SECRET

MODEL SPECIFICATION

FOR

CF-105 MARK 1

AIRCRAFT

AVRO AIRCRAFT LIMITED

AVRO

ANALYZED

MODEL SPECIFICATION
FOR
SUPERSONIC AIRCRAFT
TYPE CF105 MARK 1

Prepared By

Model Specification Group
Standards and Specifications Section

Recommended for
Technical Approval

G. Hake
CF105 Project Designer

R.N. Lindley
Chief Design Engineer

Recommended for
Contractual Approval

J.R. Douglas
Contracts Administrator

Technical Approval

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Vice President,
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Contractual Approval

J.A. Morley
Vice President,
Sales and Service

AAMS-105/1
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AVRO AIRCRAFT LIMITED

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MODEL SPECIFICATION AMENDMENT NOTICE	TO RCAF DATE 26 SEP 57
SPECIFICATION TITLE Model Specification for Arrow 1 Supersonic Aircraft.	COPY NO. 48 SPEC. NO. AAMS-105/1 ISSUE NO. 1

Amended. 30.10.57 EP.

Attached is a copy of Amendment No. 1 together with amended pages 1, 1a, 4 and 192, for inclusion in your copy of the above noted specification.

This amendment is to be filed in Appendix IV of the specification. Pages 1, 1a, and 4, marked "REV. SEP 57", replace the existing pages, which are obsolete. Please destroy the obsolete pages in accordance with existing security regulations.

The RCAF has stated that this specification is now approved with the incorporation of the subject amendment.

R.B. Cairns
Model Specification Section

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APPENDICES

SECTION I

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 in this specification. The applicable paragraphs of this specification shall, in each case, state the extent to which the aircraft design complies with the following specifications. Failure to list non-compliance with the requirements of the specifications listed in this paragraph, which, by a reasonable engineering interpretation should apply to this specification, shall indicate the Contractor's intention to meet all such requirements even though no specific mention is made of the requirement in this specification. Where the Contractor does not intend to comply with such requirements a deviation shall be raised. 1

Contractor specifications and publications shall be approved by the RCAF prior to forming a part of this specification.

At the discretion of the Company subsequently dated RCAF approved issues may be used.

AIR 7-4 Issue 3	R.C.A.F. Specification for Supersonic All-Weather Interceptor Aircraft Type CF105
CAP 479	Manual of Aircraft Design Requirements for the Royal Canadian Air Force
ARDCM 80-1	Handbook of Instructions for Aircraft Designers
PWA Spec. No. 2605	Pratt and Whitney Aircraft, JT4A-25 Turbo-Jet Engine Specification
PWA Spec. No. 2611	Pratt and Whitney Aircraft, JT4A-23 Turbo-Jet Engine Specification
RCAF Spec. C-28-96	Luminescent Material, Fluorescent - Radioactive
EL-5040-1	Aircraft Doppler Radar System
INST 11-1	Regulator, Oxygen
MIL-B-5087A	Bonding, Electrical (for Aircraft)

1.1 Referenced Specifications (Cont'd)

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	Design, Installation and Tests of Aircraft Hydraulic Systems
MIL-F-5572A	Fuel, Aircraft Reciprocating Engine
MIL-O-5606 (2)	Oil, Hydraulic, Aircraft, Petroleum Base
MIL-F-5616	Fuel, Aircraft Engine, Grade JP-1
MIL-F-5624 (c)	Fuel, Aircraft Turbine and Jet Engine, Grades JP-3 and JP-4

cont'd on page 2

1.1 Referenced Specifications (Cont'd)

MIL-S-5700	Stress Analysis Criteria
MIL-S-5702	Structural Criteria, Basic Flight Criteria
MIL-S-5710	Structural Criteria, Structural Tests, Static
MIL-S-5711	Structural Criteria, Structural Test, Flight
MIL-N-5877A	Nozzle, Pressure Fuel Servicing, Locking Type D-1
MIL-I-5997	Instruments and Instrument Panels, Aircraft Installation Of
MIL-I-6051	Interference Limits and Methods of Measurement, Aircraft Radio and Electronic Installations
MIL-I-6181	Interference Limits, Test and Design Requirements, Aircraft Electrical and Electronic Equipment
MIL-L-6503A	Lighting Equipment, Aircraft, General Specification for Installation Of
MIL-T-6736A	Tubing, Chrome-Molybdenum, 4130 Steel, Seamless, Aircraft
MIL-C-6818A	Clamp, Mounting, Aircraft Instruments
MIL-E-7080 (1)	Electrical Equipment, Installation of Aircraft, General Specification
MIL-T-7081A	Tubing, Aluminum Alloy (615) Seamless, Round, Hydraulic System
MIL-E-7563 (1)	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	Electric Power, Aircraft, Characteristics Of

1.1 Referenced Specifications (Cont'd)

MIL-M-7911	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	Installation and Test of Electronic Equipment in Aircraft, General Specification for
MIL-J-8711	Jack Pads, Aircraft, Design and Installation
MIL-F-8785	Flying Qualities of Piloted Airplanes
USAF Spec. 1817	Flutter, Divergence and Reversal of Aircraft, Prevention of
ANC-2	Ground Loads Criteria (Oct. 1952)
US Radium R410AB	
98-24105-5	Marking of Airplanes and Airplane Parts
CGSB 3-GP-22	Aviation Turbine Fuel - Type II
CGSB 3-GP-23	Fuel, Aviation Turbine, Type I
CGSB 3-GP-25C	Aviation Fuel
CGSB 3-GP-26A	Oil, Hydraulic, Petroleum Base
AN-L-1A	Luminescent Material, Fluorescent
AN 3114	Receptacle, External Power
CS-D-2	Protective Treatment Schedule Landplanes
CS-T-135	Tube Assembly, Hydraulic and Pneumatic, High Pressure
CS-T-148	Tube Assembly, Oxygen System
M-7-6	Tubing, Stainless Steel, Round, Close Tolerances, Hydraulic System

1.1 Referenced Specifications (Cont'd)

Avrocan E-266	Environmental Testing, Aeronautical and Associated Equipment, General Specification for CF105 Aircraft
Avro Report QC-E-9	Interchangeability - Working Lists
Avro Report P/AERO DATA/89	Detailed Analysis of Flying Qualities of CF-105
Avro Report SR-4	Compliance with ABC Standards
Dowcan 200 (Issue 1)	Silicone Based Fluid (26 Oct. 1954)

1.2 Precedence of Requirements

From the date of RCAF approval of this Model Specification the requirements of this Specification shall possess the first order of precedence. Otherwise, in the event of conflict between the requirements of the specifications, publications, and documents referenced in this Model Specification, the order of precedence for compliance shall be as follows:

(a) AIR 7-4 - R.C.A.F. Specification for Supersonic All-Weather Interceptor Aircraft Type CF-105

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

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

(d) The remaining specifications referenced in this specification.

1.3 Specification Amendments

Any alteration to this Model Specification, whether or not such alteration results in a physical change to the aircraft, shall be submitted by the Company to the R.C.A.F. in the form of a "Specification Amendment".

1.4 Deviations

Deviations are set forth in Appendix II to this document and are indicated throughout the text by the appropriate deviation number encircled in the left-hand margin. A definition of "Deviation" appears in paragraph 6.2. From the date of approval by the RCAF of the Model Specification, required additional deviations from the requirements of the specifications listed in paragraph 1.1 shall be submitted in the form of Specification Amendments. 1

SECTION 2

SCOPE

2.1 Aircraft

This Model Specification describes the CF-105 aircraft, designated CF-105 Mark 1, which shall be interim aircraft designed to the requirements of R.C.A.F. Specification AIR 7-4 Issue 3 and such additional requirements as may be specified and agreed upon between the R.C.A.F. and the Company.

The iterim aircraft shall be predicated on the development of an operational all-weather aircraft defined in AIR 7-4 Issue 3.

The first CF-105 Mark 1 aircraft, Serial Number 25201, shall be defined by this Specification, together with applicable Specification Amendments issued in accordance with paragraph 1.3 of this Specification. The design detail documents of aircraft Serial Number 25201 are identified in the effectivity schedule of the Master Record Index which forms Appendix IIIA to this Specification.

The second and subsequent CF-105 Mark 1 aircraft shall be defined by this Specification, together with applicable amendments issued in accordance with paragraph 1.3 of this Specification. The design detail documents of the second and subsequent CF-105 Mark 1 aircraft are identified in the effectivity schedule of the Master Record Index.

2.1.1 This specification describes the following aircraft:

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

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 - CF-105 Mark I

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 or J75 Model JT4A-25

2.1.1.10 Engine Specification number - PWA 2611 or PWA 2605

2.2 Role

The role of the aircraft shall be that of a flight test vehicle leading to the production of high altitude, all-weather, night and day supersonic interceptor aircraft as defined by R.C.A.F. Specification AIR 7-4.

2.3 Crew

The crew shall normally consist of a pilot and an observer.

SECTION 3

REQUIREMENTS

3.1 Characteristics

3.1.1 Three-View Drawing

See Figure 1 Page 9.

3.1.2 Interior Arrangement Drawing

See Figures 2 and 3 Pages 10 and 11.

3.1.3 Performance

The performance shall be estimated assuming:

(a) The aircraft configuration as described by this specification with all access panels, doors, and canopy in the closed position.

49 (b) I.C.A.O. Standard Atmosphere conditions except where otherwise specified.

(c) Engine performance in accordance with the Engine Specification PWA 2605 (J75 model JT4A-25).

For current performance status refer to the latest issue of CF-105 periodic performance report.

