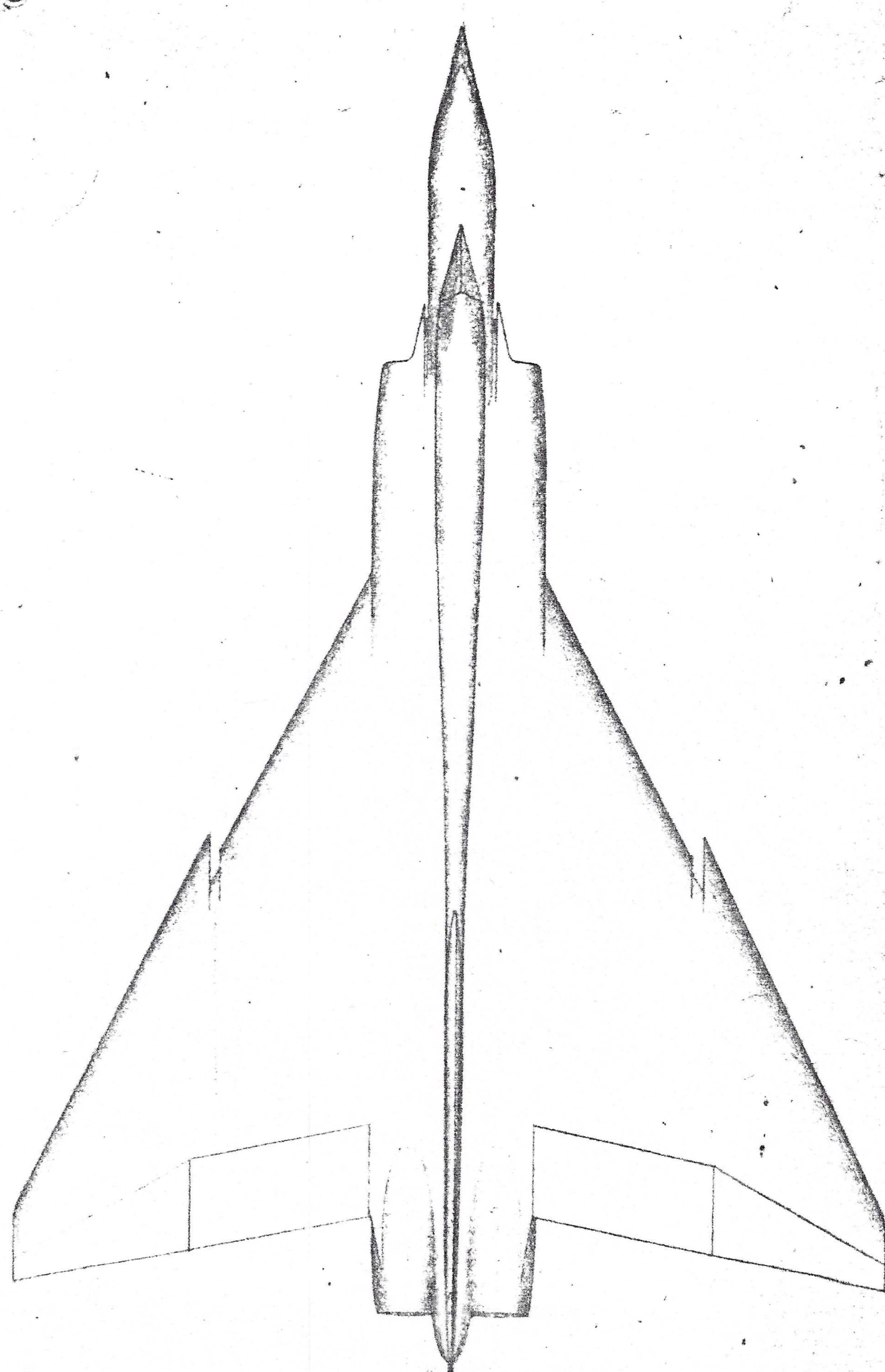

C 105 J75

Scale

RN 48

AAMS-105/1
NOVEMBER 1955

AVRO AIRCRAFT LIMITED

MODEL SPECIFICATION

FOR

SUPERSONIC ALL-WEATHER INTERCEPTOR AIRCRAFT

TYPE CF105

DRAFT COPY

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Number of Pages 176

Copy Number 62

Assigned To R.C.A.F.

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TABLE OF CONTENTS

APPENDICES

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1. Applicable Specifications and Publications

1.1 Referenced Specifications

The following specifications and publications, of the issue in effect on 23 April 1954, shall form a part of this specification to the extent stated herein. At the discretion of the Company subsequently dated issues may be used.

AIR 7-4	R.C.A.F. Specification for Supersonic All-Weather Interceptor Aircraft Type CF105 (31 August 1955)
✓ CAP 479	Manual of Aircraft Requirements for the Royal Canadian Air Force
✓ ARDCM 80-1	Handbook of Instructions for Aircraft Designers (July 1954)
PWA-2611	Model Specification, JT4A-23 Turbo-Jet Engine, Pratt and Whitney
ORI/4-2	R.C.A.F. Standard Flight Panel
ORI/4-5	Operational Requirements, Exterior Lighting
EL-5040-1	Aircraft Doppler Radar System
✓ MIL-B-5087A	Bonding - Electrical (for Aircraft)
✓ MIL-W-5088A	Wiring, Aircraft, Installation of
✓ MIL-I-5099A	Indicator, Cabin Air Pressure, 1-7/8 Inch Dial, Type MA-1
✓ MIL-H-5440A B	Design, Installation and Tests of Aircraft Hydraulic Systems
✓ MIL-F-5572A	Fuel, Aircraft Reciprocating Engine
✓ MIL-O-5606	Oil, Hydraulic, Aircraft, Petroleum Base
✓ MIL-F-5624C	Fuel, Aircraft Turbine and Jet Engine, Grades JP-3 and JP-4
-? ✓ MIL-S-5700 c-1803-E	Structural Criteria, Piloted Airplanes ~~Stress Analysis Criteria~~
✓ MIL-N-5877A	Nozzle, Pressure Fuel Servicing, Locking, Type D-1

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1.1 Referenced Specifications (Cont'd)

MIL-I-5997 A	Instruments and Instrument Panels, Aircraft Installation of
MIL-I-6181B	Interference Limits, Test and Design Requirements, Aircraft Electrical and Electronic Equipment.
MIL-L-6503A	Lighting Equipment, Aircraft, General Specification for Installation of
MIL-C-6818 A	Clamp, Mounting, Aircraft Instruments
MIL-E-7080	Electrical Equipment, Installation of Aircraft, General Specification
MIL-E-7563	Electrical Equipment, Aircraft, Installation of, General
MIL-E-7614	Electrical Equipment, Alternating Current, Aircraft, Installation of, General Specification
MIL-P-7788	Plate, Plastic, Cockpit and Interior Controls Lighting.
MIL-E-~~7894~~ 7984	Electric Power, Aircraft, Characteristics of
MIL-M-7911 MIL-STD-130 PROC 100-4	Marking, Identification of Aeronautical Equipment, Assemblies and Parts
MIL-T-7935	Towing Requirements and Provisions for Land and Carrier Type Military Aircraft
MIL-I-8500A	Interchangeability and Replaceability of Component Parts for Aircraft
MIL-I-8700 (ASG)	Installation and Test of Electronic Equipment in Aircraft, General Specification for
MIL-J-8711 (ASG)	Jack Pads, Aircraft, Design and Installation
AN-L-1A-1 MIL-L-25142	Luminescent Material, Fluorescent
Spec. Bulletin ANC-2a	Ground Loads
USAF Spec. 1815B MIL-F-8785	Flying Qualities of Piloted Airplanes

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1.1 Referenced Specifications (Cont'd)

~~USAF Spec. 1817~~ MIL-F-25352	Flutter, Divergence and Reversal of Aircraft, Prevention of
98-24105-S MIL-M-25047	Marking of Airplanes and Airplane Parts
CGSB 3-GP-22a	Aviation Turbine Fuel - Type II
CGSB 3-GP-25c	Aviation Fuel
RCAF Specification C-28-96 MAT 1-2 (1)	Luminescent Material, Fluorescent - Radioactive

superseded by MAT

1.2 Precedence of Requirements

The requirements of all applicable specifications shall be superseded by those of the approved Model Specification. Otherwise, the following order of precedence shall apply:

(a) AIR 7-4 - R.C.A.F. Specification for Supersonic All-Weather Interceptor Aircraft Type CF105

(b) Contractor Specifications when approved by the R.C.A.F.

(c) CAP 479 - Manual of Aircraft Design Requirements for the Royal Canadian Air Force

(d) ARDCM 80-1 - Handbook of Instructions for Aircraft Designers

(e) The remaining specifications listed in paragraph 1.1 of this specification.

1.3 Deviations

Deviations are set forth in Appendix II of this document and are indicated throughout the text by the appropriate deviation number in parenthesis in the right hand margin. A definition of 'Deviation' appears in paragraph 6.2.

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2. Scope

2.1 Aircraft

The aircraft defined herein shall be the first aircraft of the contract and shall be designed to the requirements of the Royal Canadian Air Force Type Specification AIR 7-4 and such additional requirements as may be specified and agreed upon between the R.C.A.F. and the Company.

2.1.1 This specification describes the following aircraft:

2.1.1.1 R.C.A.F. name and mark number - CF105

2.1.1.2 R.C.A.F. aircraft specification number - AIR 7-4 (Issue 3)

2.1.1.3 Manufacturer's name - Avro Aircraft Limited

2.1.1.4 Manufacturer's model designation - CF105

2.1.1.5 Number of engines - two

2.1.1.6 R.C.A.F. name and mark number of engine -

2.1.1.7 R.C.A.F. engine specification number -

2.1.1.8 Engine manufacturer's name - Pratt and Whitney Aircraft Division of United Aircraft Corp.

2.1.1.9 Engine manufacturer's model designation - J75 Model JT4A-23

2.1.1.10 Engine Model Specification number - PWA 2611

2.2 Role

2.2.1 Primary Role

The primary role of the aircraft shall be high altitude, all-weather, night and day interception and destruction of airborne enemy bomber aircraft. (1)

2.2.2 Secondary Role

The secondary role of the aircraft shall be low altitude, all-weather, night and day interception and destruction of enemy bomber aircraft. However, the aircraft shall be designed to fulfill its primary role and limitations will be accepted in the fulfillment of its secondary role. (1)

2.3 Crew

The crew shall consist of a pilot and an airborne interception radar operator.

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3. Requirements

3.1 Characteristics

3.1.1 Three View Drawing

See Figure 1 Page 7

3.1.2 Interior Arrangement Drawing

See Figure 2 Page 8

3.1.3 Performance

The performance shall be estimated assuming:

(a) All items of removable primary role service load installed and operative.

(b) N.A.C.A. Standard Atmosphere conditions except where otherwise specified.

(c) Engine performance in accordance with the Engine Model Specification (J75 model JT4A-25 engine performance data (as of November 1955) utilized. This represents a 3% optimism in fuel consumption relative to J75 model JT4A-23 data).

(d) All access panels, doors and canopy in the closed position, and landing gear retracted.

3.1.3.1 Tabulated Performance

	Estimated	
Combat Load Factor at Combat Speed of Mach 1.5, Combat Altitude of 50,000 feet, and Combat Weight (51,326 lb.):	1.5	(4)
Maximum Level Speed at 50,000 feet and at Combat Weight (51,326 lb.):	Mach 1.99	
Maximum Allowable Speed with Overload Fuel Tank installed:	Mach 0.95	
Combat Ceiling at Combat Weight (51,326 lb.):	57,200 feet	(6)
With aircraft at Normal Gross Weight (58,975 lb.), and positioned at end of runway, elapsed time from pushing first button to start first engine until aircraft becomes airborne:	0.75 min.	

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3.1.3.1 Tabulated Performance (Cont'd)

	Estimated	
Elapsed time to reach a level flight combat speed of Mach 1.5 and a combat altitude of 50,000 feet from the time aircraft becomes airborne during take-off at normal gross weight (58,975 lb.) under sea level conditions:	3.7 min.	
Take-off distance in still air at maximum gross weight (67,730 lb.) at sea level, and standard summer temperature of 38°C, to clear 50 foot obstacle (Maximum Thrust with afterburning):	6300 feet	(5)
Landing distance from 50 foot obstacle in still air under NACA Standard Atmosphere conditions, at a maximum landing weight of 55,000 lb., at sea level (landing parachute operative after touchdown):	5500 feet	(103)
Touchdown Speed	148 knots	

3.1.4 Performance Curves

3.1.4.1 Stalling Speed vs Weight - Figure 3 Page 9

3.1.4.2 Speed, Rate of Climb, and Time to Height vs Altitude

3.1.4.2.1 Speed vs Altitude - Figure 4 Page 10

3.1.4.2.2 Rate of Climb vs Altitude - Figure 5 Page 11

3.1.4.2.3 Time to Height vs Altitude - Figure 6 Page 12

3.1.4.3 Take-off Performance vs Weight - Figure 7 Page 13

3.1.4.4 Mission Diagrams

3.1.4.4.1 Combat Radius of Action - Figure 8 Page 14

3.1.4.4.2 Cruising Radius of Action - Figure 9 Page 15

3.1.4.4.3 Overload Range - Figure 10 Page 16

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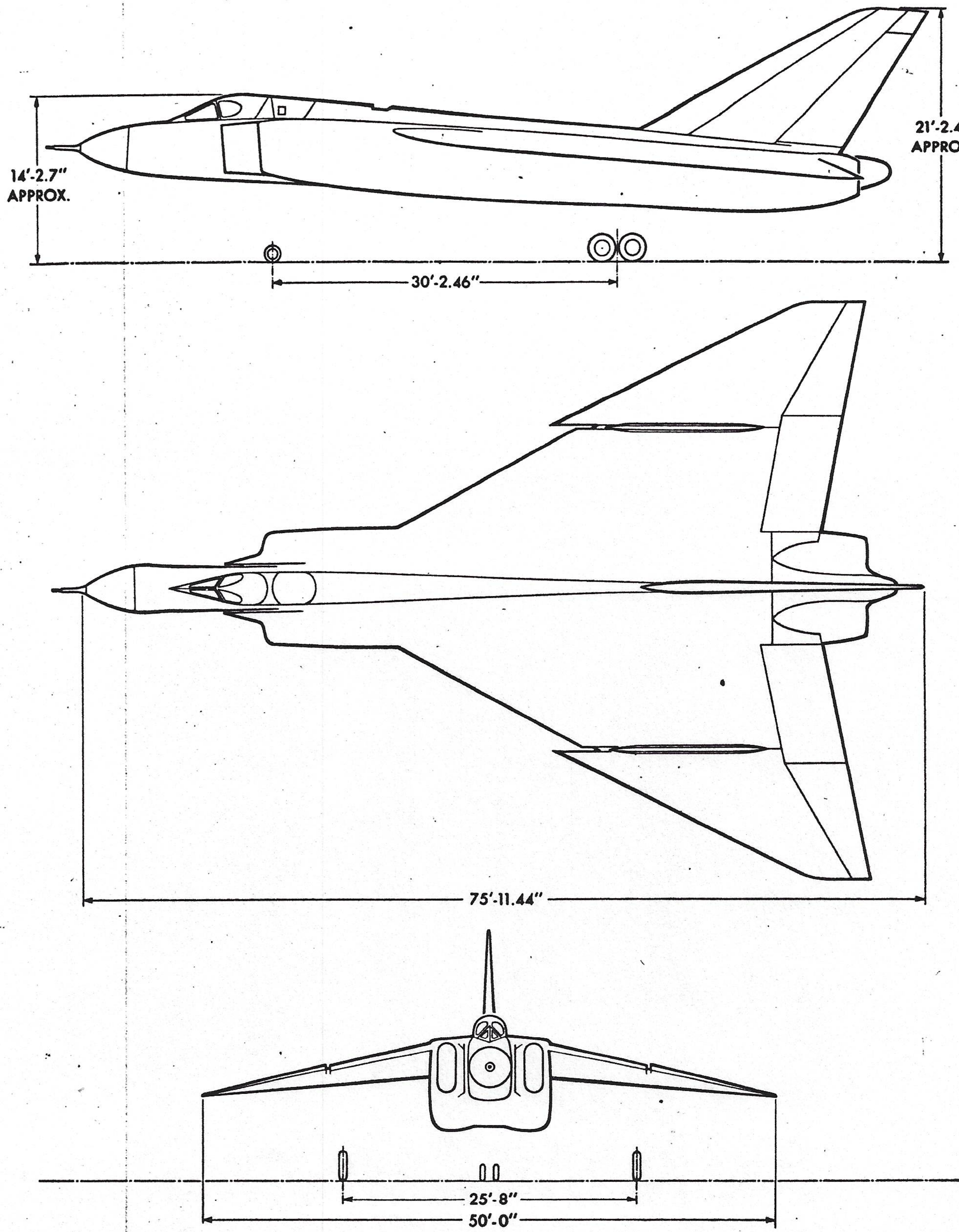

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FIG. 1

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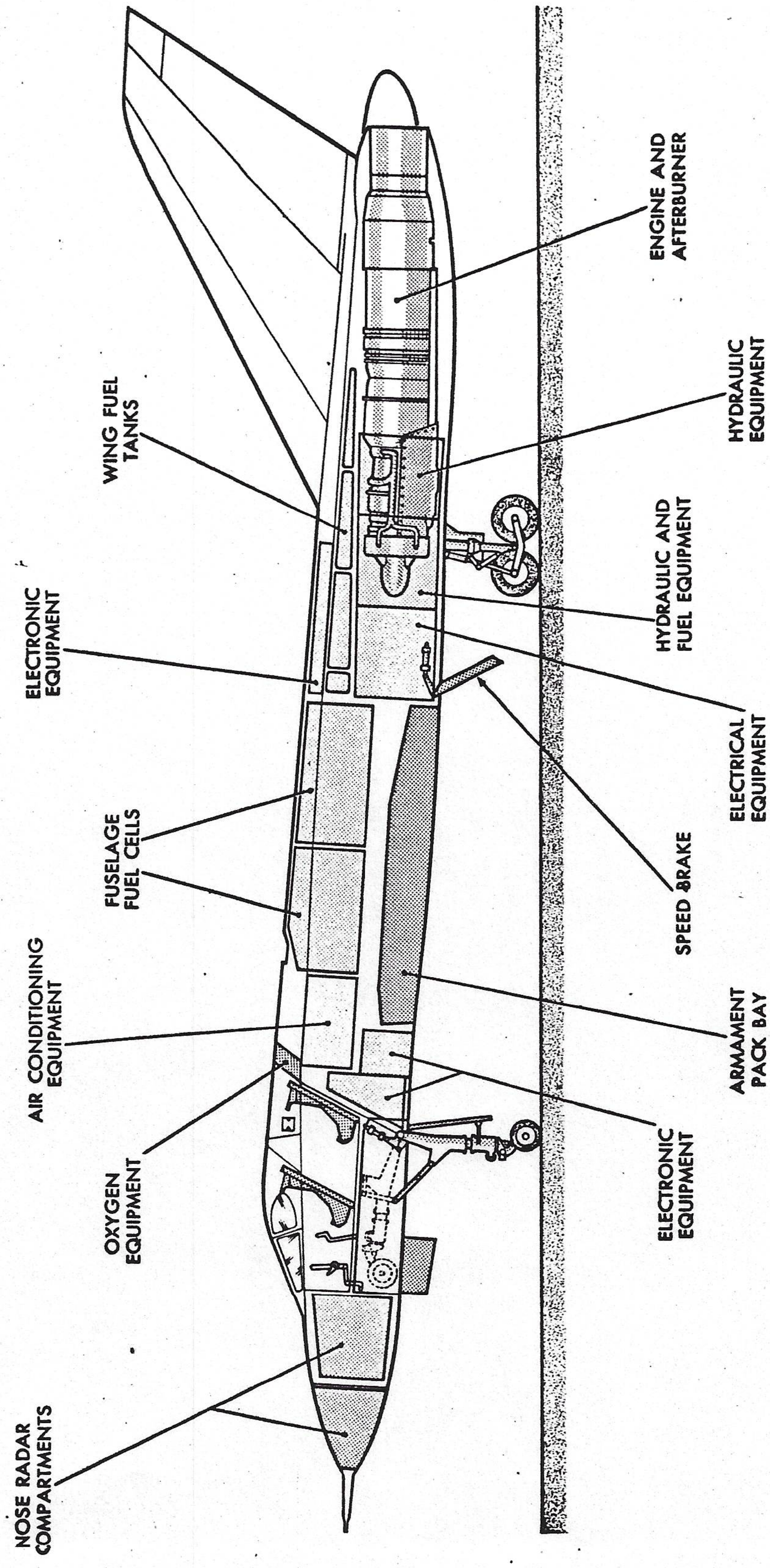

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FIG. 2

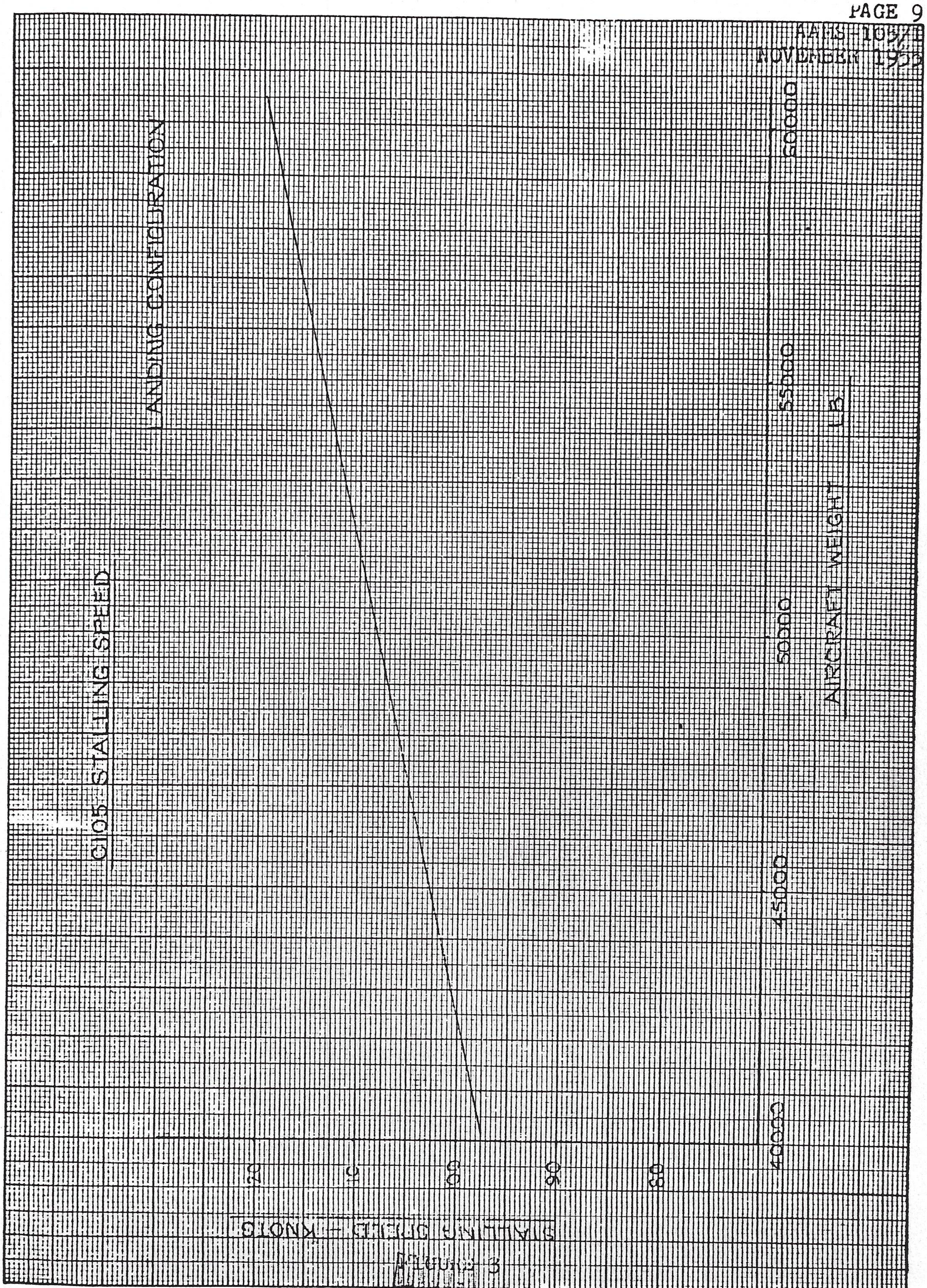

FIGURE 3

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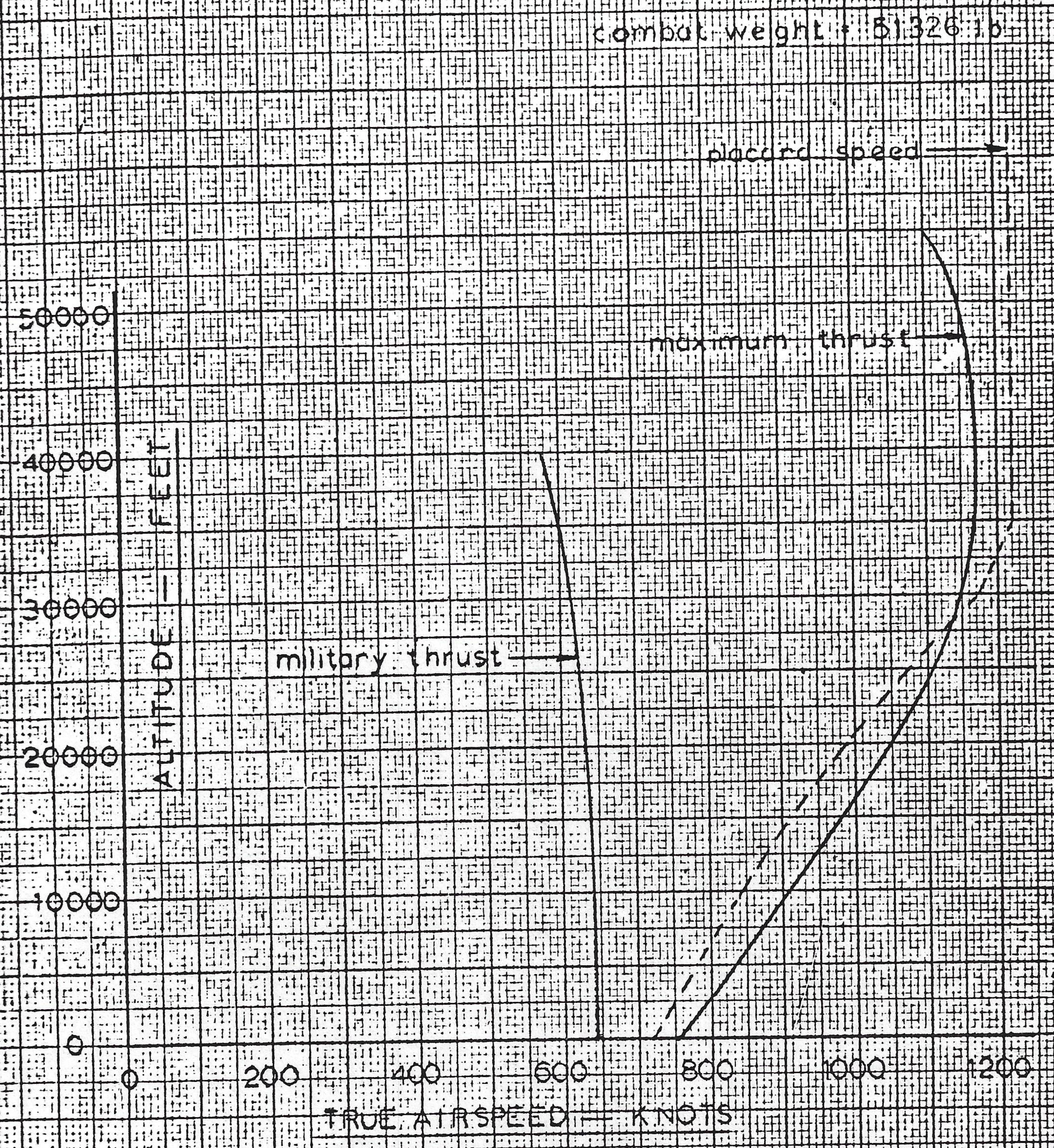
LEVEL FLIGHT TRUE AIRSPEED



KE 10 X 10 TO THE ½ INCH 359-12
KEUFFEL & ESSER CO.

