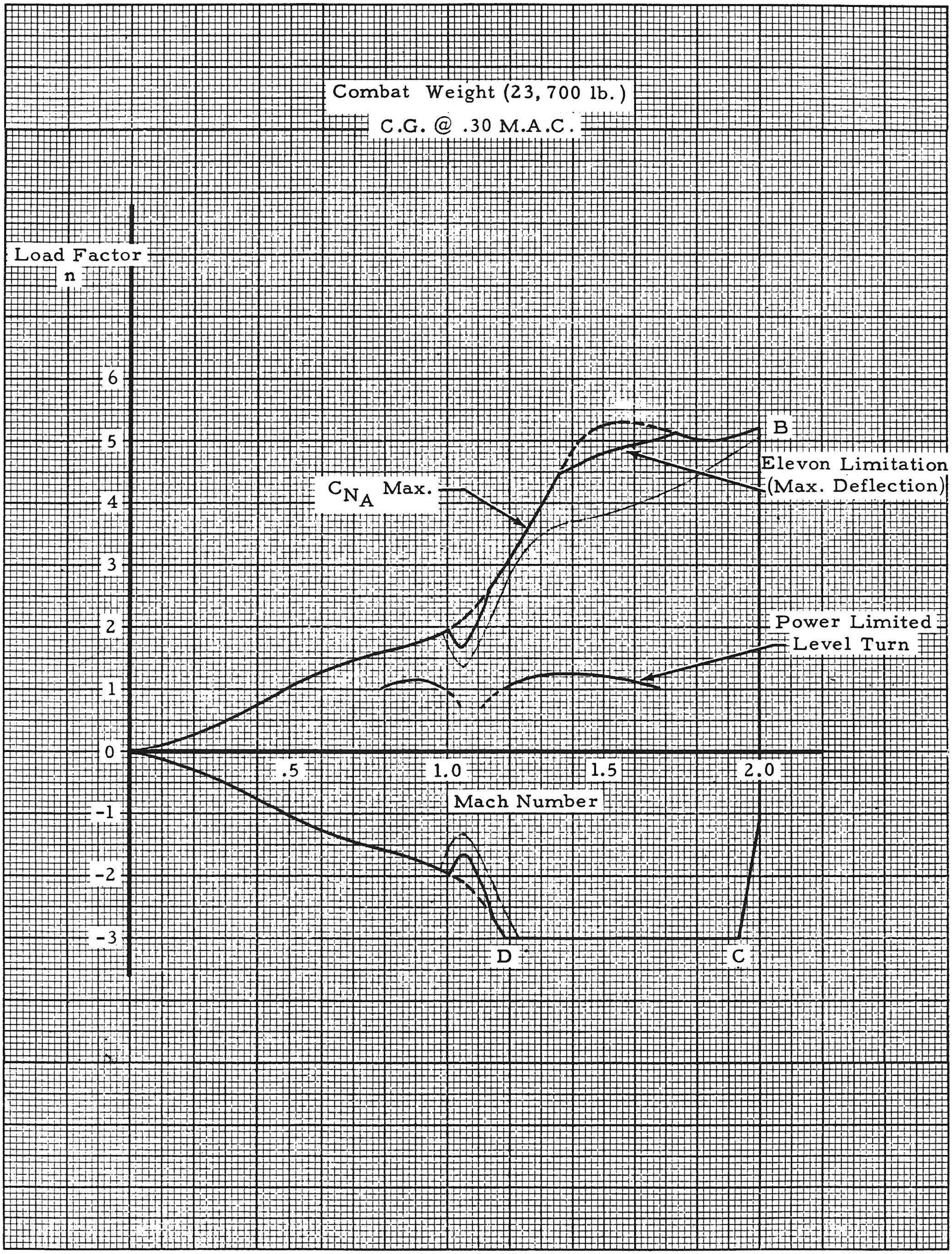


 Fig. 19 Performance Flight Envelope at 50,000 ft. Altitude

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 end of this brochure.

TABLE 6 - STRUCTURE AND POWERPLANT

Item	Weight (lb.)
WING GROUP:	
Wing - including elevons	4,060
TAIL GROUP:	
Fin and rudder	525
BODY GROUP:	
Fuselage - including engine rails and rear nacelle	3,000
Speed brakes	120
LANDING GEAR:	
Main landing gear - including jacks	900
Nose landing gear - including jacks and steering unit	295
Tail skid	10
ENGINE SECTION:	
Shrouds and firewalls	210
Engine mounting	20
POWER PLANT GROUP:	
Engine, complete	3,772
Engine controls	10
FUEL SYSTEM:	
Tanks	217
Piping, pumps, etc.	250
ENGINE DE-ICING SYSTEM	95
AFTERBURNER - COMPLETE	936
TOTAL:	14,420

TABLE 7 - FIXED EQUIPMENT GROUP

Item	Weight (lb.)
INSTRUMENTS	46
SURFACE CONTROLS:	
for elevons, rudder, speed brakes - including jacks and artificial feel	400
HYDRAULIC SYSTEM	599
ELECTRICAL SYSTEM	550
HUGHES ELECTRONIC EQUIPMENT - MX 1179:	
Radome equipment	210
Cockpit equipment	190
Main equipment bay	1,400
ARMAMENT PROVISIONS	200
FURNISHINGS:	
Ejector seat	132
Emergency accommodations	15
Oxygen	20
AIR CONDITIONING AND PNEUMATIC SYSTEMS	215
ANTI-ICING EQUIPMENT	115
BRAKE PARACHUTE	45
EXTERIOR FINISH	43
TOTAL:	4,180
TOTAL BROUGHT FWD:	14,420
WEIGHT EMPTY:	18,600

TABLE 8 - UNIT WEIGHTS

Component	Unit Weight
WING GROUP (Gross Area 617.5 sq. ft.) lbs. per sq. ft.	6.57
TAIL GROUP (Gross Area 81.6 sq. ft.) lbs. per sq. ft.	6.43
WEIGHT OF FUEL SYSTEM PER GAL. CAP. (980 Imperial gal. of fuel)	.477

TABLE 9 - USEFUL LOAD

Item	Weight (lb.)
CREW - ONE (1) PILOT	207
FUEL:	
Usable	8,000
Total residual	123
OIL	20
ARMAMENT:	
Guided missiles (6)	660
Rockets (24) - including jettisonable container	590
USEFUL LOAD:	9,600
WEIGHT EMPTY:	18,600
GROSS WEIGHT:	28,200

TABLE 10 - DESIGN INFORMATION

Length - max.	58 ft. 10 in.
Height - max.	18 ft. 6 in.
Span	32 ft. 6 in.
Thickness - root chord	.03
Thickness - tip chord	.03
Wing area - net	436.15 sq.ft.
Taper ratio (Root chord/tip chord)	11.65:1
Length - root chord	35 ft. 0 in.
Length - tip chord	3 ft. 0 in.
Maximum fuselage depth	6 ft. 3 in.
Maximum fuselage width	6 ft. 4 in.

Design gross weight	28,200 lb.
Stressing weight and load factor	
At combat weight	23,700 lb.
Ultimate load factor	11.00 g.
Limit load factor	7.33 g.
Factor of safety	1.50
Airplane weight immediately after take-off	27,000 lb.
Ultimate load factor	9.67 g.
Limit load factor	7.33 g.
Factor of safety	1.33

3.1.3.1 Alternate Loading: Alternate loading arrangements are not applicable.

3.1.3.2 Gross weight estimation is as follows:-
Design gross weight .... 28,200 lb.

3.1.4 Centre of Gravity Locations:
Design gross weight c.g. location, wheels up:
Aft l.e. of m.a.c. .... 28.01% m.a.c.
Above l.e. of m.a.c. .... 1.28 ft.

Design gross weight, c.g. location, wheels down:	
Aft l.e. of m.a.c.	28.27% m.a.c.
Above l.e. of m.a.c.	1.12 ft.
Extreme forward position, c.g. possible in flight regardless of loading at take-off - wheels up:	
Aft l.e. of m.a.c.	27.29% m.a.c.
Above l.e. of m.a.c.	1.55 ft.
Gross weight for this condition	20,200 lb.
Extreme rearward position, c.g. possible in flight regardless of loading at take-off - wheels down:	
Aft l.e. of m.a.c.	30.86% m.a.c.
Above l.e. of m.a.c.	1.41 ft.
Gross weight for this condition	18,950 lb.

3.1.5 Areas:

Wing area, total, including elevons	617.5 sq. ft.
Elevon area aft of hinge line (each)	43.8 sq. ft.
Vertical tail area, total	81.6 sq. ft.
Fin - to rudder hinge	61.5 sq. ft.
Rudder, aft of hinge	20.1 sq. ft.
Speed brakes:	
Upper (each) - projected area	6.05 sq. ft.
Lower (one) - projected area	12.5 sq. ft.

NOTE: The rudder and elevons are power controlled and do not incorporate aerodynamic balance aids.

3.1.6 Dimensions and General Data:

Wings:	
Span, maximum	32 ft. 6 in.
Chord:	
At root	35 ft. 0 in.
At construction tip (theoretical extended section at tip)	3 ft. 0 in.
Mean aerodynamic	23 ft. 5.89 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	0 deg.
At construction tip	0 deg.
Sweepback at 25% chord	52 deg. 15 min.
Dihedral	1 deg. 45 min.
Aspect ratio	1.71
Elevons:	
Span (each)	13 ft. 6 in.
Chord (average percent wing chord)	20%
Speed brakes (fuselage):	
Span - upper (each)	2 ft. 3 in.
Span - lower	5 ft. 6 in.
Chord - upper	2 ft. 9 in.
Chord - lower	2 ft. 3 in.
Location:	
Upper	One each side of fin on fuselage, above wing
Lower	One below fin on lower surface of fuselage
Tail - vertical:	
Airfoil section designation and thickness	NACA (Modified) 0003-63
Sweep of leading edge	60 deg. 45 min.
Aspect ratio	1.71
Height over highest fixed part of airplane - fin ..	18 ft. 6 in.
Height - wing tip	5 ft. 6 in.
Height - to top of cockpit	12 ft. 7 in.

NOTE: The above heights are taken with the airplane in its normal 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	58 ft. 10 in.
Three point attitude	59 ft. 6 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 ...	14 ft. 2 in.
Angle between reference line and wing zero lift line ...	0 deg.
Ground angle ...	3 deg.
Wheel rim size:	
Main wheels ...	29 x 7.7
Nose wheel ...	18 x 5.5
Tire size:	
Main wheels ...	29 x 7.7
Nose wheel ...	18 x 5.5
Tread of main wheels ...	12 ft. 2 in.
Wheel base ...	22 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 1W (front elevation) ...	81 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 1W (side elevation) ...	15 deg.

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 11

Surfaces	Control	Movement
Rudder	Surface	15 deg. RIGHT, 15 deg. LEFT
-	Pedals	According to USAF Spec. AMC80-1
Elevons:		
Pitch control	Surface	20 deg. UP 20 deg. DOWN
Roll control	Surface	7 deg. UP 7 deg. DOWN
Maximum deflection	Surface	27 deg. UP 27 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, power plant 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	Power Plant Installation

In order to achieve the quoted performance of the airplane it has been essential to keep the weight of the equipment to a minimum consistent with reasonable safety and, furthermore, to house it within the smallest possible fuselage. Special attention has been given to ease of access for those installations which require frequent servicing; in particular, this applies to the electronics. The proposed arrangement is compact, yet orderly, and all equipment can be serviced from outside the fuselage. Further information on this latter subject is given in later paragraphs.

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 powerplant and equipment, and ease of general maintenance. Special attention shall be 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 five main sections. These comprise:-

(a) The radome

(b) The cockpit section

(c) The armament and equipment section

(d) The power plant section, and

(e) The rear nacelle section.

Item (d) forms the largest section and is approximately 21 feet long by 5.5 feet wide.