3.1.3.1 Tabulated Performance

	Estimated
Combat Load Factor at a speed of Mach 1.5, an altitude of 50,000 feet, and a half fuel weight of 54,705 lb.	1.15
Maximum Level Speed at 50,000 feet and a half fuel weight of 54,705 lb.	936 Kts.
Combat Ceiling at a half fuel weight of 54,705 lb.	52,000 ft.
With aircraft at Maximum Gross Weight (64,626 lb.), and positioned at end of runway; elapsed time from pushing first button to start first engine until aircraft becomes airborne:	0.77 min.

3.1.3.1 Tabulated Performance (Cont'd)

	Estimated
Elapsed time to reach a level flight speed of Mach 1.5 and an altitude of 50,000 feet from the time aircraft becomes airborne during take-off at Maximum Gross Weight (64,626 lb.) under sea level conditions:	5.77 min.
Take-off distance in still air at Maximum Gross Weight (64,626 lb.) at sea level, and ambient temperature of 38°C, to clear 50 feet obstacle (Maximum Thrust with afterburning):	6,000 ft.
(94) Landing distance from 50 foot obstacle in still air, at a maximum landing weight of 55,000 lb., at sea level (drag parachute operative after touchdown):	5,540 ft.
Touchdown Speed:	149 Kts.

2097-105-1

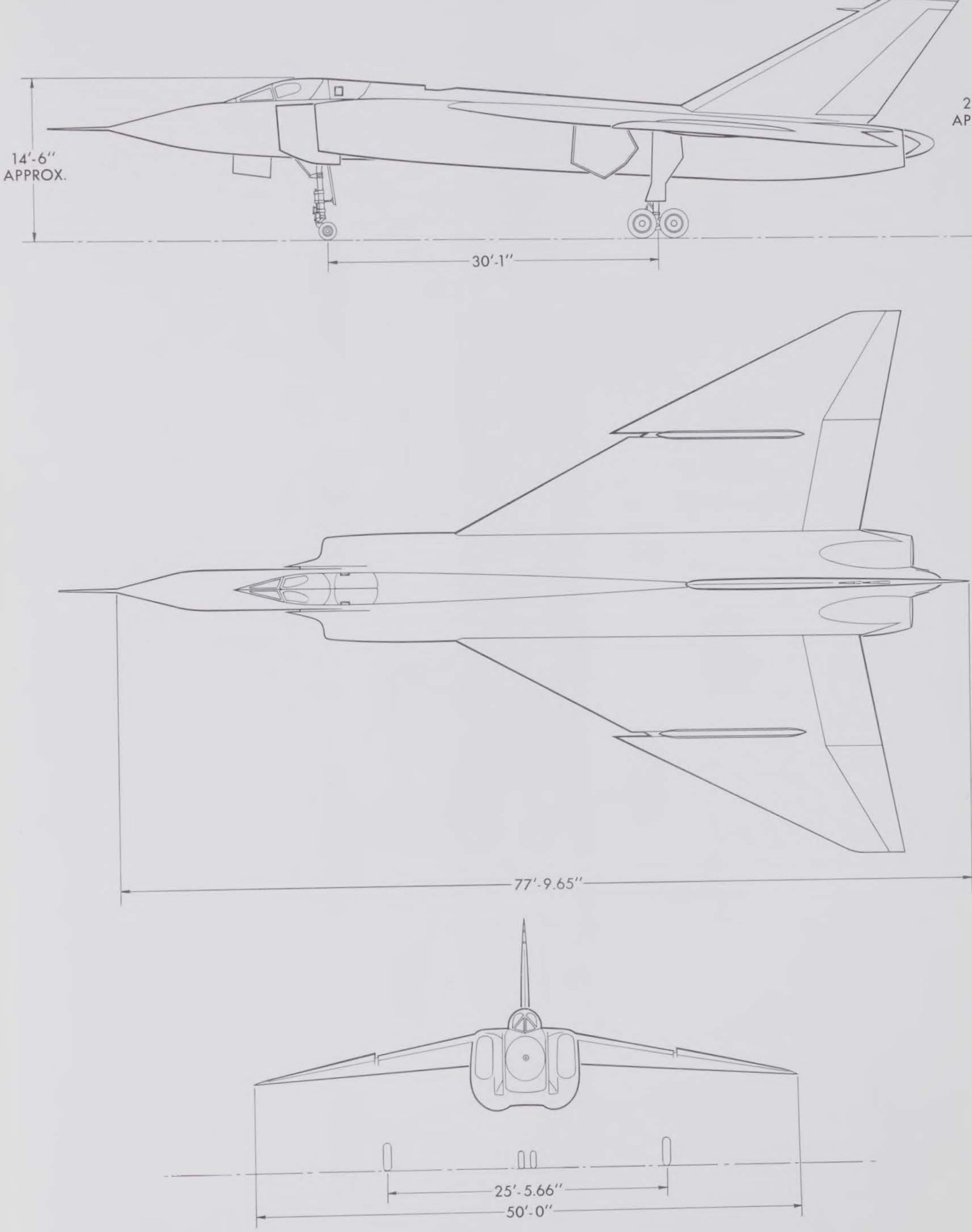

FIG. 1

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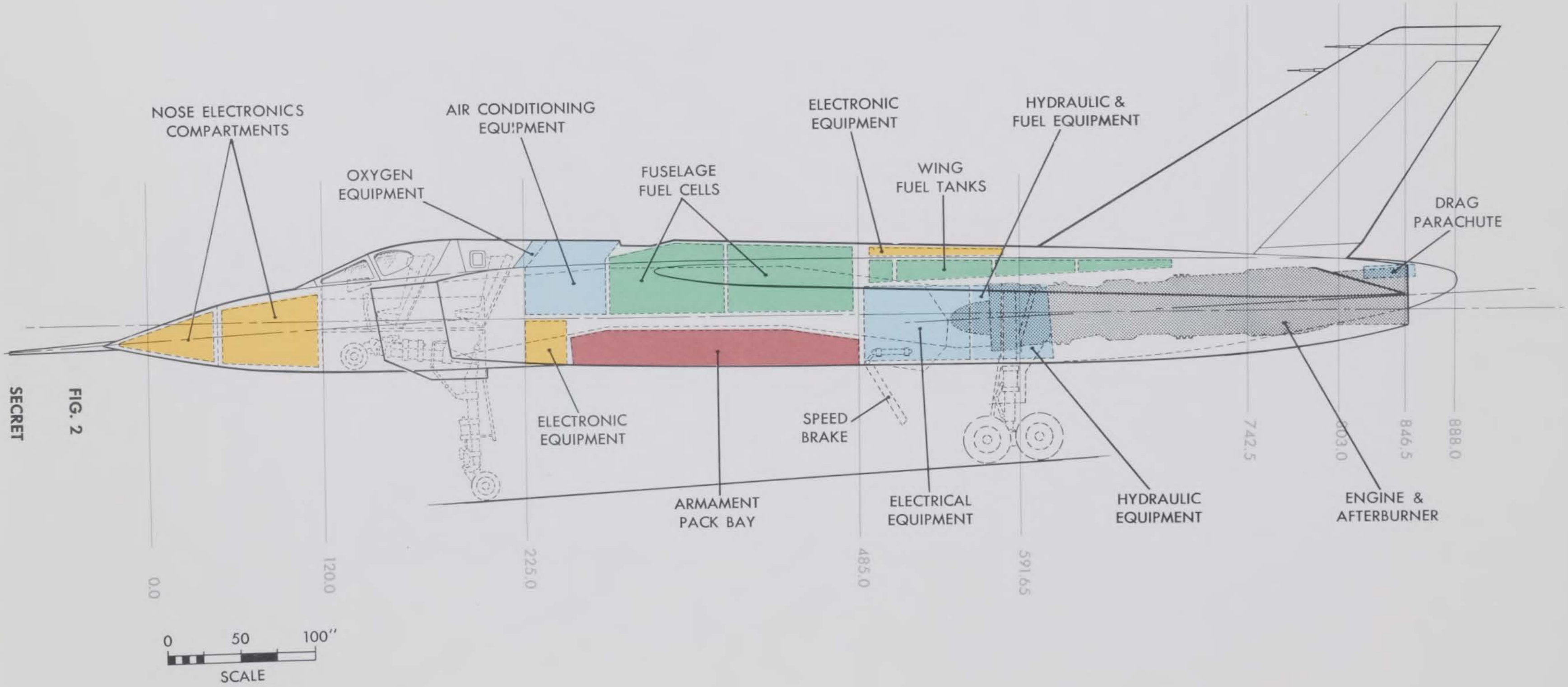