FIGURE

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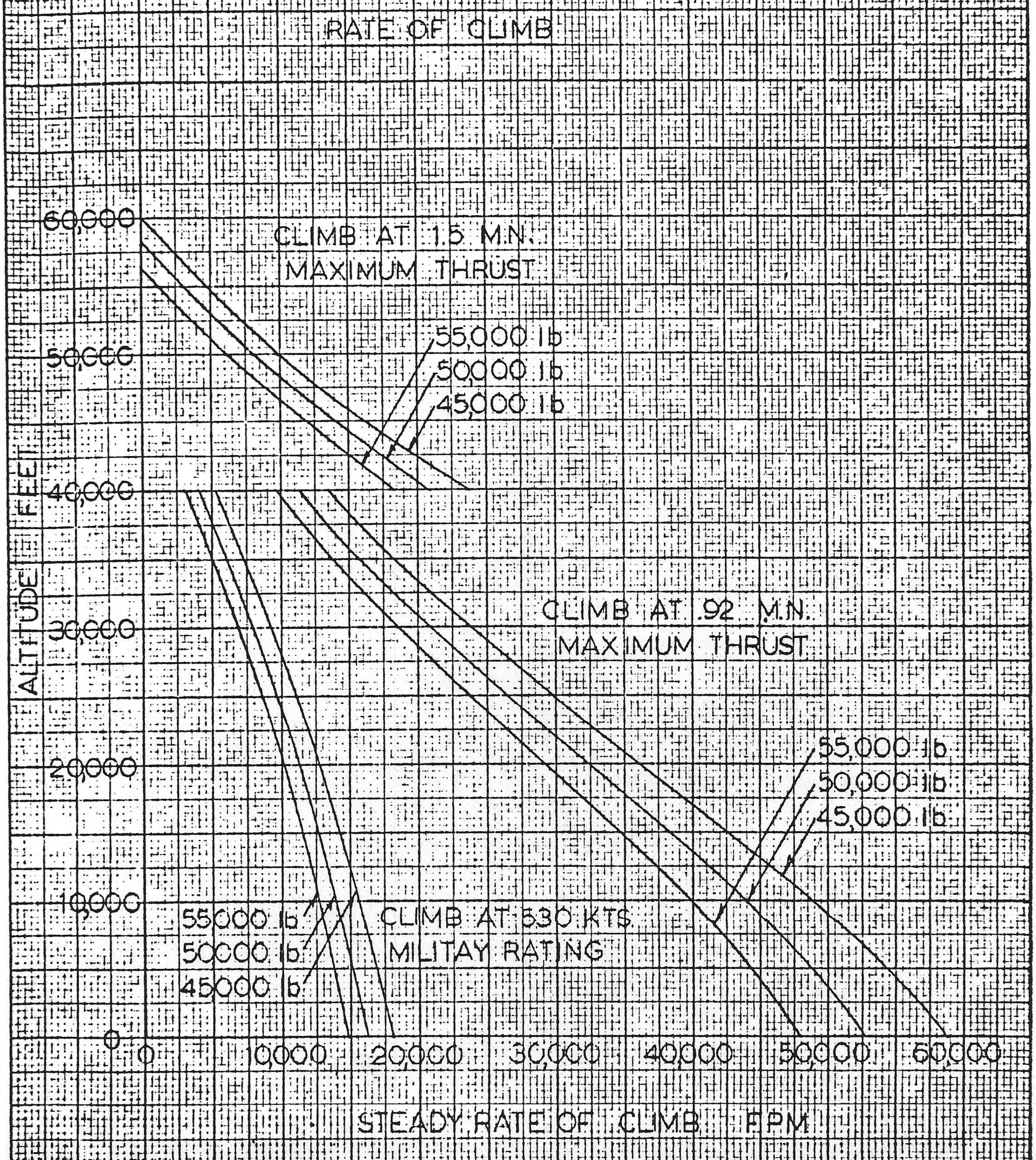

FIGURE 5

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TIME TO HEIGHT

takeoff weight - 58975 lb

one half minute allowed from engine start to military rating

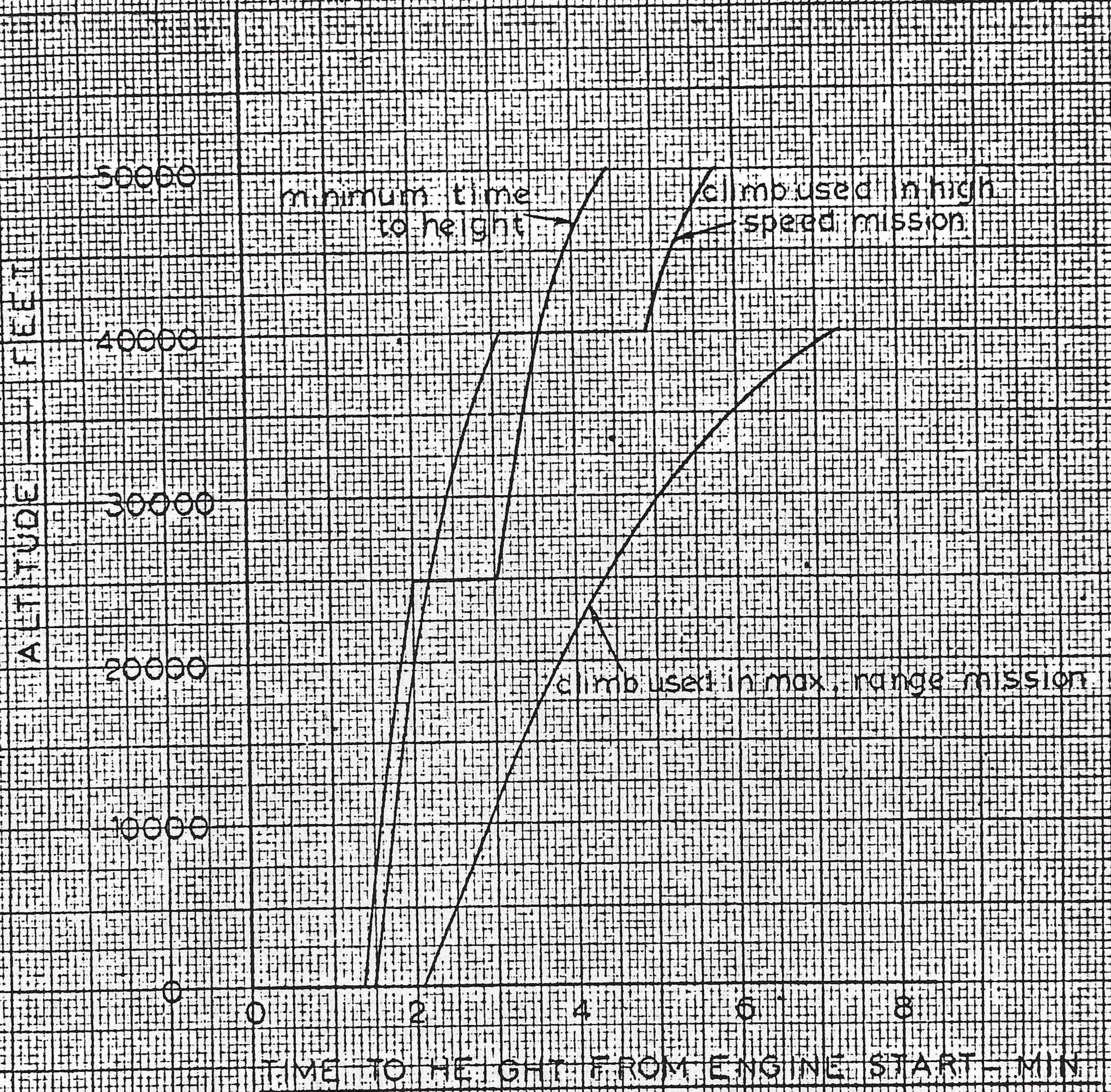

FIGURE 6

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K+E 10 X 10 TO THE 1/2 INCH 359-12 KEUFFEL & ESSER CO. MADE IN U.S.A.

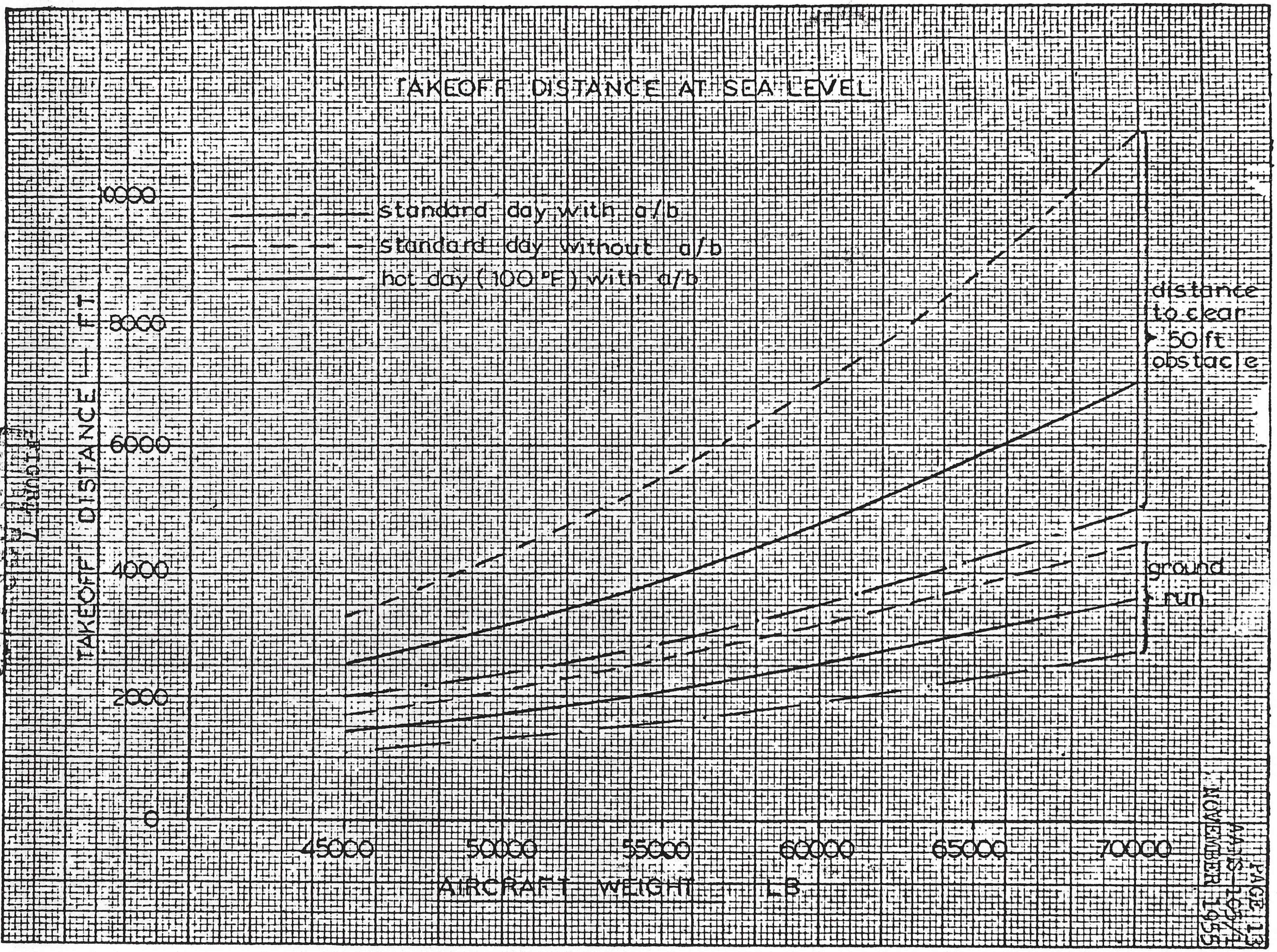

FIGURE 7

359-12 KEUFFEL & ESSER CO.
10 X 10 to the 1/2 inch, 5th lines accented.
MADE IN U.S.A.

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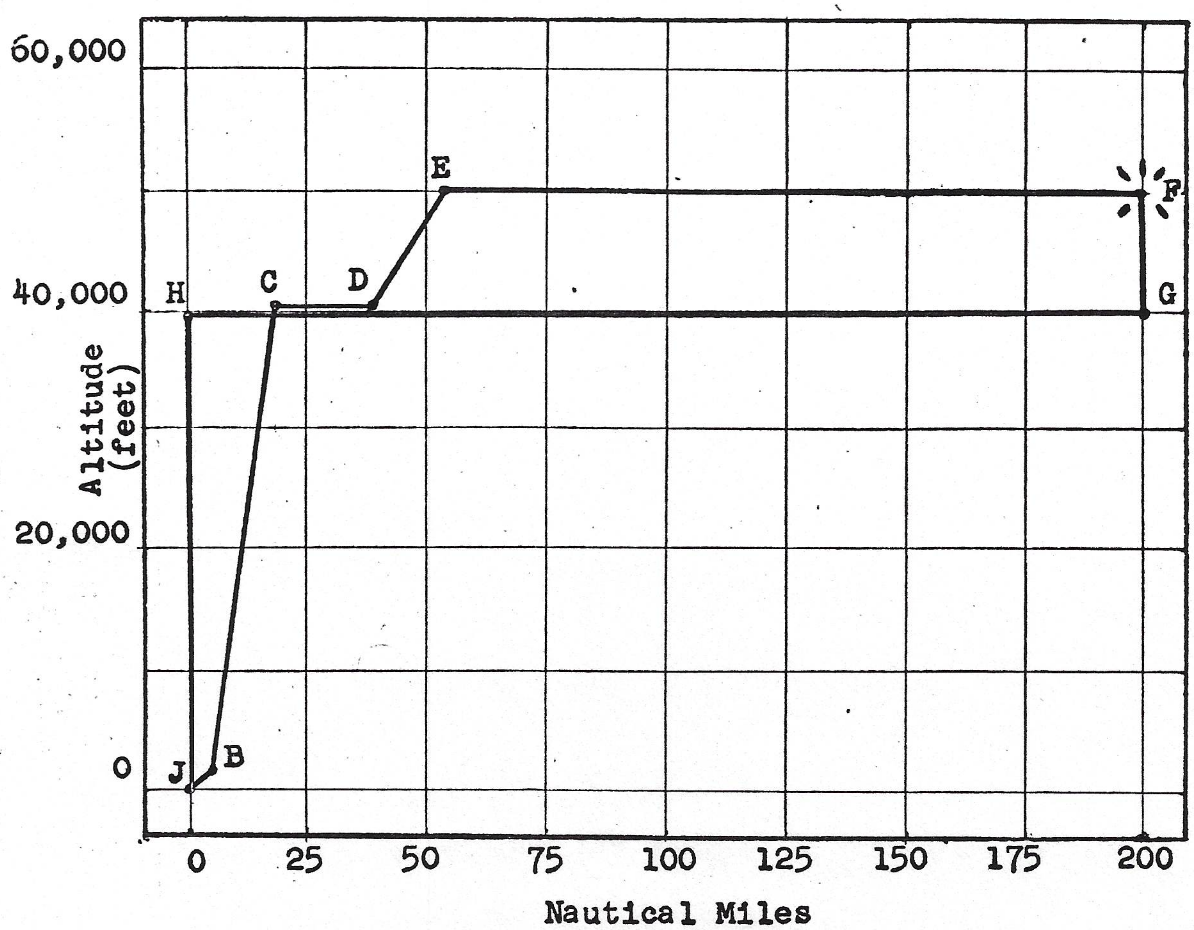
FORM 1746

MISSION DIAGRAM

COMBAT RADIUS OF ACTION



CONDITION		DIST. N.M.	TIME MINS.	FUEL LB.
A	Engine Start	-	.50	100
A	Take-Off to Unstick (Mil Rating)	-	.41	170
A-B	Accel. to Mach .92 (A/B Lit)	3.6	.60	1040
B-C	Climb to 40,000' at Mach .92 (A/B Lit)	14.8	1.60	1570
C-D	Accel. to Mach 1.5 (A/B Lit)	21.5	1.80	1290
D-E	Climb to 50,000' at Mach 1.5 (A/B Lit)	12.5	.87	650
E-F	Cruise Out 50,000' at Mach 1.5	147.6	10.25	4350
F	Combat 50,000' at Mach 1.5 (A/B Lit)	-	5.00	2650
F-G	Descend to 40,000'	-	1.20	40
G-H	Cruise Back (Max. Range)	200	22.64	1750
H	Stack at 40,000' (Max. End.)	-	15.00	1100
H-J	Descend to S.L.	-	3.80	128
J	Land with 5 Min. Fuel (Max. End.)	-	5.00	460
TOTAL		400	68.67	15298

FIGURE 8

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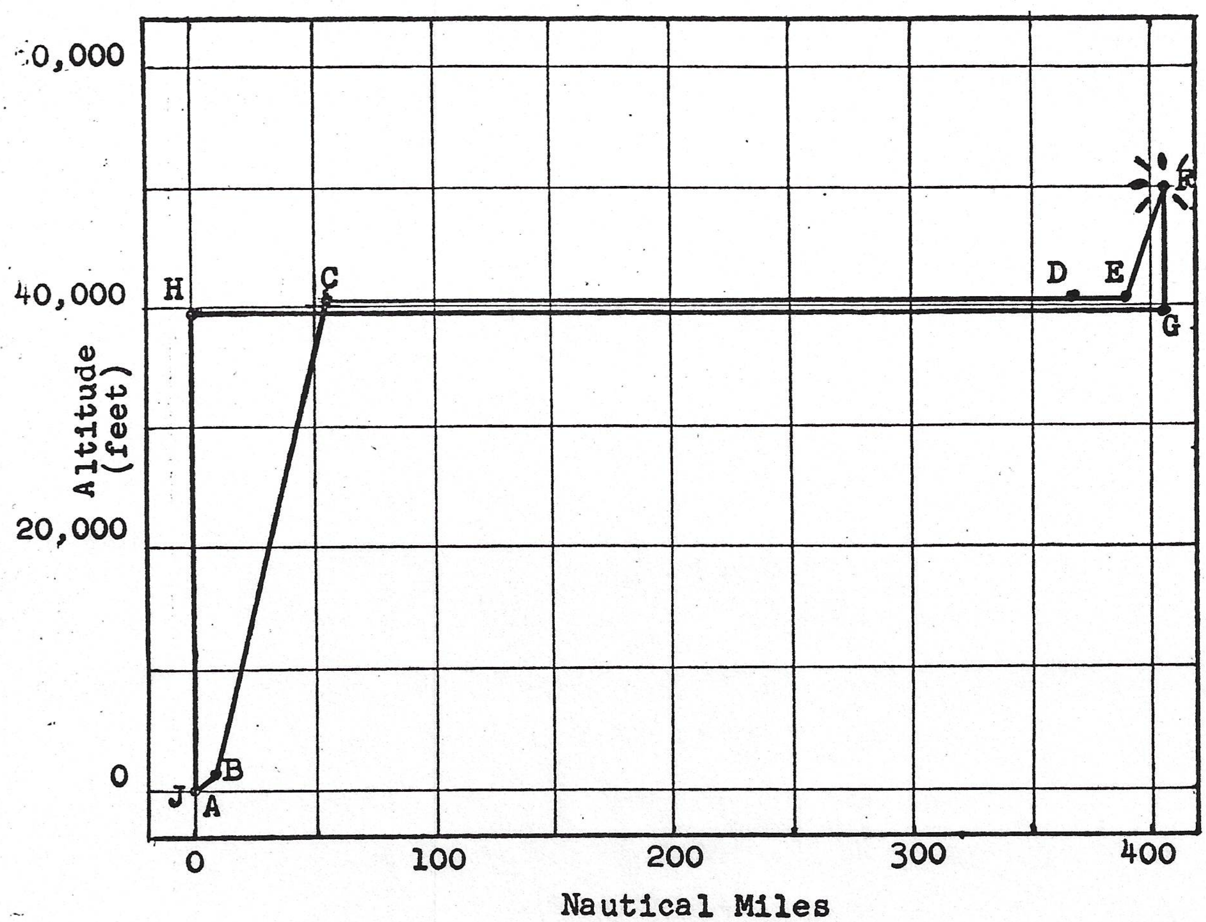
MISSION DIAGRAM

CRUISING RADIUS OF ACTION



	CONDITION	DIST. N.M.	TIME MINS.	FUEL LB.
A	Engine Start	-	.50	100
A	Take-Off to Unstick (Mil Rating)	-	.41	170
A-B	Accel. to 530 Kts. (Mil Rating)	7.0	1.20	520
B-C	Climb to 40,000' at .530 Kts. (Mil Rating)	45.5	5.17	1350
C-D	Cruise out 40,000' (Max. Range)	322.0	36.50	3410
D-E	Accel. to Mach 1.5 40,000' (A/B Lit)	19.7	1.65	1190
E-F	Climb to 50,000' at Mach 1.5(A/B Lit)	11.5	.80	590
F	Combat 50,000' at Mach 1.5 (A/B Lit)	-	5.00	2650
F-G	Descend to 40,000'	-	1.20	40
G-H	Cruise Back 40,000' (Max. Range)	406	45.96	3590
H	Stack at 40,000' (Max. Endurance)	-	15.00	1100
H-J	Descend to Sea Level	-	3.80	128
J	Land with 5 Min. Fuel (Max. End.)	-	5.00	460
	TOTAL	812	122.00	15298

FIGURE 9

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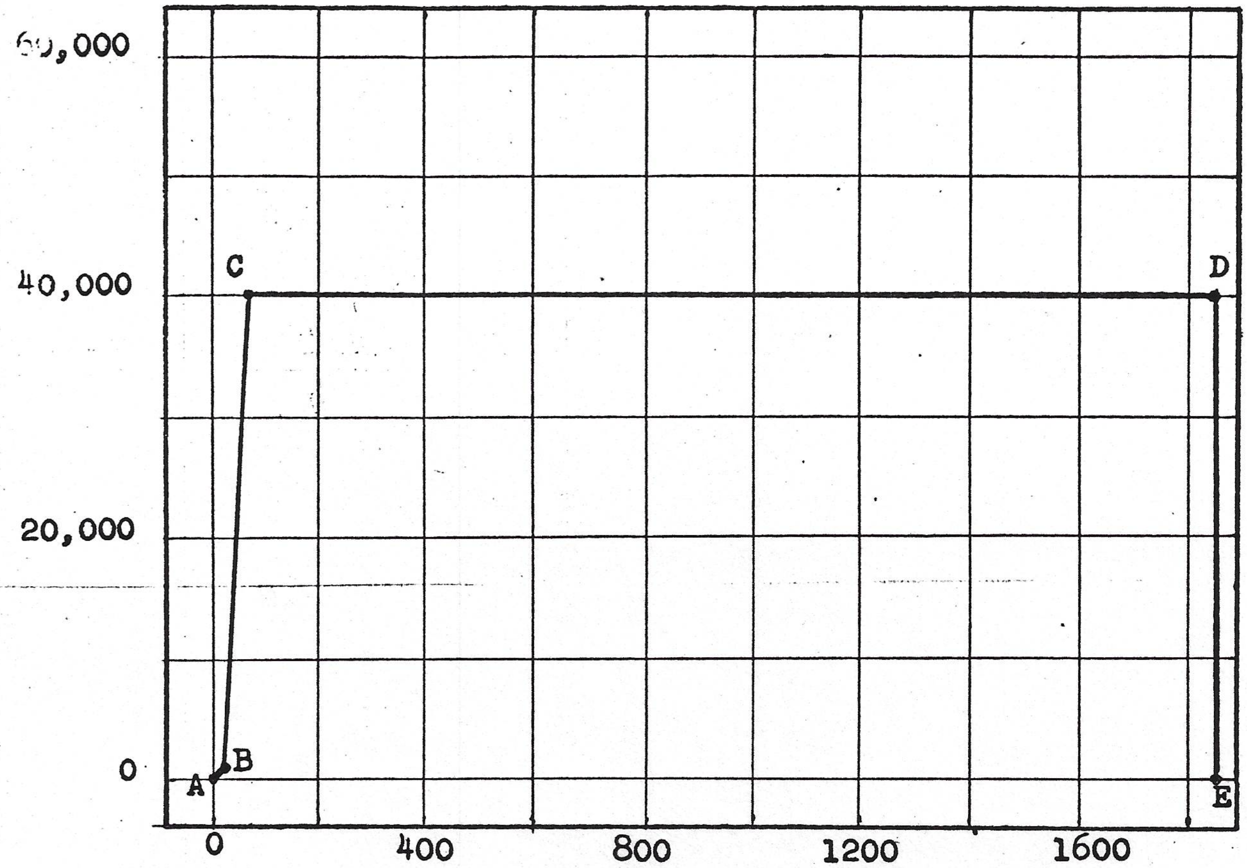
MISSION DIAGRAM

OVERLOAD RANGE



CONDITION		DIST. N.M.	TIME MINS.	FUEL LB
A	Engine Start	-	.50	100
A	Take-Off Unstick (Mil Rating)		.52	212
A-B	Accel. to 530 Kts. (Mil Rating)	8.2	1.40	631
B-C	Climb to 40,000' 530 Kts.(Mil Rating)	61.0	6.90	1740
C-D	Cruise 40,000' (Max. Range)	1790	203.00	19313
D	Stack at 40,000' (Max. Endurance)	-	15.00	1139
D-E	Descend to S.L. (Idling)	-	3.80	132
E	Land with 5 Min. Fuel (Max. Endurance)	-	5.00	476
TOTAL		1859	246.12	23743

FIGURE 10

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3.1.5 Weight and Balance

The estimated weight for the first aircraft is as follows. For current weight status refer to the latest issue of Avro Aircraft Report No. 7-0400-05.