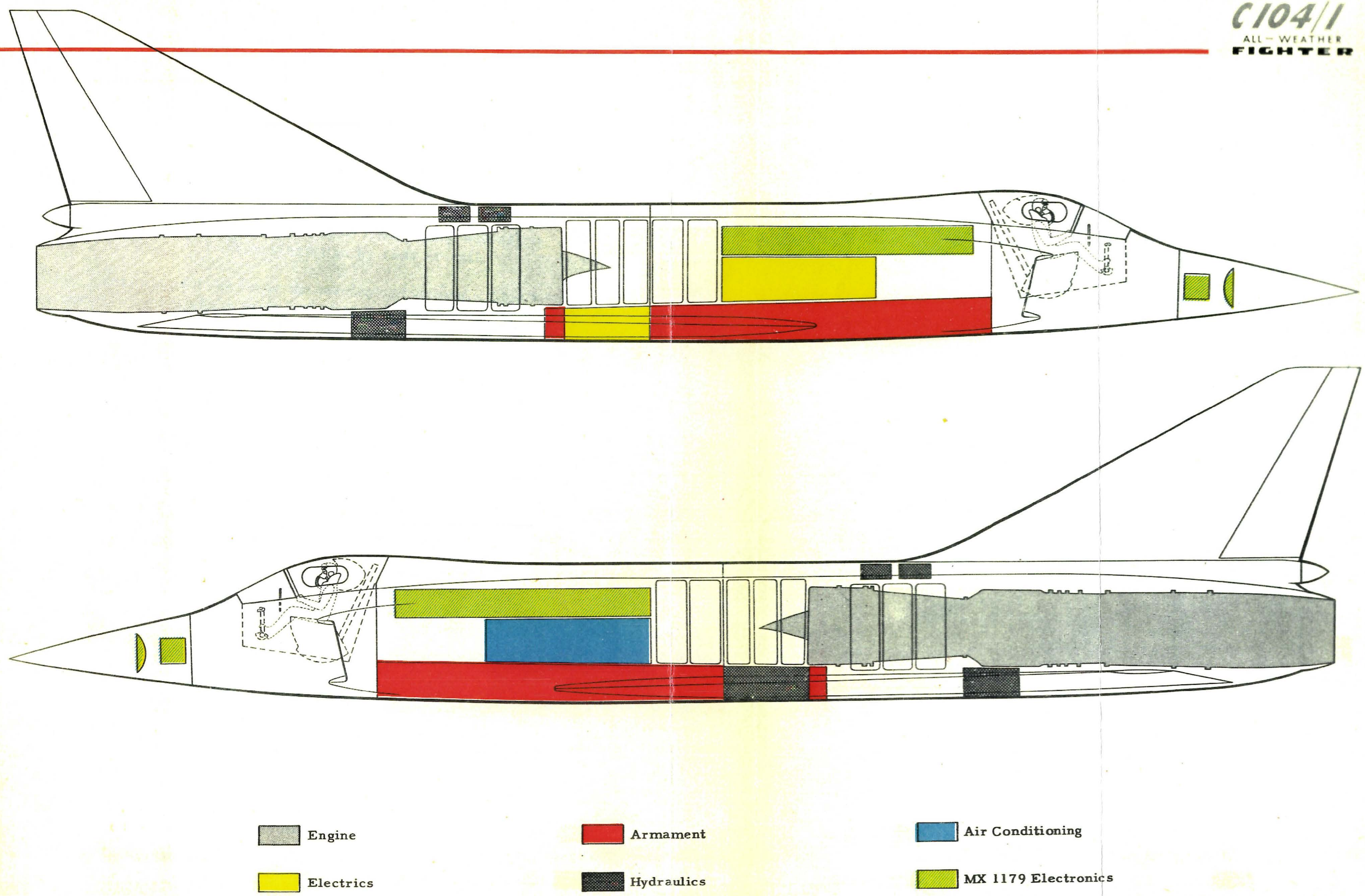


FIG. 20 EQUIPMENT ARRANGEMENT

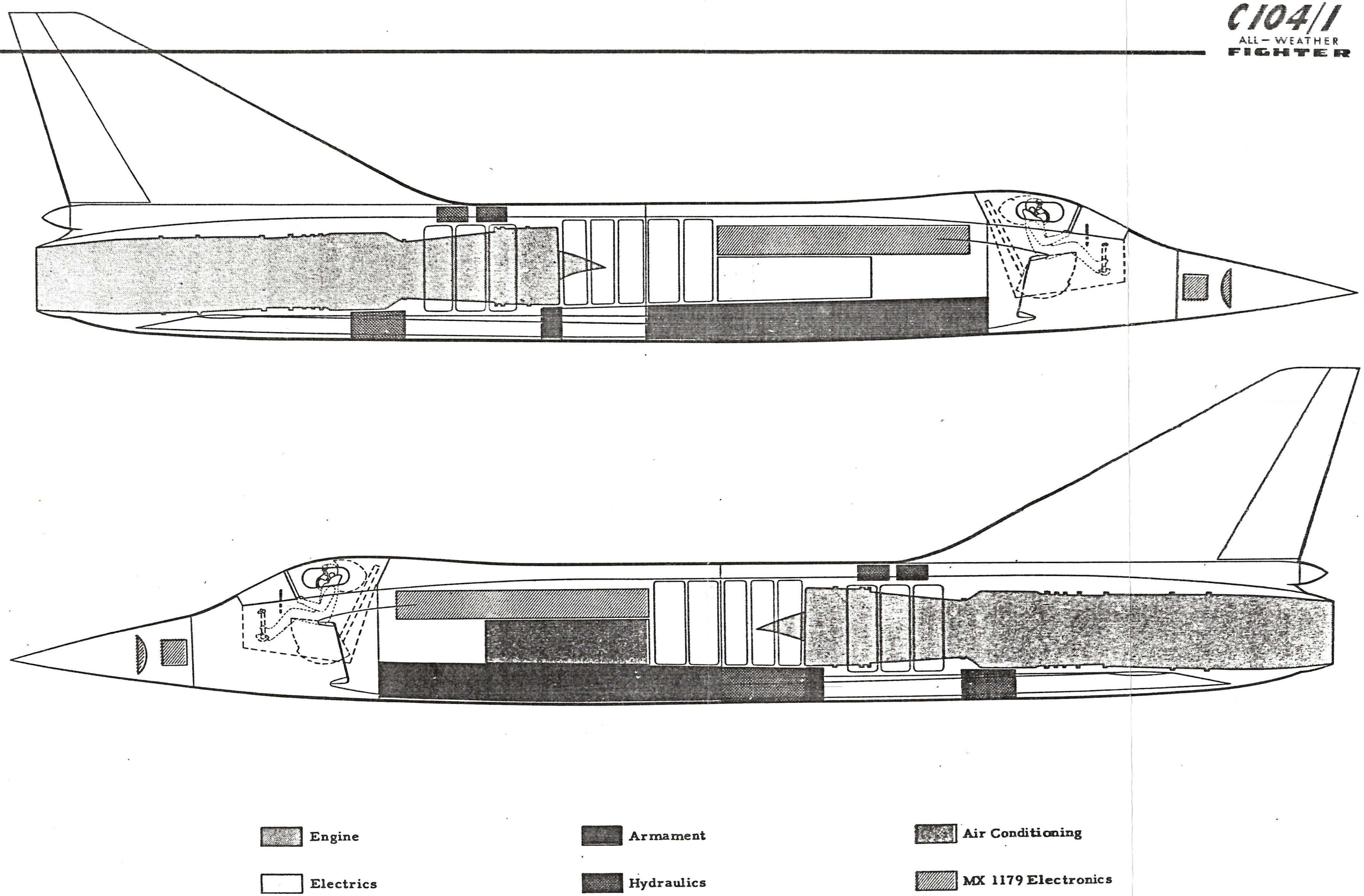


FIG. 20 EQUIPMENT ARRANGEMENT

The wing structure, which passes below the fuselage, is divided into two separate parts joined by bolted fittings at the center line of the airplane. Each wing half incorporates a detachable wing tip; additionally, a single-piece elevon is hinged to each wing rear spar. The dimensions of a half wing are approximately 14.5 feet by 27 feet, when resting on its leading edge with elevon and wing tip detached.

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

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

3.2.5 Interchangeability and Replaceability: The component parts of all aircraft of the same model, exclusive of experimental and service test aircraft, shall be interchangeable or replaceable in accordance with and to the extent required by Spec. AN-1-21, 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 all temperature 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 aircraft 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 specifically 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 Installation: 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

Refer to paragraph 3.18 for detailed descriptions.

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 as specified in the preceding sub-paragraph. An additional function of the equipment is armament firing control. Details of this latter procedure are outlined in sub-paragraph 3.18.

3.2.13.2.2 Interception navigation is effected by referring the G.C.I. broadcast data to the aircraft 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 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 a 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 computor gives the necessary instructions to reach a marshalling area - defined by its co-ordinates in the memory of the machine. 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 computor and applied, from the severage point onwards, to the A.P.I. without change. The error involved by this method is often quite small. The path of the target may also be computed from G.C.I. data stored magnetically in the event of failure of this link; reasonable accuracy is maintained 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

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

Landing is accomplished automatically.

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

Item	Reference Para
Turbo-jet engine and after-burner	3.12
Emergency ram air turbine	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 trans-sonic 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 validity 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	.00405	41.4
Body - including canopy and side air intakes	.00394	40.1
Vertical tail	.00067	6.8
Armament	.00029	3.0
Interference	.00085	8.7
TOTAL:	.0098	100.0

The aerodynamic cleanness and cleanness ratio of several jet airplanes - including the C104/1 - 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
C104/1	.00316	1.40

The aerodynamic cleanness required for the C104/1 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 C104/1 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 AERD 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

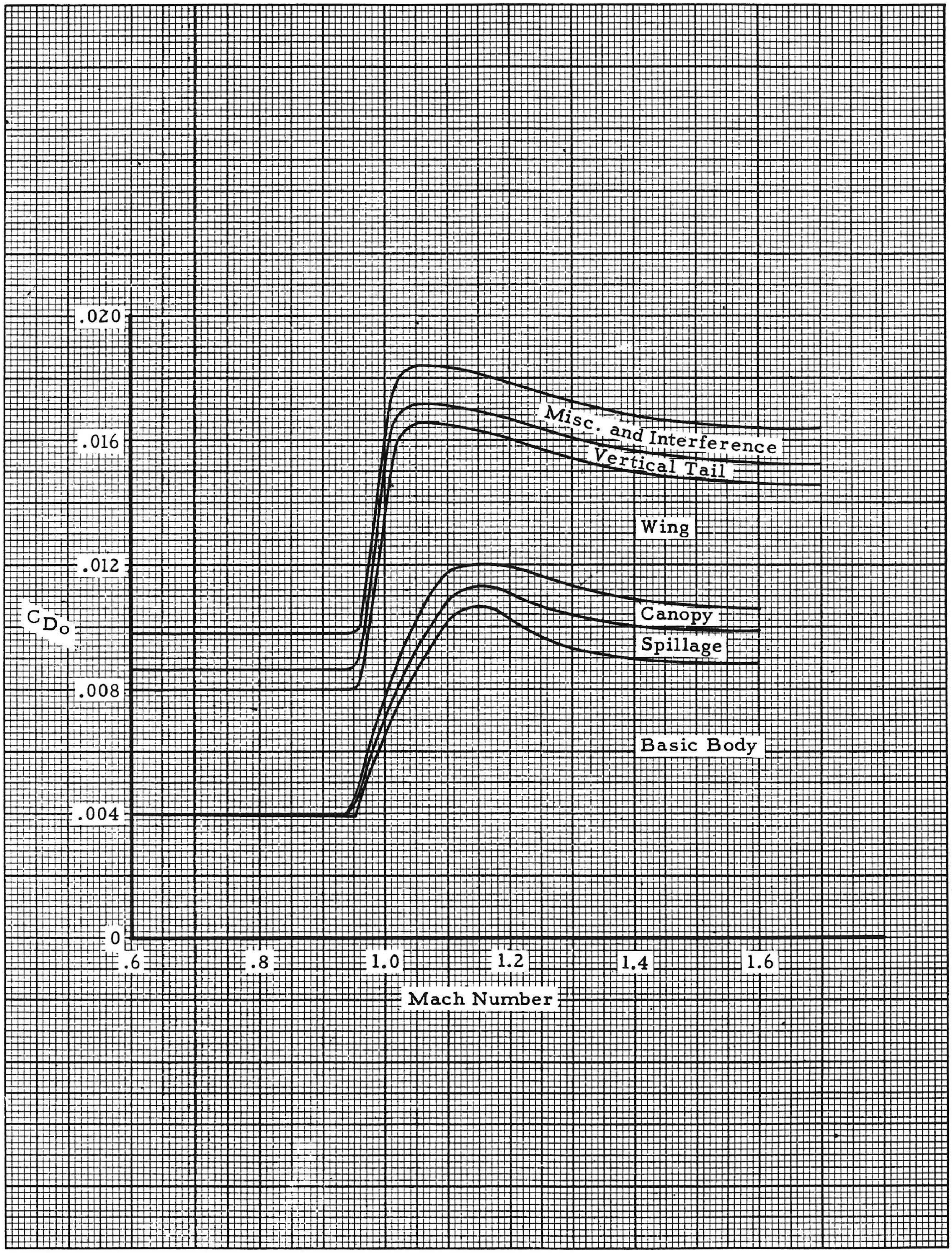


Fig. 21 Profile Drag Coefficient

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).

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 Elevon 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 curve 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 elevons, 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 L9I30: 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. 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. Losses are not expected to exceed 3% of the total head. Both intakes run together aft of the cockpit but are separated by the deep structural keel member up to a point just in front of the engine. At this position the two passages are joined together, the combined air flow being fed into the engine. This arrangement prevents flow instability in the ducts.

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 arranged for de-icing, using either infra-red radiation or hot air circulation.

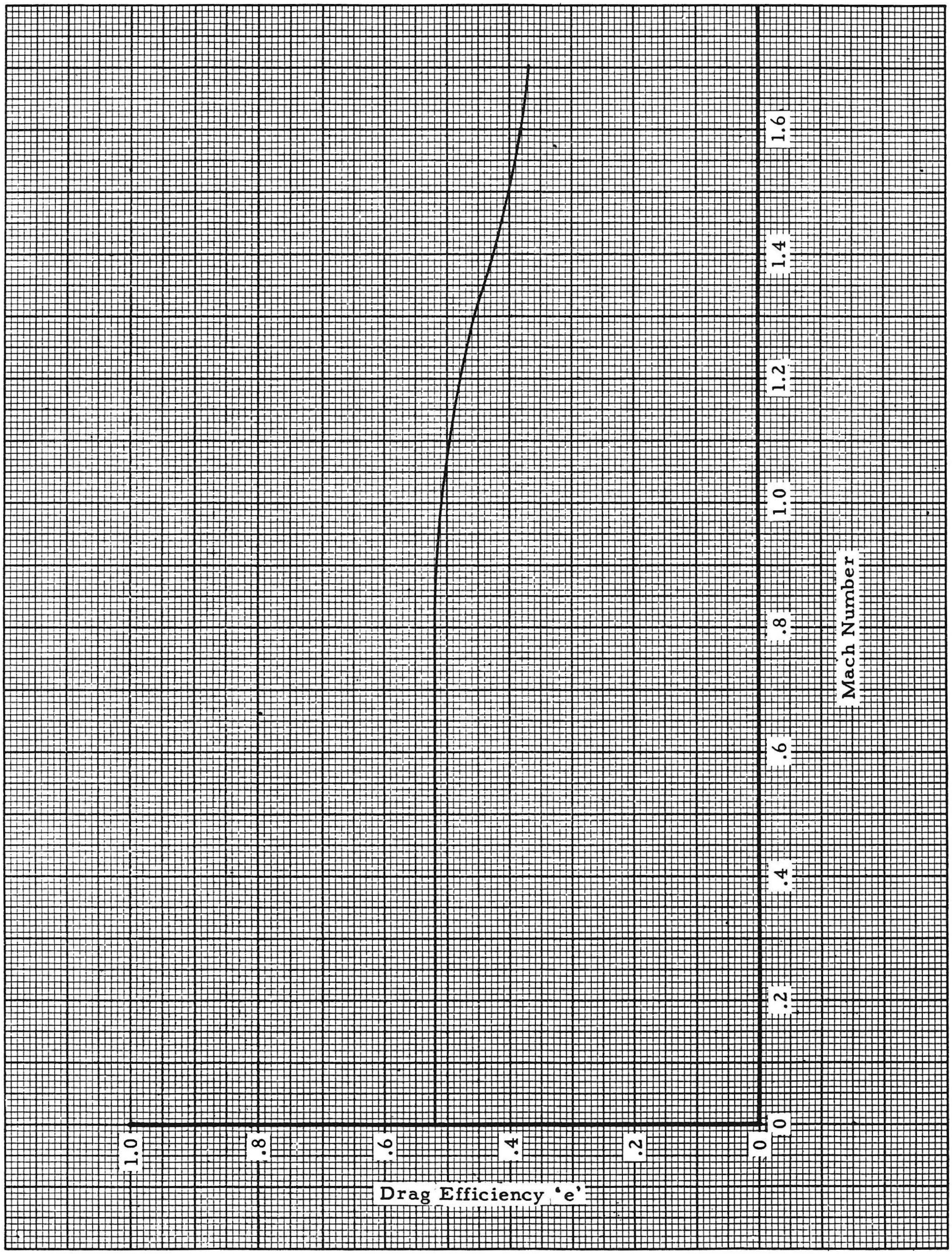


Fig. 22 Drag Efficiency

3.3.2 Stability and Control:

3.3.2.1 Longitudinal Stability and Control: The aircraft 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 elevons 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 elevons at a cost of 25,000 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 3 degrees elevon 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 147 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 pipe. 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 number, 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), elevon 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 elevon 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 a 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 pipe. 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 12 - 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 147 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 possess 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 elevons and the rudder. All the surfaces are fully irreversible and are power operated. The elevons are capable of giving a maximum rolling performance of 220 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 elevon 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 elevons on the C104 are nearly four times more powerful than the above requirement.