FIG. 2

3.1.4 Performance Curves

3.1.4.1 Stalling Speed vs Weight - Figure 4 Page 13

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

3.1.4.2.1 Speed vs Altitude - Figure 5 Page 14

3.1.4.2.2 Rate of Climb vs Altitude - Figure 6 Page 15

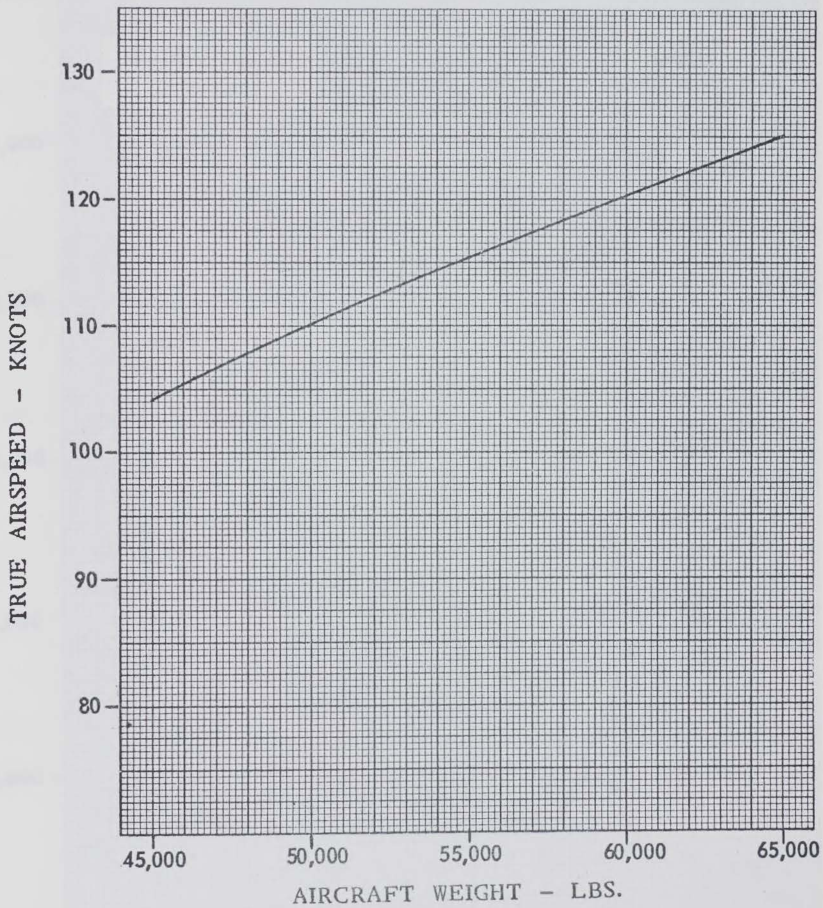
3.1.4.2.3 Time to Height vs Altitude - Figure 7 Page 16

3.1.4.3 Maneuverability - Figure 8 Page 17

3.1.4.4 Take-Off Distance vs Weight - Figure 9 Page 18

3.1.4.5 Landing Distance vs Weight - Figure 10 Page 19

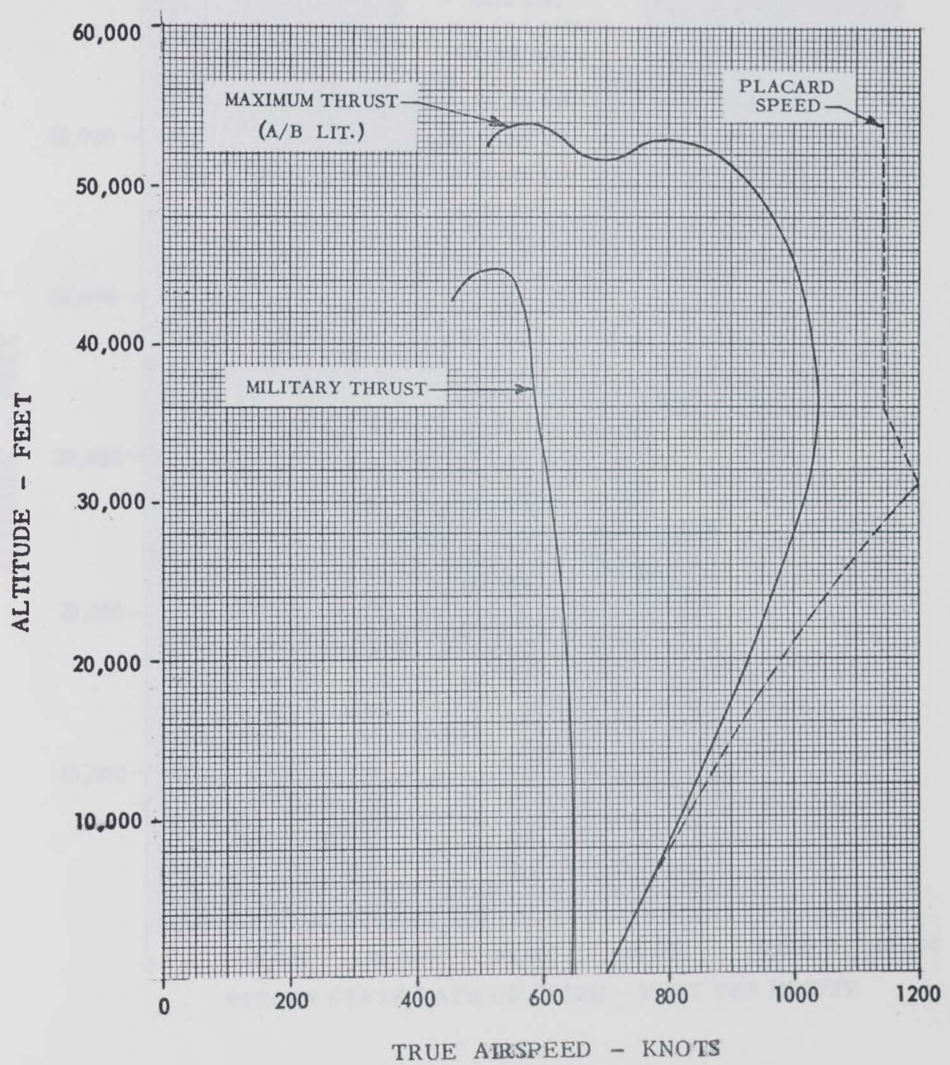
CF105 PERFORMANCE
STALLING SPEED
- POWER OFF -



CF105 PERFORMANCE
TRUE AIRSPEED

LEVEL FLIGHT

HALF FUEL WEIGHT – 54,705 LB.