3.1.5.1 Basic Weight

DESCRIPTION	WEIGHT LB.	
STRUCTURE		16,830
Wing	9,542.56	
Fin and Rudder	912.02	
Fuselage Structure Fwd. 255"	2,200.09	
255" to 485"	1,523.08	
Aft 485"	2,652.27	
UNDERCARRIAGE		2,861
Main Undercarriage	1,839.6	
Main U/C Doors & Fairings	287.32	
Main U/C Hydraulics	295.56	
Nose Wheel Undercarriage	294.00	
Nose U/C Doors & Fairings	25.92	
Nose U/C Hydraulics	118.95	
POWER PLANT & SERVICES		13,889
Engines J75	12,647.00	
Gear Box & Drive	150.00	
Engine Controls	25.10	
Pneumatic Starting System	70.00	
Engine De-Icing	65.75	
Fire Extinguishing System	64.27	
Engine Mountings & Brackets	221.11	
Fuel System	645.35	
FLYING CONTROLS GROUP		1,724
Mechanical Flying Controls	784.89	
Flying Controls Electronics	108.00	
Flying Controls Hydraulics	830.87	
EQUIPMENT FIXED & REMOVABLE		6,535
Instruments	57.30	
Probe	15.00	
Oxygen System	46.12	
Air Conditioning System	624.95	
Hydraulic Main System	215.66	
Brake Parachute	68.03	
Electrical System	767.74	
Low Pressure Pneumatics	16.60	
Oil & Hydraulic Fluid Cooling	119.80	

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3.1.5.1 Basic Weight (Cont'd)

DESCRIPTION	WEIGHT LB.	
EQUIPMENT FIXED & REMOVABLE (Cont'd)		
Intake De-Icing	101.72	
Radio & Radar Fixed	150.00	
Radio & Radar Fixed Allowance	771.10	
Canopy Actuation	47.00	
Cabin Consoles	20.85	
Radar Door Actuation	10.00	
Radome Anti-Icing	16.80	
Cabin Insulation	11.91	
Cockpit Pressure Sealing	20.00	
Ejector Seats	204.00	
Emergency Provision	16.95	
Radar Removable	380.7	
Radar Removable, Allowance for	879.00	
Radio Removable & I.F.F.	113.50	
Radio Removable, Allowance for	162.70	
Missile Pack Structure, Allowance for	676.17	
Missile Pack Mechanism Allowance for	410.48	
Missile Pack Hydraulics, Allowance for	293.00	
Missile Pack Electronics Allowance for	318.00	
AIRCRAFT WEIGHT EMPTY		41,839
USEFUL LOAD		17,136
Crew	430.00	
Oil	85.08	
Alcohol for Radome De-Icing	22.00	
Residual Fuel	219.80	
Fuel for Combat Mission	15,298.00	
Missiles (Armament) Allowance for	1,042.40	
Oxygen Charge	13.39	
Engine Fire Extinguisher Fluid	25.00	
Normal Combat Mission	58,975.00	
Half Combat Mission Fuel 980 @ 7.8 lb./gal.	7,649.00	
Combat Weight (Half Combat Mission Fuel)	51,326.00	
Operational Weight Empty	43,677.00	

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3.1.5.1 Basic Weight (Cont'd)

DESCRIPTION	WEIGHT LB.
Maximum Internal Fuel 2,544 gal. @ 7.8 lb./gal.	19,843.00
A.U.W. Max. Internal Fuel	63,520.00
Max. External Fuel, 500 gal. @ 7.8 lb./gal. & Drop Tank	4,210.00
A.U.W. Max. Internal & External Fuel	67,730.00

3.1.5.2 Unit Weights

(a) Wing Group (Gross Area 1225 sq. ft.) 7.778 lb./sq. ft.

(b) Vertical Tail (Gross Area 158.79 sq. ft.) 5.737 lb./ sq. ft.

(c) Fuel System (Capacity 2544 Imp. Gal.) .253 lb./Imp. Gal.

(d) Lubrication System - Not applicable.

(e) Cooling System - Not applicable.

3.1.5.3 Balance

The c.g. limits of the aircraft are estimated to be:

Forward Limit 28% of the M.A.C. (limited structurally)

Aft Limit 31% of the M.A.C. (limited aerodynamically)

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3.1.6 Areas (Not to be used for inspection purposes)

Wing area (including ailerons, elevators and 390.50 sq. ft. of fuselage and not including extended leading edge)	1225.00 sq. ft.
Aileron area (aft of hinge line)	66.55 sq. ft.
Elevator area (aft of hinge line)	106.90 sq. ft.
Vertical tail area (including rudder)	158.79 sq. ft.
Fin area	120.62 sq. ft.
Rudder area (aft of hinge line)	38.17 sq. ft.
Speed brake area - 2 - (Projected)	14.367 sq. ft.

3.1.7 Dimensions and General Data (Not to be used for inspection purposes)

3.1.7.1 Wings

Span	50 ft. 0.00 in.
Chord - Root	45 ft. 0.00 in.
- Construction Tip	4 ft. 0.00 in.
Mean Aerodynamic Chord	30 ft. 2.61 in.
Airfoil Section - Inner Wing	
- Profile	.0003.5-6-3.7 (Modified)
- Camber	.0075 (Modified)
- Leading Edge Droop	8.0 degrees
- Outer Wing	
- Profile	.0003.5-6-3.7 (Modified)
	.0003.8-6-3.7
- Camber	.0075 (Modified)
- Leading Edge Droop	4.0 degrees
Incidence - Root	0 degrees
- Construction Tip	0 degrees
Sweep back - Leading Edge	61.45 degrees
- Trailing Edge	11.21 degrees
- ¼ Chord	55.00 degrees
Anhedral	4.00 degrees
Aspect Ratio	2.04
Ailerons - Span (each)	10 ft. 0.00 in.
- Chord (average percent wing chord)	
- Root	25.735
- Tip	35.000
Elevators - Span (each)	10 ft. 2.00 in.
- Chord (average percent wing chord)	
- Root	14.109
- Tip	25.735

3.1.7.2 Horizontal Tail

Not applicable.

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3.1.7.3 Vertical Tail

Span	12 ft. 10.50 in.
Chord - Root	19 ft. 0.00 in.
- Construction Tip	5 ft. 8.00 in.
Mean Aerodynamic Chord	13 ft. 6.41 in.
Airfoil Section	.0004-6-3.7 (Modified)
Sweep back - Leading Edge	59.34 degrees
- Trailing Edge	33.08 degrees
- ¼ Chord	55.00 degrees
Aspect Ratio	1.04
Rudder - Span (average)	9 ft. 11.00 in.
- Chord (average percent vertical tail chord)	30.00 in.

3.1.7.4 Speed Brakes

Span (each)	2 ft. 1.08 in.
Chord	4 ft. 1.00 in.

3.1.7.5 Height of Aircraft

Reference to ground static line .... 21 ft. 2.07 in.

3.1.7.6 Length of Aircraft (Not including 3 ft. (approx.) probe)

Aircraft reference line level .... 75 ft. 11.45 in.

3.1.7.7 Propeller

Not applicable.

3.1.7.8 Landing Gear

Tread	25 ft. 7.34 in.
Wheel Base	30 ft. 2.46 in.

3.1.7.9 Ground Angle

Angle between aircraft reference line and ground static line .... 4.55 degrees

3.1.8 Control Surfaces and Corresponding Control Movements (Not to be used for inspection purposes)

		Surface Movement	Control Movement
Ailerons:	up and down	19°	14.20°
Elevators:	up	30°	14.5°
	down	20°	9.67°
Rudder:	left and right	30°	3.25 in.
Speed Brake		60°	-

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3.2 Construction

3.2.1 General Interior

A pressurized compartment for the accommodation of the crew and incorporating instruments, controls, and stowages as described in the appropriate sections of this specification shall be located in the nose section of the aircraft. The front cockpit shall be equipped to accommodate the pilot, and the rear cockpit equipped to accommodate the radar operator. It shall be possible for the pilot to perform all the normal and emergency functions required to fly the aircraft without the assistance or presence of the second crew member. The cockpit compartment shall be enclosed by a fixed windshield and a canopy with two electrically actuated split clamshell type hatches. Equipment and service compartments shall be as described in paragraph 3.7.5.

3.2.2 Materials

Materials used in construction of the airframe and contractor furnished equipment shall conform to:

(a) Requirements issued by D.N.D. or approved by the R.C.A.F. as covered by CAP 479 Part 5; or

(b) Requirements covered by ARDCM 80-1; or

(c) R.C.A.F. approved Company specifications.

3.2.3 Standards

Standards used in construction of the airfram shall conform to:

(a) Standards issued or approved by the R.C.A.F., as covered by CAP 479 Part 5; or

(b) Company design standards; or

(c) Standards covered by ARDCM 80-1; or (2) (3)

(d) ABC standards as agreed between the R.C.A.F. and the Company.

3.2.4 Workmanship

All workmanship and shop practice shall be in accordance with accepted standards of aeronautical engineering practice.

3.2.5 Production, Maintenance and Repair

The design of the aircraft shall be such as to be suitable for large scale production. Consideration shall be given during the design to provide access to the aircraft and installed equipment to facilitate ease of replacement, maintenance, and repair. Maintenance provisions incorporated in the aircraft and the equipment installed therein shall conform to:

(a) Requirements issued by D.N.D. or approved by the R.C.A.F. as covered by CAP 479 Chapter 4; or

(b) Requirements covered by ARDCM 80-1.

The above considerations and requirements shall be subordinate only to the fulfillment of the primary role of the aircraft and to the safety of the crew.

3.2.6 Climatic Conditions

The aircraft shall be designed to meet the environmental conditions given in ARDCM 80-1, and additionally for operation within the design flight conditions.

All contractor furnished equipment installed in the aircraft shall be designed to meet the environmental conditions given in ARDCM 80-1 and additionally, where the pressure altitude and/or temperature is in excess of that covered by ARDCM 80-1 the requirements in the Company Equipment Specifications and Company Specification Avrocan E-266, as applicable, shall govern.

3.2.7 Noise and Vibration

The aircraft shall be designed so that the local vibration characteristics shall not exceed the limits specified in the applicable equipment specification.

Noise levels at the head positions of the occupants at their respective stations, during flight under cruising conditions shall not normally exceed the values as given in CAP 479 paragraph 20.04.

3.2.8 Processes

Processes used in construction of the airframe and contractor furnished equipment incorporated in the various systems installed therein shall conform to:

(a) Requirements issued by D.N.D. or approved by the R.C.A.F. as covered by CAP 479 Part 5 (including

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3.2.8 Processes (Cont'd)

(a) (Cont'd)
Processes covered by R.C.A.F. approved Company Specifications; or

(b) Requirements covered by ARDCM 80-1.

3.2.9 Finish

The finish on all parts and components shall be in accordance with Avro Aircraft Company Standard CSD 2, and CAP 479 Part 5.

3.2.10 Colour Scheme and Identification Markings

Aircraft components and aircraft parts shall be marked in accordance with MIL-M-7911 and Specification 98-24105-5. The aircraft exterior colour scheme and markings shall conform to:

(a) CAP 479 Chapter 6; or

(b) Company Drawings, where not covered by (a).

3.2.11 Pipeline Identification

All pipeline and electrical conduit in the aircraft shall be marked in accordance with CAP 479 Chapter 6, together with such additional markings as may be required by the specification governing each system.

3.2.12 Electrical Circuit Identification

Identification of electrical circuits shall be in accordance with the requirements of Specification MIL-W-5088A, and as additionally agreed between the R.C.A.F. and the Company.

3.2.13 Interchangeability

Interchangeability and replaceability shall conform to the requirements of Specification MIL-I-8500A and Avro Aircraft Report QC-E-8.

3.2.14 Lubrication

The lubrication schedule and types of lubricants to be used shall be as detailed in the Description and Maintenance Instructions for the CF105 aircraft.

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3.2.15 Equipment

Contractor furnished equipment incorporated in the various systems installed in the aircraft shall conform to:

(a) Requirements issued by D.N.D. or approved by the R.C.A.F., as covered by CAP 479 Part 5; or

(b) Requirements covered by ARDCM 80-1; or

(c) R.C.A.F. approved Company specifications.

Government supplied equipment shall be suitable for use under the conditions appropriate to its purpose in the aircraft and shall be installed in the aircraft without modification by the Company.

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3.3 Aerodynamics

3.3.1 General

The aircraft shall be a high wing, delta planform with 4° anhedral, and of moderate wing loading. The utmost consideration shall be given to cleanness of design with all antennas flush mounted and proturberances kept to a minimum.

The aerodynamic characteristics of the aircraft shall be such as to permit the accomplishment of the primary role as stated in paragraph 2.2.1. These characteristics, including performance, controllability, stability and flutter shall conform to the requirements of ARDCM 80-1, except as stated in the Deviations given in Appendix II.

3.3.1.1 Special Characteristics

The wing leading edge shall be slotted, extended and drooped (as described in paragraph 3.5.2.3) to prevent "pitch-up" at high lift coefficients and to extend the aerodynamic aft center of gravity limit.

A maximum camber of .75%C (Negative) shall be incorporated in the wing design in order to reduce the required elevator deflection and to increase the airplane ceiling.

The air intake for the air induction system to the engine shall be preceded by a fixed wedge shaped ramp adjacent to the fuselage. The wedge angle of the ramp shall be designed so as to:

(a) Induce (at supersonic mach numbers) an oblique shock wave near the lip of the ramp and a shock wave normal to the ramp in order to reduce intake pressure losses.

(b) Prevent formation of a shock wave within the engine air intake.

(c) Provide for a minimum amount of spillage drag at supersonic mach numbers.

A boundary layer bleed shall be installed between the fixed ramp and the fuselage to prevent boundary layer breakaway and to improve intake efficiency.

A two position annular by-pass around the engines shall be provided to increase the intake stable mass flow range, improve intake efficiency, reduce spillage drag and supply air to the exhaust nozzle ejector.

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3.3.1.2 Aerodynamic Data

Aerodynamic data, including lift, moment, drag, yaw, thrust, take-off and landing, stability and controllability characteristics of the aircraft will be found in the reports listed in Appendix III.

3.3.2 Stability and Control

The aircraft shall be designed to meet the stability and control requirements of U.S.A.F. Specification 1815B except as stated in Avro Aircraft Report Number Aero Data 66. MIL-F-8785

3.3.3 Aero-Elasticity

Flutter and divergence calculations shall be computed in accordance with the requirements of U.S.A.F. Specification 1817. MIL-F-2356

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3.4 Structural Design Criteria

The structural design of the aircraft shall be in accordance (100)
with the requirements of Specification MIL-S-5700 and limit (101)
load factors as stated below, and shall be based on a weight (102)
for stress analysis of 47,000 lb. (103)

3.4.1 Limit Flight Load Factors

3.4.1.1 Gross Weight for Stress Analysis 47,000 Lb.

		Clean Configuration	Missile Firing	Aux. Tank Inst
Maneuver	Positive	+7.33 *	+4.00	+4.50 **
	Negative	-3.00	0.	-1.50 **
Gust	Positive	+3.9		
	Negative	-1.9		

* The limit load factor above Mach 1.0 shall decrease (99)
from 7.33 as a function of skin temperature.

** Up to the structural speed limitation of Mach 0.95.

3.4.1.2 Weight in Excess of 47,000 Lb.

In the clean configuration and at weights in excess of 47,000 lb., the aircraft limit flight maneuver load factor shall be:

$$n_1 = \frac{47000}{W} \times n$$

3.4.1.3 Flight Envelopes

In addition to the above the limit flight load factors shall be as shown in the following flight envelopes.

Sea Level:	Figure 11 Page 30
10,000 feet:	Figure 12 Page 31
20,000 feet:	Figure 13 Page 32
30,000 feet:	Figure 14 Page 33
Tropopause:	Figure 15 Page 34
40,000 feet:	Figure 16 Page 35
50,000 feet:	Figure 17 Page 36
60,000 feet:	Figure 18 Page 37
70,000 feet:	Figure 19 Page 38

The following symbols, as utilized on the above noted flight envelopes, shall be defined as:

Line A-B = Positive Structural Maneuver Limit
Line D-C = Negative Structural Maneuver Limit

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3.4.1.3 Flight Envelopes (Cont'd)

Line B-C = Limit Dive Speed
Lines H = Positive Gust
Lines J = Negative Gust
$V_E = V\sqrt{\sigma}$ KTS. = Equivalent Air Speed (Knots)
M = Mach Number

3.4.1.4 Load Factors in Roll

The limit flight maneuver load factors in rolling shall be in accordance with the requirements of MIL-S-5702 except that the maximum load factor in a rolling pull-out shall be 4.89. (9)

3.4.1.5 Load Factors in Spin

The limit flight maneuver load factors in spin shall be in accordance with the requirements of MIL-S-5702 except that yawing velocity in a flat spin shall be reduced from 5 radians per second to 3.5 radians per second. (8)

3.4.2 Limit Ground Load Factors

The design limit ground load factors shall be computed in accordance with the requirements of Specification Bulletin ANC-2a. (7)

3.4.3 Limit Diving Speed

The limit diving speed shall be as shown on the flight envelopes (Reference paragraph 3.4.1).

3.4.4 Ditching Criteria

Not applicable.

3.4.5 Ultimate Loads

All limit loads derived from the above criteria shall be multiplied by 1.365 to obtain ultimate loads.

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359-12 KEUFFEL & ESSER CO.
10 X 10 to the ½ inch, 5th lines accented.
MADE IN U.S.A.

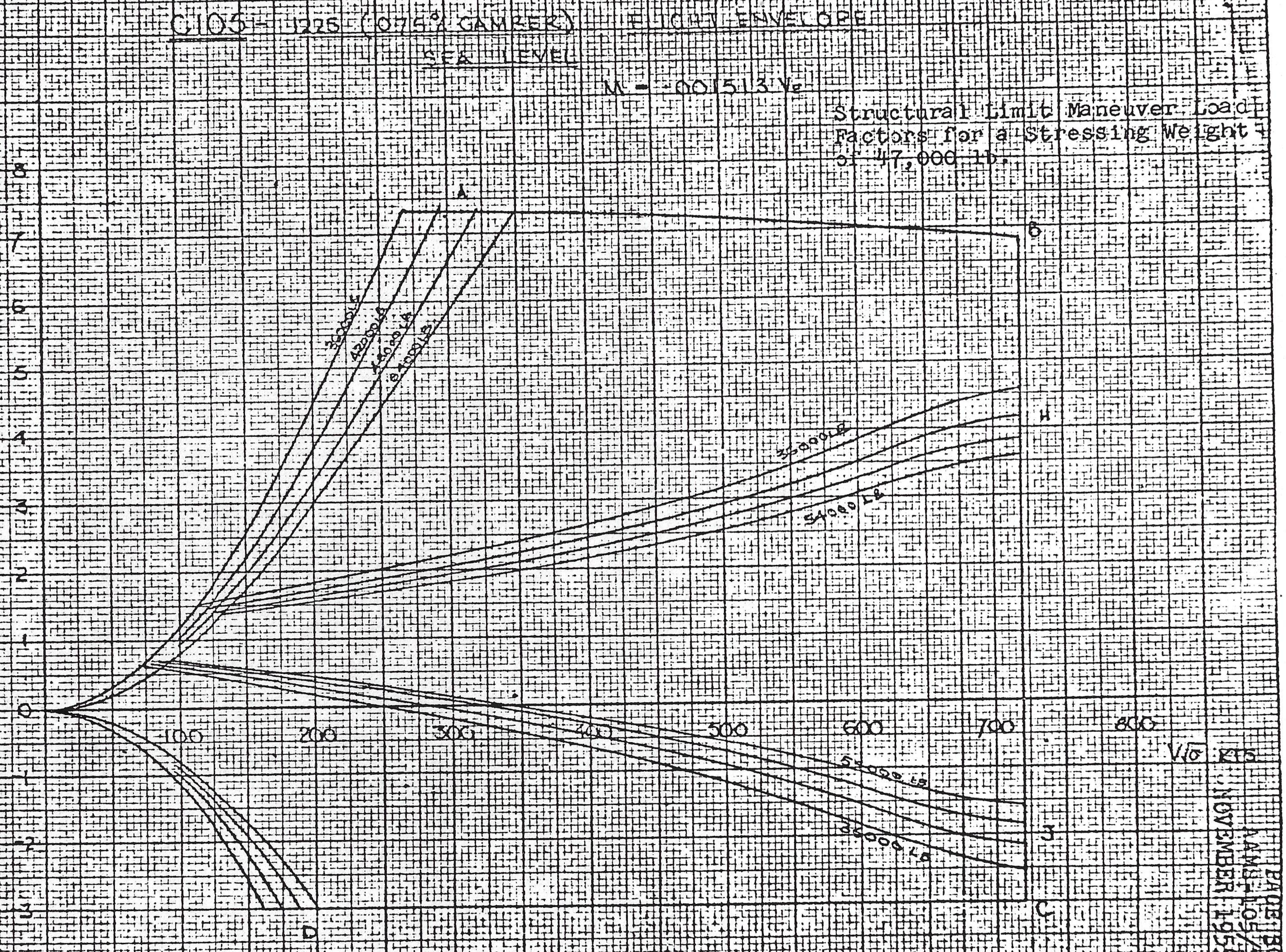


FIGURE 11

FORM 1746

359-12 KEUFFEL & ESSER CO.
10 X 10 to the ½ inch, 5th lines accented.
MADE IN U.S.A.