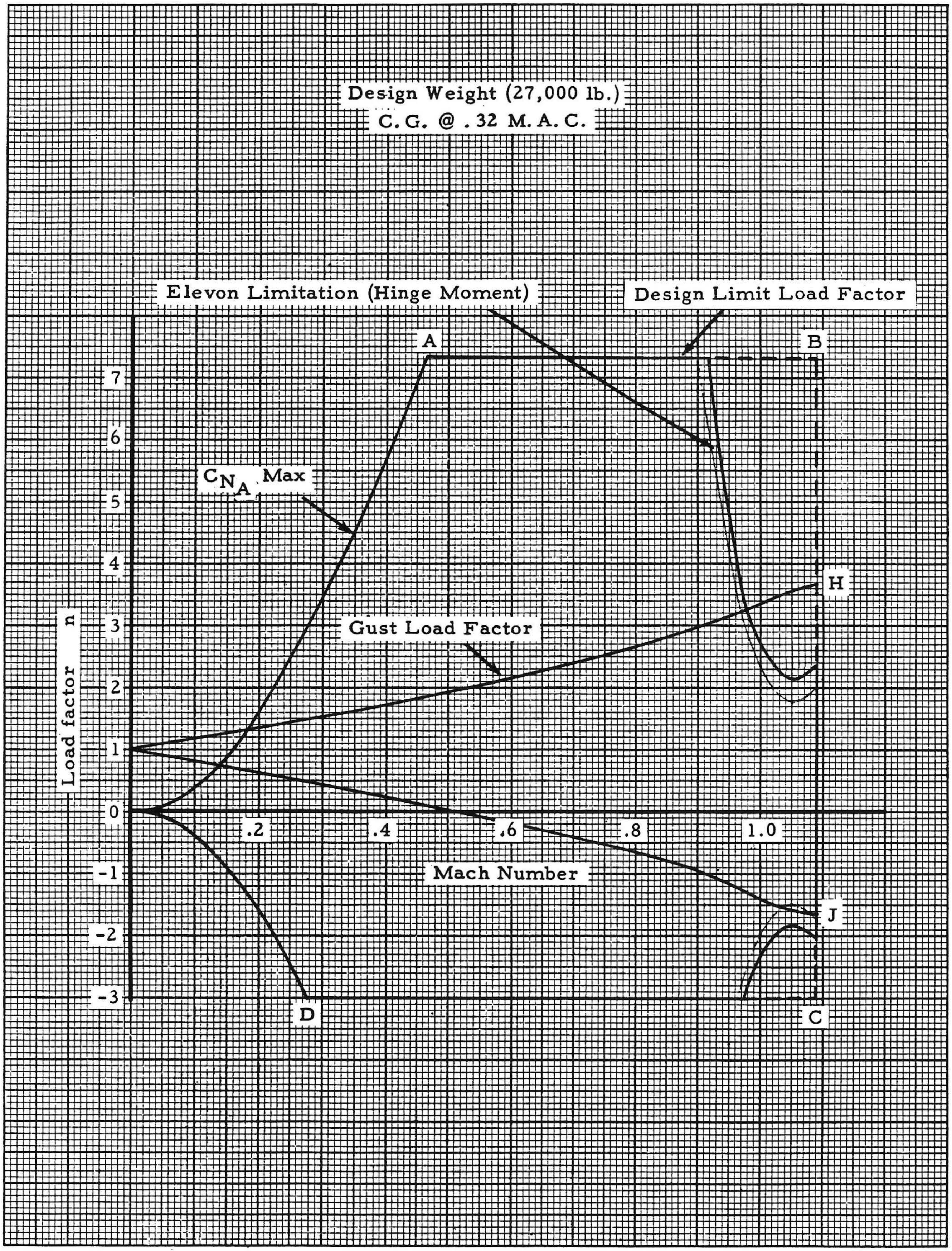


Fig. 23 Structural Flight Envelope at Sea Level

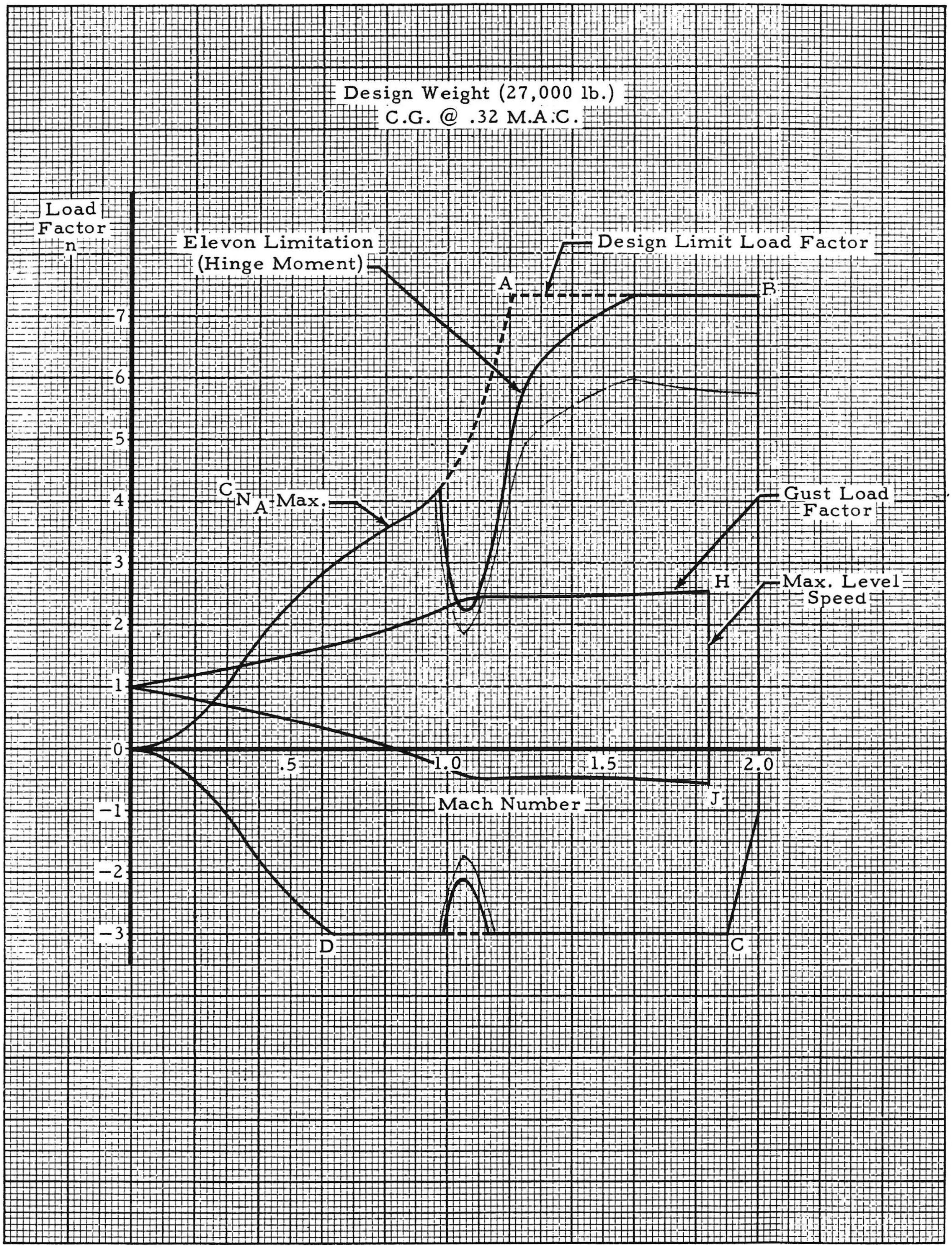


Fig. 24 Structural Flight Envelope at 30,000 ft. Altitude

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 elevon 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 = 23,700 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 has a triangular shape in plan (so called delta-shape), passes through the lower part of the fuselage. The reason for adopting the low wing configuration is that it forms the only possible arrangement for this airplane which gives a satisfactory undercarriage stowage and engine installation while also keeping the structure weight to a minimum. The maximum wing thickness is three percent of the chord at all stations along the span. The wing consists of a left and right hand panel which are joined at the airplane center line inside the fuselage, by means of bolted transport joints. Each panel has a detachable wing tip and a detachable elevon. The leading edge portion of each panel is manufactured as a separate sub-assembly but it is not intended to make this a detachable component.

3.5.2 Construction: Basically the wing structure consists of:-

(a) Four spars which run at right angles to the airplane center line

(b) A leading edge spar which is located at the 25% chord station along the span

(c) Ribs located parallel to the airplane center line intercostal between spars, and

(d) Aluminum alloy skins which are reinforced by spanwise stringers located intercostal between ribs.

The spars will take all the bending-loads and normal shear loads acting on the wing while the skin will cater for torsional shear loads only. The top skin of the wing terminates at the fuselage sides and the shear loads in the skins are transferred at this point to the fuselage skins through a bolted connection between the fuselage and wing skins. Root ribs transfer shear loads from the wing bottom skin to the fuselage sides. All four spars and the bottom skin between them extend across the fuselage below the engine and afterburner and are connected by bolted joints at the center line of the airplane. The leading edge spar is attached by fittings to the side of the fuselage at the transport joint bulkhead which connects two fuselage sections. The wing structure is broken into a fore-and-aft part by the main landing gear. These parts are connected and made integral by the rugged leading edge spar. The greater portion of the normal bending moment on the wing at the root is taken by those spars which are located on either side of the landing gear; this, therefore, concentrates the wing structure where it can also be used to cater for the landing gear loads. The structural arrangement places the flexural axis close to the extremes of the center of pressure range of the wing, thus resulting in relatively low torque loads. Due to the large wing chord in the root region of the wing, provision must be made to cater for large chordwise bending moments; this has been accomplished by designing rugged ribs with extruded light alloy booms between the spars. The wing structure aft of the landing gear is designed to cater to integral fuel tanks and the proposal is to make the top and bottom skin panels over the integral tanks detachable and replaceable. These skin panels are stressed only by relatively light shear loads from wing torsion, the wing bending being catered for entirely by the wing spars. These detachable wing skin panels are stiffened by stringers and boundary angles which will be spot welded to the skin. The panels will be bolted around the edges to the rugged sparbooms and rib booms and it is proposed to seal the panel boundaries against fuel leakage by means of a rubber beading similar to that employed on the 'Comet' and a method by which sealing material is forced under pressure (by means of a type of grease-gun) between the skin and the extruded spar- and rib-booms; it is known that satisfactory sealing can be achieved in this way. In the case of battle damage to a skin panel, this would then be removed and replaced by a new skin panel, a stock of which would be kept in store at the fighter base. The fuel cells forward of the landing gear will consist of bladder tanks. The leading edge is a monocoque structure consisting of closely spaced ribs with relatively thick skins.

Provision is made in the leading edge for hot air anti-icing. 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 aircraft. The effect of this temperature, which may reach a value of 160 deg. C 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:-

		Spec.
All skin panels and ribs	24ST clad	AN-A-11
Sparboom extrusions	75ST	AN-A-10
Spar webs	75ST	AN-A-10

3.5.3 Elevons:

3.5.3.1 Refer to Fig. 25. The elevons which combine elevator motion with aileron motion, are full span plain flaps extending from the fuselage sides to the wing tips and are fully power-operated by hydraulic jacks. The motion is as follows:-

(a) Elevator - 20 degrees up and down

(b) Aileron - 7 degrees up and down.

The aileron motion is superimposed on the elevator motion so that the maximum angular motion of the elevons is 27 degrees up to 27 degrees down. The distance from the airplane center line to the centroid of elevon surface area is 8.65 feet.

3.5.3.2 The elevons are attached to the wing rear spar by six hinges. The hinge-line is at right angles to the airplane center line; this is necessary because, due to the extremely thin airfoil section, the elevon operating horns will protrude into the airstream and it is hence necessary to shroud these horns and the driving mechanism with fairings which are in line with the airstream. Each elevon is actuated by two sets of two jacks. The inboard jacks are installed in tandem within the fuselage contour while the outboard jacks are mounted externally on either side of the surface at 70% of the elevon span and are covered by fairings. The elevon structure is a simple monocoque, using thick skin, a leading edge spar and a small number of ribs. A blunt trailing edge is used which will consist of a light alloy extrusion; the reason for this is that a blunt trailing edge will improve the torsional stiffness considerably and, from the aerodynamic point of view, a blunt trailing edge will not cause additional drag at supersonic design speed. All bearings will be of the roller type. No mass balance or aerodynamic balance is incorporated and no tabs will be fitted.

3.5.4 Lift and Drag-increasing Devices: These are not fitted on the wing.

3.5.5 Speed Brakes: These are not fitted on the wing.

LEGEND

(1) Radome
(2) Transport joint - radome/cockpit section
(3) Cockpit section
(4) Pilot's seat/sloping bulkhead
(5) Central beam
(6) Hinged access doors
(7) Equipment and armament section
(8) Power plant section
(9) Control trough
(10) Typical spar joint in fuselage
(11) Access panel to front engine steady
(12) Detachable wing tip
(13) Elevon outboard jack fairing
(14) Elevon
(15) Fuel cell access panels
(16) Typical transport joint - fin spars to fuselage formers
(17) Fin front spar
(18) Fin center spar
(19) Rudder operating jacks - upper
(20) Fin rear spar
(21) Rudder
(22) Brake parachute housing
(23) Variable afterburner nozzle
(24) Rudder operating jack - lower
(25) Rear nacelle section
(26) Transport joint - power plant/rear nacelle section
(27) Fuselage speed brake - upper
(28) Elevon operating jack - inboard
(29) Elevon hinges - 5 off
(30) Wing rear spar
(31) Elevon operating jacks - outboard
(32) Wing intermediate spar - aft
(33) Integral fuel tanks - 6 off
(34) Wing center spar
(35) Main gear pivot axle
(36) Main gear bay
(37) Wing intermediate spar - forward
(38) Fuselage fuel bladder cell compartments - 7 off
(39) Leading edge spar
(40) Fuel bladder cell compartments - 3 off
(41) Front spar
(42) Transport joint bulkhead
(43) Electronic equipment bay
(44) Armament compartment
(45) Armament compartment roof
(46) Air intake duct
(47) Transport joint - nose section/ equipment section
(48) Air Intake
(49) Boundary layer bleed
(50) Pilot's floor