CF105 PERFORMANCE
RATE OF CLIMB

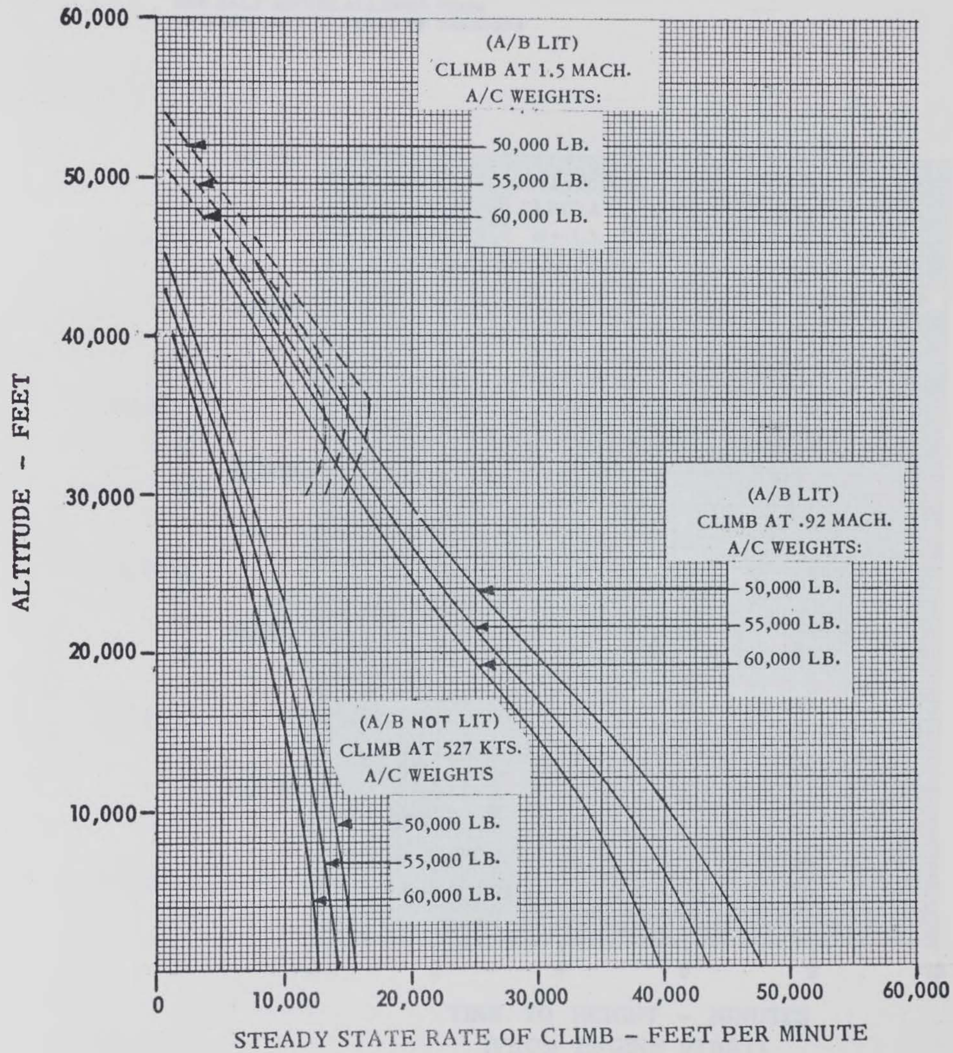


FIGURE 6

CF105 PERFORMANCE
TIME TO HEIGHT

STARTING WEIGHT = 64,626 LB.
ONE HALF MINUTE ALLOWED FROM
ENGINE START TO START OF TAKE-OFF

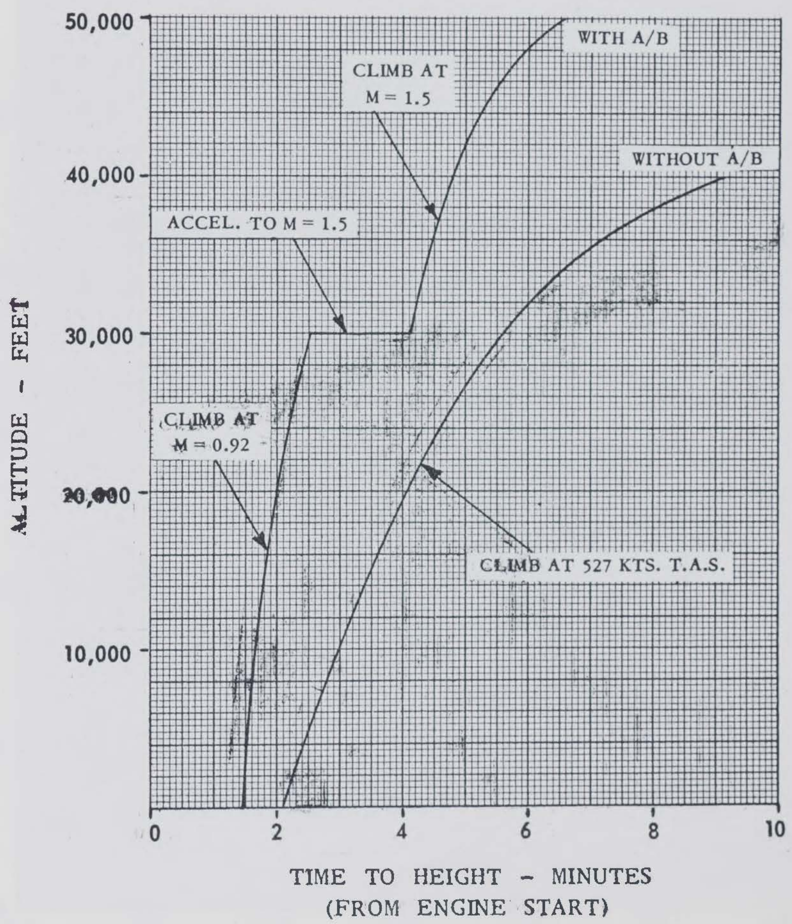
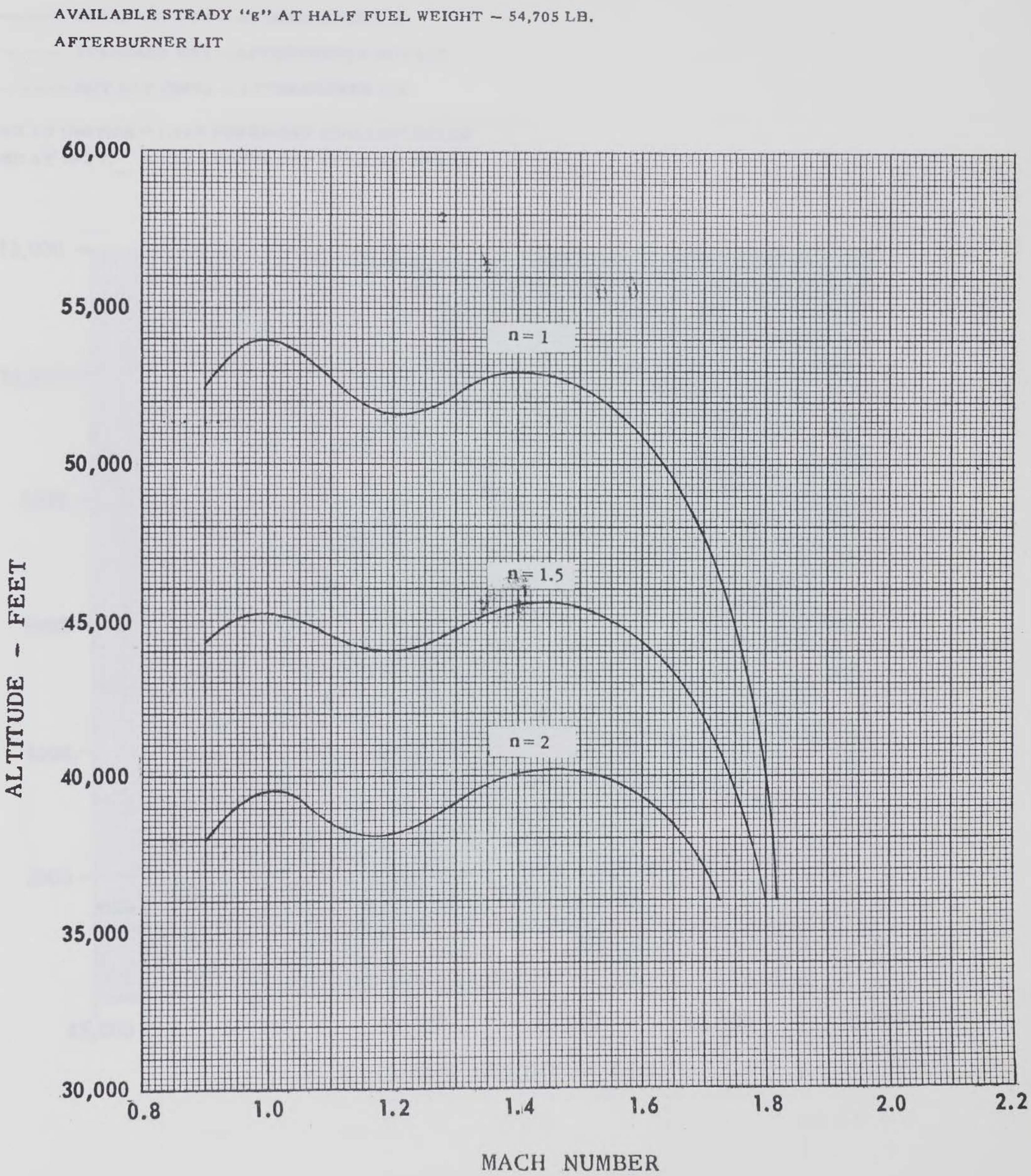