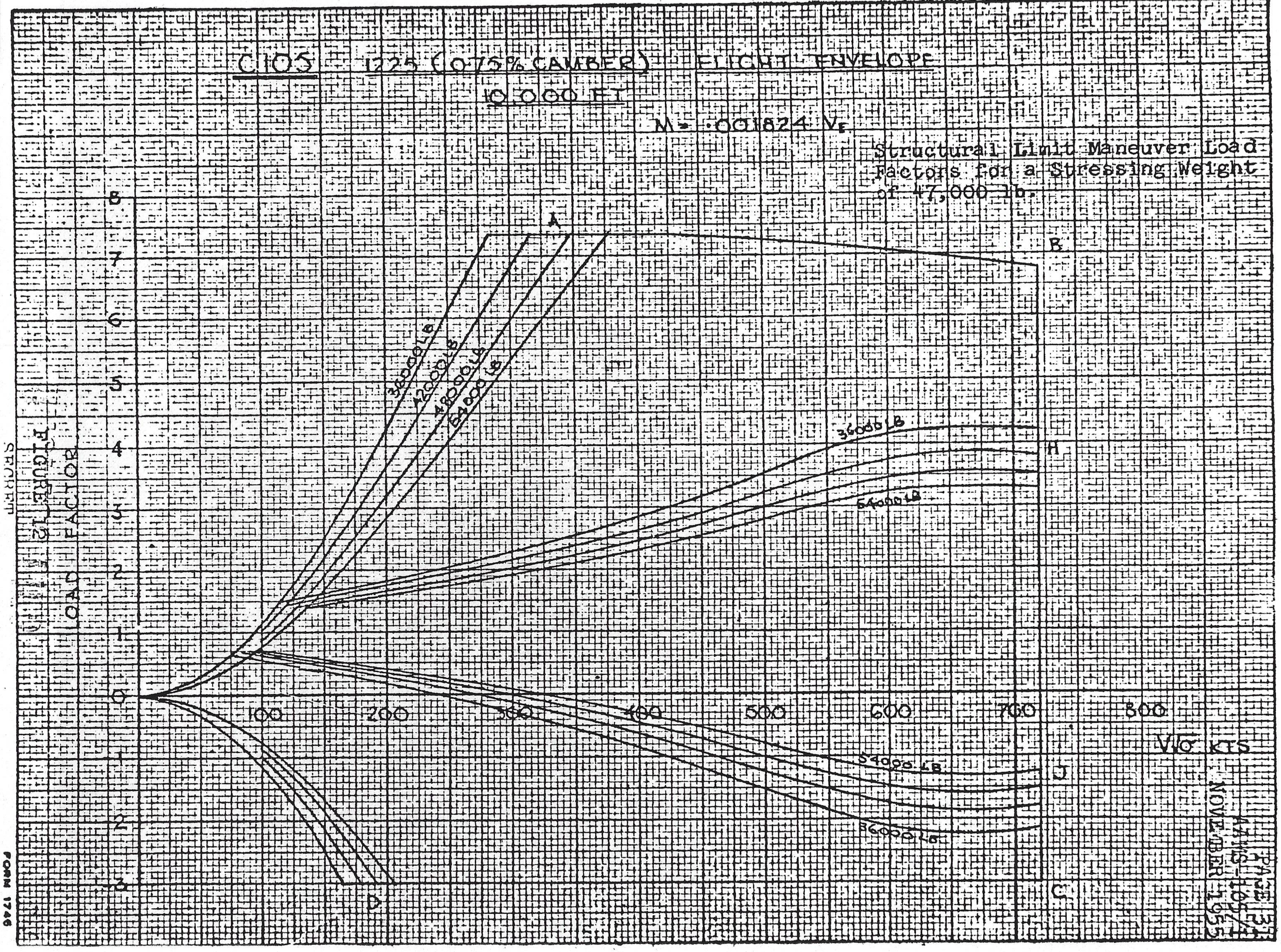


FIGURE 12

FORM 1746

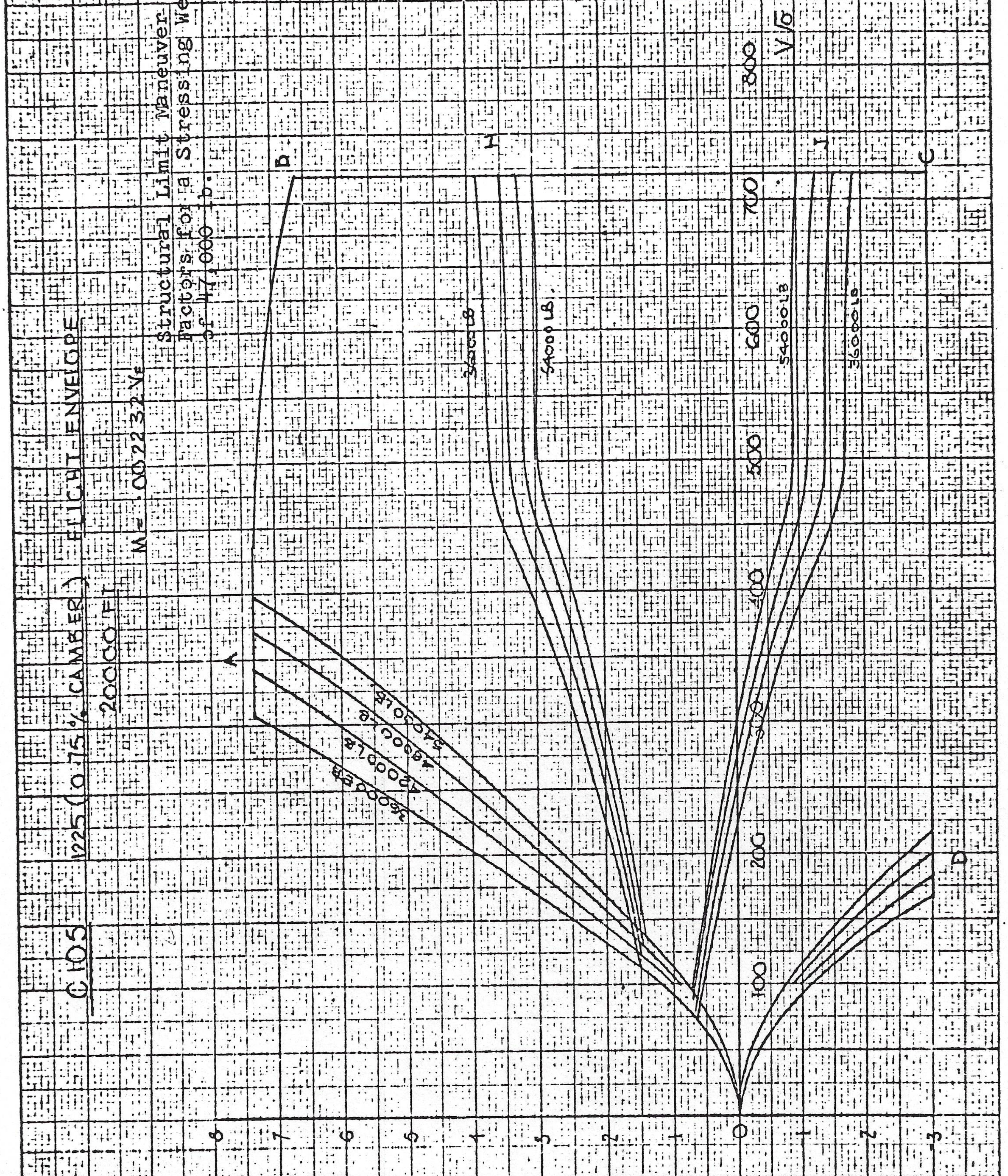
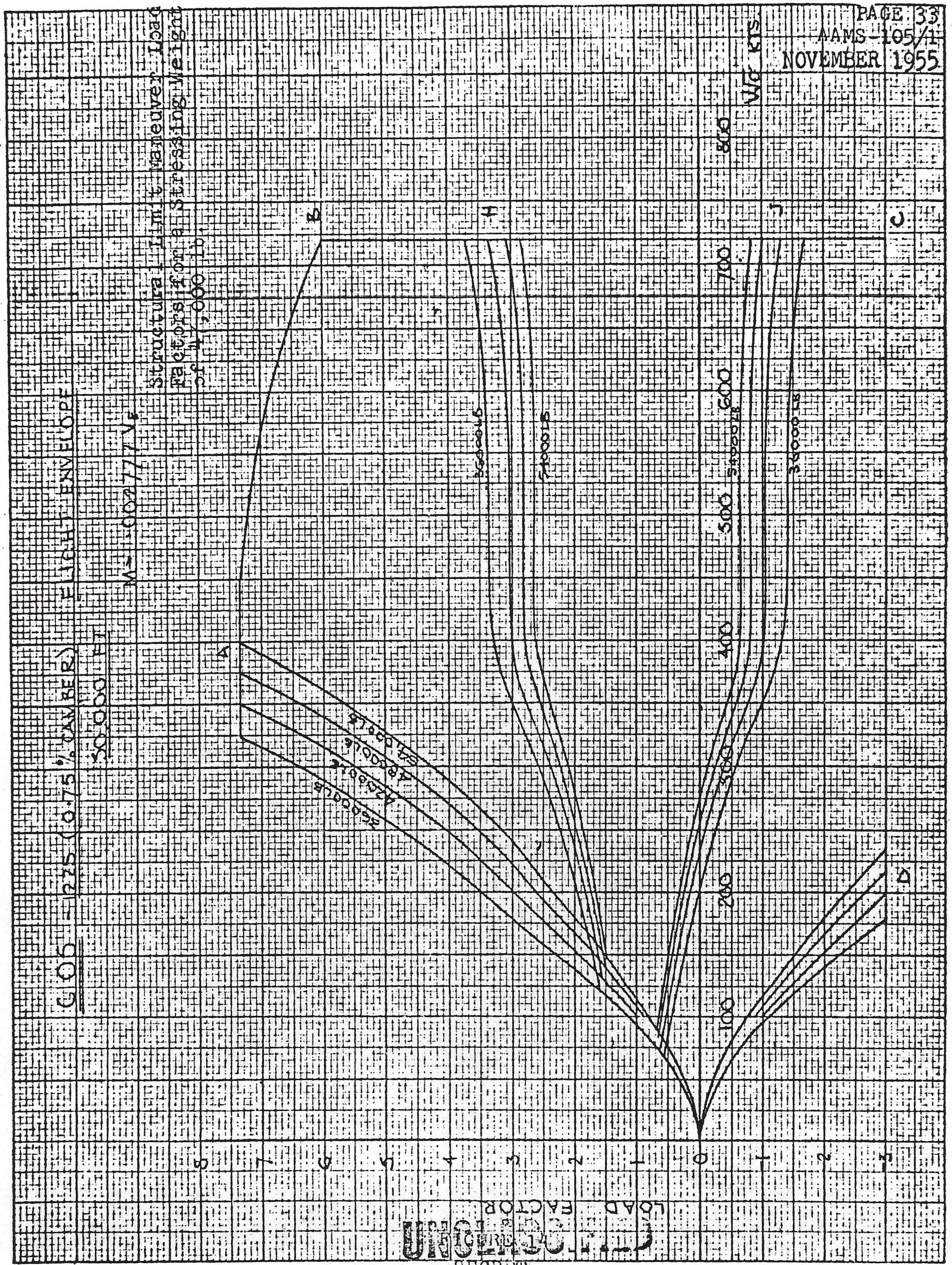


FIGURE 13

UNCLASSIFIED

FORM 1746
KEUFFEL & ESSER CO. 10 X 10 to the ½ inch, 5th lines accented. MADE IN U.S.A.



359-12 KEUFFEL & ESSER CO.
10 X 10 to the 1/2 inch, 5th lines accented.
MADE IN U.S.A.

SECRET

FORM 1746

359-12 KEUFFEL & ESSER CO.
10 X 10 to the ½ inch, 5th lines accented.
MADE IN U.S.A.

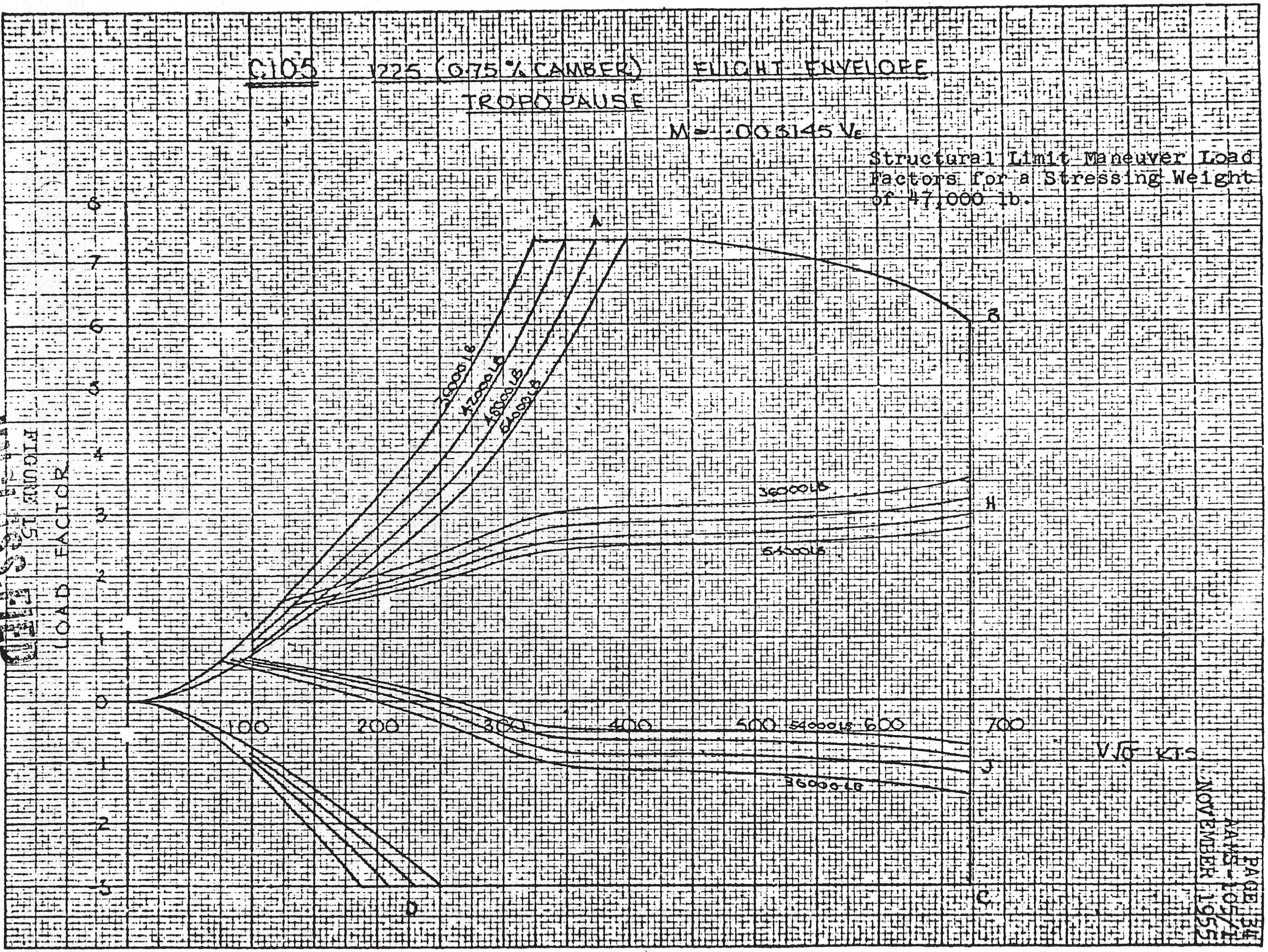


FIGURE 15

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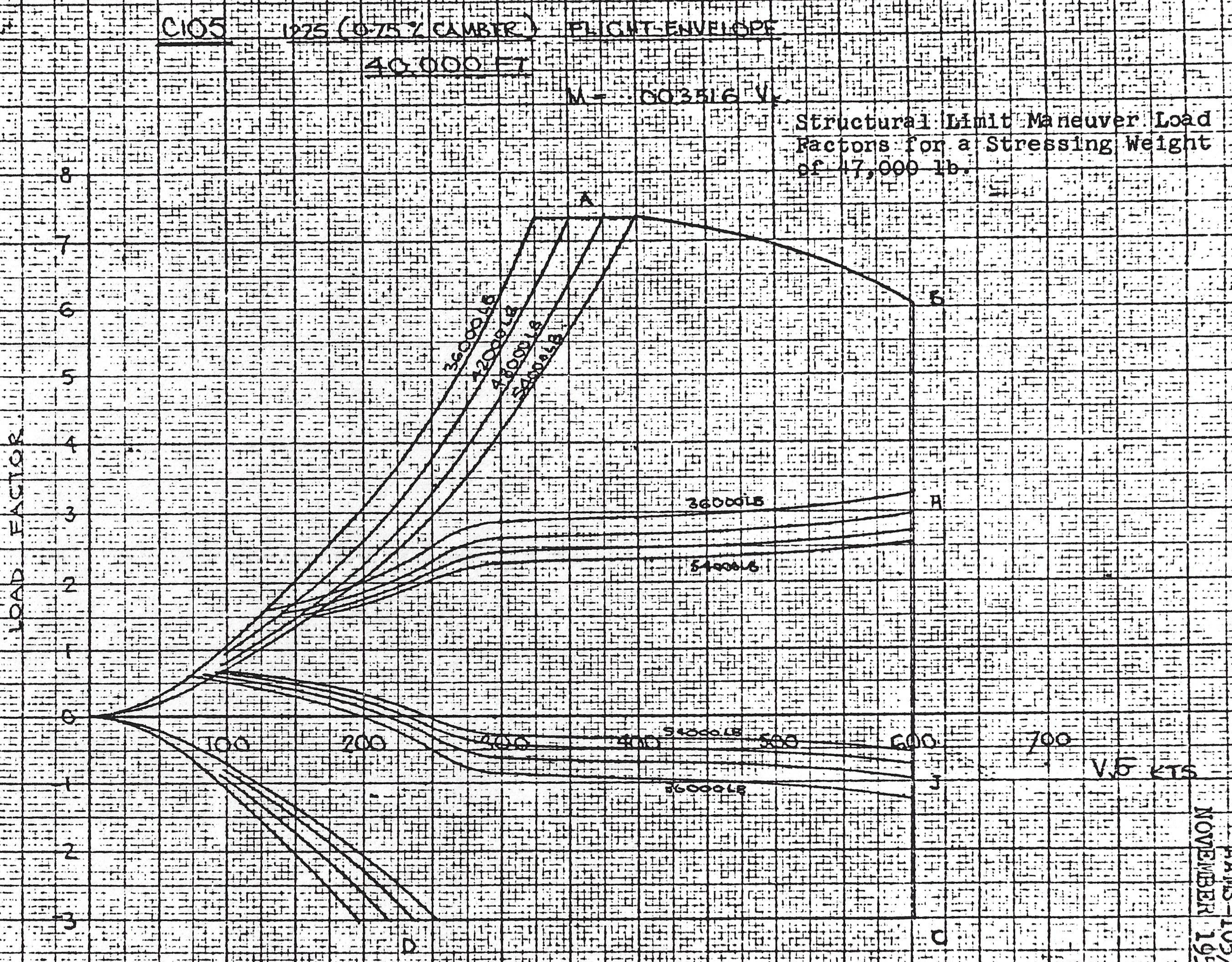

FIGURE 16

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359-12 KEUFFEL & ESSER CO.
10 X 10 to the ½ inch. 5th lines accented.
MADE IN U.S.A.
FORM 1746

C105 T25 (0.75% CAMBER) FLIGHT ENVELOPE
50,000 FT

M = .00446 V_e

Structural Limit Maneuver Load Factors for a Stressing Weight of 47,000 lb.

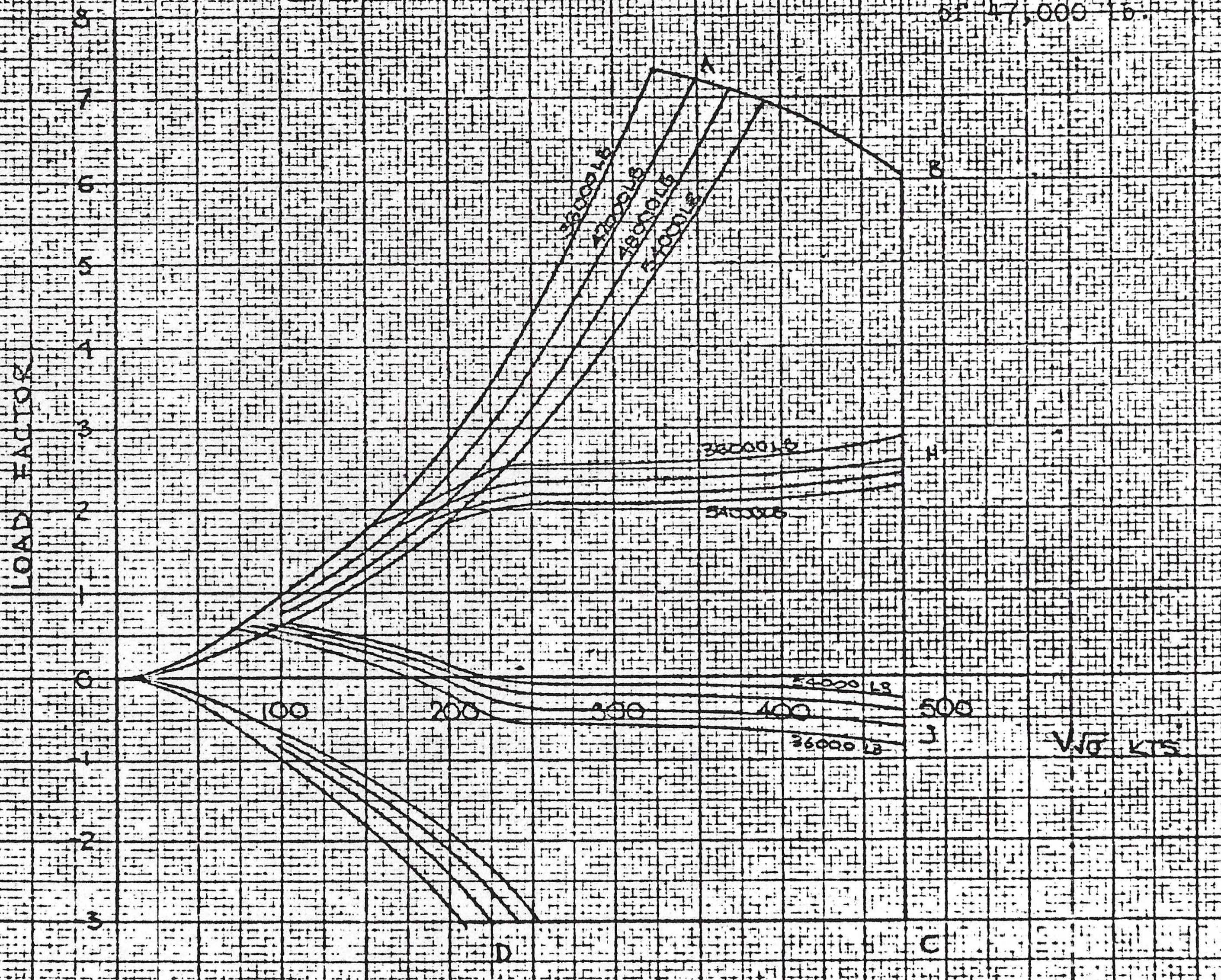

FIGURE 17

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389-L KEUFFEL & ESSER CO.
10 X 10 to the 1/2 inch, 5th lines accented.
MADE IN U.S.A.

FORM 1746

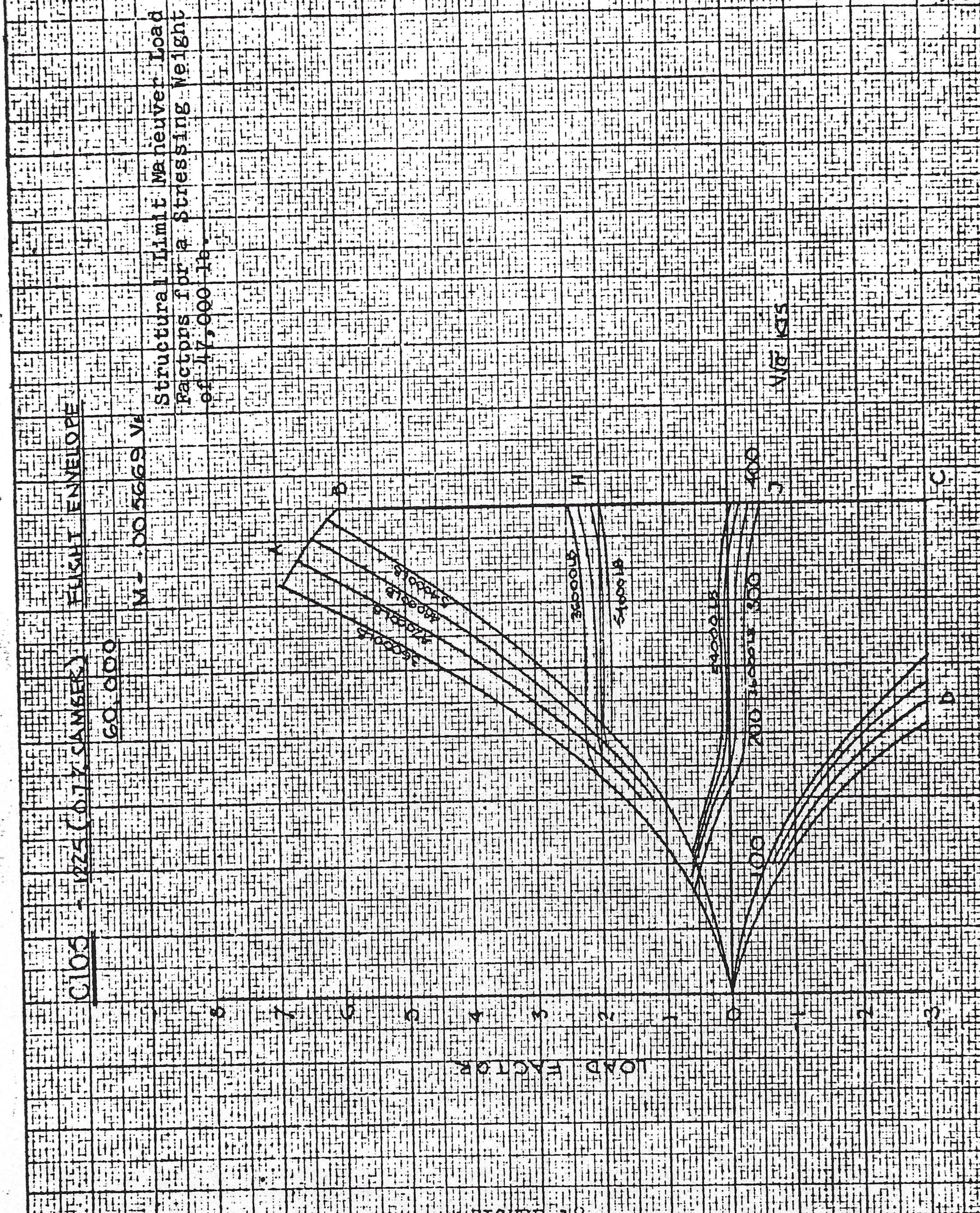

FIGURE 18

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359-12 KEUFFEL & ESSER CO.
10 X 10 to the ½ inch, 5th lines accented.
MADE IN U.S.A.
FORM 1746

C105 - 1225 (0.75% CAMBER) FLIGHT ENVELOPE
70,000 FT
M = .007198 V_e

Structural Limit Maneuver Load Factors for A Stressing Weight of 47,000 lb.

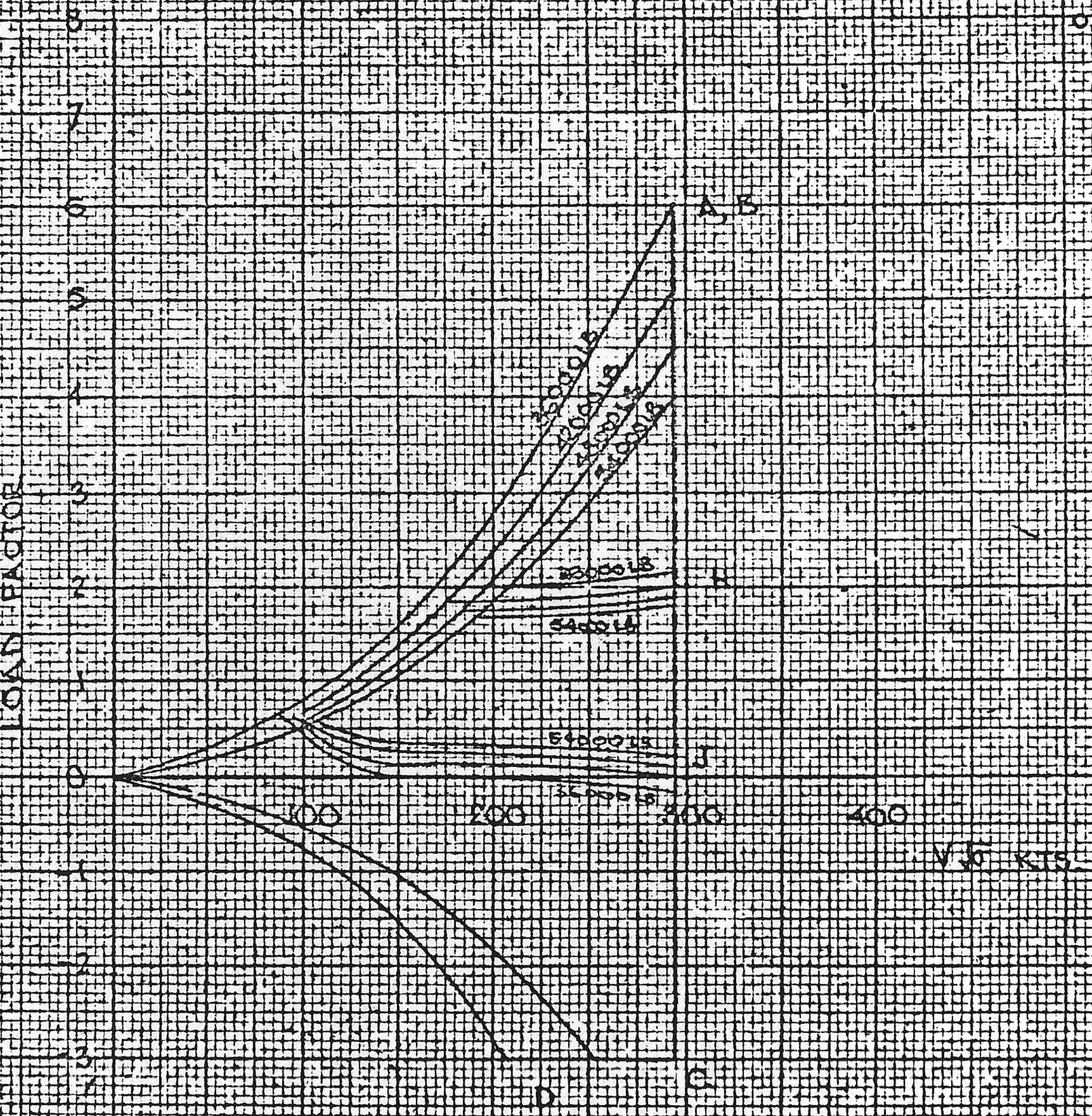

FIGURE 19

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FORM 1746
359-12 KEUFFEL & ESSER CO.
10 X 10 to the 1/2 inch, 5th lines accented.
MADE IN U. S. A.

3.5 Wing Group

3.5.1 Description and Components

The wing shall be a delta type of full cantilever, all metal, stressed skin construction, comprising six main sections on each side of the aircraft center line:

Inner Wing
Outer Wing
Leading Edge
Trailing Edge
Elevator
Aileron

Access doors shall be provided for inspection, maintenance and repair, of aircraft services.

3.5.2 Construction

3.5.2.1 Inner Wing

The inner wing shall consist of two sections.