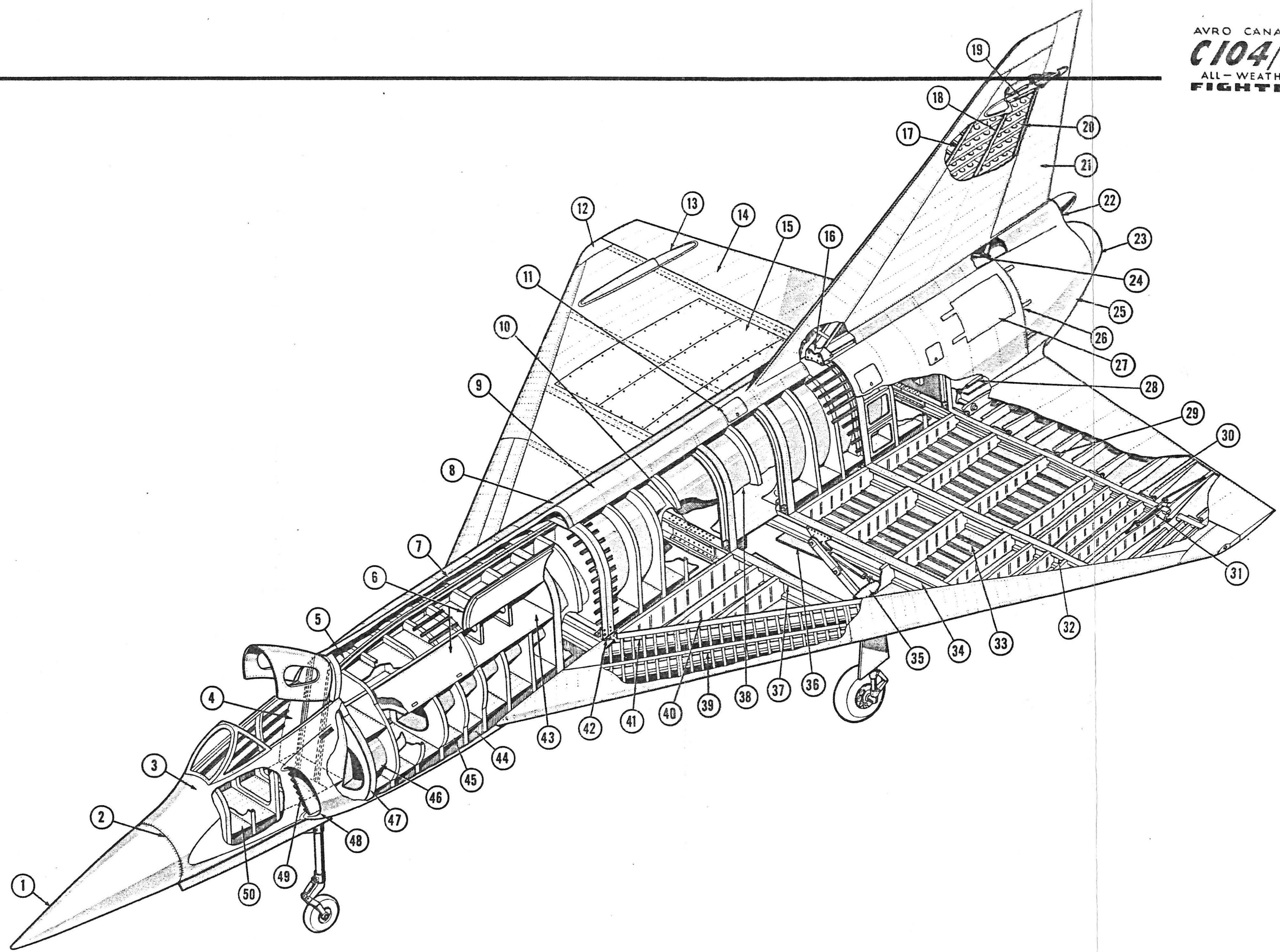


FIG. 25 STRUCTURE

3.6 Tail Group:

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

3.6.2 Stabilizer: Stabilizers are not fitted to this airplane.

3.6.3 Elevator: Elevators are not fitted. Refer to sub-paragraph 3.5.3 for details of the elevons.

3.6.4 Fin: The fin is of three spar construction with closely spaced chordwise ribs and relatively thin skin without stringers. All bending and normal shear is taken by the spars, the skin only catering for torsional shear loads. The three spars attach at the top of the fuselage to three rugged fuselage formers by means of bolted fittings which are housed inside a dome-shaped fairing. This extends along the top of the fuselage and blends into the brake parachute container. The torque on the fin is distributed into the fuselage by means of a bolted connection between the skin and the fuselage skin. The large angle of sweep of the fin spars makes it necessary to fit a rugged rib at the root of the fin to distribute the horizontal component of the spar bending moments into the skin, the vertical component of the spar bending moments is fed into the fuselage formers. The leading edge consists of a simple monocoque structure with closely spaced ribs at right angles to the front spar to support the skin. Hot air anti-icing will be used in the leading edge. Material will be 24ST aluminum alloy.

3.6.5 Rudder: The rudder is attached to the fin rear spar by means of five hinge brackets and is actuated by two sets of two hydraulic jacks. The inboard jacks are installed in tandem within the fuselage contour at the root of the rudder, the outboard jacks are mounted externally on the fin on either side of the rudder surfaces at approximately 70 percent of the rudder span and are covered by fairings. In order to keep the external fairings as small as possible, the outboard jacks are positioned so that their center lines are in line with the airflow; since the hinge line of the rudder is inclined relative to the airflow, there will be a misalignment of the jack force. This misalignment, however, is less than 10 degrees and can be catered for by a self aligning roller bearing. All other hinges have simple roller bearings. The rudder structure is of a simple monocoque construction with chordwise ribs covered by a relatively thick skin. Stringers are not fitted and a blunt trailing edge extrusion, similar to that used on the elevon, completes the assembly. All material will be 24ST clad aluminum alloy (AN-A-11).

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 circular over the top half with flat sides extending downwards to form a right angle junction with the top surface of the wing. Accommodated in the fuselage are the pilot, equipment (including armament), nose landing gear, engine intake ducts, the engine and afterburner, flying and power plant controls, speed brakes, a brake parachute, tail skid, and fuselage fuel tanks. The fuselage is to be manufactured in five 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 power plant section, and

(e) The rear nacelle section.

The engine intake duct is split some distance forward of the engine and the two halves diverge forward to form two 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 is a detachable dome-shaped fairing which fairs into the canopy at the forward end and into the brake parachute container 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 three separate flaps, one on each shoulder and one on the bottom of the rear nacelle section of the fuselage. A sharp pointed 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. Special attention has been given to the housing of equipment from the point of access for servicing and maintenance; further details of this are contained in the following paragraph and in the paragraphs dealing with the various equipment items. Much thought has also been given to the accessibility of the engine and afterburner. The low wing arrangement, which is mandatory for this size of airplane from considerations of main landing gear stowage and the delta wing configuration together with the large fin on top of the fuselage, make it impossible to break the fuselage in a way to give ideal accessibility such as can be done on airplanes of more conventional design. Every effort has therefore been made to provide as many large detachable access panels, in the sides and bottom of the fuselage, as possible, for engine and afterburner servicing. This aspect is enlarged upon in the paragraphs dealing with the engine installation.

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 intake for the engine. The general structure follows conventional practice with skin, stringer, 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 metal (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 compartment houses the air-intake ducts, the electronic, electric, pneumatic and air conditioning equipment, the guided missiles and the front 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) A centrally located beam, extending from the top of the fuselage to the armament bay roof, which splits the intake duct into two halves.

(b) The armament bay roof, which extends right across the fuselage below the intake ducts and this, in effect, forms the main bottom structural boundary of this fuselage section since the armament bay structure below this roof diaphragm is designed as being a subsidiary structure.

(c) The electronic equipment bay floor which extends across the fuselage above the intake ducts. The floor is split into halves by the centrally located beam to which the two halves are attached.

(d) The fuselage sides between the armament bay roof and the electronic equipment bay floor.

(e) The front transport joint bulkhead, to which are attached the nose landing gear leg axle, the centrally located beam, the electronic equipment bay floor, the armament bay roof and the cockpit section floor. The latter two are bolted together at this bulkhead.

(f) A rear transport joint bulkhead, to which are attached the centrally located beam, the armament bay roof and the power plant section floor. The latter two items are bolted together at this bulkhead, which also picks up the wing leading edge spars.

(g) The front fuel tank bulkhead, to which is attached the electronic equipment bay floor, and

(h) The fuel tank baffle bulkheads.

It will be seen that these main scantlings form a structural box with the centrally located beam passing through the middle of it. This box is bounded by the front and rear transport joint bulkheads through which pass the air intake ducts. Located above the box is the electronic equipment; which is accessible through long, unstressed, quick-release type doors which hinge upwards. Below the box is the armament bay which consists of subsidiary structure. Inside the box between the ducts and the outside skin is the electrical equipment (right hand) and the pneumatic and air conditioning equipment (left hand). This equipment is accessible through large stressed side-skin panels which are located between intermediate formers and attached by screws. For routine maintenance, small, quickly-detachable handhole doors are provided in the screwed-on skin panels. The vertical bending moment and shear on the fuselage is taken by the central beam and the side skins of the box. The horizontal shear and bending moment is taken by the electronic equipment floor and the armament bay roof. Fuselage torque is taken by the walls of the box in shear. The general construction follows conventional practice with skins, stringers, diaphragms, formers and longerons. The material will be clad 24ST light alloy throughout. The 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 bulkhead 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 for the cell housings. Access holes are provided in the fuselage stressed side skins for installation and maintenance of the cells. These holes are covered by screwed-on doors.

3.7.1.2.4 Powerplant Section: In this section of fuselage the following items are accommodated:-

(a) The engine air intake duct

(b) The engine and afterburner

mounted on the center line, there would be no room to house the required structure and retracting mechanism. The speed brakes, which are described in sub-paragraph 3.7.1.6, are flush with the outside contour of the fuselage when retracted. The brake hinges are mounted on the last but one bulkhead. This member will also be used for the attachment of the hydraulic actuating jacks. Airloads acting on the speed brakes are transferred into the bulkhead and fuselage skin.

3.7.1.2.5 Rear Nacelle Section: This section is of simple monocoque construction and incorporates a cut-out for the retractable tail skid. It carries no appreciable load. Consideration will be given to the use of magnesium alloy for constructional purposes.

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 of this brochure 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 instruments, switches 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 15 degrees. Further, 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 be made to the equipment illustrations in this brochure and the paragraphs dealing with equipment.

3.7.1.6 Speed Brakes: Three separate speed brakes are fitted and comprise two flaps mounted one each side in the shoulder of the rear fuselage (termed upper speed brakes) and one large flap mounted in the bottom of the rear fuselage (the lower speed brake).

3.7.1.6.1 Upper Speed Brakes: These are mounted flush with the fuselage contour and are each actuated by two hydraulic jacks which extend the flaps into the air flow about two hinges attached to a fuselage bulkhead. The structure of the flaps consists of a conventional reinforced skin panel made of light-alloy material.

3.7.1.6.2 Lower Speed Brake: This component is mounted flush with the fuselage contour and is actuated by two hydraulic jacks which lower the flap into the air flow about hinges attached to the wing rear spar. Flap structure is of conventional light alloy construction.

3.8 Alighting Gear:

3.8.1 General Description and Components: Refer to Fig. 25. The alighting gear is the conventional type of tri-cycle 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 inboard of its pivot-axis into a compartment in the wing below the engine-floor. The tail skid retracts backwards into the left side of the rear nacelle. 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 main wheels; this line between the aft c.g. limit and the wheel axle makes an angle of 15 degrees with the wing-chord. 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 will be of conventional design in every respect and similar to the CF-100 gear. Axle travel for shock absorption will be approximately eleven inches and a Dowty liquid-spring shock absorber will be used. A side-stay consisting of two links will be used; retraction will be by means of one hydraulic jack for each main gear. The retraction pivot axle will be supported by means of fittings attached to two of the wing spars. Emergency lowering of the gear, in case of hydraulic failure, will be provided by means of compressed air stored in a 3000 p.s.i. air bottle. The design of the attachments of the gear to the airplane structure will be such as to permit rapid replacement of a complete unit.

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 links will be connected to hydraulic valves which meter the hydraulic fluid to the brakes. A hand operated parking brake will be installed.

3.8.2.3 Tires and Tubes: The tire size will be 29 x 7.7 in accordance with U.S.A.F. drawing SF51F601 type VII - military aircraft. The air pressure will be 220 p.s.i.

3.8.2.4 Shock Absorbers: These will be of the liquid spring type, manufactured by Dowty Equipment Limited.

3.8.2.5 Retracting, Extending and Locking Systems: These systems will be similar to those used on the CF-100.

3.8.2.6 Doors and Fairings: Two doors will cover the well into which the main gear retracts. The wheels will be enclosed by a mechanically-operated door hinged from the fuselage structure on the center line of the airplane. The legs and part of the wheels will be covered by fairing doors which are mounted on the legs of the landing gear.

3.8.2.7 Inspection and Maintenance: Provisions 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 para. 3.8.1.

3.8.4 Auxiliary Landing Gear (Nose Wheel):

3.8.4.1 Description: This will be of conventional design in every respect. The gear retracts forward and a single wheel will be used. Axle travel for shock absorption will be approximately eleven inches and a Dowty 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 will be 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 by means of compressed air stored in a 3000 p.s.i. 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 p.s.i.

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. The doors will be hinged from the sides of the wheel-well fuselage structure.

3.8.4.7 Steering Control: It is proposed to make the nose-wheel 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 paragraph 3.8.1.

3.9 Alighting gear (water type): Not applicable to this airplane.

3.10 Surface Control System:

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

(a) The elevons, which span the entire length of the wing trailing edge, and

(b) A single-piece rudder hinged to the swept-back fin.

Both installations are illustrated in Fig. 25. All the surfaces are actuated by irreversible hydraulic controls, which pick up on each surface at two points, one at the inboard or, in the case of the rudder, at the lower end, the other approximately three quarters along the surface span. This support ensures satisfactory rigidity for adequate control and eliminates the necessity for mass balance devices. Two hydraulic jacks, each actuated by one of two independent hydraulic systems are located at each pick up point. The outboard (or upper) system comprises two jacks symmetrically disposed in external blisters. This eliminates any actuating load from being applied to the hinge fittings. The inboard (or lower) jack pistons are mounted in one body which is located inside the fuselage structure, this arrangement obviating the necessity for external blisters. Dual hydraulic control valves, serving both sets of jacks, are located inside the fuselage adjacent to the inner (or lower) jacks. Normally, the valves are operated by electric actuators which are controlled by the auto-pilot, but a straight-run push-pull rod leads from the valves to the cockpit to link up with the pilot's conventional controls. Artificial 'feel' for the pilot is obtained through a suitable spring system which can be biased to give trim conditions as stated in sub-paragraph 3.10.3. A bob weight is provided in the longitudinal control circuit in order to sense 'g' application. In general, the system is similar to that used on F86E airplanes.

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 would 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 stowed in a housing, incorporating 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 11 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. 1, 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 buffeting 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 close when a fore-and-aft acceleration of 1G is exceeded, are used to operate the brakes. Maximum time allowance 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 to 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 3.18.3.4.

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. Force application necessary to accomplish this 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 stowed in the racks provided in the fuselage for electronic apparatus.

3.11 Engine Section:

3.11.1 Description and Components: Refer to Fig. 25 and 26. The engine and afterburner are housed in the power plant section of the fuselage as has been described in sub-paragraph 3.7.1.2.4. Refer also to sub-paragraph 3.12.3.

3.11.2 Construction: Refer to sub-paragraph 3.7.1.2.4.

3.11.3 Engine Mounts: The engine is supported on two trunnions located near its center of gravity on either side of the engine center line and on top of the compressor casing near the air intake by a steady mounting. The afterburner is supported on two trunnions located near its center of gravity on either side of its center line and further by the engine exhaust casing (to which the afterburner is attached so that shear loads only can be transmitted). The design of these mountings is such as to allow the engine and afterburner to expand laterally and longitudinally.

3.11.4 Vibration Isolators: These shall not be fitted.

3.11.5 Firewalls: The complete engine and afterburner are housed in a tunnel-shaped shroud. This shroud shall be of aluminum alloy sheet over the engine compressor and of stainless steel sheet over the engine combustion chamber and afterburner. A firewall shall isolate the compressor compartment from the combustion chamber compartment; this firewall shall be of stainless steel sheet.