FIGURE 7

CF105 PERFORMANCE
ALTITUDE MANEUVERABILITY



CF105 PERFORMANCE
TAKEOFF DISTANCE

SEA LEVEL

——— STANDARD DAY – AFTERBURNER LIT

– – – – – STANDARD DAY – AFTERBURNER NOT LIT

——– – —— HOT DAY (38°C) – AFTERBURNER LIT

SPEED AT UNSTICK = 1.24 × POWER-OFF STALLING SPEED

SPEED AT 50 FT. = 1.58 × POWER-OFF STALLING SPEED

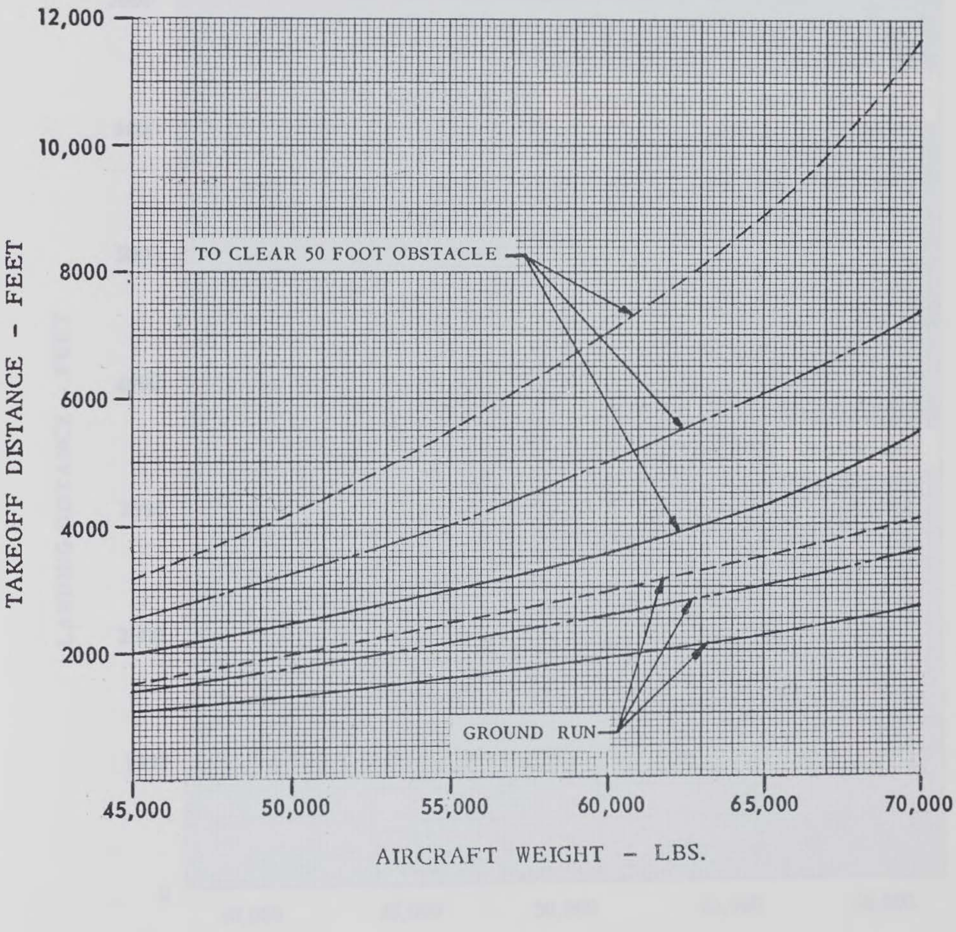


FIGURE 9

CF105 PERFORMANCE
LANDING DISTANCE

SEA LEVEL
ICAO STANDARD DAY

24 FOOT DIAMETER BRAKE PARACHUTE
ASSUMED FULLY EFFECTIVE ONE SECOND
AFTER TOUCHDOWN

APPROACH SPEED = 1.49 × POWER-OFF STALLING SPEED

TOUCHDOWN SPEED = 1.29 × POWER-OFF STALLING SPEED

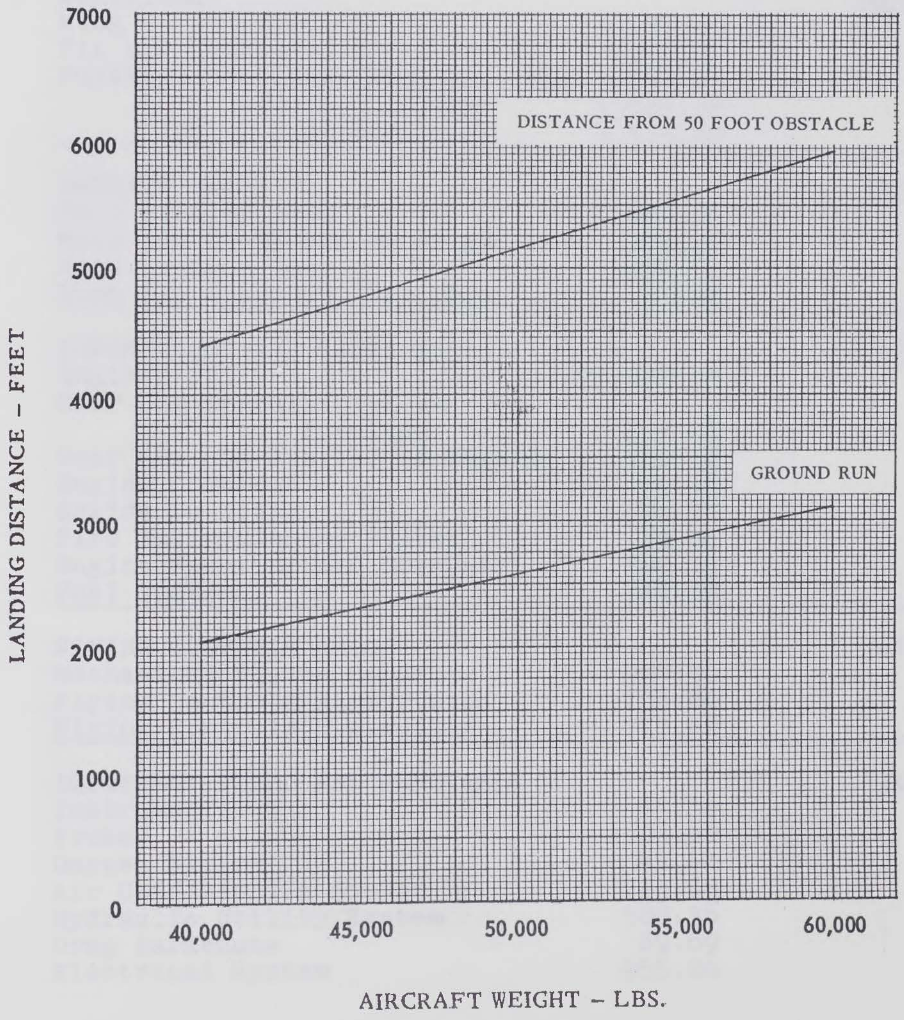


FIGURE 10

3.1.5 Weight and Balance

The estimated weight for the first aircraft is as follows and includes test instrumentation and additional fire protection not described in this specification. (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		18,327
Wing	9,989.07	
Fin and Rudder	999.96	
Fuselage Structure Fwd. STA. 255	2,493.41	
STA 255 to 485	1,664.90	
Aft STA. 485	3,179.60	
LANDING GEAR		2,612
Main Landing Gear	1,959.62	
Main L/G Doors and Fairings	294.36	
Nose Landing Gear	333.81	
Nose L/G Doors and Fairings	24.54	
POWER PLANT AND SERVICES		14,169
Engines J75	12,672.03	
Gear Box Installation on Fuselage	237.06	
Gear Box and Starter on Engine	150.18	
Engine Controls	29.19	
Engine De-Icing	70.37	
Fire Extinguishing System	70.52	
Engine Mounting and Brackets	189.19	
Fuel System	749.99	
FLYING CONTROLS GROUP		1,678
Mechanical Flying Controls	904.40	
Flying Controls Electronics	108.00	
Flying Controls Hydraulics	773.42	
EQUIPMENT FIXED AND REMOVABLE		7,033
Instruments	53.30	
Probe	23.00	
Oxygen System	43.44	
Air Conditioning System	712.69	
Hydraulic Utility System	588.36	
Drag Parachute	69.69	
Electrical System	955.20	

3.1.5.1 Basic Weight (Cont'd)

DESCRIPTION	WEIGHT LB.	
EQUIPMENT FIXED AND REMOVABLE (Cont'd)		
Low Pressure Pneumatics	39.01	
Oil and Hydraulic Fluid Cooling	22.00	
Surface Finish	100.00	
Intake De-Icing	85.84	
Interim Radio and Radar	756.70	
Canopy Actuation	54.41	
Cabin Consoles	20.65	
Radar Door Actuation	10.00	
Radome Anti-Icing	23.46	
Cabin Insulation	11.91	
Cockpit Pressure Sealing	5.00	
Ejector Seats	186.00	
Instrument Pack Structure	670.61	
Instrumentation	2447.00	
Additional Fire Protection	155.00	
BASIC WEIGHT		43,819

3.1.5.2 Maximum Gross Weight

BASIC WEIGHT		43,819
OPERATIONAL LOAD (less fuel)		964
Crew	430.00	
Oil	130.39	
Alcohol for Radome De-Icing	22.00	
Residual Fuel	218.40	
Oxygen Charge	13.39	
Engine Fire Extinguisher Fluid	25.00	
Water for Air Conditioning	125.00	
OPERATIONAL WEIGHT EMPTY		44,783
Maximum Internal Fuel 2,544 Gal. @ 7.8 Lb./Gal.		19,843
MAXIMUM GROSS WEIGHT		64,626
One Half Internal Fuel		9,921
HALF FUEL WEIGHT		54;705

3.1.5.3 Unit Weights

(a) Wing Group (Gross Area 1225 sq. ft.) 8.154 lb./sq. ft.

(b) Vertical Tail (Gross Area 158.79 sq. ft.) 6.298 lb./sq. ft.

(c) Fuel System (Capacity 2544 Imp. Gal.) .295 lb./Imp. Gal.

(d) Lubrication System - Not applicable.

(e) Cooling System - Not applicable.

3.1.5.4 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)

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 28.63 sq. ft. of 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 (Reference Figure 13)

Span.. 50 ft. 0.00 in.
Chord - Root................................ 45 ft. 0.00 in.
- Construction Tip.......................... 4 ft. 4.98 in.
Mean Aerodynamic Chord...................... 30 ft. 2.61 in.
Airfoil Section - Inner Wing
- Profile.... .0003.5-6-3.7 (Modified)
- Camber............ .0075 (Modified)
- Outer Wing
- Profile.... .0003.5-6-3.7 (Modified)
.... .0003.8-6-3.7
- Camber............ .0075 (Modified)
Incidence - Root............................ 0 degrees
- Construction Tip.......................... 0 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.

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
- $\frac{1}{4}$ 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 (approx)... 21 ft. 3.00 in.

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

Aircraft reference line level .... 77 ft. 9.65 in.

3.1.7.7 Propeller

Not applicable.

3.1.7.8 Landing Gear

Tread	25 ft. 1.56 in.
Wheel Base	30 ft. 1.00 in.