An inner wing torque box section shall form the main structural support of the wing assembly and shall consist of four spars, machined skins with spanwise integral stiffeners, and chordwise ribs. The three bays formed by the spars shall constitute four integral fuel tanks. The wing torque box section root shall be attached to the corresponding root of the opposite wing and to a wing center box at a manufacturing joint.

The forward section of the inner wing shall comprise a front spar, a transverse auxiliary spar, chordwise ribs forward of the auxiliary spar, and ribs aft of the auxiliary spar running parallel and normal to the axis of the retracted undercarriage leg. The forward section shall incorporate two integral fuel tanks and shall house the main landing gear assembly. This section shall be attached at manufacturing joints to the main spar, the opposite wing root, and the center fuselage which is indented into the delta configuration.

3.5.2.2 Outer Wing

The outer wing shall comprise five spars, stringers, chordwise inboard and tip ribs, and ribs running normal to a wing tangency line. It shall be attached to the inner wing by bolts at a transport joint with loads transmitted to the inner wing through the skin attachment, at a front spar joint, a rear spar joint, and three intermediate vertical shear joints. The outer

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3.5.2.2 Outer Wing (Cont'd)

wing shall house the aileron control unit forward of the rear spar.

3.5.2.3 Leading Edge

The leading edge of the wing shall comprise three sections with structural ribs running normal to the front spar line. As a structural assembly the leading edge shall supplement the structure of the inner and outer wing panels. At the outer chord of the inner wing panel the leading edge shall be slotted 5% of the chord and 6½" spanwise. The leading edge from outboard of the slot to the wing tip shall be extended forward along the chord line 10% of the chord. The leading edge assembly shall be attached to the inner and outer wing panels at manufacturing joints.

3.5.2.4 Trailing Edge

The trailing edge shall be divided into three sections for the purpose of manufacturing.

An inner trailing edge shall extend outboard from the wing center box to the inboard chordline of the elevator and shall be a manufacturing detail build-up section of six spanwise beams and machined skins bolted at manufacturing joints to the rear spar, the wing center box, and the center trailing edge. The inner trailing edge shall house the elevator control unit.

A center trailing edge shall extend the full span of the elevator and shall comprise an elevator hinge spar and chordwise ribs, six of which shall support the elevator control linkage. The center trailing edge shall be bolted to the inner and outer wing panels and to the outer trailing edge.

An outer trailing edge shall extend the full span of the aileron and shall comprise an aileron hinge spar, and internal ribs at approximately 74° to the spar, seven of which shall support the aileron control linkage. The outer trailing edge shall be attached to the rear spar of the outer wing panel at a manufacturing joint.

3.5.3 Ailerons

The ailerons shall be of stressed skin construction, utilizing aluminum alloy skins, a hinge spar, and ribs running normal to the spar line. Seven main ribs shall connect to

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3.5.3 Ailerons (Cont'd)

the aileron linkage in the outer trailing edge of the outer wing. The aileron shall be hinged to the wing trailing edge by a piano hinge along the topside for the full (10) span of the movable surface and shall be fully shrouded along the underside.

The angular motion of the aileron shall be 19° up and 19° down from the hinge center line. The centroid of the aileron area shall be 19.036 feet from the aircraft center line. The design of the flying control system and its power operation shall obviate the necessity for aerodynamic and static balance.

3.5.4 Aileron Tabs

Not applicable.

3.5.5 Lift and Drag Increasing Devices

Not applicable.

3.5.6 Speed Brakes

Speed brakes installed on fuselage (Reference paragraph 3.7.6).

3.5.7 Elevator

The elevator shall be of stressed skin construction, utilizing aluminum alloy skin, a hinge spar, and ribs running normal to the spar line. Six main ribs shall connect to the elevator linkage in the wing center trailing edge. The elevator shall be hinged to the wing trailing edge by a piano hinge along the top side for the full span of the (10) movable surface and shall be fully shrouded along the underside. The angular motion of the elevator shall be 30° up and 20° down from the hinge center line. The design of the flying control system and its power operation shall obviate the necessity for aerodynamic and static balance.

3.5.8 Elevator Tab

Not applicable.

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3.6 Tail Group

3.6.1 Description and Components

The tail group shall comprise a fin and rudder. Due to the delta wing configuration there shall be no horizontal stabilizer and the elevator shall be included as a section of the wing group.

3.6.2 Stabilizer

Not applicable.

3.6.3 Elevator

Elevators installed on wing surface (Reference paragraph 3.5.7).

3.6.4 Elevator Tab

Not applicable.

3.6.5 Fin

The fin shall be of aluminum alloy stressed skin construction and shall consist of two sections, a main structural assembly and an interchangeable rudder control linkage box aft of the rear spar. The main structure shall comprise five spars, spanwise compression ribs, and ribs running normal to the rudder hinge line. Loads shall be transmitted to the wing center box where the fin is attached at a manufacturing joint. A detachable fin tip of fibrous material shall be installed to house radio antennas. The rudder control linkage shall be housed in the control linkage box, with the control unit installed in the fin forward of the main structural assembly rear spar. Access doors shall be provided for inspection, repair, and maintenance of the rudder control unit and aircraft services.

3.6.6 Rudder

The rudder shall be of stressed skin construction and shall comprise a hinge spar, an intermediate spar, and ribs running normal to the hinge spar line. The rudder shall be supported from the fin by seven hinge ribs, five of which connect to the rudder control linkage. The design of the flying control system and its power operation shall obviate the necessity for aerodynamic and static balance.

3.6.7 Rudder Tab

Not applicable.

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3.7 Fuselage

3.7.1 Description

The fuselage shall be arranged below and extend forward of the wing and shall be designed to house two turbo-jet engines, armament, a crew of two, and the major proportion of the aircraft service components. The fuselage shall be of rounded cross-section from the nose probe to the cockpit and engine air intakes where it shall evolve into a slab-sided horizontally oblong cross-section. A pilot's V type windshield and a semicircular cockpit canopy shall protrude above the fuselage lines and shall fair into a dorsal fairing extending aft over the fuselage and wing upper surface to the vertical tail and rear fuselage.

3.7.2 Construction

The fuselage shall comprise a radar nose, nose fuselage, center fuselage, duct bay, engine bay, and rear fuselage, joined at manufacturing joints. The fuselage shall be of stressed skin construction utilizing aluminum alloy and magnesium alloy skins, with vertical bulkheads, frames, and longitudinal stringers in the radar nose, nose fuselage and center fuselage sections, and close pitched frames and longerons in the duct bay, engine bay and rear fuselage. Steel, magnesium, inconel 'X', and titanium shall be utilized in both primary and secondary structure, as required. Loads shall be transmitted between the fuselage and inner wing, by internal center struts between the fuselage main frames and inner wing spars, and through fuselage skin - underwing skin joints.

3.7.3 Crew Stations

The crew stations shall provide for a pilot and radar operator seated in tandem cockpits in the nose fuselage. The cockpits shall be pressurized in accordance with paragraph 3.22.1 and suitable insulation shall be installed to minimize heat transfer from the adjacent skin. A bulkhead shall render the two crew stations incommunicable. The cockpits
shall be enclosed by a pilot's windshield and a canopy in- (77)
corporating a split clamshell type hatch for each cockpit. (78)

The pilot's windshield and canopy windows shall comprise (12)
optically flat panels of electrically heated tempered glass, and the radar operator's canopy windows shall comprise curved panels of tempered glass. Each canopy hatch shall be normally actuated electrically and locked by a manually operated latch. To provide for single crew member operation, access to the latch handle in the radar operator's cockpit shall be gained through a door in the cockpit

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3.7.3 Crew Stations (Cont'd)

separating bulkhead. A canopy actuating switch shall be installed in each cockpit for the respective canopy hatch. A switch for each canopy hatch shall be installed in a location accessable from the aircraft exterior. Emergency open- (11)
ing of the canopy hatches shall be by means of gas pressure (79)
from an explosive cartridge. Internal and external hand operated mechanical control for emergency opening shall be provided.

The line of vision from the pilot's cockpit shall be directly forward to a line $12\frac{1}{2}$ degrees below the horizon, aft to (80)
120 degrees on both sides from a line directly forward, and (13)
with reasonable pilot movement vertical vision on each side to a line 30 degrees below the horizon. The pilot's cockpit shall provide 25 inches clearance across the normal shoulder location and 36 inches clearance across the normal elbow location.

3.7.3.1 Pilot's Cockpit

Manual and automatic flying controls, instruments, warning indicators, and the following items of functional equipment shall be installed in the pilot's cockpit.

Switches

Canopy Opening (Normal)
Air Conditioning
Pressurization Dump Valve
Air Conditioning Defog
Rain Repellant
High Altitude Light
Fire Warning and Extinguishing
Second Shot Fire Extinguishing
Taxi and Landing Lights
Navigation Lights
Engine Starting
Master Electrical
Alternator
External Tank Jettison
Master Warning Test
Master Warning Reset
Artificial Horizon Erection
Engine Relight
Speed Brake
Press-to-transmit
Damping System:
  Selector
  Emergency Selector
  Pitch Axis Re-engage
  Roll Axis Re-engage

Controls

Air Conditioning
Cockpit Lighting Intensity
Landing Gear
Power Control (throttle)
Anti-G Valve
Emergency Oxygen Starting
Brake Parachute
Parking Brake
Rudder Pedal Adjustment
J4 Compass
Canopy Opening (Emergency)
Radio Compass AN/ARN-6
Intercom AN/AIC-10
I.F.F. AN/APX-6
Press-to-test Oxygen Pressure
UHF Radio AN/ARC-34
Seat Raising
Manual Harness Release

3.7.3.1 Pilot's Cockpit (Cont'd)

Switches (cont'd)

UHF Antenna Selector
RMI Function Selector
L-Band Antenna Transfer
Nose Wheel Steering
Elevator, Aileron and Rudder Trim
Missiles Firing Trigger
Armament On-Off
Armament Mode Selector
Master Emergency Jettison

3.7.3.2 Radar Operator Cockpit

To be added.

3.7.4 Cargo Compartments

Not applicable.

3.7.5 Equipment Compartments

Equipment compartments and bays as listed in the following sub-paragraphs shall be provided for aircraft propulsion, armament, electronics and services equipment and components. Compartments and bays housing equipment and/or components requiring a maintained temperature and/or pressure shall be suitably insulated, sealed, and vented as required. (Reference Section 3.22 Air Conditioning.)

3.7.5.1 Nose Radar Compartments

The radar nose shall comprise two compartments. The forward compartment shall be constructed of plastic bonded fibreglass and shall be detachable. The aft compartment shall comprise the fuselage structural area forward of the pilot's cockpit. Access to the aft compartment shall be provided by two doors, one on either side of the fuselage. Conditioned air shall be supplied to the compartments and equipment housed in the compartments.

3.7.5.2 Nose Wheel Well Compartment

Space in the nose wheel well shall be utilized for the installation of electrical and associated equipment. Conditioned air shall be supplied to maintain the temperature of the forward end of the well. Access shall be gained through the open wheel well door.

3.7.5.3 Battery Compartment

A battery compartment shall be located in the nose wheel well with access provided to the compartment through the open wheel well door. The compartment shall be provided with conditioned air and shall be suitably treated against corrosion.

3.7.5.4 Forward Fuselage Electronics Bay

The area immediately aft of the nose wheel well shall comprise a forward fuselage electronics bay. Access shall be provided through a panel on the underside of the fuselage.

3.7.5.5 Forward Fuselage Electronics Compartment

The forward fuselage electronics compartment shall be located between the armament pack bay and the forward fuselage electronics bay. Conditioned air shall be pro-
vided to the compartment and installed equipment, with (91)
access gained through a door on the underside of the fuselage.

3.7.5.6 Air Conditioning Equipment Bay

An air conditioning equipment bay shall be located immediately aft of the radar operator's cockpit. The bay shall be supplied with conditioned air, and access to the bay shall be provided by removal of a section of the dorsal fairing, an air outlet duct, and a shear panel.

3.7.5.7 Oxygen Bay

The oxygen bay shall comprise the dorsal fairing area immediately aft of the radar operator's cockpit. Conditioned air shall be provided to the bay and installed equipment. Access shall be gained through the same opening as for the air conditioning equipment bay.

3.7.5.8 Armament Pack Bay

The armament pack bay shall comprise a recess in the underside of the fuselage designed to permit the installation of interchangeable missile packs. Conditioned air shall be provided to maintain the internal air temperature of an installed missile pack.

3.7.5.9 Service Bays

The fuselage areas between the left and right hand air intakes and engines shall comprise three service bays. The

3.7.5.9 Service Bays (Cont'd)

forward bay shall primarily house electrical equipment, the center bay shall primarily house fuel system and hydraulic system components, and the aft bay shall house hydraulic system components and aircraft accessories gear boxes. Access doors and panels for the three bays shall be installed on the underside of the fuselage.

3.7.5.10 Dorsal Electronics Compartment

The dorsal electronics compartment shall be located in the dorsal fairing at approximately a mid-wing position. The compartment and equipment shall be supplied with conditioned air and access to the compartment provided by the removal of a section of the dorsal fairing.

3.7.6 Speed Brakes

Two symmetrical speed brakes, of box panel construction of aluminum alloy and magnesium alloy, shall be installed on the underside of the duct bay section of the fuselage. The brakes shall be installed outboard of the frontal area of the nose landing gear fairing. The brakes shall be actuated by a continuous positioning lever powered by the utility hydraulics system, and shall retract into a sealed well recessed into the undrside of the fuselage.

3.7.7 Fuselage Power Plant Installation

Reference Section 3.10.

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3.8 Landing Gear

3.8.1 Description

The landing gear shall be an electrically controlled, hydraulically actuated tricycle type. The main landing gear shall retract inward and forward into the inner wing on a line at 50° to the aircraft center line. The nose gear shall be steerable and shall retract forward into the nose fuselage. The hydraulic actuating system shall be designed to retract the gear, including door operation, in 5 seconds at -20°F and 30 seconds at -65°F. When completely retracted the alighting gear shall be enclosed within the faired lines of the wing and front fuselage. (92) (14)

A mechanically releasable and jettisonable drag parachute shall be installed within the faired lines of the rear fuselage.

The landing gear shall be designed in accordance with the requirements of R.C.A.F. Specification AIR 7-4, and ARDCM 80-1, except as stated in Appendix II and as additionally stated herein.

3.8.2 Main Landing Gear

3.8.2.1 Description

Each main landing gear shall comprise a two-wheel tandem bogie pivoted to a shock absorber installed in the lower end of a main strut. A mechanical linkage to draw the shock absorber into the main strut and rotate the bogie, and a telescopic spring strut to position the unloaded bogie in a front wheel down attitude, shall be installed to permit stowage of the retracted gear within a wing wheel well.

The upper end of each main landing gear main strut shall be pivoted between the front and main spars at the outer end of the inner wing. The strut shall be braced by a drag strut in the plane of the pivot line and by a telescopic downlock strut in the plane of retraction.

3.8.2.2 Wheels, Brakes and Brake Controls

The main wheels shall be demountable and fitted with anti-friction bearings and hydraulically operated multiple disc brakes. A skid detector shall be installed in the outer end of each wheel axle to govern a brake anti-skid hydraulic control valve. The hydraulic pressure available for normal brake operation shall be a maximum of 2130 psi, and for emergency operation a maximum of 1500 psi (Reference paragraph 3.14.1.1.4).

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3.8.2.5.4 Controls

A pilot operated landing gear retraction and extension selector shall be installed to control the actuation of both main gears and the nose gear. A lock shall be incorporated in the selector to prevent UP selection until micro-switches have been closed by full extension of the shock absorbers. The actuation of the main gears and the main gear locks shall be hydraulically sequenced in relation to the actuation of the main gear doors and door locks (Reference Section 3.14). It shall be possible to reverse the motion of the landing gear, during the retraction or extension cycle, by reselection. A warning light shall be installed in the knob of the selector to indicate when the landing gear is in motion and not locked up or down, or when both engine throttle levers are retarded to 1/3 full throttle or less with the landing gear retracted. A three-way landing gear position indicator for each main gear shall be installed near the selector.

3.8.2.5.5 Emergency Retraction

An override push switch for the UP selection prevention lock shall be installed on the landing gear selector. It shall be necessary to continuously depress the override button during operation of the selector lever.

3.8.2.5.6 Emergency Extension

Operation of a push button shall release a gate and permit the landing gear selector lever to be depressed below the normal DOWN position. This action shall release a pneumatic charge into the landing gear sub-system to release all locks, actuate the doors and permit the main landing gear to extend and automatically lock in the fully extended position.

3.8.2.6 Doors and Fairings

Each retracted main gear shall be faired in conformity with the aircraft skin line by a main door, a fairing attached to the main strut and a door for the pivoted end of the main strut. The main door shall be hinged parallel to the aircraft center line and hydraulically actuated. The pivot door shall be hinged parallel to the main gear pivot line and actuated by a linkage to the main strut. The main door shall be locked in, and unlocked from, the down position by a lock within the door actuator, and locked in the up position by mechanically engaged locks which shall be released by hydraulic actuators.

3.8.2.6 Doors and Fairings (Cont'd)

The main door and door lock actuation shall be hydraulically sequenced with the main gear and main gear locks (Reference Section 3.14).

3.8.2.7 Inspection and Maintenance

Access doors shall be installed on the underside of each inner wing to provide access to the main landing gear retraction jack.

3.8.3 Auxiliary Landing Gear (Nose Gear)

3.8.3.1 Description

The steerable nose landing gear shall consist of a Y shaped main strut incorporating a liquid spring shock absorber which shall act in conjunction with a suspension lever carrying a live axle and co-rotating wheels. The two upper arms of the main strut shall be pivoted to the fuselage bulkhead aft of the navigator's cockpit. The strut shall be braced by a folding, lockable drag strut.

A hydraulic self centering actuator shall be installed on the gear main strut and linked to the nose wheel suspension lever to provide for castoring with self centering of the wheels, or for steering when steering is selected. The nose wheels shall castor, or be steerable, up to 55° (93)
on either side. Shimmy damping restrictor valves shall be installed in the steering actuator hydraulic circuit to operate when steering is not selected.

3.8.3.2 Wheels

The wheels shall be demountable and retained on the live axle by splines and lockable axle nuts.

3.8.3.3 Tires

A tubeless tire (U.S.A.F. 18 x 5.5 Type VII E.H.P.) rated at 5,050 lb. static load when inflated to 170 psi, shall be installed on each nose wheel.

3.8.3.4 Shock Absorbers

The shock absorbers shall be of liquid spring design with provision for topping up in situ. The shock absorber fluid shall be a blend of silicone oil and hydraulic oil.

3.8.3.5 Retracting, Extending and Locking Systems

3.8.3.5.1 Retraction

The nose landing gear shall be retracted forward and up

3.8.3.5.1 Retraction (Cont'd)

by a hydraulic actuator until an uplock is engaged.

3.8.3.5.2 Extension

The nose landing gear shall be extended aft by gravity and drag until a downlock is engaged.

3.8.3.5.3 Locking

The uplock and downlock for the nose gear shall engage mechanically and be released by hydraulic actuators. The gear downlock shall be part of the folding drag strut. The actuation of the nose landing gear shall be hydraulically sequenced with the nose landing gear door (Reference Section 3.14).

3.8.3.5.4 Controls

The nose landing gear shall be controlled in conjunction with the main gear (Reference paragraph 3.8.2.5.4). A three-way landing gear position indicator for the nose gear shall be installed in conjunction with the main gear position indicators.

3.8.3.5.5 Emergency Retraction

Emergency retraction of the nose gear shall be effected in conjunction with that of the main gear (Reference paragraph 3.8.2.5.5).

3.8.3.5.6 Emergency Extension

The emergency extension of the nose landing gear shall be effected by the means employed for the main landing gear (Reference paragraph 3.8.2.5.6).

3.8.3.6 Steering Control

Steering selection shall be by continuous pressure on a push button on the pilot's control column. A microswitch shall be installed on the nose gear suspension lever, to prevent selection of steering unless the nose wheels are in a loaded attitude. The rudder pedals shall be mechanically linked to the steering control valve through a hydraulically operated clutch integral with the valve and synchronization of the rudder pedals with the nose wheel deflection shall be necessary to permit the hydraulic clutch to engage. A follow-up type steering control valve shall be installed to permit control of the steering actuator when steering has been selected.

3.8.3.6 Steering Control (Cont'd)

The steering actuator shall be designed to be self-centering by the action of internal springs and the hydraulic system return pressure.

3.8.3.7 Doors and Fairings

The retracted nose landing gear shall be enclosed within faired lines of the front fuselage by a door and a fairing. The nose gear door shall be hinged to the right hand edge of the nose wheel well and shall be hydraulically actuated; the nose gear fairing shall be hinged to the aft edge of the nose wheel well and actuated by the nose gear. The door shall be locked in, and unlocked from, the down position by a lock within the door actuator, and locked in the up position by mechanically engaged locks which shall be released by a hydraulic actuator.

3.8.3.8 Inspection and Maintenance

Access to the nose landing gear for inspection and maintenance shall be possible when the gear is extended.

3.8.4 Drag Parachute

3.8.4.1 Description

A "fist" type ribbon drag parachute pack, including a spring opened pilot parachute, shall be stowed in a compartment in the top of the tail cone between the two engine jet pipe fairings. The pack shall be retained by two doors which shall maintain the skin line when closed and receed below the skin line when opened.

3.8.4.2 Release Gear

The parachute shall be streamed or jettisoned as required (76
by a mechanical release controlled by the pilot. The parachute shall be attached to the aircraft structure by a shear pin to permit breakaway of the attachment at a predetermined load.

3.9 Surface Control System

The surface control system shall be a fully powered, hydraulically actuated, irreversible system, and shall be designed (16) to meet the requirements of ARDCM 80-1 except as specified herein and as additionally stated in the deviations (Appendix II).

The primary flight controls shall be powered by two independent hydraulic power circuits with normal Manual Mode control (15) effected through command electro-hydraulic servos, and Emergency Mode control effected through mechanical linkages installed to control the surface actuator valves. Artificial pilot feel systems shall be provided for both the Manual and Emergency Modes of control. Space provision shall be made for an automatic flight control system. An artificial damping system shall be installed to provide flight damping and stabilization about all three axis, turn co-ordination, and spiral stability in the Manual Mode of control; and yaw axis damping and stabilization only in the Emergency Mode of control.

Speed brakes powered by the Utility Hydraulic System shall be installed for use within the subsonic speed range.

3.9.1 Primary Flight Control System

The primary flight control surfaces shall comprise ailerons, elevators, and a rudder with surface displacement controlled by conventional movement of a pilot's control column and rudder pedals.

A selector switch shall be installed in the pilot's cockpit for selection of manual or emergency mode of control. The emergency mode of control shall be in an operable condition at all times when the surface control hydraulic system is charged, and shall automatically become the effective mode of control in the event of failure of the manual mode.

3.9.1.1 Elevators

The control column shall be linked to elevator actuator control valves by bell cranks, quadrants, cables, and push rods with stick force transducers installed in the (18) linkage. In the manual mode of control the transducers (19) shall transmit the pilot's input stick forces as electrical signals through an amplifier to the elevator parallel (command) servo. The parallel servo shall be connected by a mechanical linkage to the control valves of the elevator hydraulic actuators. In the emergency mode of control the parallel servo shall be by-passed, with

3.9.1.1 Elevators (Cont'd)

the control column movement transmitted through the mechanical linkage direct to the actuator control valves. (17)

3.9.1.2 Ailerons

The control column shall be linked to aileron actuator control valves by bell cranks, quadrants, cables, and push rods with stick force transducers installed in the linkage. In the manual mode of control the transducers shall transmit the pilot's input stick forces as electrical signals through an amplifier to the aileron parallel (command) servo. Stick force command signals shall be limited such that roll rates in excess of 180°/second shall not be transmitted. The parallel servo shall be connected by a mechanical linkage to the control valves of the aileron hydraulic actuators. In the emergency mode of control the parallel servo shall be by-passed, with the control column movement transmitted through the mechanical linkage direct to the actuator control valves. (18)

3.9.1.3 Rudder

Co-ordinated rudder control in the manual mode of control shall be provided by the damping system (Reference paragraph 3.9.4.1). A mechanical linkage comprising bell cranks, quadrants, cables, and push rods shall connect the rudder pedals to the rudder hydraulic actuator control valves for the emergency mode of control, and to permit the pilot to override the damping system rudder co-ordination functions during maneuvers requiring unco-ordinated control. (18) (19)

3.9.1.4 Artificial Feel

3.9.1.4.1 Manual Mode Artificial Feel

Manual mode artificial feel for the elevators and ailerons shall be provided by electronic control of the parallel servos which shall provide feel reaction against control column movement.

3.9.1.4.2 Emergency Mode Artificial Feel

The emergency mode artificial feel units shall comprise positional spring units installed between the control linkage and the aircraft structure with electrical trimming devices incorporated between the feel units and the structure. A bob-weight installed on an elevator control linkage torque tube shall supplement the

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3.9.1.4.2 Emergency Mode Artificial Feel (Cont'd)

elevator feel unit providing additional feel in proportion to the "g" load in the pitching axis. The rudder feel unit shall incorporate an adjustment linkage which shall automatically govern the rudder surface deflection at a given rudder pedal load as a function of the compressible dynamic pressure.

3.9.1.5 Cable Tensioning Devices

Cable tension regulators shall be installed in each control axis cable system at the forward fuselage end of the cable runs with additional aileron control cable tension regulators installed in the aft fuselage inner wing area.

3.9.1.6 Vulnerability and Duplication

Vulnerability of the flying control system to anticipated
types of aircraft damage shall be kept to the lowest de- (20)
gree possible by utilization of inherent protection afforded by aircraft structural components. The flying control hydraulic system shall be a duplicate system up to control surface actuators.

3.9.2 Secondary Flight Control System

3.9.2.1 Lift and Drag Increasing Devices

Not applicable.

3.9.2.2 Speed Brakes

Two rectangular speed brake panels shall be installed on the underside of the fuselage, the panels being lowered by hydraulic actuators to present a braking area to the slipstream. The hydraulic actuators shall be controlled by a manually operated switch incorporated in the right hand engine throttle lever in the pilot's cockpit. The switch shall be of the 3 position type with EXTEND, HOLD and RETRACT positions and shall control the jacks through a hydraulic selector valve. The selector valve shall limit speed brake extension to speeds below Mach 1, and
shall control the degree of deflection in relation to (75)
speed brake air loads (Reference paragraph 3.14.1.1.5).

3.9.3 Trim Control Systems

Trim for the manual mode of control shall be automatically and continuously supplied by the damping system. Trim adjustment for the emergency mode of control shall be provided by trim devices, integral with the artificial feel units

3.9.3 Trim Control Systems (Cont'd)

and linkages, which unload the artificial feel springs. A trim selector button shall be installed on the pilot's control column grip.

3.9.4 Automatic Flight Control Systems

3.9.4.1 Automatic Pilot System

Not applicable.