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 Discription and Components: The airplane will be propelled by a single turbo-jet engine with an afterburner. These are installed in the rear of the fuselage as shown in Fig. 1 and 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)

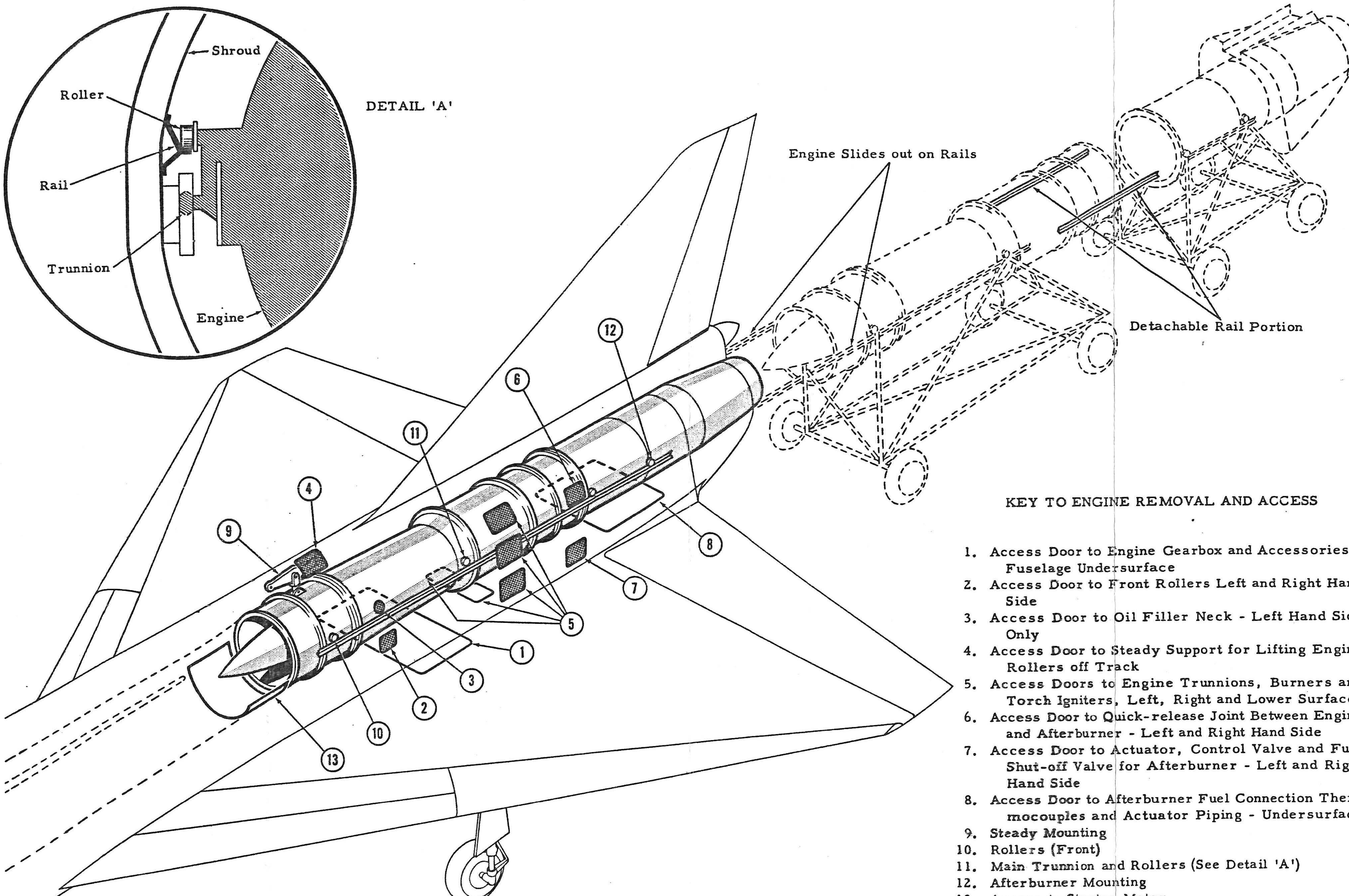
An afterburner will be fitted to whichever engine is used. All these engines will give approximately the same performance. As the engine sizes and weights are very similar, the installation details will not differ greatly. Maximum sea level static thrust of the engines will be 15,000 lb. in their fully 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. 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 to be 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 Afterburner: 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 to achieve a negligible base drag.

3.12.3 Mounting, Access and Removal:

3.12.3.1 Mounting: Refer to Fig. 26, when reading the following text. The engine is mounted on two main trunnions located near its c.g. (item 11). A steady point (item 9) is mounted on the front of the compressor. The afterburner is supported at its c.g. by rollers (item 12) which rest on the rails used for engine removal purposes. This arrangement permits free expansion of the afterburners. On removing the front steady connection and unclamping the main trunnion bearings, the rollers mounted on



KEY TO ENGINE REMOVAL AND ACCESS

1. Access Door to Engine Gearbox and Accessories - Fuselage Undersurface
2. Access Door to Front Rollers Left and Right Hand Side
3. Access Door to Oil Filler Neck - Left Hand Side Only
4. Access Door to Steady Support for Lifting Engine Rollers off Track
5. Access Doors to Engine Trunnions, Burners and Torch Igniters, Left, Right and Lower Surfaces
6. Access Door to Quick-release Joint Between Engine and Afterburner - Left and Right Hand Side
7. Access Door to Actuator, Control Valve and Fuel Shut-off Valve for Afterburner - Left and Right Hand Side
8. Access Door to Afterburner Fuel Connection Thermocouples and Actuator Piping - Undersurface
9. Steady Mounting
10. Rollers (Front)
11. Main Trunnion and Rollers (See Detail 'A')
12. Afterburner Mounting
13. Access to Starter Motor

FIG. 26 ENGINE AND AFTERBURNER INSTALLATION the engine (items 10 and 11) are permitted to rest on the rails which run along the inside of the fuselage aft end. The engine and afterburner may then be rolled along these rails as described in sub-paragraph 3.12.3.2.

3.12.3.2 Access Provisions: Access to the engine and afterburner is gained through the openings shown in Fig. 26. These doors are adequate for all inspection and servicing which does not necessitate the removal of the engine.

3.12.3.3 Removal: Following are instructions for the removal of the engine and afterburner. Refer to Fig. 26.

(1) Release the tail fairing (quick-release latches will be used here)

(2) Remove the appropriate access panels and disconnect:

On the main engine -

(i) The compressor bleed connections

(ii) The air starter supply line

(iii) The throttle and H. P. cock controls

(iv) The multi-point connections to the electrical junction box

(v) The fuel pipe from the airframe to the engine

(vi) All relevant fire extinguisher connections

(vii) The fuel and oil drain outlet pipes.

On the afterburner -

(i) The fuel supply line

(ii) All relevant ignition leads and thermocouple cables

(iii) The total pressure rake pipe.

(3) Disconnect the forward engine steady support (item 4) to allow the rollers (item 10) to rest on the rails.

(4) Release the main trunnion clamps (see detail, Fig 26) to allow the engine to be completely supported on the rails.

(5) Withdraw the engine and afterburner from the tunnel on to a suitable stand.

(6) Separate the afterburner from the engine (if required) by:

(i) Disconnecting the compressor bleed supply to the nozzle actuator and fuel pumps.

(ii) Undoing the quick-release connections between the afterburner and the engine and allowing the afterburner forward steady supports to rest on the rails.

Finally, the afterburner stand should be disconnected from the main engine stand.

3.12.4 Engine Driven Accessories: All accessories which are not integral with the engine are driven by air bled from the compressor. Further details are contained 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 intake ducts described below.

3.12.5.2 Air Intakes: The air enters through D-shaped ducts located on either side of the fuselage just aft of the cockpit. From these points it is led, via two separate ducts, to a position just forward of the engine. Here, the ducts merge into a single, circular-section passage. The combined air flow then enters the engine air inlet. For details of the aerodynamic features of this system refer to sub-paragraphs 3.3.1.2 and 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 engine 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 to 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 8000 pounds of usable kerosene which is contained in a number of tanks situated in the wing and in the fuselage. The tanks are of the 'bladder-cell' type except the aft wing tanks which are of the 'integral' type; the reason for using some integral tankage in the wing is that the required capacity in this size of airplane can only be obtained that way. If bladder cells were used throughout, the size of the airplane would have to be increased. This would jeopardize performance and mission requirements. The fuel system will supply fuel to the engine and the afterburner under all flight conditions including inverted flight. Two immersed-type booster pumps, which are located in the fuselage cells forward of the engine, supply the engine and afterburner in the normal way, the wing fuel being fed into these fuselage tanks by transfer pumps. In case of failure of either booster pump, the remaining pump is capable of supplying the engine. In the event of failure of both fuselage booster pumps, it is possible for the pilot to direct fuel from the wing tanks directly to the engine.

3.12.9.2 Pumps: There are two immersed 'booster' pumps in a forward fuselage bladder cell. It is proposed to drive these pumps by air-turbines fed from the pneumatic system. The reason for adopting pneumatic power to drive these large pumps instead of electric power is that our 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 four transfer pumps which transfer the fuel in the wing tanks to the fuselage tank in normal circumstances, or directly to the engine in an emergency. These transfer pumps are driven by immersed a.c. electric motors.

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

Two integral wing tanks	1670 lbs.
Ten bladder cell wing tanks	2255 lbs.
Thirteen bladder cell fuselage tanks	4075 lbs.
Total:	8000 lbs.

The bladder cells will be of conventional construction similar to the ones installed in the CF-100. The integral tanks have already been described in the description of the wing structure (refer to sub-paragraph 3.5.2). The proposed wing structure lends itself admirably to the incorporation of an integral tank since all the bending moment acting on the wing will be taken by rugged concentrated spars and the skin of the integral tank would cater to relatively light torsional shear loads only. The integral tanks in the left and right wing are covered by three fairly large skin panels, top and bottom. These are secured by countersunk head bolts to the spars and ribs. The panels are reinforced by short spanwise stringers and boundary members spot welded to the skin.

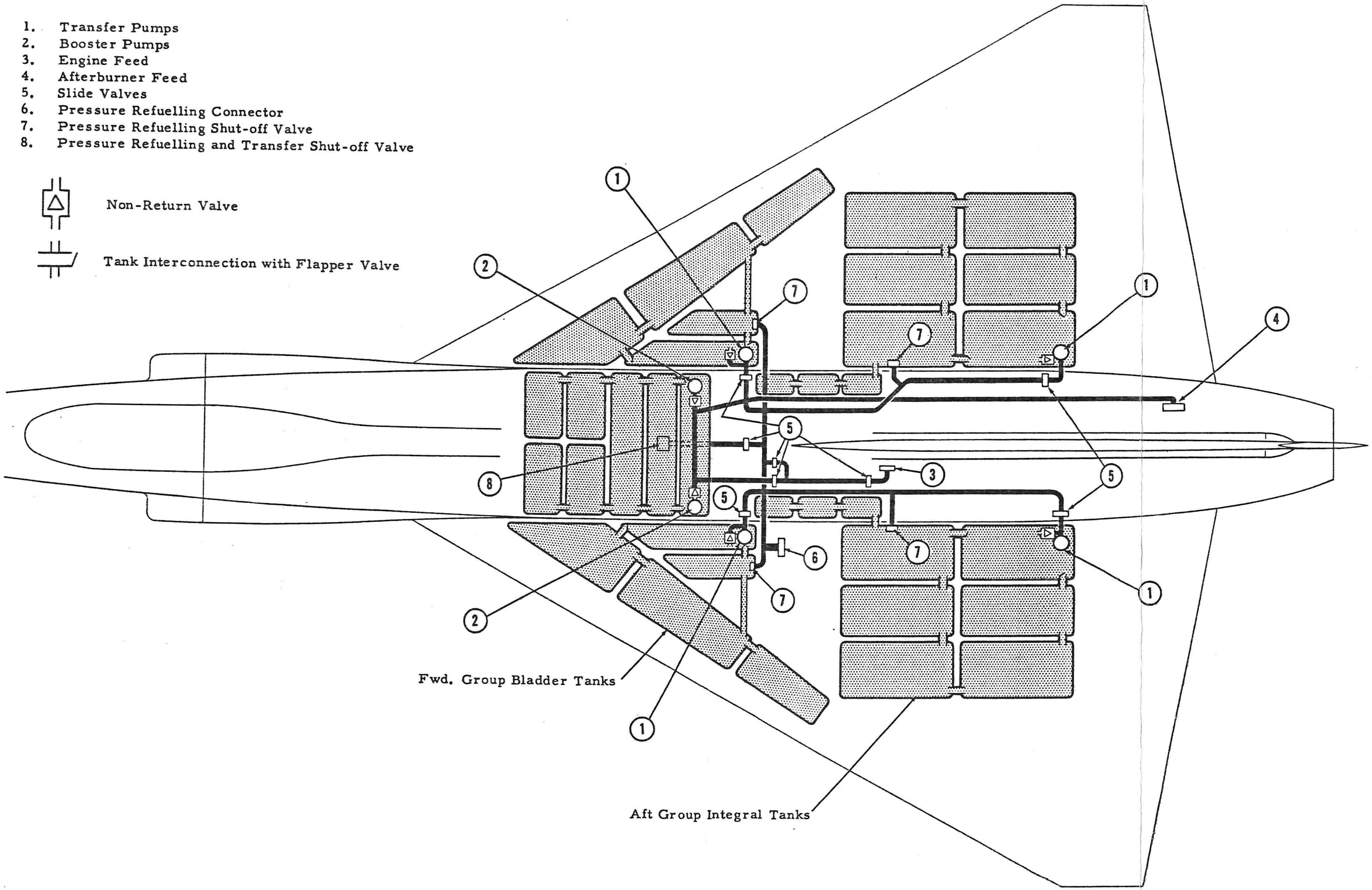


FIG. 27 FUEL SYSTEM

Sealing around the edges of each detachable panel will be accomplished by means of rubber beading clamped to the spars and ribs. In addition, it is proposed to insert sealing material, under pressure, (by means of a type of greasegun) between the panel edges and the spar and rib extrusions after the panel is screwed on. In case of battle-damage to a panel it may be replaced by a new one, an adequate quantity of which will be kept in stock at the fighter base.

3.12.9.4 Tanks (droppable): These will not be carried on this airplane as all fuel for the required mission is housed internally.

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.

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 p.s.i. 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 engine and afterburner 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 an air-turbine starter motor located in the nose-bullet of the engine which will be energized by an external ground compressor unit. This will plug 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: A ram-air turbine is fitted which derives its energy from the forward speed of the airplane, air being bled from the engine's air intake duct. This auxiliary power plant is used only to drive an emergency hydraulic pump for energizing the power controls in case of engine failure. In normal circumstances this power plant will not be operative; it may be started up by the pilot opening a valve, or alternatively by an automatic valve, if the main engine power should fail. It is designed to provide adequate power to land the airplane at normal speeds.

3.13.2 Installation: This ram-air turbine, together with its emergency hydraulic pump, is installed at the side of the armament rocket pack in the forward part of the power plant section of the fuselage. Access is gained through a door at the side of the fuselage.

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

(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 on top of the air intake ducts where it is easily accessible from the outside of the fuselage through large non-stressed, hinging doors, as has been described elsewhere in this brochure.

3.15 Hydraulic, Emergency Air and Pneumatic Systems:

3.15.1 Hydraulic and Emergency Air Systems:

3.15.1.1 Description and Components: Refer to Fig. 28 and 29. A dual hydraulic system is used to actuate the power flying controls and airplane services, each system being hydraulically independent of the other with separate pump, accumulator, reservoir, etc. Each hydraulic pump is driven from a separate air turbine, actuated, in turn by air from the engine compressor. This method of drive results in the pumps running at constant speed and will deliver full horsepower for the entire normal engine speed range. 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 through a ram air turbine and operated if engine failure should occur. In this instance both engine compressor driven pumps would not function. Nominal operating pressure of each system is 3,000 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. It is possible, throughout, to operate each system on the ground either by a hydraulic rig through conventional ground connections or through actuation of the air turbines from a ground compressor unit.

3.15.1.2 Summary of Actuated Items:

3.15.1.2.1 Flying Control Circuits: The control surfaces to be actuated are the two elevons and the rudder. Each surface is actuated by a double piston jack located at the root of the surface, and an outboard set of two jacks located one on either side of the surface, at 70% of the surface span. The utility circuit supplies one piston of the double piston jacks and one of the outboard jacks of each control surface. The flying control circuit supplies the other piston of the double piston jacks and the other outboard jack of each control surface. Fluid supply to the jacks is controlled by one dual sliding valve for each control surface. Each valve in the dual sliding valve body serves one of the two independent circuits. The design of the dual sliding valve is such that in the event of failure of one of the two circuits, the fluid between each side of the jacks 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 and jack satisfactorily and without restriction. The actuation of the dual sliding valve is described in sub-paragraph 3.10.1.