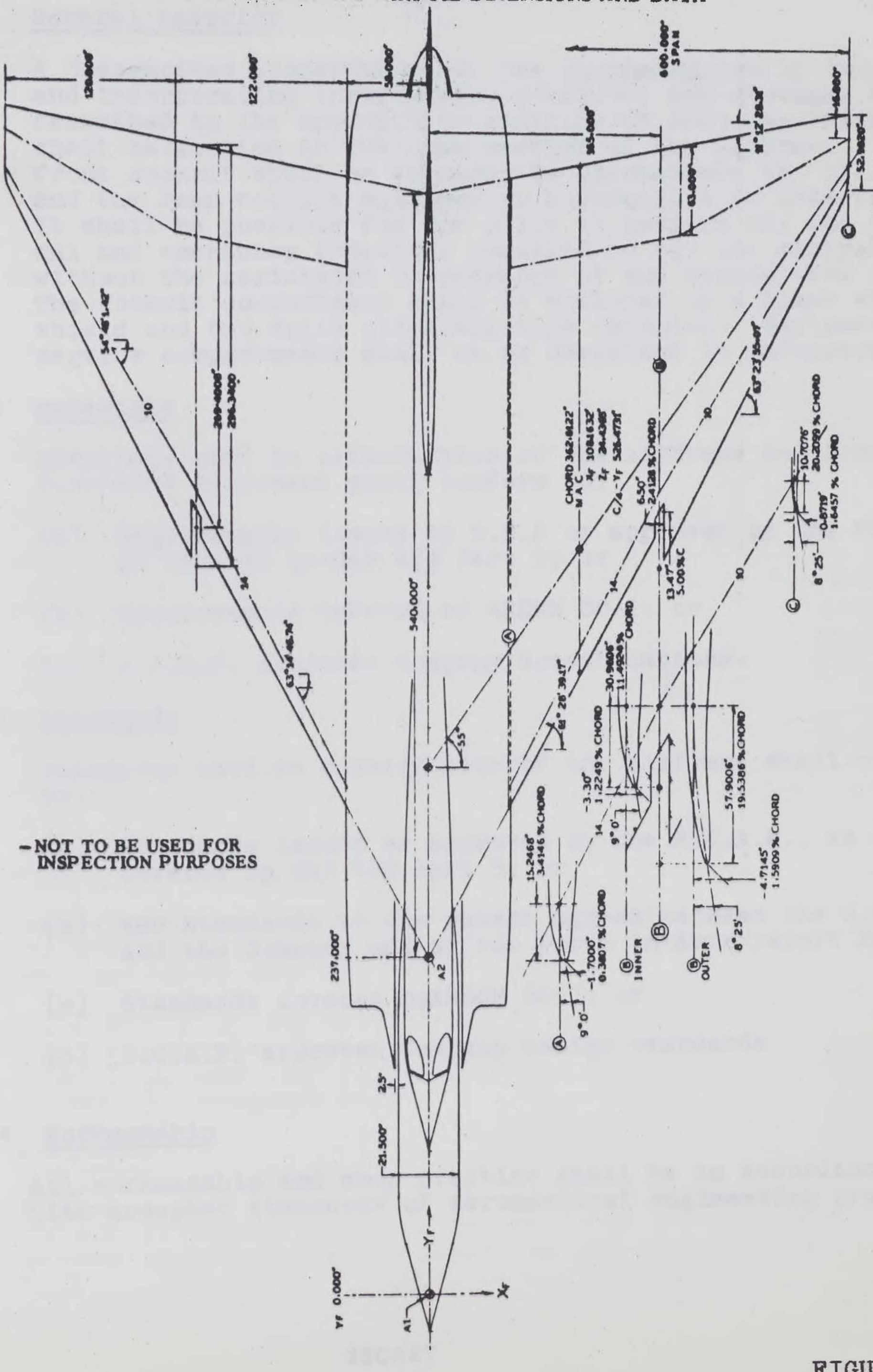
3.1.7.9 Ground Angle

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°	-

WING PLANFORM AND AIRFOIL DIMENSIONS AND DATA



-NOT TO BE USED FOR INSPECTION PURPOSES

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FIGURE 11

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 an observer. 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 two split clamshell type canopies. 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 RCAF 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 airframe shall conform to:

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

(b) ABC Standards to the extent agreed between the R.C.A.F. and the Company and as set forth in Avro report SR-4.

114 (c) Standards covered byARDCM 80-1; or

1 (d) R.C.A.F. approved Company design standards

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 covered by CAP 479 or as otherwise agreed between the R.C.A.F. and the Company.

(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:

3.2.8 Processes (Cont'd)

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

120 (b) Requirements covered by ARDCM 80-1.

3.2.9 Finish

The finish on all parts and components shall be in accordance with R.C.A.F. approved Avro Aircraft Company Standard CS-D-2, and CAP 479 Part 5.

3.2.10 Colour Scheme and Identification Markings

3.2.10.1 Colour Scheme

The aircraft exterior colour scheme and markings shall conform to (a) CAP 479 Chapter 6; or
(b) Company drawings, where not covered by (a).

The cockpit interior colour scheme shall conform to ARDCM 80-1.

3.2.10.2 Identification Markings

Aircraft components and aircraft parts shall be marked in accordance with MIL-M-7911 and Specification 98-24105-5.

3.2.11 Pipeline Identification

119 All removable 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 as set forth in Company Reports listed in Appendix III.

3.2.13 Interchangeability

Interchangeability and replaceability shall conform to the requirements of Specification MIL-I-8500A and as set forth in Avro Aircraft Report QC-E-9.

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 CF-105 aircraft.

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, specified as a mandatory requirement by the R.C.A.F., shall be installed without modification or adjustment, except to the extent of normal minor calibration or adjustment.

Where a degree of performance to meet the specification requirements is obtainable only by a process of testing and/or selection or selective matching of Government supplied equipment, then such testing or selective process shall not be considered a part of the requirements of this specification and upon the subsequent interchange of related components the aircraft performance obtained by such testing and selective process is no longer assured.

Where failure of Government supplied equipment to provide a degree of performance necessary to meet the specification requirements for the aircraft is indicative of a modification requirement of such equipment, then the Company shall notify the R.C.A.F. and recommend such modification for such equipment as it may deem necessary, but shall not undertake such a program of modification as a part of the requirements of this specification.

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 protuberances kept to a minimum.

The aerodynamic characteristics of the aircraft shall be such as to permit the accomplishment of the primary role as defined in R.C.A.F. Specification AIR 7-4, except as limited by the J-75 engine installation.

3.3.1.1 Special Characteristics

The wing leading edge shall be slotted, extended and drooped (as described in paragraphs 3.1.7 and 3.5.2.3) to prevent "pitch-up" at high lift coefficients and to produce favourable air-flow conditions in the fuselage - tail - wing root region.

A maximum camber of 0.75% C (negative) shall be incorporated in the wing design in order to effectively reduce the required elevator deflection at design speed and altitude, thus effectively reducing the trim drag.

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 each fixed ramp and the fuselage, and on each intake ramp face, to prevent boundary air from the forward fuselage and intake ramps from entering the intakes, thus improving intake efficiency.

3.3.1.1 Special Characteristics (Cont'd)

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.

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 Specification MIL-F-8785 except as stated in Avro Aircraft Report Number Aero Data 89.

3.3.3 Aero-Elasticity

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

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3.4 Structural Design Criteria

91 92 93 94 117 The structural design of the aircraft shall be in accordance with the requirements of Specification MIL-S-5700 and limit load factors as stated below, and shall be based on a weight for stress analysis of 47,000 lb.

3.4.1 Limit Flight Load Factors

3.4.1.1 Gross Weight for Stress Analysis 47,000 Lb. (clean configuration)

Maneuver --	Positive	+7.33*
	Negative	-3.00
Gust **	Positive	+5.2
	Negative	-3.2

90 *The limit load factor shall decrease from 7.33 due to the effect of skin temperature rise as shown on flight envelopes 12 to 20 inclusive.

**Vertical Velocity of 55 feet per second.

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 load factor shall be:

Maneuver

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

Gust

$$n_G = \frac{(n-1)\ 47000}{W} + 1$$

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 12 Page 35
10,000 feet:	Figure 13 Page 36
20,000 feet:	Figure 14 Page 37
30,000 Feet:	Figure 15 Page 38
Tropopause:	Figure 16 Page 39
40,000 feet:	Figure 17 Page 40
50,000 feet:	Figure 18 Page 41
60,000 feet:	Figure 19 Page 42
70,000 feet:	Figure 20 Page 43

3.4.1.3 Flight Envelopes (Cont'd)

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
Line B-C = Limit Dive Speed
Line H = Positive Gust-Vertical Velocity of 55 ft. per second
Line J = Negative Gust-Vertical Velocity of 55 ft. per second

3.4.1.4 Load Factors in Roll

4 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.

3.4.1.5 Load Factors in Spin

3 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.

3.4.2 Limit Ground Load Factors

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

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.