3.9.4.2 Artificial Damping System

The damping system shall operate through the flying control hydraulic system with normal damping operable in conjunction with the Manual Mode of control and emergency damping operable in conjunction with the Emergency Mode of control.

*Electrical power for operation of the damping system shall be provided by the aircraft power supply. A system failure in either the primary flight controls manual mode of control or the damping system normal operation shall automatically transfer the damping system to emergency operation. A STANDBY-OPERATE-OFF switch for control of the entire damping system, a switch for selection of the damping system to normal or emergency operation, and roll and pitch axes cut-out re-engagement switches shall be installed in the pilot's cockpit. A warning light shall be installed in the pilot's cockpit to indicate when the damping system is in emergency operation.

3.9.4.2.1 Normal Damping

Normal operation shall provide automatic damping of short period oscillations about the three axes, turn co-ordination, control of spiral stability so that the amplitude shall not double in less than 20 seconds, and sideslip minimumization in any maneuvers up to 6g in pullout, 4g in turns and negatively to -2g.

A rudder pedal force switch shall provide means for disconnecting the damping system automatic co-ordination control of rudder and ailerons, and allow the pilot to produce intentional sideslip. A "g" cut-out switch shall be installed to automatically disengage the pitch

*If necessary a separate power supply will be installed for the damping system.

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3.9.4.2.1 Normal Damping (Cont'd)

axis of the damping system and engage the corresponding trim feel system to prevent the aircraft from exceeding the structural integrity "g" limits about the pitch axis. A roll rate cut-out switch shall be installed to automatically disengage the damping system roll axis channel and engage the aileron trim feel system to prevent the aircraft from exceeding the structural integrity rate of roll limit. It shall be possible to re-engage the roll and pitch axes by means of the re-engagement switches in the pilot's cockpit.

Air data from air sensors, pitot and static systems, etc. (Reference paragraph 3.13.3.4), shall be combined with data from the damping system flight sensing instruments (gyros, accelerometers) for scheduling of aileron, elevator and rudder position control signals. These scheduled signals shall be continuously transmitted by magnetic amplifiers to the appropriate differential servos located on the actuation jacks of the control surfaces. In response to the applied signals the differential servos shall operate the hydraulic valves which control the surface actuation jacks resulting in adjustment of the control surfaces according to the sensed aircraft stability requirements. Mechanical and electrical feedbacks in the system provide closed loop stabilization.

3.9.4.2.2 Emergency Damping

The emergency damping shall consist of a duplication of the normal yaw axis damping channel components, to provide a limited structural integrity protection in the event of normal damping system failure.

3.9.5 Inspection, Maintenance and Repair

Means of access to the control systems shall be through panels in the underside of the fuselage and panels provided, where necessary, in the wings and front fuselage. Repair shall be possible by replacement of damaged components.

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3.10 Engine Section

3.10.1 Description and Components

The engine bay shall form an integral part of the fuselage structure and shall house two power plants, together with various accessories.

3.10.2 Construction

Construction shall be in accordance with paragraph 3.7.2.

3.10.3 Engine Mounts

The engine mounts shall provide two planes of attachment for securing the engine to the aircraft structure.

The forward engine mounting shall consist of two ball-and-socket joints. A spanwise beam, suspended from the wing structure, shall provide mating attachment points for the inboard mount of each engine. The male half of the inboard mount shall be attached to the pad provided on the engine forward housing. The inboard mounts shall be designed to absorb longitudinal, vertical and side loads, and to permit the engine to pivot about the mounting point. Separate engine supports suspended from the wing structure shall provide attachment points for each outboard engine mount. These supports shall contain the ball and socket joint, which is secured to a conical spigot attached to the pad provided on the engine forward housing. The outboard mounts shall be designed to absorb vertical loads only.

The hangar-type rear engine mount shall have three attachment points, one on the engine vertical center line, and two (one inboard and one outboard) on the horizontal center line. The center point shall be secured to the wing structure by a pinned joint, designed to absorb side loads only. The inboard and outboard mounts shall be secured by hangar linkages to supporting brackets on the wing structure where they are spherically jointed to an interconnecting horizontal member. The hangar linkage shall be designed to accept vertical loads only, and to prevent unacceptable deflections from being transmitted to the engine turbine housing. Provision shall be made in each outboard linkage for vertical adjustment of the engine during installation, with lateral adjustment provided at the center mount.

3.10.4 Vibration Isolation

Not applicable.

3.10.5 Fire Walls

The engine shrouds and steel liners described in paragraph 3.11.2 shall form the fire walls.

3.10.6 Cowling and Cowl Flaps

Not applicable.

3.10.7 Inspection and Maintenance

Access doors shall be provided for inspection, maintenance, removal and installation of engines and accessories. The fuselage tail section shall be removable for engine change. (94)

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3.11 Propulsion

The propulsion system shall be designed to the requirements of R.C.A.F. Specification AIR 7-4 and ARDCM 80-1 except as stated herein and in Appendix II (Deviations).

3.11.1 Engines

The aircraft shall be powered by two Pratt and Whitney J75 turbojet engines. Each engine shall have a static sea level military thrust rating of 15,500 pounds and a maximum thrust rating, with afterburner, of 23,500 pounds.

3.11.2 Engine Installation

Each engine and afterburner shall be contained in a shroud (95)
fabricated from aluminum alloy and titanium. A fireproof, (24)
ceramically insulated, stainless steel liner shall be in- (22)
stalled in sections around the afterburner and the "hot" (23)
region of the engine.

Engine mounts shall be in accordance with paragraph 3.10.3 and the afterburner shall be mounted rigidly to the engine.

All services, except for engine air bleeds, shall enter the engine shrouds in the region of the access doors in the underside of the engine bays, and shall be quickly detachable to facilitate engine removal.

Engine installation and removal shall be carried out using (104
an engine stand. Rails installed from the stand to brackets within the engine shroud, shall, on engine installation, automatically locate the engine in the mounting position. Securing of the engine mounts shall lift the engine sufficiently to permit removal of the rails.

3.11.3 Engine Driven Accessories

3.11.3.1 Description

An alternator, with constant speed hydraulic drive, shall be mounted on the accessories pad at each engine inlet face. The constant speed drive shall consist of a combination hydraulic pump and motor, the fluid for which shall be supplied from the accessories gear box oil system.

One gear box shall be mounted on the accessories drive of each engine beneath the high pressure compressor housing. The starter and generator pads shall be utilized for mounting the gear boxes which shall be driven from the starter pads only. Each gear box shall provide a mounting for the engine starter, and a power take-off for an aircraft accessories gear box.

3.11.3.2 Remote Gear Boxes and Drives

Two aircraft accessories gear boxes shall be located between the engines, each shaft-driven from the adjacent power take-off. Each gear box shall drive three hydraulic pumps and provide a power take-off to drive the fuel booster pump.

A drive shall be provided on each gear box for motoring from a ground power supply for hydraulic system checking.

3.11.4 Air Induction System

3.11.4.1 Description and Components

The air intakes shall be located outboard of the crew (21)
stations. They shall be approximately "D" shaped and external compression shall be achieved by a two dimensional ramp with a 12° wedge attached to the side of the fuselage. The duct from inlet to engine shall diffuse from 5.6 square feet at the inlet to 7.1 square feet at 9 feet from the inlet, then hold a constant diameter circular section back to the compressor face.

The boundary layer air of the fuselage shall pass beneath the ramp leaving the "clean" air to approach the intake. This air in turn builds up its own boundary layer which shall be sucked through a porous strip on the ramp parallel to the duct intake face. The air then entering the intake shall have the least possible turbulence therefore the maximum relative speed, i.e. ram pressure.

3.11.4.2 Air Filters

Not applicable.

3.11.4.3 Intercoolers

Not applicable.

3.11.5 Exhaust System

The turbine exhaust shall be forced rearward through a nozzle whose orifice is automatically increased in area when afterburning is selected. To improve cooling air flow and provide an increase in thrust, the steel liner surrounding the nozzle shall form an afterburner ejector.

Thermocouples shall be installed in each jet pipe to provide turbine discharge temperature sensing.

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3.11.5 Exhaust System (Cont'd)

The design of the exhaust system shall permit longitudinal and radial expansion and contraction of the exhaust system components.

3.11.6 Cooling System

3.11.6.1 Engine Cooling

At speeds less than Mach 0.5 ventilating and cooling air shall be drawn through spring loaded, inwardly opening doors at the forward end of the rear engine zone. The air shall be induced through the afterburner section by the main engine nozzle ejector, and through the accessories section by an engine air supplied ejector. (Engine compressor air shall not be supplied to this ejector at speeds in excess of Mach. 0.5).

At speeds in excess of Mach 0.5 cooling shall be provided by air entering the upper segment of a peripheral intake immediately upstream of the compressor face and circulating around the engine and accessories. The flow of air (109 shall be controlled by gills which close at speeds below Mach 0.5.

Engine cooling air shall be exhausted at the circumference of the afterburner tail pipe, and accessories cooling air through an ejector nozzle on the underside of the aircraft.

3.11.6.2 Heat Exchangers

There shall be three air cooled heat exchangers beneath each engine which are cooled by intake air entering the (109) lower segment of the peripheral intake. Additionally, three fuel cooled heat exchangers shall be installed ahead of the front face of each engine. In the case of both the air and fuel cooled exchangers one shall be used for cooling engine oil, supplementing the oil cooling system which is part of the engine, one for cooling flying control hydraulic oil, and one for cooling gear box and constant speed drive oil.

3.11.7 Lubrication System

3.11.7.1 Description and Components

Lubrication of the engine shall be a closed system except for supplementary cooling as described in paragraph 3.11.6.2. Low pressure warning lights shall be installed on the pilot's warning indicator panel.

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3.11.7.1 Description and Components (Cont'd)

For further details of engine lubrication see applicable Pratt and Whitney Engine Specification.

3.11.8 Fuel System

A pressurized fuel system of sufficient capacity to meet engine requirements, shall be installed in the aircraft. (28)

In the event of a single strike not more than 20% of the fuel remaining in the tanks shall be lost, unless a collector tank is ruptured in which case not more than 50% of the remaining fuel shall be lost. (87)

Full fuel flow to the engines during inverted flight shall be provided for 15 seconds at sea level, or for approximately 45 seconds at combat altitude. (85)

3.11.8.1 Description and Components

The fuel system shall be basically divided into left hand and right hand sub-systems. Fuel shall be transferred from the storage tanks of each sub-system to the respective collector tank by pressurization, (Reference paragraph 3.11.8.10) and a 5-way flow proportioner shall maintain the flow from each tank at a predetermined value in order to restrict the aircraft c.g. travel within specified limits.

A booster pump submerged in each collector tank, and driven by the airframe accessories gear box, shall supply fuel to the engine feed manifolds via an engine fuel proportioning unit. This unit shall normally equalize the flow of fuel from each sub-system, regardless of throttle setting and consequent engine consumption, thus maintaining the aircraft lateral balance. (83) (89)

In the event of failure of either sub-system and/or either engine, the proportioner shall maintain fuel flow from the operative sub-system(s) to the operative engine(s). (86)

In the event of pump failure, tank pressurization and suction of the engine fuel proportioning unit shall provide fuel flow through a by-pass around the inoperative pump.

Fuel shut-off valves shall be installed adjacent to the engine fire walls to provide for isolation of each engine. Actuation of either engine fire extinguishing system shall automatically close the appropriate valve.

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3.11.8.4 Auxiliary Fuel Tank

Structural provision shall be made in the underside of the fuselage for the attachment of a jettisonable tank of 500 Imperial gallons useable fuel capacity.

3.11.8.5 Piping and Fittings

The piping, couplings and fittings for the fuel and pressurization systems shall be aluminum alloy. Quickly detachable connectors shall be provided in the engine supply lines at the points of connection to the engine. (111)

3.11.8.6 Valves

Valves and other components of the fuel system shall be designed to function in the presence of air at temperatures between -65°F and +350°F, and the fuel designated in paragraph 3.11.8.2 at temperatures between -65°F and +185°F.

3.11.8.7 Strainers and Filters

An eight mesh strainer shall be installed in each inlet to the booster pumps. (105)

A two hundred mesh screen filter shall be installed in the pressurization line from the pneumatics system to filter the pressurization air.

3.11.8.8 Quantity Gauges, Flowmeters and Indicators

A capacitance type fuel contents system shall be installed in the aircraft. Two quantity gauges, indicating in pounds, the quantity of fuel in each sub-system shall be installed in the front cockpit. (90)

Four warning lights shall be installed in the front cockpit, one to indicate failure of either booster pump, one to indicate failure (by-pass open) of either 5-way flow proportioner, and two, one left hand and one right hand, to indicate low fuel level in the respective collector tank.

3.11.8.9 Purging and Explosion Suppression System

Not applicable. (25)

3.11.8.10 Pressurization

The fuel system shall be pressurized, using air from the pneumatics system (Reference paragraph 3.15.1.2) to

3.11.8.10 Pressurization (Cont'd)

transfer fuel, to prevent fuel boiling at altitude and to provide pressure for defueling.

The fuselage tanks shall be maintained at 10 psig and the wing tanks, (except collector tanks) at 25 psi abs. by pressure regulating valves. Pressure relief valves shall be installed in the main pressurization lines to the fuel tanks in the fuselage and each wing to prevent over-pressurization in the event of a regulating valve failure. A negative "g" and low level air admission valve shall be installed at each collector tank inlet to permit the entry of air during final emptying of the tanks and also during periods of negative "g".

Flow limiters shall be installed in each wing pressurization system, and in the fuselage pressurization system to limit the flow of air required to be handled by the pressure relief valves to a value commensurate with initial pressurization build-up requirements. To prevent excess spillage of air in case of tank damage the pressurization lines to individual wing tanks shall be appropriately sized, and flow limiters shall be installed in the line to each fuselage tank.

3.11.8.11 Vent System

In flight venting of the differentially pressurized fuselage tanks shall be accomplished through a differential pressure relief valve installed in each tank. Venting of the wing storage tanks, which are maintained at an absolute pressure, shall be unnecessary.

When the pressure in the collector tanks exceeds that required for negative "g" and low level air admission (Reference paragraph 3.11.8.10), accumulated air shall be vented through a fuel-level-sensitive air release valve installed in each collector tank.

3.11.8.12 Refueling System (Ground)

The refueling system shall provide for pressure refueling (and defueling) of the internal tanks through two pressure fuel servicing adaptors. One adaptor to mate
with refueling nozzle Type D1 (MIL-N-5877) shall be in- (88)
stalled in each main landing gear well. The system (29)
shall permit refueling to the combat radius of action fuel load within five minutes.

From each adaptor fuel shall pass to the respective 5-way proportioner and then along the fuel transfer

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3.11.8.12 Refueling System (Ground) (Cont'd)

lines to all tanks except collector tanks. A separate line shall be installed to fill each collector tank, as the transfer lines from the proportioners to the collector tanks will be closed during refueling to prevent overfilling the collector tanks. During full refueling a by-pass on the proportioners shall be opened providing a minimum restriction to filling. A dual shut-off valve, servo controlled by a dual level sensing unit, shall be installed in each tank. During partial refueling the (26)
by-pass valve shall be closed and the proportioners shall operate in reverse, controlling the amount of fuel entering each tank so as to maintain the aircraft c.g. within specified limits. The pressurization relief valves shall be opened to provide venting of all storage tanks, with venting of the collector tanks provided through the air release valves.

Controls and indicators located adjacent to the left-hand speed brake and each refueling adaptor, shall provide for selection and indication of the refueling or defueling operation.

3.11.8.13 Refueling System (In Flight)

Not applicable.

3.11.8.14 Drainage

Combination condensate and drain valves shall be in- (30)
stalled at the low point in each wing tank, except tank number 4, to permit ground purging of water or drainage of fuel from each tank. Wing tank number 4 and the fuselage tanks shall be provided with condensate drain valves only.

3.11.8.15 Defueling Provisions (Ground)

Defueling shall be accomplished through the two fuel servicing adaptors. Fuel from all internal tanks, except the collector tanks shall be transferred to the adaptors through the normal fuel transfer lines by pressurizing the tanks from a ground service unit (Reference paragraph 3.22.2). Fuel shall be removed from the collector tanks through the lines used for filling, by suction of the ground service unit. All valves shall be appropriately positioned by selection of "defuel" on the selector switch (Reference paragraph 3.11.8.12).

3.11.8.16 Fuel Jettisoning

Applicable to auxiliary fuel tank only. Reference paragraph 3.11.8.4.

3.11.8.17 Maintenance and Inspection

Hand holes shall be provided for access to the interior of each tank for inspection and maintenance of all equipment requiring such attention.

3.11.9 Water Injection System

Not applicable.

3.11.10 Propulsion System Controls

3.11.10.1 Description and Components

The power plant controls shall comprise a throttle lever for each engine and afterburner, engine starting switches, and engine re-light buttons.

3.11.10.2 Engine Control System

The throttle levers shall be mounted on a quadrant on the left hand console and shall provide for selection of a full range of engine power with positions for "off" "idle" and "military" thrust. Initial movement of the throttle levers shall open the high pressure fuel cocks. (98)

Depression of the throttle levers in any position forward of two-thirds throttle shall operate microswitches for selection of afterburning. Variation of thrust with the afterburners operating shall be achieved by variation of engine power, with the afterburners operating at constant power. Rearward movement of the throttle levers to one-third throttle or less shall automatically cut out the afterburners. (74)

The throttle levers shall be connected to the automatic fuel metering controls, provided as part of the engine, by a system of cables and pulleys.

A normal/emergency switch shall be installed so that in the event of failure of the automatic fuel controls on the engine the pilot may have manual control.

3.11.10.3 Induction Air Controls

Not applicable.

3.11.10.4 Starter Controls

Two "start-off-reset" switches shall be installed on the right hand console in the front cockpit to control the electrical power supply to the engine igniter systems, and to the starting external air supply control

3.11.10.4 Starter Controls (Cont'd)

valves. Centrifugal switches shall be installed in each starter system to complete the circuits to the ignitors when the engines reach 700 rpm, and to break the circuits when the engines reach 3000 rpm. Indicator lights adjacent to the starter switches shall be installed to signal the pilot to move the throttles from "off" to "idle" at 700 rpm.

A relight button shall be installed in each throttle lever for the purpose of relighting the engines in flight.

3.11.10.5 Propeller Controls

Not applicable.

3.11.10.6 Cooling Air Controls

Cooling air control shall be a function of Mach No. (Reference paragraph 3.11.6.1).

3.11.10.7 Water Injection Controls

Not applicable.

3.11.11 Starting System

An air turbine starter shall be mounted on each engine, (Reference paragraph 3.11.3.1). The starters shall be powered from a ground source and shall be capable of meeting the scramble requirement of paragraph 3.1.3.1. Automatic quick disconnects shall be provided for the ground air supply.

3.11.12 Propeller

Not applicable.

3.11.13 Rocket Propulsion System

Not applicable.

3.12 Auxiliary Power Plant

Not applicable.

3.13 Instruments and Navigational Equipment

Instrument arrangement shall be subject to agreement between the R.C.A.F. and the Company, based on the operational role of the aircraft and on ORI/4-2. Navigational equipment in accordance with the requirements of R.C.A.F. Specification AIR 7-4 shall be installed.

3.13.1 Instruments

3.13.1.1 Pilot's Instruments

3.13.1.1.1 Flight Instruments

Mach Air Speed Indicator
Artificial Horizon Indicator
Rate of Climb Indicator
Turn and Bank Indicator
Pressure Altimeter
Radio Magnetic Indicator (Ref. Para. 3.13.3.1 & 3.17.2.8)
Accelerometer
Radio Altimeter (Ref. Para. 3.17.2.2)

3.13.1.1.2 Navigational Instruments

Clock
Standby Magnetic Compass
R-Theta DR Repeater (Ref. Para. 3.13.3.2)
Tacan Deviation Indicator (Ref. Para. 3.17.2.8)

3.13.1.1.3 Engine Instruments

Turbine Discharge Temperature Gauge (2) (Ref. Para. 3.11.5)
Turbine Discharge Pressure Ratio Indicator (2) (Ref. Para. 3.11.5)
Fuel Contents Indicator (2) (Ref. Para. 3.11.8.8).

All engine instruments shall be of the electrical remote single indicating type and of 2 inch case size to drawing AND 10412.

3.13.1.1.4 Miscellaneous Instruments

Skin Temperature Gauge
Cabin Pressure Gauge (Ref. Para. 3.22.1.1.1)
Oxygen Pressure Gauge (Ref. Para. 3.21.1.4)
Oxygen Quantity Gauge (Ref. Para. 3.21.1.4)
Landing Gear Position Indicator (Ref. Para. 3.8.2.5.4)

3.13.1.2 Radar Operator's Instruments

To be determined.

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3.13.2 Air Data Sensing Equipment

3.13.2.1 Pitot Static System

To be determined.

3.13.2.2 Relative Wind Sensors

To be determined.

3.13.2.3 Air Temperature Sensors

To be determined.

3.13.2.4 Air Data Scheduling Equipment

To be determined.

3.13.3 Navigational Equipment

3.13.3.1 J-4 Compass

A J-4 type compass system shall be installed to provide indication of the magnetic heading of the aircraft on a radio magnetic indicator in each cockpit and to an R-Theta DR Computer. A controller incorporating the system switches and controls shall be installed in the pilot's cockpit and power for the system shall be supplied from the 115V AC bus and the 27.5V DC bus.

3.13.3.2 R-Theta DR Navigation System

The R-Theta dead reckoning navigation system shall be installed in the aircraft for the purpose of automatically and continually computing the position of the aircraft in relation to a selected datum point. The system shall also provide true airspeed reading, track data and a selection of heading indications.

A pilot's DR repeater shall be installed in the front cockpit and an indicating DR computer and an indicating ground speed computer shall be installed in the rear cockpit.

True airspeed information shall be obtained from the air data sensing equipment and aircraft headings from the J-4 compass system. Wind information shall be fed in manually.

3.13.3.3 Navigation Radio Aids

The navigation radio aids shall be as described in paragraph 3.17.2.

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3.13.4 Installation

The instruments and main instrument panels shall be installed in accordance with the requirements of AIR 7-4 and specifications MIL-I-5997 and MIL-C-6818A as applicable. The connections to the instruments and instrument panels shall be flexible to the extent that free action of the shock absorbers is not restrained. All hoses and electrical leads shall be of sufficient length to permit the instruments to be withdrawn from the panel for disconnection. (32) (97) (96)

3.13.4.1 Instrument Markings

On all contractor furnished instruments the major scale markings and pointers shall be treated with Specification C-28-96 or U.S. Radium R410AB self-luminous compound, and all minor scale markings shall be treated with fluorescent compound to specification AN-L-1A. Range and limit markings shall be applied to all instruments requiring such markings.

3.13.4.2 Inspection and Maintenance

All instruments and connections thereto shall be accessible without removal of other instruments or equipment. Four knurled nuts at the mounting brackets shall permit quick removal of the main instrument panel for inspection and maintenance.

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3.14 Hydraulic Systems

3.14.1 Description and Components

Three separate 4000 psi hydraulic systems shall be installed in the aircraft:-

A utility services system, to operate the landing gear, nose wheel steering, wheel brakes, speed brakes and a removable armament pack.

Two flying control systems, each providing sufficient power for limited control of the aircraft in the event of failure of the other.

The systems shall be designed in accordance with the requirements of Specification MIL-H-5440 except as stated in Appendix II (Deviations) and herein. System design shall permit a maximum operating fluid temperature of 250°F, with local rises to 275°F.

Six engine driven hydraulic pumps shall be installed, two in the utility services system power circuits, and two in each flying control system power circuit. Three compensators, one for each system, shall provide fluid reserve, pump inlet pressurization, and system ground pressurization. Hydraulic power for emergency operation of the brakes, and pressurized air for emergency extension of the landing gear shall be provided.

Ground operation for system testing with the engines inoperative shall be effected by use of the aircraft pumps, the aircraft accessories gear boxes being designed to permit use of a mobile plug-in mechanical power unit.

3.14.1.1 Utility Services System

3.14.1.1.1 Utility Services System Power Circuit

Two 4,000 psi variable delivery hydraulic pumps shall be installed, one on each aircraft accessories gear box with the combined output from both pumps utilized to power the utility services and charge an accumulator. The accumulator output shall be reduced to 1,500 psi and utilized for the emergency brake supply, and to pressurize the compensator in each hydraulic system.

Two warning lights shall be installed on the pilot's warning indicator panel, one to indicate when the utility services pressure falls below 1,000 psi and one to indicate when the pressure stored in the accumulator falls below 1,600 psi.

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3.14.1.1.2 Landing Gear Sub-System

The landing gear and landing gear door actuation shall be hydraulically sequenced during retraction and normal extension. Actuation shall be controlled by a selector valve in conjunction with a manually operated selector lever installed in the pilot's cockpit.

Emergency extension shall be by air from a 5000 psi storage bottle with release of the air controlled by the landing gear selector lever (Reference paragraphs 3.8.2.5.6 and 3.8.3.5.6).

3.14.1.1.2.1 Retraction

Up selection shall hydraulically release all gear downlocks and raise the gear until the uplocks engage mechanically. In the last stages of the engagement of each gear uplock, a controllable check valve shall permit hydraulic pressure to release the door downlock and cause the actuator to raise the door until the door uplocks engage mechanically.

3.14.1.1.2.2 Extension

Down selection shall hydraulically release all landing gear door uplocks and lower all doors until the downlocks are engaged. As each door is locked down, a controllable check valve shall permit hydraulic release of the gear uplock, and hydraulically operate a transfer valve. The transfer valve shall release the hydraulic pressure from the landing gear actuator, permitting the gear to fall by gravity and drag forces until a downlock engages mechanically.

3.14.1.1.2.3 Emergency Extension

Emergency down selection shall permit a supply of air from the emergency air storage bottle to enter an emergency extension circuit. The emergency circuit shall permit the compressed air to simultaneously release all gear and door uplocks and operate the door actuators. The landing gear shall extend by gravity and drag, and lock in the down position.

3.14.1.1.3 Nose Wheel Steering Sub-System

A double ended hydraulic actuator shall be installed for nose wheel steering. A selector valve controlled by a push button on the pilot's control column shall be installed for selection or release of hydraulic pressure for steering. A follow-up type steering control

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3.14.1.1.3 Nose Wheel Steering Sub-System (Cont'd)

valve shall be mechanically linked to the rudder pedals through a hydraulic clutch to prevent transmission of rudder pedal movement to the valve until rudder pedal deflection has been synchronized with nose wheel deflection. Release of steering hydraulic pressure shall permit a restricted run-around hydraulic circuit to provide shimmy damping and hydraulic assist to nose wheel centering (Reference paragraph 3.8.3.6).