3.15.1.2.2 Landing Gear Circuit: The landing gear circuit comprises a hydraulic circuit, for normal operation supplied from the utility system, and 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 to the two main gear jacks. In each case the fluid first passes through a shuttle valve and, on

	Description	No.
[illegible]	Pumps	3
2	Flying Control System Reservoir	1
3	Utility System Reservoir	1
4	Oil Filter - Low Pressure	2
5	Oil Cooler	2
6	Ground Test Connection Suction	2
7	Ground Test Connection Pressure	2
8	Non-Return Valves	7
9	Pump Relief Valve	2
10	Pressure Warning Switch	2
11	Pressure Gauge	2
12	Charging Gauge	4
13	Accumulators	2
14	Outboard Elevon Jacks	4
15	Inboard Elevon Jacks	2
16	Rudder Jack	1
17	Elevon Follow-up Valve	2
18	Rudder Follow-up Valve	1
19	Dive Brake Control Valve	1
20	Dive Brake Pressure Relief Valve	1
21	Dive Brake Jacks Upper	4
22	Undercarriage Control Valve	1
23	Main Undercarriage Shuttle Valve	1
24	Pressure Sequence Valve	1
25	Main Undercarriage Up locks	2
26	Main Undercarriage Jacks	2
27	Variable Flow Valve	1
28	Restrictor - Nose Undercarriage	1
29	Nose Undercarriage Jack	1
30	Nose Undercarriage Shuttle Valve	1
31	Undercarriage Jettison Valve	1
32	High Pressure Air Bottle	1
33	High Pressure Air Release Valve	1
34	Nose Wheel Steering Stop Valve	1
35	Nose Wheel Steering Control Valve	1
36	Nose Wheel Steering Motor	1
37	Nose Wheel Steering Restrictor	1
38	Nose Wheel Steering Pressure Relief Valve	1
39	Brake Relay Valve	1
40	Brake Accumulator	1
41	Signal Pumps	2
42	Brake Gauges	4
43	Brake Units L. H.	1
[illegible]	Brake Units R. H.	1
[illegible]	Dive Brake Jack Lower	2
[illegible]	Pressure Reducing Valve, Brakes	1

LEGEND

1. Armament Bay Air Conditioning Duct
2. Electronic Bay Air Conditioning Duct
3. Electrical Starter Panel
4. Electrical Distribution Box
5. Electrical Relay Panel
6. 28v Rectifier Units
7. Alternator Synchronizing Unit
8. Alternator Control Panels
9. Alternator Circuit Breaker Panels
10. Battery
11. Electrical Fuel Contents Units
12. Hydraulic Reservoirs
13. Pressure Air Collector Ring
14. Hydraulic Accumulators
15. Hydraulic Filters
16. Air-driven Turbines and Hydraulic Pumps
17. Emergency Air Bottles (3000 psi)
18. Main Pressure Air Supply Duct
19. Emergency Hydraulic Pump
20. Ram Air Turbine
21. Ram Air Intake
22. Access Panels
23. Pneumatic Ground Connection
24. Air-to-Air Heat Exchanger
25. Air Mass Flow Control Unit
26. Heat Exchange Air Exhaust
27. Air Turbine Driven Fan
28. Cockpit Air Temperature Control Valve
29. Cockpit Air Conditioning Duct
30. Alternators Cooling Duct
31. Alternators
32. Air Driven Turbines for Alternators
33. Air Duct to Turbines
34. Emergency Ram Air to Cockpit from Engine Intake

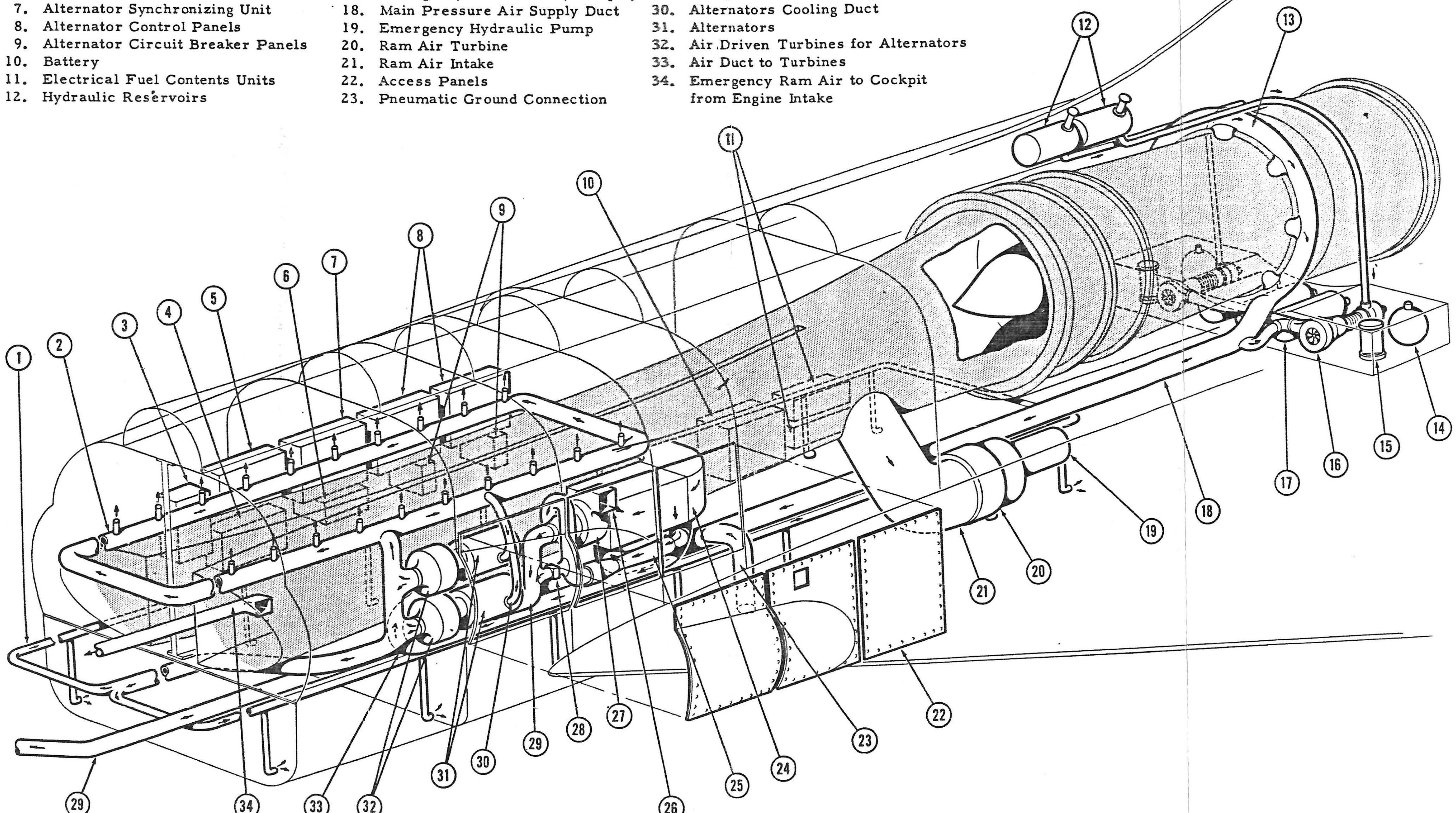


FIG. 28 ELECTRICAL, PNEUMATIC, AIR CONDITIONING AND HYDRAULIC EQUIPMENT

the main legs, a pressure sequence valve before entering the main jacks. The shuttle valve, when in use, admits air to the line for emergency operation only; the pressure sequence valve ensures that the main up-locks are released before pressure is applied to the main jacks. For UP selection the fluid is directed straight to the jacks, the down locks being located internally in the jacks and are broken by initial hydraulic pressure. A variable flow valve in the main gear UP line controls the extension rate without restricting the retraction rate. For emergency operation (DOWN selection only), air is stored at 3,000 lb. per sq. in. and is released by an emergency air control valve, the supply passing through a jettison valve and the shuttle valves before entering the landing gear system to release the up-locks and jacks. Fluid displaced in the up lines by this operation is routed back to the reservoir through the jettison valve.

3.15.1.2.3 Speed Brakes Circuit: The speed brake circuit consists of six hydraulically connected jacks, two operating each 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 dive brakes are 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 1,500 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 wheel brake cylinders 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.

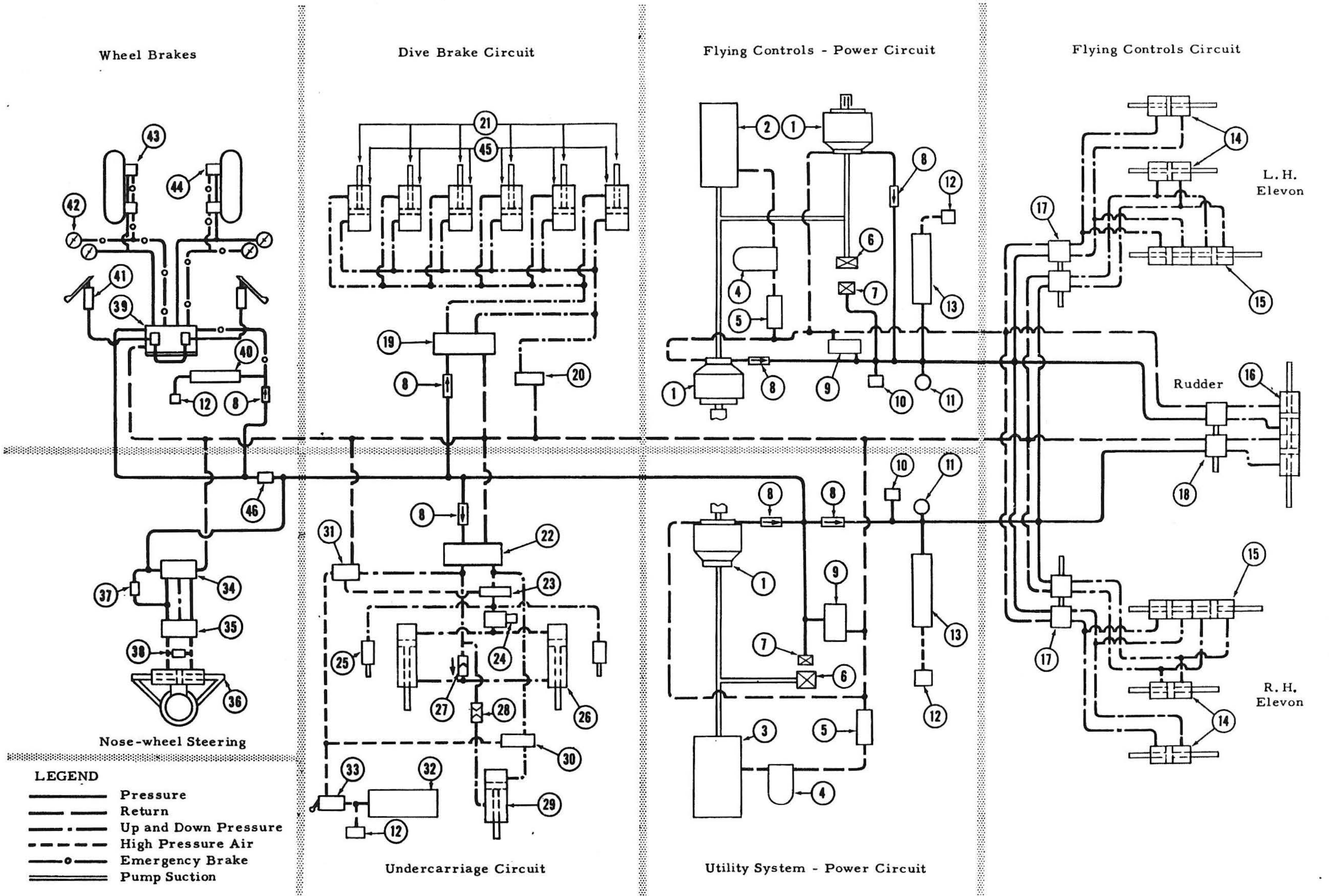


FIG. 29 HYDRAULIC SYSTEM DIAGRAM

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 about 100 lb. per sq. in. In flight, the system will function on air bled from the main compressor; for ground operation the system will draw compressed air from a portable gas turbine compressor unit. 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:- The two fuel booster pumps delivering main fuel supply will be driven by air turbines.

(c) Air Conditioning System:- An air cycle refrigeration unit, which supplies heated or cooled air to the cockpit, will be dependent on the pneumatic system for operation (refer to sub-paragraph 3.20.1).

(d) Engine Starting:- Engine ground starting will be achieved through an air turbine driven starter motor.

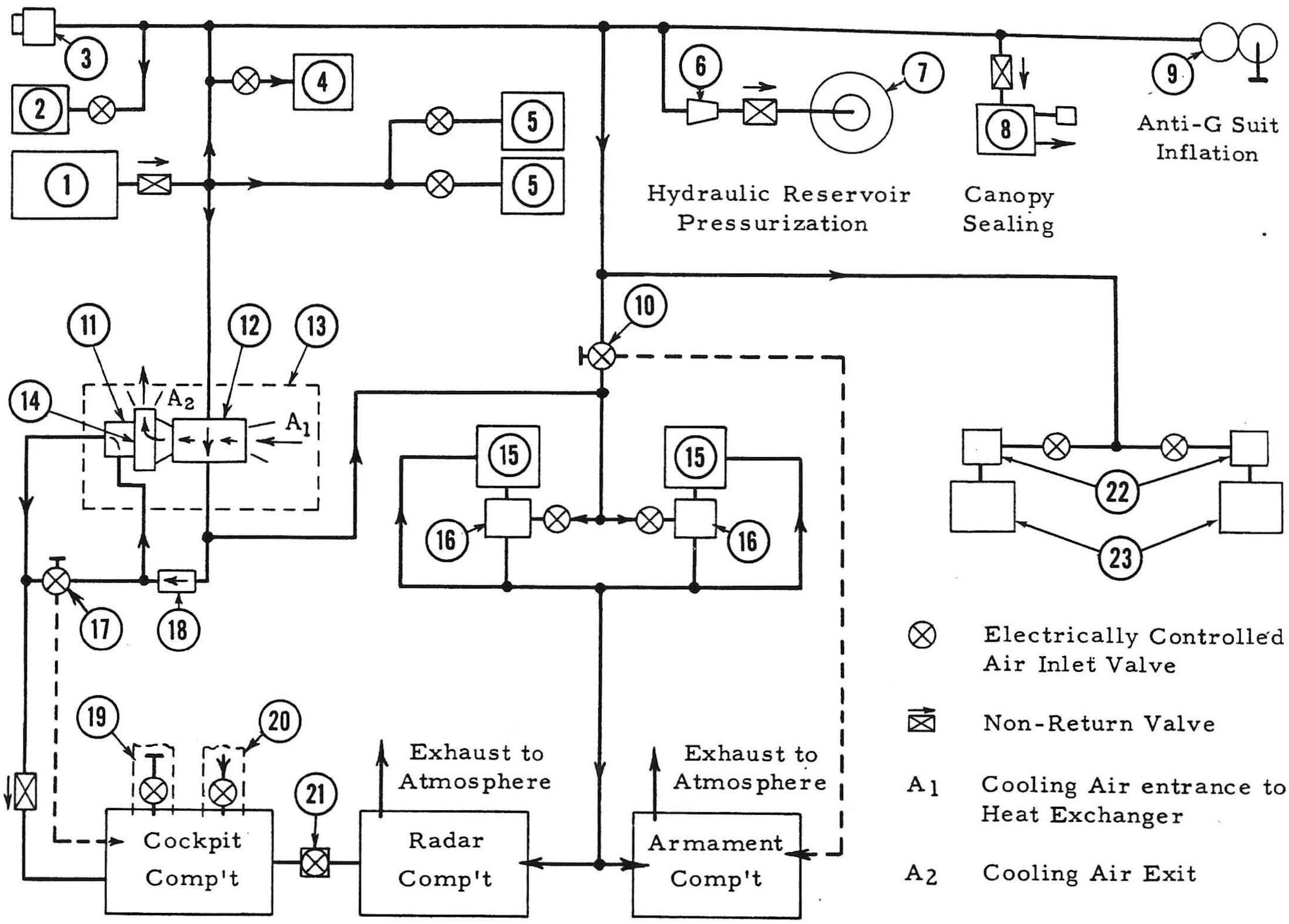
(e) Miscellaneous Services:- In addition to the above, the system will supply air to the pilot's anti-'g' suit, the canopy sealing, hydraulic reservoir pressurization and windscreen de-misting.