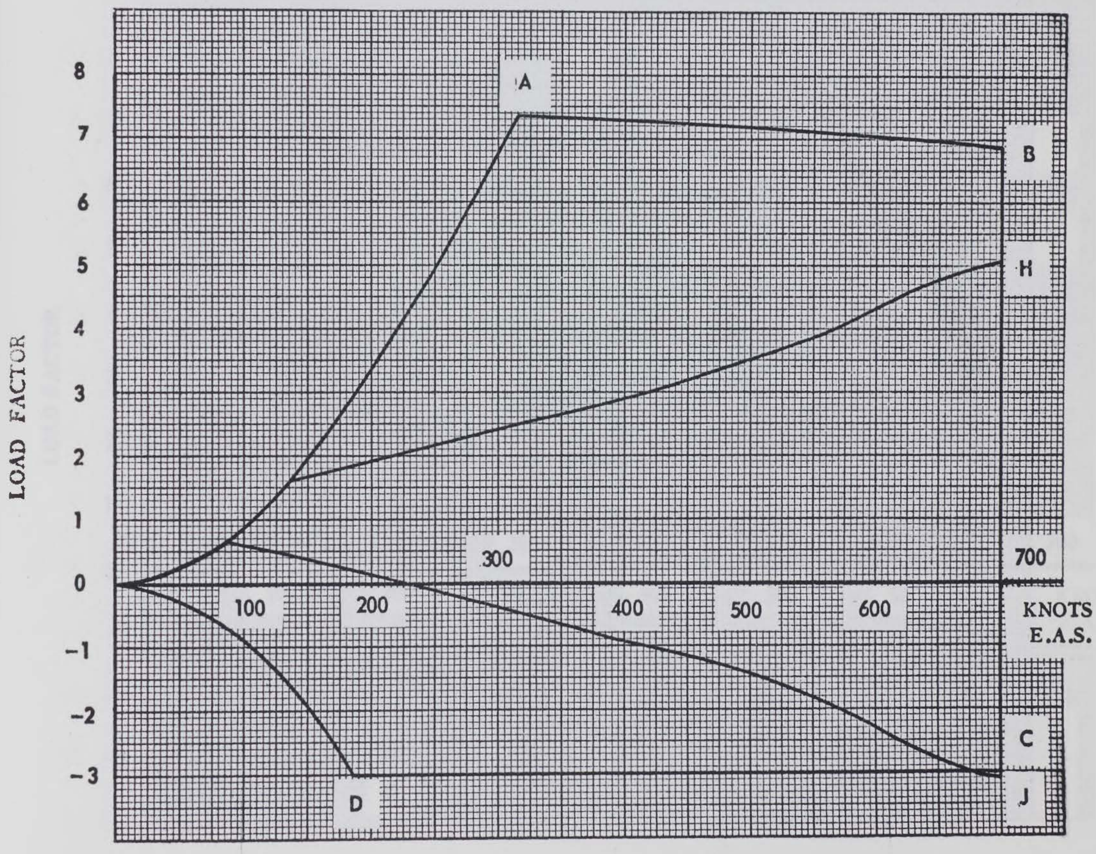
3.4.5 Ultimate Loads

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

CF105 FLIGHT ENVELOPE

SEA LEVEL

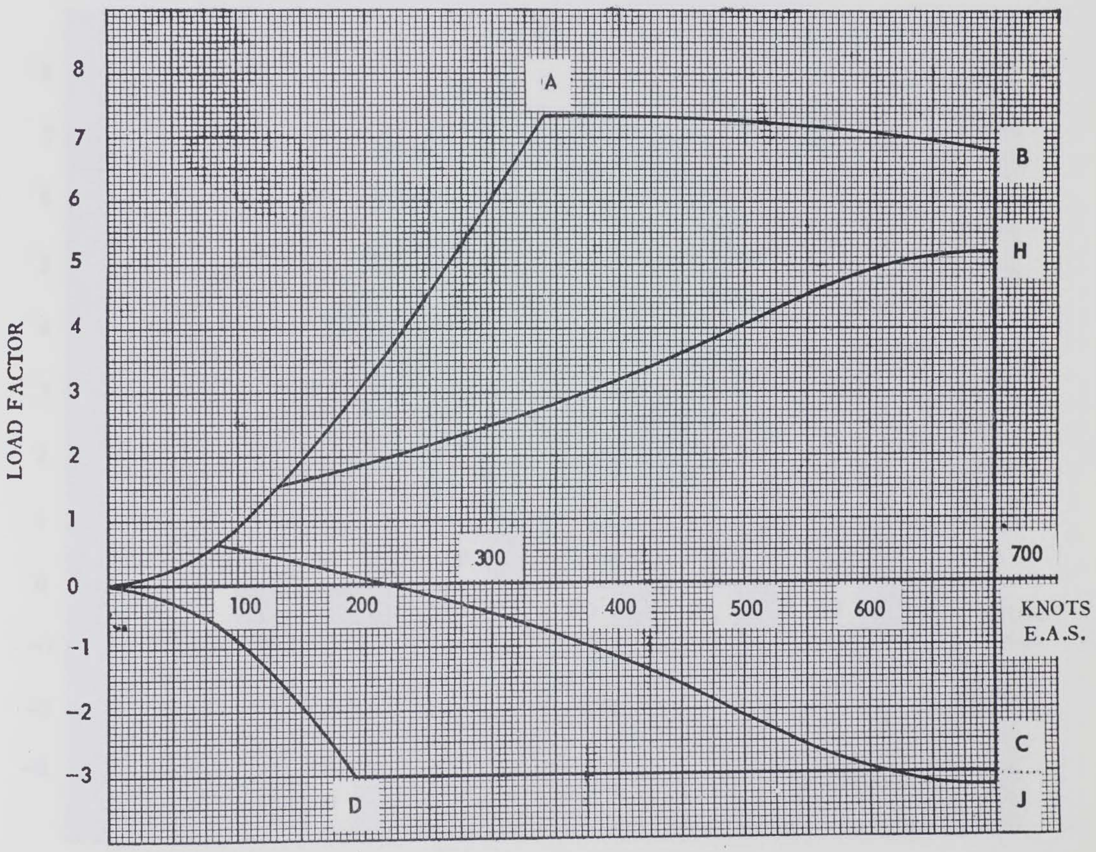
Structural Limit Maneuver
Load Factor for a Stressing
Weight of 47,000 lb.



CF105 FLIGHT ENVELOPE

10,000 FT.

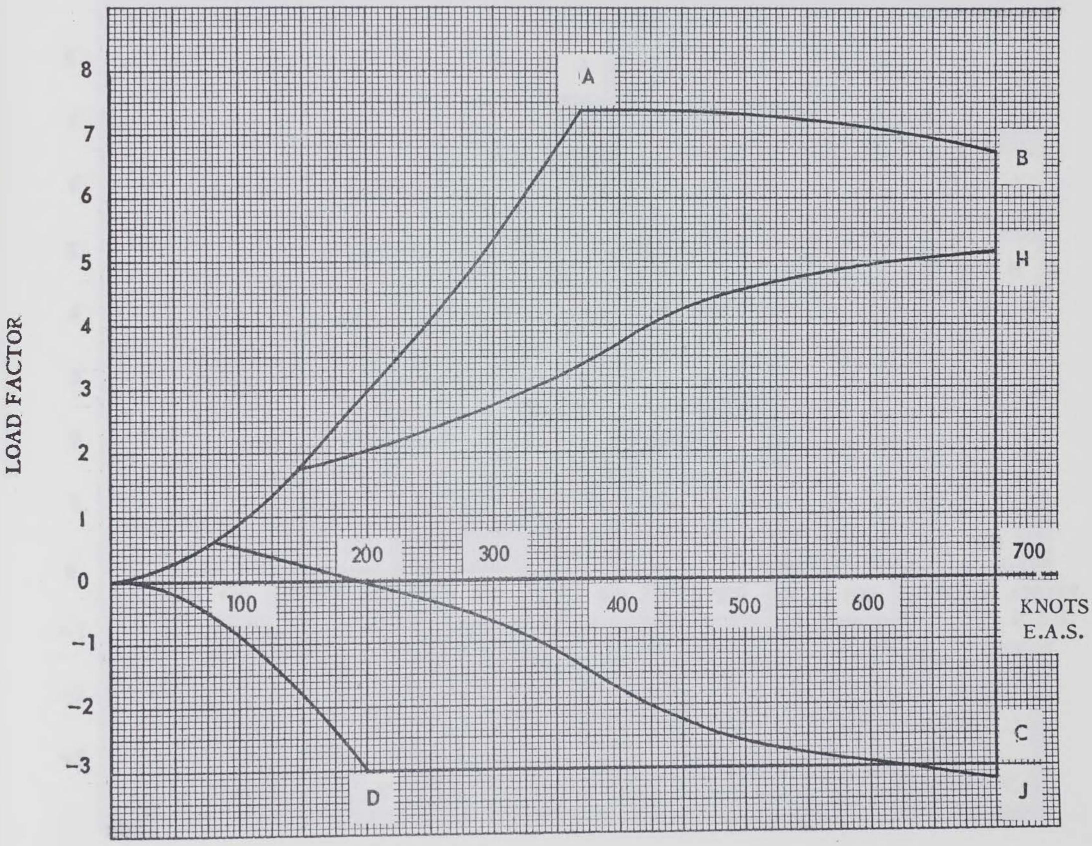
Structural Limit Maneuver
Load Factor for a Stressing
Weight of 47,000 lb.



CF105 FLIGHT ENVELOPE

20,000 FT.

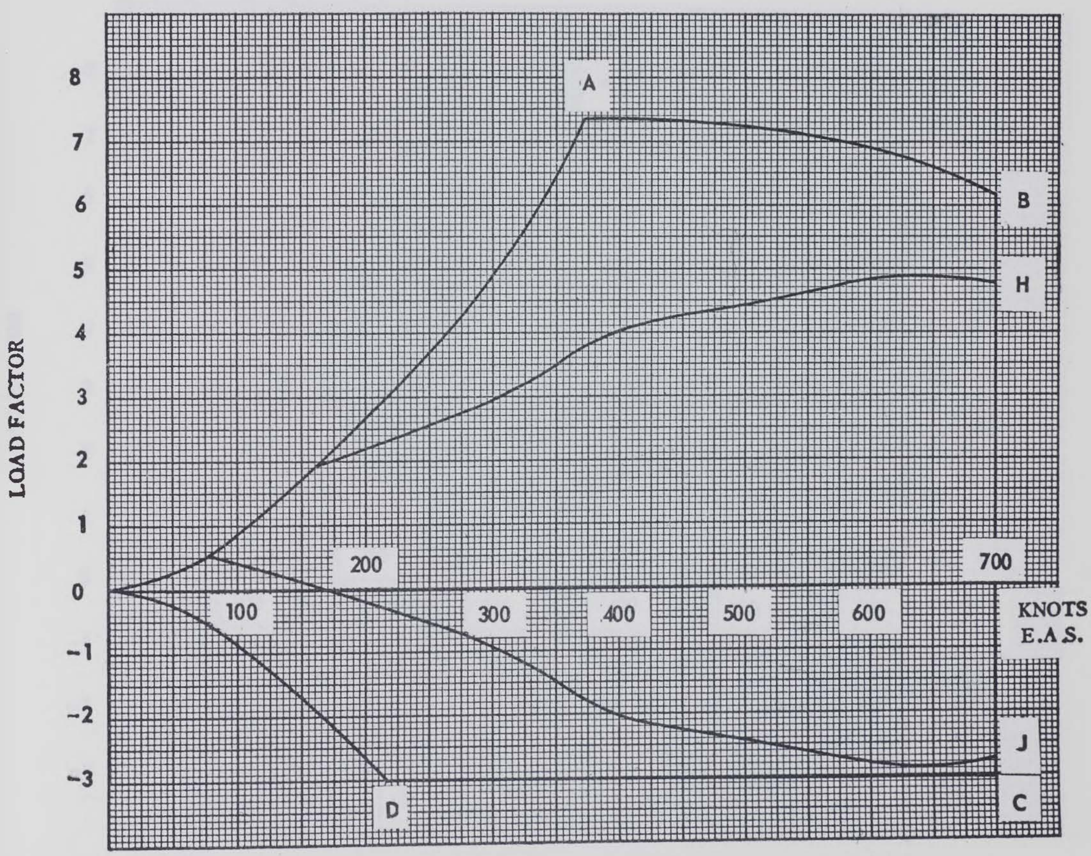
Structural Limit Maneuver
Load Factor for a Stressing
Weight of 47,000 lb.