3.14.1.1.4 Wheel Brakes Sub-System

The hydraulic pressure available for normal brake application shall be a maximum of 2130 psi reduced from the 4000 psi utility hydraulic system. Pressure available for emergency brake application shall be a maximum of 1500 psi reduced from a 400 psi accumulator. Two control valves shall be linked, one to each brake pedal, to permit metered differential control of the brakes. Each valve shall incorporate a transfer component for automatic changeover to the emergency brake supply. A solenoid operated valve shall be incorporated in each control valve to permit automatic brake operation during main gear retraction. Locking the brakes for parking shall be achieved by means of a mechanical linkage controlled from the front cockpit.

The normal pressure control outputs shall be conveyed to the wheel brakes through an anti-skid and shuttle valve assembly installed on each main gear and governed by a skid detector installed on each wheel of the associated bogie.

In the event of normal supply pressure failure, the emergency brake pressure shall be routed through an anti-skid return line to the shuttle valve and to the brakes. The anti-skid valve shall be by-passed during emergency actuation of the brakes. (34)

3.14.1.1.5 Speed Brakes Sub-System

Two hydraulic actuators, one for each of the two speed brakes, shall be controlled by a selector valve in conjunction with a three position switch. A relief valve to limit the degree of speed brake extension in relation to speed brake air loads (Reference paragraph 3.9.2.2). A check valve shall be incorporated in the pressure line to prevent excessive back pressures, set up by high hinge moments on the speed brakes, from entering the pressure lines of the utility system. (75)

3.14.1.1.6 Armament Pack Supply

A pressure and a return line shall be installed and shall terminate in a self-sealing half coupling at the aft end of the armament pack bay. The couplings shall correspond with half couplings carried by any armament pack designed to be fitted to the aircraft.

3.14.1.2 Flying Control Systems

3.14.1.2.1 Flying Control Systems Power Circuits

The two flying control hydraulic systems shall comprise an "A" and "B" system, each powered by two 4000 psi variable delivery pumps. One pump of each system shall be installed on each of the two aircraft accessories gear boxes. The output of the two pumps for each system shall be combined and utilized to power control surface actuators and servo units. (108

Two warning lights, one for each flying control system power circuit shall be installed on the pilot's warning indicator panel to indicate loss of pressure in either power circuit to 1000 psi or less. A red and an amber Master Warning shall indicate failure of both circuits. (Reference paragraph 3.16.11.1).

3.14.1.2.2 Control Actuators and Servo Units

Tandem dual cylinder and piston type actuators shall be installed to permit hydraulic actuation of the control surfaces from the two independent "A" and "B" hydraulic systems. Single differential servo control units shall be installed on the aileron and elevator actuators to permit damping system signalled hydraulic operation from the "B" system. A dual differential servo control unit shall be installed on the rudder actuator to permit rudder damping signalled hydraulic operation from both "A" and "B" systems.

Two command (parallel) servo control units shall be installed and powered from system "B" to permit pilot command signal controlled hydraulic operation of the control valves of the aileron and elevator hydraulic actuators.

3.14.1.2.3 Flying Control Systems Return Circuits

An air cooled heat exchanger and a fuel cooled heat exchanger shall be installed to limit the temperature of hydraulic fluid at the pump inlets to 225°F (approx.).

3.14.1.2.3 Flying Control Systems Return Circuits (Cont'd)

Each heat exchanger shall comprise two sections to permit each flying control hydraulic system to be kept functionally separate.

A compensator designed to pressurize the return fluid at 90-100 psi and to separate air from the fluid, shall be installed. It shall be possible to manually ground bleed the separated air from the compensator.

In each hydraulic system return circuit a self displacing type accumulator shall be installed to damp out surges set up by the continual operation of the control actuators, and reduce return pressure fluctuations in the compensator.

3.14.1.2.4 Compensator Pressure Supplies

The compensator of each flying control system return circuit shall be pressurized by a 1,500 psi supply from the Utility Services System Power Circuit. Emergency pressurization of the compensators, at 1,250 psi shall be automatically available from the respective flying control power circuit.

3.14.1.3 Filters

High and low pressure ten micron filters shall be installed in the main pressure and return lines respectively of all three main hydraulic systems. The filters shall embody pressure differential by-pass valves set at approximately 50 psi.

In-line type filters shall be installed in the pressure line of the nose wheel steering sub-system and in the supply lines to the aileron and rudder control actuators.

3.14.1.4 Inspection and Maintenance

Access panels and doors shall be installed to facilitate inspection and maintenance.

Three separate filling connections, one for each hydraulic system shall be installed on the aircraft. (35)

Provision for disconnecting the aircraft accessories gear box drives from the engine gear boxes shall permit each gear box to be driven independently by a ground power unit

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3.14.2 Hydraulic Fluid

The hydraulic systems shall be designed for the use of hydraulic fluid to Specification MIL-O-5606.

3.14.3 Piping and Fittings

High pressure lines shall be of stainless steel. Low pressure lines shall be of aluminum alloy. All piping joints shall be made with flareless fittings to Avro Aircraft Company Standards. The use of flexible rubber hose shall be kept to a minimum. To cater for component motions, expandable and/or swivel type couplings shall be used. (33)

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3.15 Pneumatic System

The pneumatic system shall comprise several sub-systems which shall utilize air pressure from the air conditioning system (Reference Section 3.22). Complete provision shall be included for an air supply to be taken from the heat exchanger input to power a turbine driven generator, when installed, for an integrated electronics system. Structural provision shall be made for an air supply to power a turbine driven generator in a Sparrow missile pack (when fitted). Ground operation of the pneumatic services shall be possible by utilization of air supplied by the ground air conditioning service unit (Reference Section 3.22).

A ground chargeable air storage bottle shall be installed in the aircraft to supply air for emergency extension of the landing gear (Reference Section 3.14).

The system shall be designed in accordance with the requirements of ARDCM 80-1 except as stated in Appendix II (Deviations) or as additionally stated herein.

3.15.1 Description and Components

The sub-systems constituting the pneumatic system shall be as follows:

(1) A low pressure services sub-system for:-

(a) Canopy and windshield seal inflation.
(b) Anti-G suit inflation.
(c) Radome anti-ice system air supply.

(2) Fuel tank pressurization air supply.

(3) Windshield rain repelling system.

(4) Complete provision for turbine air supply.

3.15.1.1 Low Pressure Services Sub-System

Air at 85 psi (max.) pressure from the air conditioning system heat exchanger shall be filtered and utilized to inflate the canopy and windshield seals, anti-G suits, and to operate the radome anti-ice fluid system. The filter shall incorporate a drainable moisture trap.

3.15.1.1.1 Canopy and Windshield Seal Inflation

Air shall be ducted to a solenoid operated, pressure regulating, pressure relieving and check valve. The valve shall be designed to provide seal inflation air reduced at 20 psig pressure, to relieve seal pressure

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3.15.1.1.1 Canopy and Windshield Seal Inflation (Cont'd)

exceeding 25 psig, and to prevent back flow from the seals when the solenoid is energized. When the solenoid is de-energized the valve shall vent seal pressures. The control solenoid shall be electrically linked to the canopy latches of both cockpits.

A male threaded fitting for connecting an external source of pressure for canopy and windshield seal inflation during cockpit leakage tests shall be installed.

3.15.1.1.2 Anti-G Suit Inflation

A branch duct shall convey air from the filter to an anti-G valve in each cockpit. The anti-G valves shall automatically control anti-G suit inflation. (82)

3.15.1.1.3 Radome Anti-Ice System Air Supply

A branch from the anti-G suit inflation air supply ducting shall convey air to a pressure reducing valve which shall reduce the pressure to 10 psi. The reduced pressure air shall be utilized to:-

(1) Pressurize the radome anti-ice fluid tank through a check valve.

(2) Provide purging air for the anti-ice fluid distributor supply line at the end of each fluid distribution period.

3.15.1.2 Fuel System Pressurization Supply

Air at 85 psi (max.) shall be ducted from the air conditioning system heat exchanger to a hot air filter. The output from the filter shall be ducted to the pressure reducing valves of the fuel tank pressurizing system (Reference Section 3.11.8.10).

3.15.1.3 Windshield Rain Repelling System

Air shall be ducted from the air conditioning system heat exchanger to a distributor installed outside at the base of the windshield. The duct shall incorporate an electrically operated on-off valve controlled by a switch in the pilot's cockpit and thermostatically controlled to shut off the supply when air temperatures exceed 250°F.

3.15.1.4 Piping and Fittings

Low temperature pipes or ducting shall be of aluminum alloy and high temperature or highly stressed ducting

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3.15.1.4 Piping and Fittings (Cont'd)

shall be of stainless steel. Couplings below 1 inch diameter shall be flareless type couplings to Avro Aircraft Company Standards. Couplings of 1 inch and above diameter shall be band type couplings.

3.15.1.5 Inspection and Maintenance

Equipment components of the pneumatic system shall be made accessible for inspection and maintenance. (36)

3.15.2 Ground Operation

Air at a pressure of approximately 45 psi shall be available for operation of the pneumatic services from the air conditioning ground air supply.

3.16 Electrical System

The aircraft electrical system shall consist of a 208/120 volt 400 cycle, 3 phase AC system and a 27.5 volt DC system. Two engine driven alternators shall be the prime source of electrical power with provision for the conversion of AC to DC power by means of two transformer-rectifiers.

The electrical system shall be designed to the requirements of R.C.A.F. Specification AIR 7-4 except as stated herein and in Appendix II (Deviations).

3.16.1 General Description

3.16.1.1 AC System

The AC system shall be a three wire, star connected, neutral grounded system. The primary and secondary AC loads shall normally be connected to the right and left alternator systems respectively. Provision shall be made for cockpit indication of power failure and automatic transfer of the primary load to the left alternator and for disconnection of the secondary AC services in the event of the right alternator failure.

3.16.1.2 DC System

The DC system shall be a single wire, negative ground return system. The DC loads shall be distributed among the main, shedding, emergency and battery buses. Provision shall be made in the system for discontinuing the power supply to the shedding bus in the event of a single engine failure.

3.16.1.3 Emergency DC System

A battery shall be installed to supply the emergency DC power, with distribution of power to the emergency services through the emergency and battery buses.

Provision shall be made in the system for the isolation of the battery and emergency buses from the main DC bus in the event both transformer-rectifiers fail.

3.16.1.4 Distribution

An electrical power junction box, containing bus bars, relays and protective devices shall be installed in the electrical equipment compartment for interconnection and distribution of AC and DC power to the various aircraft services.

3.16.2 Electrical Power Supply

3.16.2.1 Alternators

A 20 KVA 208/120 volt 3 phase, 400 cycle alternator driven by a constant speed regulating mechanism shall be installed in the nose bullet of each engine. Cooling for the alternators shall be provided by ram air.

3.16.2.2 Voltage Regulators

Two voltage regulators, part of the control/transformer rectifier panels, shall be installed in the electrical equipment compartment to provide voltage regulation of each alternator output. Cooling of the control/transformer-rectifier panels shall be provided by the air conditioning system (Reference paragraph 3.22.1.2.1).

3.16.2.3 Controls

A master ON-OFF power supply switch shall be installed in the front cockpit. Two alternator failure warning lights, and two ON-RESET-OFF switches to permit individual control of each alternator output shall be installed in the front cockpit. Two DC failure warning lights, and a DC RESET switch to permit restoration of DC power output in the event of short duration failures due to transient faults shall be located in the front cockpit.

3.16.3 Electrical Power Conversion

3.16.3.1 Transformer Rectifier Units

Two 3KW transformer rectifiers, one located in each control/transformer-rectifier panel, shall provide for the conversion of AC power to 27.5 volt DC power.

3.16.3.2 Protective Devices

Over voltage and reverse current protection shall be provided in accordance with the requirements of MIL-E-~~7894~~. 7954

3.16.3.3 Battery

A 24 volt, 15 amp hr., nickel cadmium, hermetically sealed storage battery, shall be installed in the nose wheel well. The battery shall normally be connected to the battery bus with provisions for automatic disconnection when external DC power is used. Cooling air for the battery shall be supplied by the air conditioning system.

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3.16.4 Equipment Installation

The electrical equipment shall be installed in accordance with the requirements of specifications MIL-E-7563, MIL-E-7080 and MIL-E-7614. (72)

3.16.5 Wiring

The installation of all aircraft wiring shall be in accordance with specification MIL-W-5088A. (39) (40) (41) (42)

3.16.6 Bonding and Shielding

Bonding and ground returns shall be installed in accordance with specification MIL-B-5087. Shielded wire shall be used where required.

3.16.7 Controls

Rheostats, resistors and switches shall be installed in accordance with the requirements of specifications MIL-E-7563 and/or MIL-E-7080. (38)

Circuit breakers shall be installed in accordance with the requirements of specification MIL-E-7614 and located on a circuit breaker panel in the nose wheel well.

Current limiters shall provide circuit protection in locations where high ambient temperatures, preclude the use of circuit breakers.

Fuses shall be installed in the console panel of each cockpit to provide protection for the cockpit lighting circuits.

3.16.8 Lighting

3.16.8.1 Interior Lighting

The interior lighting comprising instrument panel, console panel and map lighting, shall be installed in accordance with specifications CAP 479, MIL-P-7788 and MIL-L-6503.

3.16.8.1.1 Instrument Panel Lighting

The instrument panel lighting shall consist of red edge panel lights and post type flood lights.

3.16.8.1.2 Console Panel Lighting

The console panel lighting shall consist of red edge panel lights and hooded type red flood lights. Wiring

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3.16.8.1.2 Console Panel Lighting (Cont'd)

provisions shall be made for installation of high altitude white flood lighting.

3.16.8.1.3 Interior Illumination Controls

Three continuously variable transformers (0-27.5 volts) shall be installed in each cockpit to provide illumination control of post-type red flood lights, red edge lights, and the console flood lights respectively. An ON/OFF switch shall be provided in each cockpit for high altitude white flood lights.

3.16.8.1.4 Map Lighting

An amber flood lamp with an integral intensity control shall be installed in each cockpit to provide illumination for map reading. These lights shall be connected directly to the emergency DC bus and may be used for emergency lighting purposes.

3.16.8.2 Exterior Lighting

The exterior lighting comprising the navigation, taxi, and landing lights, shall be installed in accordance with R.C.A.F. specifications AIR 7-4 and CAP 479.

3.16.8.2.1 Navigation Lights

Navigation lights shall consist of a red port wing tip light, green starboard wing tip light, and one red and one white light in the trailing edge of the fin. A flasher unit shall be installed in accordance with Specification MIL-L-6503.

3.16.8.2.2 Taxi Light

A taxi light shall be installed on the nose landing gear assembly such that it will follow the direction of the nose wheel steering.

3.16.8.2.3 Landing Light

A landing light shall be installed on the nose landing gear main strut.

3.16.8.2.4 Exterior Lighting Controls

Exterior lighting controls shall be located in the pilot's cockpit. The control for the navigation lights shall provide for selection of steady, off or flashing.

3.16.8.2.4 Exterior Lighting Controls (Cont'd)

A switch shall be installed to provide for selection of the landing and/or taxi light.

3.16.9 Ignition System

The engine ignition system shall be in accordance with Engine Model Specification No. PWA 2611. Switches, located in the front cockpit, shall provide for selection and control of engine starting.

3.16.9.1 Engine Starting

Two START-OFF-RESET switches shall be installed on the right hand console in the pilot's cockpit to provide control of the DC power to the relevant engine igniter system and to the external air control valve for engine starting. (106)

3.16.9.2 Engine Relight

A relight button shall be provided in each throttle lever for the purpose of relighting the engines in flight.

3.16.10 Receptacles

3.16.10.1 External Receptacles

An external power receptacle conforming to the outline of AN 3114, and suitable for mating with a lanyard operated self-ejecting plug, shall be installed in accordance with specification MIL-E-7563 to provide for a ground supply of AC power.

A receptacle suitable for mating with an external quick disconnect lanyard release connector shall be installed to facilitate cable connection for engine starting control and cockpit to ground intercommunication.

3.16.10.2 Static Ground

A whisker type static grounding device shall be installed on each main landing gear to automatically bring the aircraft to a ground potential on landing.

3.16.10.3 Fuel Nozzle Grounding

An electrical ground receptacle for grounding the refueling nozzle shall be installed adjacent to each refueling adaptor in accordance with the requirements of ARDCM 80-1.

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3.16.10.4 Grounding Jack

A grounding stud located on the underside of the rear fuselage shall facilitate attachment of a grounding cable incorporating a pull-away quick disconnect.

3.16.11 Indicators

3.16.11.1 Master Warning Lights

One red and one amber master warning light shall be installed at the top center of the main instrument panel. Each light assembly shall embody two bulbs which are connected in parallel. The red warning light shall indicate fire detection and the amber warning light shall indicate trouble in any of the circuits designated on a warning indicator panel. Both red and amber warning lights illuminate in the event of loss of pressure in the power circuits of both flying control hydraulic systems. (44)

3.16.11.2 Warning Indicator Panel

A panel with provision for 20 warning indicators shall be installed in the front cockpit to provide, in conjunction with the master warning lights, indication of specific system failure. (44)

The following warning lights shall be incorporated in the panel:

2	Fuel Low Level L.H. and R.H.
1	Fuel Proportioner L.H. or R.H.
1	Fuel Pump Differential Pressure L.H. or R.H.
1	Engine Fuel
2	Engine Fuel Pressure L.H. and R.H.
2	Oil Pressure L.H. and R.H.
2	Flying Control Hydraulic A and B
1	Utility Hydraulic
1	Emergency Brake Hydraulic
2	Alternator Failure L.H. and R.H.
2	DC Failure L.H. and R.H.
2	Low Rotor Overspeed L.H. and R.H.
1	Damping System

Two switches shall be installed on the indicator panel for testing the indicator bulbs and resetting the master warning lights. A two position dimming control shall permit illumination intensity control of the warning panel indicators.

3.16.11.3 Fire Warning

A fire warning light, to operate in conjunction with the master warning light, shall be incorporated in each of the fire extinguisher buttons.

3.16.11.4 Landing Gear Position and Warning Lights

Three landing gear position indicators and a red warning light shall be provided as described in paragraph 3.8.2.5.4.

3.16.12 Electric Drives

Electric drives (canopy actuators, motor operated fuel valves etc.) shall be installed in accordance with specification MIL-E-7080 and MIL-E-7614.

3.16.13 Filters

Radio interference filters shall be installed in the aircraft where necessary. Interference limits and methods of measurement for all installations shall be to the requirements of MIL-I-6051.

3.16.14 Emergency Operation

To be determined.

3.16.15 Inspection and Maintenance

Suitable provisions shall be made in the aircraft for the inspection, maintenance, removal and re-installation of electrical equipment. (37) (43) (107)

3.17 Electronics - (Interim System)

An interim system as defined in Appendix "A" to R.C.A.F. Specification AIR 7-4 shall be installed to the requirements of Specification MIL-I-8700 (ASG), except as stated herein and in Appendix II (Deviations).

The interim electronic system shall comprise the following:

Command Set	AN/ARC-34
Interphone	AN/AIC-10
Radio Compass	AN/ARN-6
Homing Adaptor	AN/ARA-25
Distance Measuring Equipment	AN/ARN-21
Identification Equipment	AN/APX-6

In addition to the above, complete provision shall be made for the installation of a radar altimeter type STR-30 and a radar beacon type RBX-1, and space provision for the installation of automatic ground position indicating equipment (Doppler).

Temperature and pressure within the electronic compartments shall be limited by the air conditioning system (Reference paragraph 3.22). Junction boxes and panels shall be provided to facilitate interconnection of wiring for related systems. All antennas shall be installed internally, either within the aircraft structure or flush with the aircraft skin. The radio controls and selector switches shall be conveniently located in the console panels of the respective cockpits. A micro-switch incorporated in each seat installation shall provide a means for switching the UHF communication transmitter and the IFF transponder to transmit on the distress fequency when the seat is ejected. An override test switch shall be installed in the front cockpit for the purpose of testing the emergency function of the UHF and IFF systems.

3.17.1 Communication Equipment

3.17.1.1 Command Set

An AN/ARC-34 type UHF transceiver shall provide air-to-air and air-to-ground communication facilities on 1750 channels, twenty of which can be preset. This equipment incorporates a guard channel receiver sub-assembly which is tuned to a preset emergency frequency while the main receiver is selected to any other frequency. Provision shall be made in the system for connection of the receiver portion of the UHF homing adaptor (Reference paragraph 3.17.2.6). A UHF remote control unit shall be installed in the front cockpit.

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3.17.1.1 Command Set (Cont'd)

The pilot's and the radar operator's press-to-transmit buttons shall be installed in the inboard throttle grip, and on the left hand console switch panel of the rear cockpit, respectively.

Two omni-directional antennas shall be provided for use with this equipment. A fan shaped vertical radiator shall be mounted under a fiberglass fairing at the top of the fin and a downward facing annular slot type antenna shall be mounted flush with the skin under the electronics bay on the right hand side of the access door. A selector switch located on the front cockpit right hand console shall provide means for connection of either antenna to the set. An indicator located on the rear cockpit right hand console shall indicate which UHF antenna is in use.

The system shall operate on 27.5 volt DC obtained from the emergency DC bus.

3.17.1.2 Liaison Set

Not applicable.

3.17.1.3 Interphone

A type AN/AIC-10 interphone system shall be installed to provide intercommunication between the crew members and a means of selection and audio signal level control of the aircraft's communication and navigational radio facilities. The interphone amplifier shall obtain power from the emergency 27.5 volt DC bus.

Ground operation of the interphone shall provide intercommunication between the crew stations and ground service personnel stations, and between the crew stations and a telescramble land telephone line. Connections for ground operating power and for the land telephone line shall be provided through the external receptacle (Reference paragraph 3.16.10.1). Electrical isolation shall be provided between the aircraft land telephone circuit and ground service circuit.

3.17.1.4 Microphones and Headsets

Complete provision for the use of a type M-32/AIC microphone and a type H-75/AIC headset, or equivalent, shall be provided for each crew member.

A mating combination microphone and headphone jack shall be installed on the right hand side of each ejection seat.

3.17.1.4 Microphones and Headsets (Cont'd)

Quick disconnects shall be provided for automatic separation of the aircrew's microphone and headset cable connections from the aircraft to seat, and from the seat to man (Reference paragraph 3.19.1).

3.17.1.5 Filters

Radio filters shall not be required. Radio interference caused by the operation of the electronic equipment installed in the aircraft shall not exceed the limits defined in Specification MIL-I-6051. (46

3.17.1.6 Recording Equipment

Not applicable.

3.17.2 Navigation Equipment

3.17.2.1 Radio Compass

A type AN/ARN-6 LF-MF radio compass system shall be installed with control facilities in each cockpit. The system shall provide visual bearing indication of a selected radio station on the radio magnetic indicators located on the front and rear cockpit instrument panels.

A non-directional sense antenna and a flush type directional loop shall be fitted.

3.17.2.2 Radar Altimeter

Complete provision shall be made for the installation of a type STR-30 radar altimeter system which shall be installed to provide continuous indication of the aircraft height above terrain within the range of 0-500 feet.

3.17.2.3 Radio Range Receiver

Not applicable.

3.17.2.4 Marker Beacon Equipment

Not applicable.

3.17.2.5 Instrument Approach Equipment

Not applicable.

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3.17.2.6 Homing Adaptor

A type AN/ARA-25 UHF homing adaptor shall be installed in the aircraft. This equipment shall be used in conjunction with the UHF communication receiver to provide a continuous visual indication, on the two radio magnetic indicators, of the direction of a selected UHF signal source.

A directional antenna assembly shall be mounted in the nose radar compartment and a solenoid relay, controlled by the function selector on the UHF controller, shall permit selection of either UHF communication antenna, or the homer directional antenna.

3.17.2.7 UHF Navigation Equipment

Not applicable.

3.17.2.8 Distance Measuring Equipment

Type AN/ARN-21 (TACAN), air navigational equipment shall by installed to provide continuous indication of distance and direction of the aircraft from a selected L-band ground beacon station.

A cross pointer (course deviation) indicator shall be located on the pilot's main instrument panel. A distance indicator and a control box shall be located in the rear cockpit. An RMI needle indication selector switch located in each cockpit shall provide a means of connecting either the UHF homer or the Tacan system to the radio magnetic indicators.

Two L-band antennas shall be installed, a fan shaped dipole located at the top of the fin and an annular slot type located in the electronic power equipment bay door, either one of which can be connected to the Tacan transmitter-receiver. A transfer switch located in the rear cockpit shall provide a means for transferring the connection of the two antennas alternately to the Tacan and the IFF systems.

3.17.2.9 Arbitrary Course Computer

Not applicable.

3.17.3 Radar

3.17.3.1 Search Equipment

Not applicable.

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3.17.3.2 Loran Equipment

Not applicable.

3.17.3.3 Automatic Ground Position Indicating Equipment

Space provision shall be made for installation of the Doppler true ground speed and ground track measuring equipment to DRB Specification EL 5040-1.

3.17.3.4 Identification Equipment

A radar identification set, type AN/APX-6, shall be installed to permit the aircraft to identify itself automatically when interrogated by a ground or airborne L-band radar. The IFF control box shall be installed in the front cockpit.

Antenna requirements shall be furnished by the L-band antenna installation used with the Tacan system (Reference paragraph 3.17.2.8). The L-band antenna transfer switch in the rear cockpit shall permit connection of either antenna to the set. This system shall obtain the DC power from the 27.5 volt DC emergency bus and the AC power from the 115 volt AC bus.

3.17.3.5 Interrogation Equipment

Not applicable.

3.17.3.6 Radar Beacon

Complete provisions shall be made for the installation of an X-band radar beacon, type RBX-1. When installed, this equipment shall provide the means for checking an X-band antenna installed in the fin. The system shall provide a reply pulse in the manner of an IFF transponder when triggered by an X-band AI system. Switching of power to the equipment shall be accomplished by closing the appropriate circuit breaker in the electronics bay before flight. A dual horn antenna, waveguide coupled to the transmitter-receiver unit, shall be installed in the fin.

3.17.4 Electronic Countermeasures

3.17.4.1 Search Equipment

Not applicable.

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3.17.4.2 Analyzing Equipment

Not applicable.

3.17.4.3 Panoramic Receiving Equipment

Not applicable.

3.17.4.4 Direction Finding Equipment

Reference paragraph 3.17.2.8.

3.17.4.5 Transmitting Equipment

Not applicable.

3.17.5 Electronic Guidance System

3.17.5.1 Guide Links and System

Not applicable.

3.17.5.2 Television and Telemetering Equipment

Not applicable.

3.17.6 Static Dischargers

Not applicable.

3.17.7 Emergency Rescue Transmitter

Not applicable.

3.17.8 Inspection and Maintenance

Doors and panels shall be installed to provide quick access into electronic equipment compartments and at antenna installation areas. Equipment shall be mounted so that removal or installation of any unit can be made without need for removal of adjacent equipment.

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3.18 Armament

3.18.1 Weight and space provision shall be made for the installation of air-to-air guided missiles. A fairing shall be installed to close the opening provided to accommodate a missile package.