(f) Hydraulic System:- Two air turbine motors will drive the two main hydraulic pumps.

The design of ducting and associated fittings will, in general, follow the requirements of AMC Manual No. 80-1.

3.15.2.2 Actuated Items:

(a) Constant speed air turbine motors:- Two of these motors are used to drive the alternators. For these units an accurate control of output shaft speed is required. This is accomplished, usually, by restricting inlet temperatures and pressures, and drawing a substantial flow of air for operating purposes. For this reason, the supply is ducted from the source via a heat exchanger from which it is led to the motors. Air discharge from the motors, at ambient atmospheric pressures, 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 at Selected Temp. within range +20°F to 100°F

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

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

LEGEND

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

FIG. 30 PNEUMATIC AND AIR CONDITIONING

(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, expansion air turbine and a cooling air fan. Heated compressed air is ducted directly to the unit from whence it is passed, through the air-to-air heat exchanger and expansion turbine, 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 of the cockpit. Discharged cockpit air will be led into the equipment compartment and circulated to provide additional cooling.

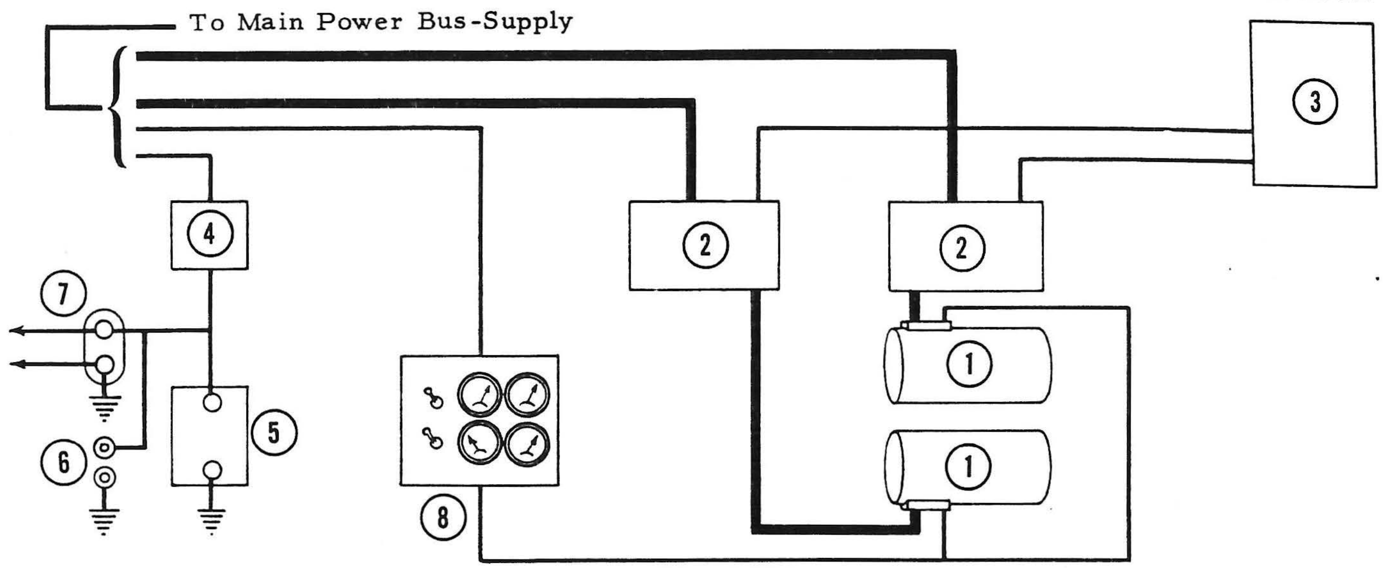
(d) Air turbine engine starter:- Operation of this unit will be required only when the airplane is grounded. The unit draws air from an external supply through the ground connection provided.

(e) Hydraulic pumps:- Operation of these pumps is by air turbine motors which receive hot compressed air directly from the pressure air collector ring attached to the engine compressor.

3.16 Electrical:

3.16.1 Description: Refer to Fig. 28. All electrical power - except that stored in the batteries - is generated by two 8 kilowatt, 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.

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



LEGEND

1 8 KW - 400 Cycle Alternator
2 Alternator Control Panel
3 Electrical Synchronizing Panel
4 Selenium Rectifiers
5 24 Volt 12 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

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
- 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

FIG. 31 ELECTRICAL CIRCUIT DIAGRAM

Supply		Demand
Electronic equipment	... ...	10 kw
Fuel pumps	... ...	1.5 kw
Variable load (Lighting, flying controls, etc.)	... ...	2 kw
Armament (Pulse load)	... ...	5 kw

In the event of failure of an alternator, the remaining unit will supply the power at 50% overload for a short period with added cooling. A control panel is provided for each alternator in the fuselage equipment 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 which require 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., 12 ampere-hour battery is connected in parallel with the 28v. bus. Battery capacity is small as the demand on this component is limited to engine ignition and initial alternator excitation. An external electrical supply is provided only for the battery. All other power from a ground source is obtained by supplying air pressure to the pneumatic system from a ground compressor unit. In turn, this pressure operates the turbines which drive the alternators. The alternators are cooled by a separate air duct from 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 in the equipment compartment.

3.16.4 Equipment - Installation: Refer to Fig. 28. The alternators are installed on the left hand side of the fuselage below the electronic equipment compartment. Side access panels are provided for removal and installation purposes. Inset in these panels are smaller, quickly-detachable panels which are intended for use when minor adjustments or servicing details are to be carried out on the equipment. The alternator control panels and rectifiers are located on the right hand side of the fuselage opposite to the alternators. Side access panels, similar to those used for the alternators, are provided. On the right hand side of the fuselage, adjacent to the rocket pack, is a stowage for the battery. Removal and servicing of this component is achieved through an access door fitted on the undersurface of the structure. The alternator switches, instruments and power buses are all located in, and accessible from, the cockpit.

3.16.5 Wiring: This 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 incorporated on 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: An air turbine motor will be used to start the engine (refer to sub-paragraph 3.12.12 and 3.15.2.2). When the engine speed reaches approximately 3,500 r.p.m., a starter fuel solenoid valve will release fuel to the torch igniters. The igniters receive high tension sparks from the booster coils.

3.16.10 Receptacles: An external power receptacle is provided for the 28v. a.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 wing transfer fuel pumps will be electrically driven.

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 majority of the equipment is installed in the electronic compartment located in the equipment section of the fuselage (Fig. 20). Hinged side access doors will be provided for easy access to the equipment which will be secured by quick-disconnect attachments. Provision will also be made for the rapid testing of each unit and, if any of these are found to be unserviceable, the faulty item will be able to be removed quickly and replaced by a new component without complicated alignment measures. 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 this 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

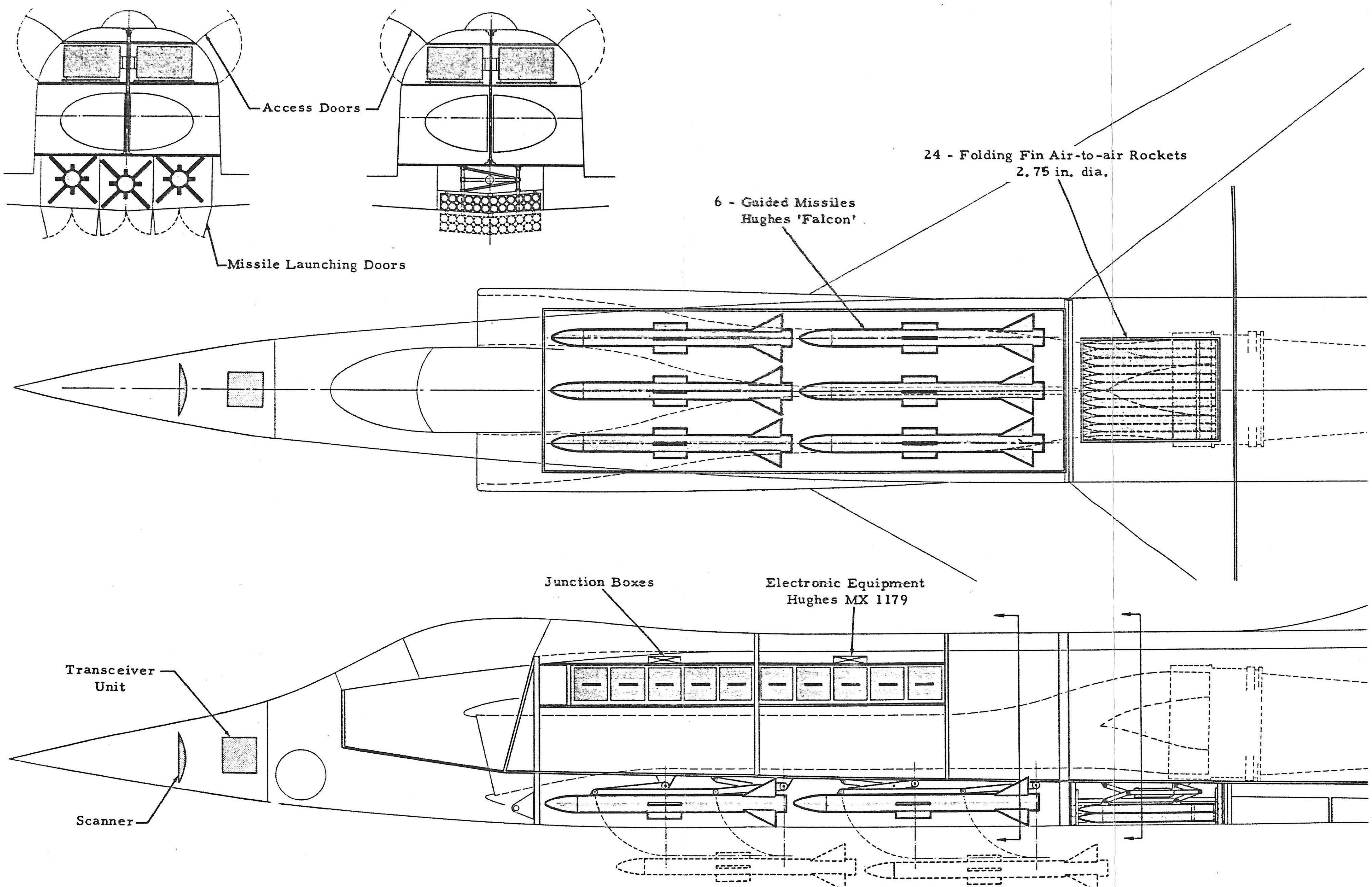


FIG. 32 ARMAMENT AND ELECTRONIC EQUIPMENT INSTALLATIONS

proposal for the CF-100, Mark 4. Extension is by means of a hydraulic jack, 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 Para. 3.20.1).

3.18.3.2 Optical Sights: At altitudes below 1,000 feet, the airborne radar tracking of the target is ineffective. Any launching 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. This type of sighting is suitable only for stern attacks.

3.18.3.3 'Falcon' Guided Missiles: Six 'Falcon' missiles are stowed in the bottom 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 Para. 3.20.1. 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 aircraft 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 with the APG40 in the event of failure of the automatic controls. Insofar as the pilot is relieved of the majority of his present duties, target discrimination cannot 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 aircraft 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 compartment 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 opened 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 check list holder,
Airplane flight report holder	and
Map case	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 engine bay.

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 Manual 80-1. The main source of air pressure will be taken from the engine compressor 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 fighter, 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. The cockpit will be pressurized according to 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 duct into the compartment. At Mach numbers greater than 1.0 this provides adequate pressurization for survival at 50,000 ft.

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

(a) The armament compartment, and

(b) The equipment compartments.

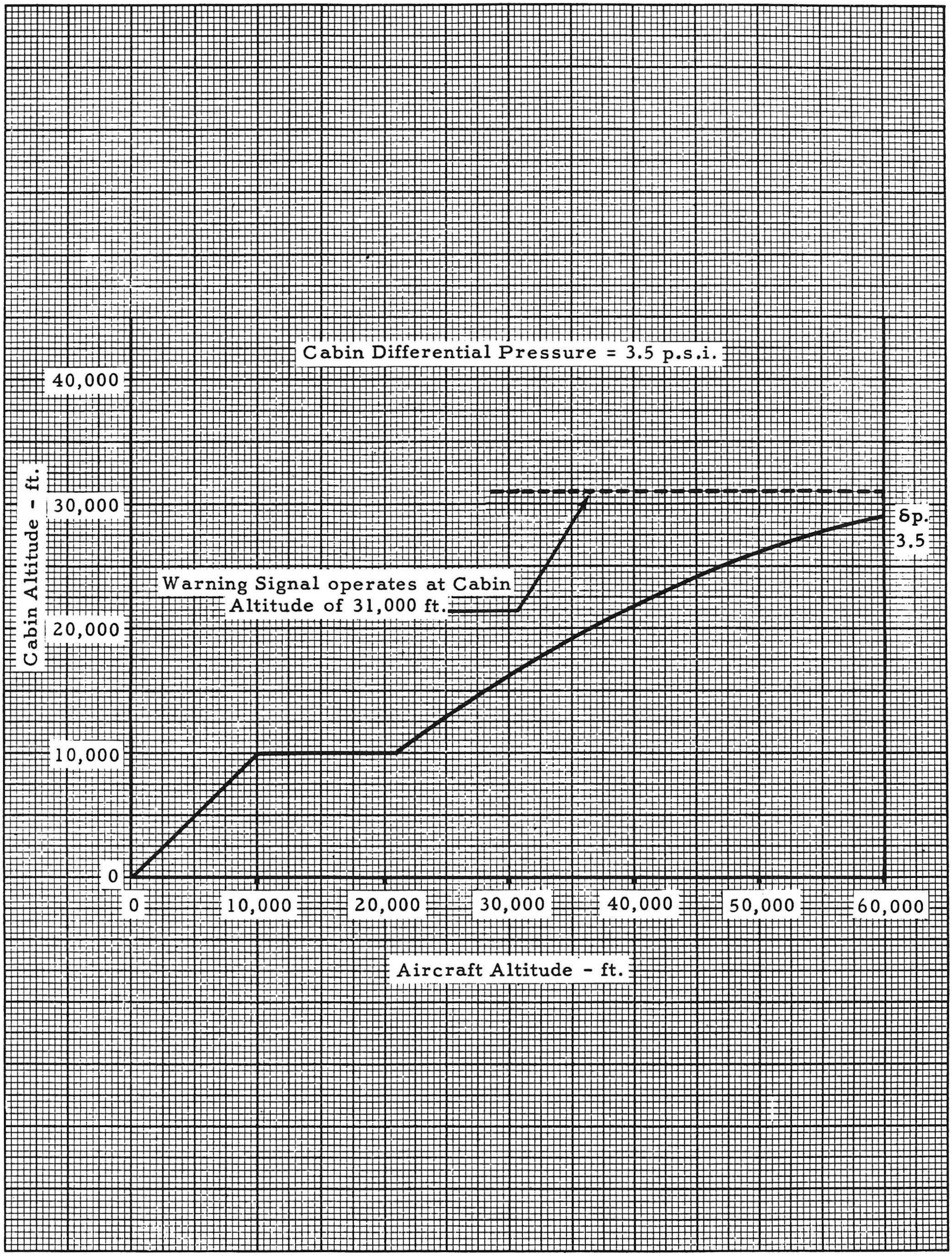


Fig. 33 Pressure vs Altitude Schedule

Dealing with item (a) first, the compartment, which contains rockets and guided missiles, 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 components 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 circulating air to atmosphere can be accomplished easily through louvres.