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3.19 Equipment and Furnishings

Equipment and furnishings appropriate to the primary role of the aircraft shall be installed in accordance with the requirements of CAP 479 and R.C.A.F. Specification AIR 7-4 except as stated in Appendix II (Deviations. (51) (71) (52)

3.19.1 Personnel Accommodation

A fully automatic ejection seat with an ejection velocity of 80 feet per second, and incorporating the necessary controls and safety harness, shall be installed for each member of the crew. Action to fire the seat shall also clear the ejection path.

A composite quick disconnect located on the right hand side of each seat shall provide connections for the following services: (47)

Oxygen
Capstan type pressure suit
Visor demisting
Telecommunications
Anti-g suit

The assembly shall provide for automatic, (on seat ejection) or individual manual disconnect of these services between the crew and the ejection seat, and between the seat and the aircraft.

3.19.2 Miscellaneous Equipment

3.19.2.1 Drinking Water Containers

Not applicable. (49)

3.19.2.2 Crew Relief Provisions

Not applicable. (48)

3.19.2.3 Compass Deviation Card Holder

A compass deviation card holder shall be installed in each cockpit.

3.19.2.4 Pilot's Check List Holder

Provision shall be made in the front cockpit for a pilot's check list.

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3.19.2.5 Map Stowage

A map stowage shall be located on the right hand side of the front cockpit. (53)

3.19.3 Windshield Wipers

A rain repeller shall be provided in accordance with paragraph 3.15.1.3.

3.19.4 Furnishings

Insulation shall be installed on the interior of the cockpit to minimize heat transfer from the adjacent skin. (50)

3.19.5 Emergency Equipment

Accommodation for a special pack parachute and a seat pack emergency kit (14 x 15 x 5.5 inches) shall be provided in each ejection seat.

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3.20 Fire Protection

The fire protection system shall provide for detecting and extinguishing fires in the hydraulics bay and each engine compartment. The detection system and the extinguishing system shall be in accordance with specification AIR 7-4 (55)
and CAP 479 respectively, except as stated herein and (57)
Appendix II (Deviations). (58)

3.20.1 Description and Components

3.20.1.1 Detection System

Continuous wire type fire detector circuits shall be installed in each of the three regions and connected to both a master warning light on the pilot's main instrument panel, and three corresponding compartment warning lights each combined with an extinguishing switch. All fire warning lights shall be red. On receipt of a signal from any of the detector circuits, the affected fire detection control unit shall illuminate the master warning light and the light in the appropriate extinguishing switch.

3.20.1.2 Extinguishing System

Two twelve pound, triple outlet fire extinguisher bottles shall be installed in the hydraulics compartment. The (54)
bottles shall be interconnected to provide a single charge to any two ot the protected regions, or two charges to any one region. The extinguishing agent shall be discharged through high rate discharge nozzles in the fore and aft zones of each engine compartment, and in the hydraulics bay.

3.20.1.3 Operation

On pressing the illuminated switch, in the case of an engine compartment fire warning, a time delay unit shall provide time for the circuit to automatically close the low pressure fuel valve thus cutting off the supply of fuel to the respective engine before discharging one charge of fluid into the compartment. Operation of the system for the hydraulics compartment shall be similar but without the time delay and fuel shut-off features.

A toggle switch, which shall remain dead until an extinguishing switch has been depressed, shall be installed adjacent to the extinguishing switches to provide for discharge of a second charge to the previously selected zone without further operation of the extinguishing switch.

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3.20.1.3 Operation (Cont'd)

Relays in the circuit shall ensure that if the same extinguishing switch is pressed a second time, a second charge will not be discharged into the same zone.

In the event of a crash landing, an inertia switch shall complete a circuit from the battery to automatically discharge the extinguishing agent to both engine compartments. (56)

3.20.1.4 Power Supply

Power for normal operation of the fire detection and extinguishing circuits shall be provided from the main DC supply. In case of failure of the main DC supply, power for the detector and second charge circuits shall be supplied from the emergency DC bus, and power for the fire extinguishing circuits shall be supplied from the battery bus.

3.20.1.5 Inspection and Maintenance

Test switches in the nose wheel well shall provide for detector circuit testing on the ground. Quick disconnects in the detector circuits and extinguisher lines shall provide for uncoupling these services for engine removal.

A pressure gauge shall be installed on each fire extinguisher bottle and shall be accessible for inspection. The bottles shall be removable for recharging.

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3.21 Oxygen System

A liquid oxygen system shall be installed to provide oxygen for breathing and pressure suit operation for both crew members. A compressed gas emergency system shall be installed to provide oxygen in the event of normal system failure or bail-out of the crew. The oxygen systems shall be designed to meet the requirements of ARDCM 80-1 except as stated herein and in the Deviations given in Appendix II.

3.21.1 Description and Components

3.21.1.1 Normal System

The normal oxygen supply shall be stored in a 5 litre, portable type, 300 psig, liquid oxygen converter. The converter capacity shall be sufficient to supply the required oxygen to both crew members for one maximum ferry mission. All components required for the oxygen conversion process and maintenance of system pressure shall be an integral part of the detachable converter assembly. The mounting and locking of the converter in the aircraft shall automatically put the system in an operable condition.

3.21.1.2 Emergency System

The emergency oxygen supply shall be stored in two oxygen bottles, one bottle installed on each ejection seat. Each bottle shall contain 100 litres NTP oxygen, stored at 1800 psig. This supply shall be sufficient for approximately twenty minutes of normal breathing and pressure suit operation for each crew member.

3.21.1.3 Distribution

A high altitude, automatic pressure demand, dual pressure, dual outlet oxygen regulator shall be installed on each ejection seat. The regulator performance shall conform to R.C.A.F. Specification INST 11-1. Normal system oxygen shall be supplied to the regulator through a three part composite quick disconnect (Reference paragraph 3.19.1) (70)

The individual emergency oxygen supply shall feed into the normal system through a trip valve installed on each ejection seat and may be selected manually or automatically at bail-out. A dual check valve shall be installed on each seat to block the normal oxygen supply line and to permit the emergency supply to flow to the regulator.

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3.21.1.4 Indicators

A capacitance type oxygen quantity gauge shall be installed in the pilot's cockpit to indicate the quantity of liquid oxygen contained in the converter. An oxygen pressure gauge shall be installed in the pilot's cockpit to indicate the gaseous oxygen pressure in the normal system.

A pressure gauge shall be installed adjacent to each emergency oxygen bottle, and shall be so located as to be easily observable from the normal cockpit entrance path and from the normal seated position.

3.21.1.5 Piping

All low pressure piping and fittings in the oxygen system up to the regulator shall be aluminum alloy, and all high pressure piping and fittings up to the regulator shall be stainless steel.

3.21.2 Ground Service

The liquid oxygen supply shall be replenished by the replacement of the oxygen converter with a fully charged unit. The converter shall be installed in the airplane through an access panel opening and shall automatically couple into the system supply line, overboard vent line, and quantity gauge coupling electrical leads, with the supply line quick disconnect self sealing when disengaged. The converter shall lock into the aircraft by a positive lock on the converter mounting tray.

The emergency oxygen bottles shall be rechargeable through a quick disconnect charging valve installed on each ejection seat and so located as to be readily accessible for ground service.

3.21.3 Inspection and Maintenance

Suitable provisions shall be made for the inspection, maintenance, removal and re-installation of the oxygen equipment.

3.22 Air Conditioning

The air conditioning system shall be of the engine compressor bleed type and shall be designed to maintain specified conditions of air temperature and/or pressure in the cockpits and equipment compartments, and to equipment. Conditioned air exhausted from the cockpit, equipment compartments and equipment shall be utilized to condition other compartments as indicated in the table below:

Primary Conditioned	Secondary Conditioned
- Cockpits	- Armament Pack Bay
- Nose Radar Equipment - Battery Compartment	- Nose Wheel Well (forward end)
- Oxygen Converter - Forward Fuselage Electronics Compartment	- Air Conditioning Bay
- Dorsal Electronics Compartment	
- Alternator Control/Transformer Rectifier Box	

Complete provision shall be installed to supply conditioned air for a stabilization platform of an integrated electronics system.

Ground air conditioning shall be possible by the use of a ground servicing unit which shall supply both high and low pressure air for the system through quick disconnects.

Air from the air conditioning system shall be utilized during both in-flight and ground operation to supply the requirements of the pneumatic services (Section 3.15).

The air conditioning and pressurizing system shall be de- (59)
signed to meet the requirements of R.C.A.F. Specification (60)
AIR 7-4 and ARDCM 80-1 except as stated in Appendix II (Deviations) and additionally herein.

3.22.1 In-Flight Air Conditioning

Supplies of air, cooled by successive stages of the system shall be mixed to provide conditioned air for the cockpit, compartments and equipment. The system shall be designed to automatically pressurize the cockpits above an ambient pressure altitude of 10,000 feet and shall incorporate limited manual control of cockpit temperature.

3.22.1 In-Flight Air Conditioning (Cont'd)

An air supply selection control shall be installed to permit either complete shut-down of the system, or utilization of ram air to ventilate unoccupied compartments (Reference paragraph 3.22.1.4), in the event of system malfunction or failure.

3.22.1.1 Occupied Compartments

3.22.1.1.1 Cockpits

A control system shall be installed to permit the pilot to select a cockpit temperature in the range of +40°F to +80°F. As a preventative against formation of fog in the cockpit, the lower temperature of the air admitted to the cockpit shall be automatically limited to a minimum of +55°F below 20,000 feet ambient pressure altitude. To disperse cockpit fog, it shall be possible for the pilot to select a cockpit temperature of +90°F.

Automatic cabin pressure control equipment shall be installed to provide:

(a) A cockpit pressure differential of zero up to an ambient pressure altitude of 10,000 feet.

(b) A linear increase in cockpit pressure differential to a maximum of 4.5 plus 0.5 minus 0 psi as the ambient pressure altitude increases from 10,000 to 60,000 feet.

(c) A constant cockpit pressure differential of 4.5 plus 0.5 minus 0 psi at ambient pressure altitudes exceeding 60,000 feet.

The cabin leak rate shall be less than two-thirds of the maximum quantity of air available for pressurization at the absolute ceiling of the aircraft

An electrically operated dump valve, controlled by a switch in the pilot's cockpit, and incorporating components to provide automatic outward and inward relief of cockpit pressure shall be installed. A cockpit altimeter conforming to Specification MIL-I-5099, and the air supply selection control shall be installed in the pilot's cockpit. (69

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3.22.1.2 Unoccupied Compartments

3.22.1.2.1 Primary Conditioning Distribution

A system of ducting incorporating restrictors to provide for control of air distribution shall convey conditioned air at 80°F ±5°F to all the primary conditioned equipment compartments and equipment to maintain the internal air temperature of each below +160°F. The three radar compartments shall be fitted with restricted vents to prevent the internal pressure of the compartments falling below the equivalent of 55,000 feet pressure altitude. An air valve, controlled by pressure sensing switches, shall be installed to stop the flow of conditioned air to the nose radar compartment in the event of a drop in supply pressure, from either engine compressor, below the system minimum requirements. Failure of the power output from either transformer-rectifier unit shall provide for the diversion of the flow of conditioning air to the box of the remaining operable unit.

3.22.1.2.2 Secondary Conditioning Distribution

The exhaust air from the cockpits shall be ducted to the armament pack bay to maintain the internal air temperature between 0 and +130°F.

Exhaust air from the nose radar and battery compartments shall be utilized to maintain a cooling air supply, at a temperature below 160°F, for the equipment in the forward end of the nose wheel well.

The air vented from the forward fuselage electronics compartment and from the oxygen converter shall be utilized to maintain the air in the air conditioning equipment bay below 250°F.

3.22.1.3 Cooling Sub-System

3.22.1.3.1 Air-to-Air Heat Exchanger

Air for conditioning shall be bled from the two upper bleed ports of each engine through a pressure reducing and check valve set at 85 psi, and ducted to a heat exchanger.

Ducting shall be installed to convey cooling air to the heat exchanger from two sources as follows:

(1) Ramp air, drawn from each engine air intake ramp air bleed by the turbine driven fan.

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3.22.1.3.1 Air-to-Air Heat Exchanger (Cont'd)

(2) Ram air, provided by the boundary layer air bleed between each engine air intake and the fuselage wall.

Both cooling air supplies shall be vented to atmosphere. The air conditioning output from the heat exchanger shall be ducted to the air cooling water evaporator.

3.22.1.3.2 Air Cooling Water Evaporator

The air cooling water evaporator shall have a nominal capacity of 125 pounds of water and shall be designed to withstand freezing and thawing of its contents under all conditions of operation. Steam generated in the evaporator shall be vented to atmosphere.

The main air conditioning output of the cooling evaporator shall be ducted to the cooling expansion turbine. The remaining output shall be utilized for quantity and temperature control.

3.22.1.3.3 Air Cooling Expansion Turbine

The turbine shall cool the conditioned air by expansion (81)
and shall power the fan to draw ramp air through the heat exchanger (Reference paragraph 3.22.1.3.1).

3.22.1.4 Ram Air Ventilation

Selection of ram air shall permit air from the ram air (62)
duct (Reference paragraph 3.22.1.3.1) to be utilized for ventilation of all the conditioned compartments excepting the cockpits, nose radar compartment and armament pack.

3.22.2 Ground Air Conditioning

Two automatic disconnect ground air conditioning couplings, (61)
each embodying a check valve, shall be installed to utilize ground service air supplies as follows:

(1) Air at a pressure of 3.5 psi and at a temperature within the range of 55 to 80°F. Ducting shall be installed to permit the air supply to enter the aircraft system at the output side of the expansion turbine, and be utilized to condition the cockpit, equipment compartments, and equipment.

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3.22.2 Ground Air Conditioning (Cont'd)

(2) Air at a pressure of 45 psi and at a temperature of 350°F or less. Ducting shall be installed to permit the air supply to enter the aircraft system upstream of the heat exchanger, and be utilized to provide temperature control air for the system, and air for the ground requirements of the pneumatic system (Reference Section 3.15).

3.22.3 Inspection and Maintenance

Provision for connecting cockpit leakage test equipment shall be installed. Access doors and detachable panels shall be installed to facilitate inspection and maintenance.

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3.23 Anti-Icing and De-Icing Systems

Fully automatic anti-icing and de-icing systems shall be provided for the following areas:

Engine Air Intakes
Engine and Accessories
Cockpit Transparencies
Pressure Heads
Radome

Except as stated in the Deviations and as additionally set forth herein, the systems shall be designed in accordance with the requirements of R.C.A.F. Specification AIR 7-4.

3.23.1 Propeller De-Icing

Not applicable.

3.23.2 Carburettor Anti-Icing and De-Icing

Not applicable.

3.23.3 Air Intakes

3.23.3.1 Engine Air Intakes

The outer surface of the shock ramps and the leading edges and inner surfaces of the engine air intakes shall be protected from excessive ice accretion by electro-thermal de-icing boots. The de-icing boots covering the leading edge of each ramp shall include a protective covering of stainless steel and a suitably perforated boot of the same type shall be used to protect the ramp air bleed area. The boots shall incorporate parting strips to prevent ice from forming an unbroken cap which would prohibit shedding. Power for heating the boots shall be supplied by the 120 volt AC system.

The de-icing cycle shall be automatically controlled by an icing detector, installed at the top of the right hand air intake, in conjunction with a de-icing controller which actuates separate shedding distributors for the left hand and right hand intakes. These components shall operate from the 28 volt DC supply.

During the cyclic period the parting strips shall dissipate 20 watts per square inch continuously, and the shedding areas 12 watts per square inch when they are energized. The boots shall be protected from overheating by thermostats and temperature control relays which override the de-icing controller signals.

3.23.3.1 Engine Air Intakes (Cont'd)

To prevent operation of the de-icing boots during engine runup the DC circuit on which the control equipment operates shall include a scissors switch mounted on the main landing gear.

3.23.3.2 Engine and Accessories

Each engine, as supplied by the engine manufacturer, shall include an integral hot air return anti-icing system, as defined in Pratt and Whitney Engine Specification PWA 2611.

Automatic selection of anti-icing air flow for both engines shall normally be controlled by the icing detector on the right hand engine air intake (Reference paragraph 3.23.3.1). An icing detector shall be installed in the left intake to protect the left hand engine during ground operation, when the right hand engine is not running.

3.23.4 Cockpit Transparencies

Anti-icing and anti-misting of the windshield and canopy windows in the forward cockpit only shall be accomplished (63
by electrically conductive transparent heating elements. These elements, dissipating approximately 5 watts per square inch, shall be applied to the inner surface of the outer ply of each panel during manufacture. Temperature sensing units embedded in the vinyl interlayer adjacent to the heating elements shall permit temperature control for each circuit.

The temperature control shall be automatic, and in order to overcome thermal lag, power shall be applied to the circuit at all times when the aircraft is operating or in a state of immediate readiness.

3.23.5 Main Plane, Stabilizer and Fins

Not applicable.

3.23.6 Antennae Masts

Not applicable.

3.23.7 Pressure Heads

An air data sensing boom, including pitot pressure heads, alpha and beta angle sensors shall be protected from the formation of ice by means of built-in electrical heaters.

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3.23.7 Pressure Heads (Cont'd)

The circuit shall be of the constant heat type protected by an overheat thermostat.

3.23.8 Undercarriage

Not applicable.

3.23.9 Panels and Doors

Not applicable.

3.23.10 Vents

Not applicable.

3.23.11 Photographic Installations

Not applicable.

3.23.12 Radome

A freezing point depressant fluid (glycol and water solution), for application to the radome as a protection against the formation of ice, shall be stored in a 2.75 Imperial Gallon pressurized tank. The solution shall be sprayed from a distributor mounted near the base of the nose boom and pneumatic system air, (Reference paragraph 3.15.1.1.3) shall provide power to pressurize the tank and operate the distributor. An ice detector, operating from the 27.5 volt DC supply and located on the underside of the radome, shall control operation of the system.

3.23.13 Inspection and Maintenance

Provision shall be made for inspection and maintenance of de-icing equipment. A ground test switch, to override the landing gear scissors switch and permit testing of the de-icing boots, shall be installed on the panel adjacent to the left hand speed brake.

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3.24 Photographic Equipment

Not applicable.

3.25 Auxiliary Gear

3.25.1 Towing Provisions

Towing provisions shall conform to the requirements of Specification MIL-T-7935. Lugs shall be provided at the nose wheel pivot for attachment of a tow bar. The turning angle will be limited to 55° either side of neutral permitting the aircraft to be turned in a 21 foot radius. Provision shall be made for interconnection between the AN/AIC-10 interphone system and the towing vehicle with connection made at the same point as for ground intercommunication (Reference paragraph 3.16.10.1). A microswitch on the tow bar shall initiate a warning through the intercom to the tractor driver when the maximum turning angle is approached. During towing, power to operate the interphone system shall be supplied by the towing vehicle.

Towing lugs shall be provided on each main landing gear unit for forward or rearward towing of the airplane by means of a towing bridle.

3.25.2 Jacking Provisions

Jacking provisions, and the design of jack pads shall conform to the requirements of MIL-J-8711, except in the case of the nose landing gear.

Provision shall be made for jacking the complete airplane at three points, one on the airplane center line aft of the nose landing gear, and one inboard of each outer wing root. A removable jack pad shall be provided for each jacking point. (65)

Each main landing gear unit shall incorporate an integral jack pad. The nose landing gear shall incorporate provisions for jacking, using a special bar with a jack pad conforming to MIL-J-8711. (64)

3.25.3 Mooring Provisions

Provision shall be made for the attachment of mooring fittings to the main landing gear towing lugs and a mooring lug on the nose landing gear leg. (66) (67)

3.25.4 Hoisting Provisions

Provision shall be made for hoisting the entire aircraft from three points, one on the aircraft center line at the nose center fuselage joint, and one on each inner wing panel adjacent to the outer wing root.

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3.25.5 Leveling

Provision shall be made in the nose wheel well for the attachment of a special fixture for use in leveling the aircraft. The fixture shall indicate a level attitude of the lateral axis and 4° 4' nose up attitude of the longitudinal axis. (68)

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4. Tests

4.1 Ground Tests

Functional ground test shall be conducted under a program established by the Company to prove ground functioning of the aircraft systems and installed equipment.

Structural integrity tests shall be demonstrated under a static test program based on the requirements of Specification MIL-S-5710 and as agreed upon by the R.C.A.F. and the Company (110) and as set forth in Avro Reports ___________

4.2 Flight Tests

Functional flight tests shall be conducted under a program established by the Company to prove in-flight functioning of the aircraft systems and installed equipment.

Structural flight tests shall be conducted under a program based on the requirements of AIR 7-4 (Issue 3) and MIL-S-5711 (110) as agreed upon by the R.C.A.F. and the Company, and as set forth in Avro Report ___________

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5. Preparation for Delivery

To be determined.

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6. Notes

6.1 Explanatory Information

Not applicable.

6.2 Definitions

6.2.1 Provisions

6.2.1.1 Complete Provision

"Complete provision for" a specific item of equipment, or assembly or installation, shall mean that all supports, brackets, tubes and fittings, electrical wiring, hydraulic lines etc. have been installed and adequate weight and space allocated so that the equipment can be installed without alteration to the specified equipment or the aircraft, and that no additional parts are required for the installation other than the item itself. Standard stock items such as nuts, bolts, cotter pins, etc. need not be furnished.

6.2.1.2 Structural Provision

"Structural provision for" a specific installation shall mean that the primary structure shall be structurally adequate for the installation, but that brackets, bolt holes, electrical wiring, hydraulic lines etc. will not be required. Structural provisions also include weight of the equipment involved as an element of alternate weight.

6.2.1.3 Space Provision

"Space provision for" a specific installation shall mean that space only shall be allocated for the installation, and that brackets, bolt holes, electric wiring, hydraulic lines etc. will not be required. Space provision does not imply that adequate attaching structure is provided unless otherwise stated.

6.2.2 Statements

6.2.2.1 Deviation

A deviation is the difference between a requirement of the R.C.A.F. Type Specification (and specifications incident thereto), and the airplane design.

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6.2.2.2 Interchangeability

Interchangeability means the capability of replacing parts, components or accessories by others made to the same drawings or specifications, without further work being done on them or on the parts to which they are assembled.

6.2.2.3 Replaceable

Replaceable parts, components, and accessories are those that can be replaced by others made to the same drawings or specifications, but that require minor operations to be performed on them or on the parts to which they are assembled to effect replacement.

6.2.3 Performance

6.2.3.1 Combat Load Factor

Combat load factor is the maximum load factor that can be sustained in a steady turn without loss of speed or altitude.

6.2.3.2 Combat Speed

Combat speed is the constant speed at which the aircraft is flying during the turn in which the combat load factor is developed.

6.2.3.3 Combat Altitude

Combat altitude is the constant altitude at which the aircraft is flying during the turn in which the combat load factor is developed.

6.2.3.4 Combat Climb and Acceleration Time

Combat climb and acceleration time is the elapsed time taken to reach combat speed and combat altitude from the time the aircraft becomes airborne during take-off at normal gross weight under sea level conditions.

6.2.3.5 Combat Ceiling

Combat ceiling is the altitude where the sustained rate of climb has fallen to 500 feet per minute.

6.2.4 Weights

6.2.4.1 Combat Weight

The combat weight shall be the weight of the aircraft fully loaded with 50 percent of the fuel required for the combat mission.

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6.2.4.2 Normal Gross Weight

The normal gross weight and the normal weight for take-off shall be the weight of the aircraft fully loaded with primary armament and fuel for the combat mission.

6.2.4.3 Gross Weight for Stress Analysis

The gross weight for stress analysis (stressing weight) shall not be less than the normal gross weight less fifty percent of the combat mission fuel.

6.2.4.4 Maximum Gross Weight

The maximum gross weight and the maximum weight for take-off shall be the weight of the aircraft fully loaded with primary armament, fuell internal fuel, and external fuel for the overload range mission.

6.2.4.5 Maximum Landing Gross Weight

The maximum landing gross weight shall not be less than the maximum gross weight less; assist take-off fuel, droppable fuel and tanks, dumpable fuel and any other items normally expended during or immediately after take-off (except bombs, rockets, missiles, and ammunition shall be retained).

6.2.4.6 Normal Design Landing Gross Weight

The normal design landing gross weight shall not be less than the applicable take-off weight less; 75% of fuel (internal and external) carried in the basic mission, oil expended consistent with fuel expended, any external fuel tanks which must be dropped by requirements of the mission, any other item which must be expended by the requirements of the mission, and bombs, rockets, missiles, and ammunition.

6.2.4.7 Weight Empty

The weight empty shall be the weight of airframe, power plant, and fixed equipment.

6.2.4.8 Basic Weight

The weight of an aircraft with fixed and removable equipment installed for the purpose of performing a specific role. The term "Basic Weight" shall be qualified as to role when referred to an aircraft in which various items of removable equipment may be installed for different roles. It includes airframe, power plant, accessories,

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6.2.4.8 Basic Weight (Cont'd)

trapped fuel and oil, and non-expendable fluid systems (hydraulic, coolant) filled to capacity, but without expendable items.

6.2.4.9 Operational Load

Operational load includes crew, passengers, parachutes, baggage, cargo, personal safety equipment, expendable items (fuel, oil, de-icing fluid, water injection fluid, catering provisions, ammunition, rockets and bombs), and residual fuel.

6.2.5 Equipment and Fluids

6.2.5.1 Fixed Equipment

Equipment installed in an aircraft and not intended to be removed for any specific role.

6.2.5.2 Removable Equipment

Readily removable equipment installed in an aircraft for the purpose of performing a specific role.

6.2.5.3 Trapped Fuel and Oil

The fuel and oil remaining in the aircraft fuel and oil systems after they have been filled and then drained by means of the tank drains, with the aircraft in the normal ground position.

6.2.5.4 Residual Fuel

Residual fuel is fuel, in excess of trapped fuel, that cannot be consumed in flight, but that can be drained by means of the tank drains (i.e. does not include trapped fuel, and is not included in Basic Weight).

6.2.6 Engine Definitions

6.2.6.1 Maximum Rated Thrust

Maximum rated thrust is the maximum thrust which the contractor specifies the engine will deliver at standard sea level static conditions for a duration of 5 minutes. In flight, maximum thrust will be the thrust developed with the power lever in the "Maximum" position. If maximum thrust is greater than military thrust, its permissible duration in flight shall be 15 minutes.

6.2.6.2 Military Rated Thrust

Military rated thrust is the maximum thrust which the contractor specifies the engine will deliver without augmentation at standard sea level static conditions for a duration of 30 minutes. In flight, military thrust will be the thrust developed with the power lever in the "military" position.

6.2.6.3 Normal Rated Thrust

Normal rated thrust is the maximum thrust which the contractor specifies the engine will deliver at standard sea level static conditions for continuous operation. In flight, normal thrust will be the thrust developed with the power lever in the "normal" position.

6.2.6.4 Idling Thrust

The idling thrust is the minimum developed thrust at which the contractor specifies the engine may be operated at standard sea level static conditions. In flight, idling thrust will be the thrust developed with the power lever in the "idle" position.

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