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, connected to the low pressure air ground connection will supply the necessary air conditioning for all conditions. Temperature ranges to be attained are given in the following subparagraphs.

3.20.1.2.1 Occupied Compartments: The cockpit will be the only occupied compartment 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 from 20 to 30 minutes. Provision will be made in the air conditioning system for the dissipation of heat generated by the equipment during the warm-up period. Ventilating air will be circulated in the compartment to ensure it 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-edges of the wing, the fin and the air intake ducts will be fitted with anti-icing means.

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, either by running the engine or by an external ground source of hot air.

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

APPENDICES I, IB AND IC

The above appendices, which cover Government and contractor furnished equipment lists, are 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 20432	C104/1, General Arrangement Drawing
SK 20427	C104/1, Armament and Electronics Installation
SK 20434	C104/1, Inboard Profile and Fixed Equipment

APPENDIX IV

OPTIONAL ARRANGEMENTS

IV A Trainer Version: (Model Designation C104/1/T)

A trainer version of the basic airplane is shown in Fig. 34. As will be seen from the drawing, 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 by 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 center of gravity of the airplane will move too far forward and it will be necessary to carry ballast in the rear nacelle of the fuselage. The resulting increase in gross weight together with the longer fuselage will, of course, detract from the performance as quoted in this brochure for the basic fighter airplane.

IV B Alternative Armament Version:

Since the basic fighter airplane's configuration must, by necessity, be tailored around one particular armament and equipment installation in order to meet the mission and performance requirements, it is impossible to make a multi-purpose airplane of this single-engined airplane and yet meet these performance requirements.

Accordingly no arrangement drawings other than the proposed Hughes MX 1179 armament and equipment installation are shown in this brochure.

IN THIS CONNECTION IT IS THEREFORE ESSENTIAL TO REALIZE THAT ONCE A COMMITMENT HAS BEEN MADE REGARDING THIS PARTICULAR ARMAMENT, NO FUTURE CHANGES CAN BE MADE WITHOUT A COMPLETE RE-DESIGN OF THE WHOLE AIRPLANE.

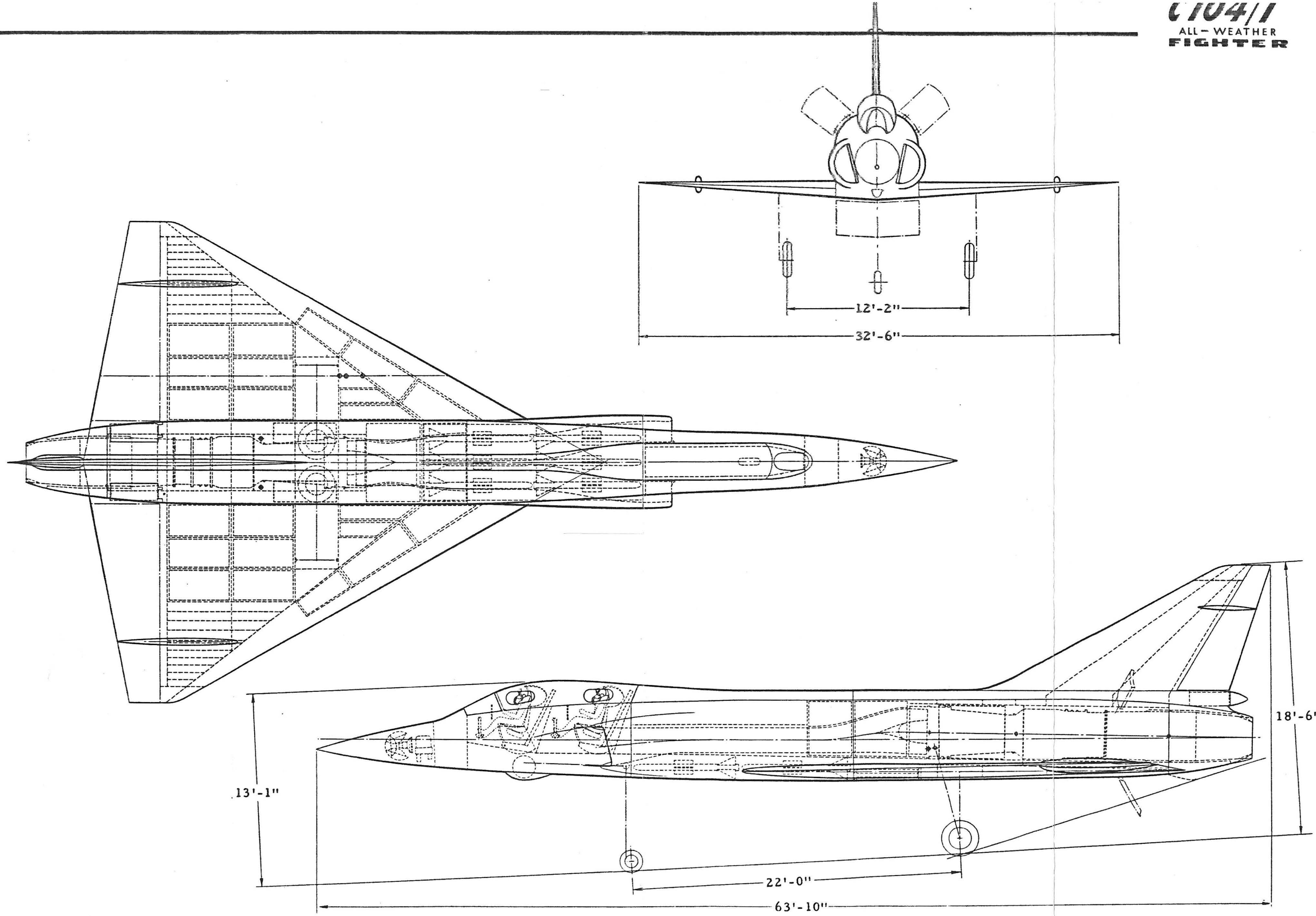


FIG. 34 - 3-VIEW G.A. OF AIRPLANE

APPENDIX V

SUMMARY OF OPTIONAL ARRANGEMENTS

Aircraft model	C104/1	C104/1/T
General arrangement drawing	See Fig. 1	See Fig. 34
Crew arrangement	1 pilot	2 pilots
Armament	6 Falcon missiles 24 F.F.A.A. rockets	6 Falcon missiles 24 F.F.A.A. rockets
Integrated navigational, communication and interception equipment	Hughes MX 1179	Hughes MX 1179
Engine type numbers	1 A.V. ROE TR9 1 Bristol OL-3 1 Curtis Wright J67	1 A.V. ROE TR9 1 Bristol OL-3 1 Curtis Wright J67
Afterburner	1 off	1 off
Fuel capacity	8,000 pounds	8,000 pounds
Ballast	None	888 pounds in rear nacelle
Empty weight	18,600 pounds	20,190 pounds
Design gross weight	28,200 pounds	30,000 pounds
Forward C.G. in flight	27.29% m.a.c.	25.03% m.a.c.
Aft C.G. in flight	30.86% m.a.c.	28.65% m.a.c.

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:-

$$\% \text{ M.A.C.} = \frac{A - 29.11}{23.49} \times 100$$

where A is the center of gravity position in feet aft of nose datum. The following table should be read in conjunction with Fig. 35.

TABLE 1

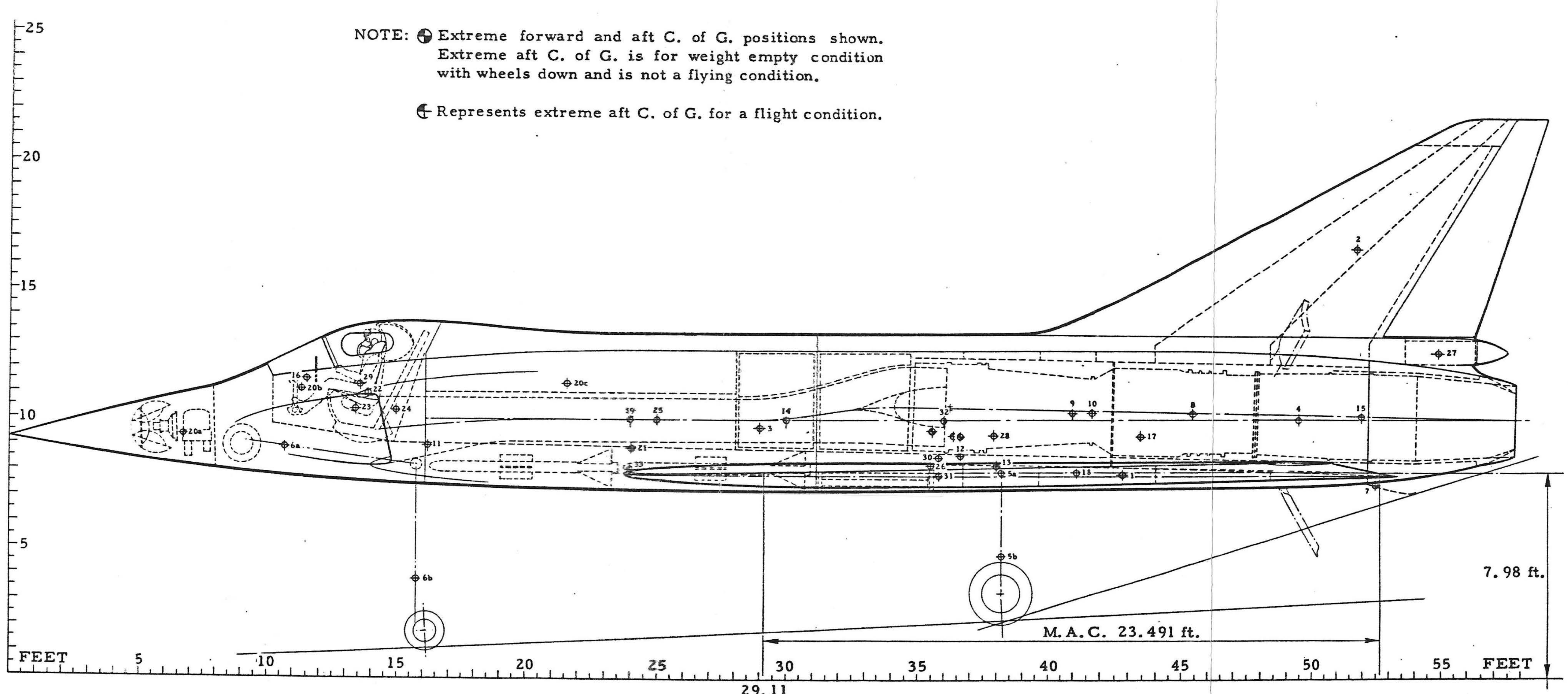
No.	Item	Weight in Pounds	Arm in Feet	Momentin Foot Pounds
	WING GROUP:			
1	Wing including elevons	4,060	42.85	173,971
	TAIL GROUP:			
2	Fin and rudder	525	51.70	27,142
	BODY GROUP:			
3	Fuselage less engine section	3,000	29.00	87,000
4	Speed brakes	120	49.50	5,940
	LANDING GEAR GROUP:			
	Retracted:			
5a	Main gear including jacks	900	38.20	34,380
6a	Nose gear including jacks	295	10.75	3,171
7	Tail skid including jack	10	52.50	525
	Extended:			
5b	Main gear		38.20	34,380
6b	Nose gear		15.80	4,661
	ENGINE SECTION:			
8	Shrouds	210	45.50	9,555
9	Engine and afterburner mounting	20	41.00	820

CONTINUATION OF TABLE 1

No.	Item	Weight in Pounds	Arm in Feet	Moment in Foot Pounds
	POWER PLANT GROUP:			
10	Engine complete	3,772	41.75	157,481
15	Afterburner complete	936	52.00	48,672
11	Power plant controls	10	16.20	162
14	Engine anti-icing	95	30.00	2,850
12	Fuel tanks	217	36.67	7,957
13	Fuel system	250	38.00	9,500
	FIXED EQUIPMENT GROUP:			
16	Instruments	46	11.50	529
17	Surface controls (including hydraulic jacks and artificial feel)	400	43.50	17,400
18	Hydraulic system	599	41.20	24,678
19	Electrical system	550	24.00	13,200
	Hughes MX 1179 system:			
20a	in radome	210	6.75	1,417
20b	in cockpit	190	11.30	2,147
20c	in equipment bay	1,400	21.55	30,170
21	Armament provisions	200	24.00	4,800
	Furnishings:			
22	ejector seat	132	14.00	1,848
23	emergency provisions	15	13.35	200
24	oxygen equipment	20	15.00	300
25	Pneumatic and air conditioning system	215	25.00	5,375
26	Anti-icing equipment	115	35.50	4,082
27	Brake parachute	45	54.90	2,470
28	Exterior finish	43	37.90	1,629
	WEIGHT EMPTY landing gear up	18,600	36.53	679,375
	landing gear down		36.61	680,865
	NON-EXPENDABLE USEFUL LOAD:			
29	Crew (one pilot)	207	13.60	2,815
31	Residual fuel	123	36.15	4,446
32	Oil	20	36.00	720
	GROSS WEIGHT LESS FUEL AND ARMAMENT	18,950		
	Landing gear up		36.28	687,356
	Landing gear down		36.36	688,846

NOTE: Extreme forward and aft C. of G. positions shown. Extreme aft C. of G. is for weight empty condition with wheels down and is not a flying condition.

Represents extreme aft C. of G. for a flight condition.



CONTINUATION OF TABLE 1

No.	Item	Weight in Pounds	Arm in Feet	Moment in Foot Pounds
	EXPENDABLE USEFUL LOAD:			
30	Fuel	8,000	36.15	289,200
33	6 Falcon missiles	660	24.00	15,840
33	24 F.F.A.A. rockets (including jettisonable container)	590	24.00	14,160
	GROSS WEIGHT	28,200		
	Landing gear up		35.69	1,006,556
	Landing gear down		35.75	1,008,046

CENTER OF GRAVITY POSITIONS

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

$$\frac{35.75 - 29.11}{23.49} \times 100 = \frac{664}{23.49} = 28.27\% \text{ M.A.C.}$$

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

$$\frac{36.61 - 29.11}{23.49} \times 100 = \frac{750}{23.49} = 31.92\% \text{ M.A.C.}$$

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

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

$$\frac{36.36 - 29.11}{23.49} \times 100 = \frac{725}{23.49} = 30.86\% \text{ M.A.C.}$$

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

(4) Design gross weight less fuel only, landing gear up:
The C.G. in percent of mean aerodynamic chord is:

$$\frac{35.52 - 29.11}{23.49} \times 100 = \frac{641}{23.49} = 27.29\% \text{ M.A.C.}$$

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

The only alternate gross weight occurs in the trainer version of this aircraft (C104/1/T) and is 30,000 pounds. The furthest forward C.G. in flight for this version is 25.03% M.A.C. with 888 pounds of ballast in the rear fuselage. The estimated limits of Center of Gravity travel as determined by aerodynamic requirements of stability and control are: From 27% to 31% for the C104/1 and from 25% to 31% for the C104/1/T.