CF105 FLIGHT ENVELOPE

30,000 FT.

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



CF105 FLIGHT ENVELOPE

TROPOPAUSE

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

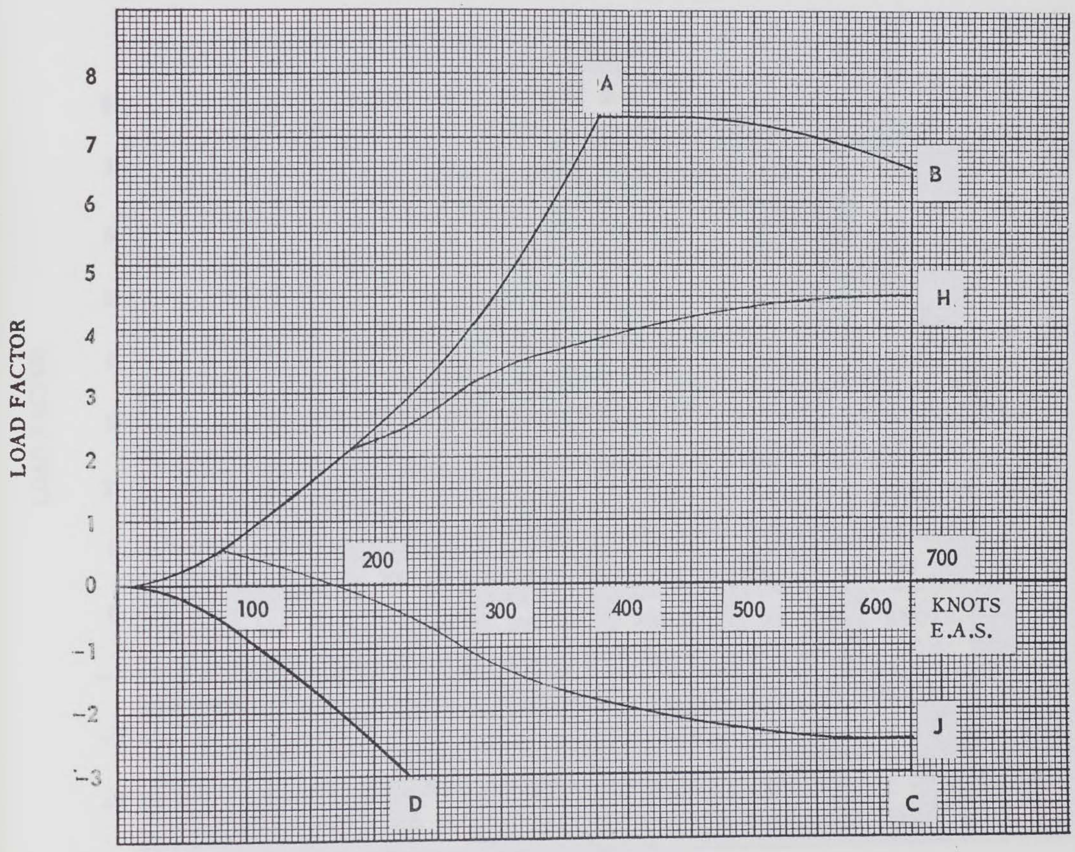
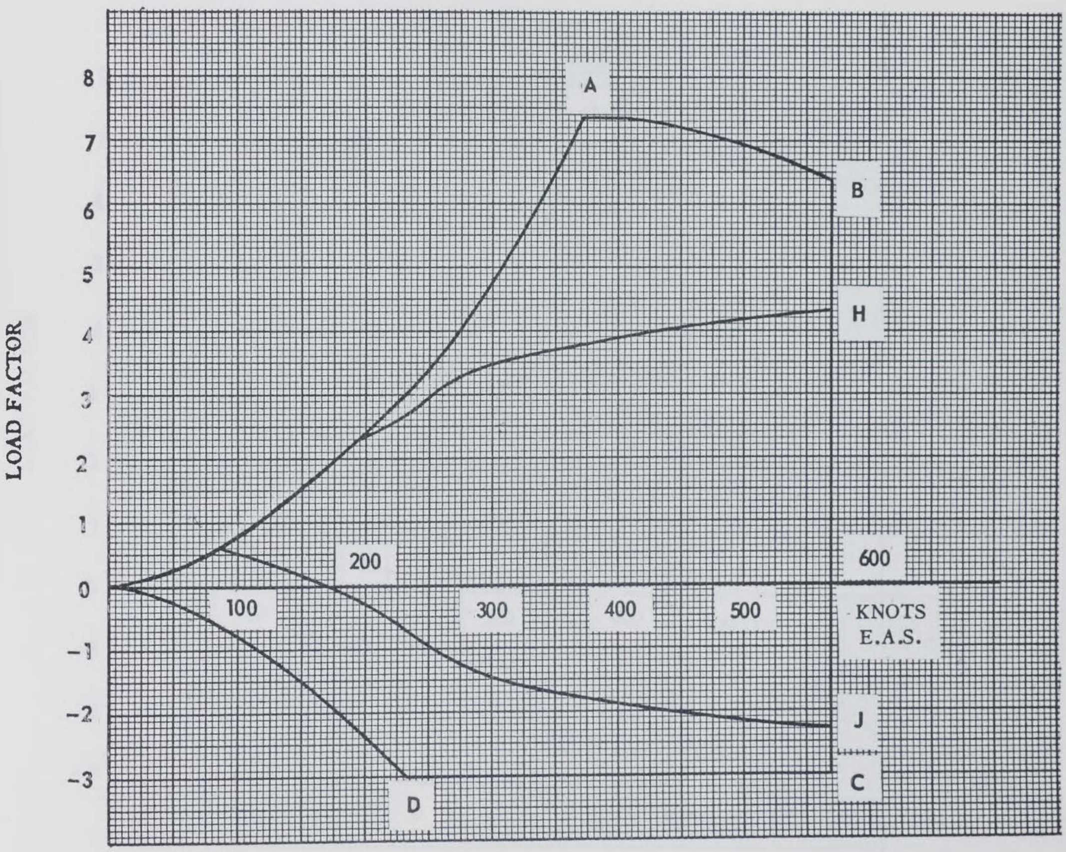


FIGURE 16

CF105 FLIGHT ENVELOPE

40,000 FT.

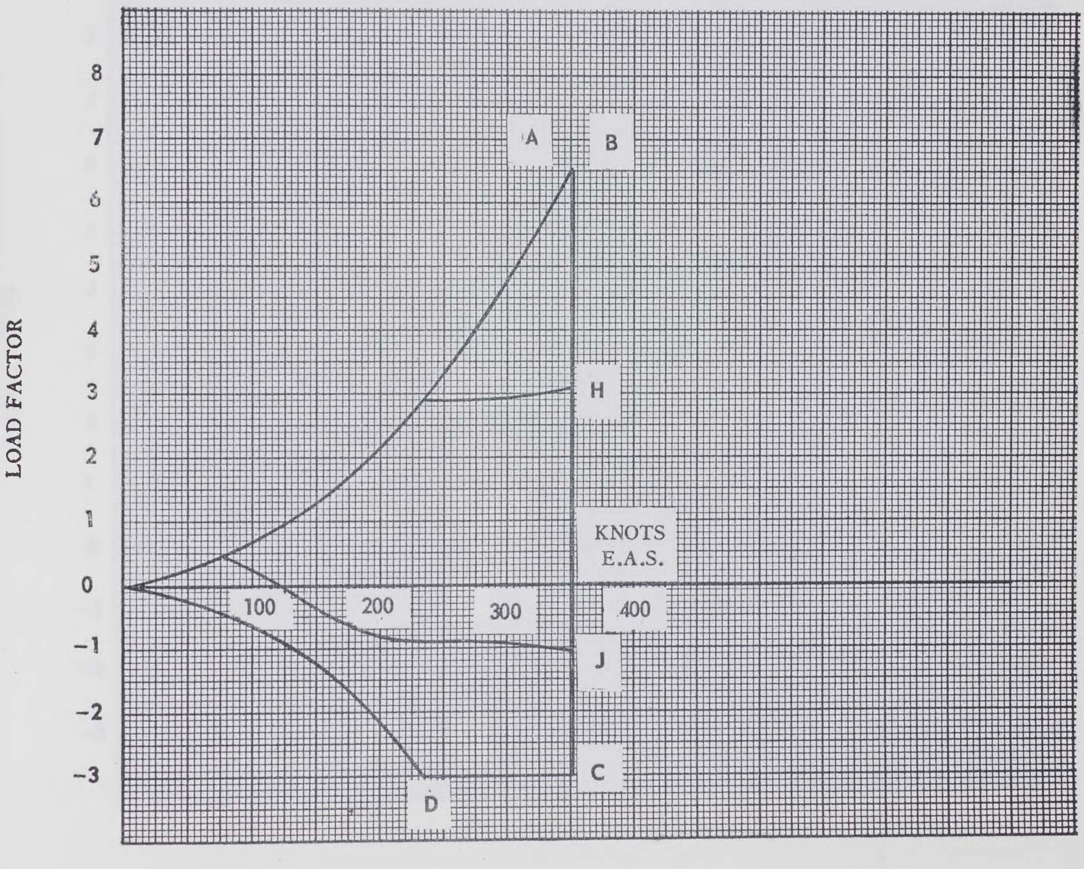
Structural Limit Maneuver
Load Factor for a Stressing
Weight of 47,000 lb.



CF105 FLIGHT ENVELOPE

60,000 FT.

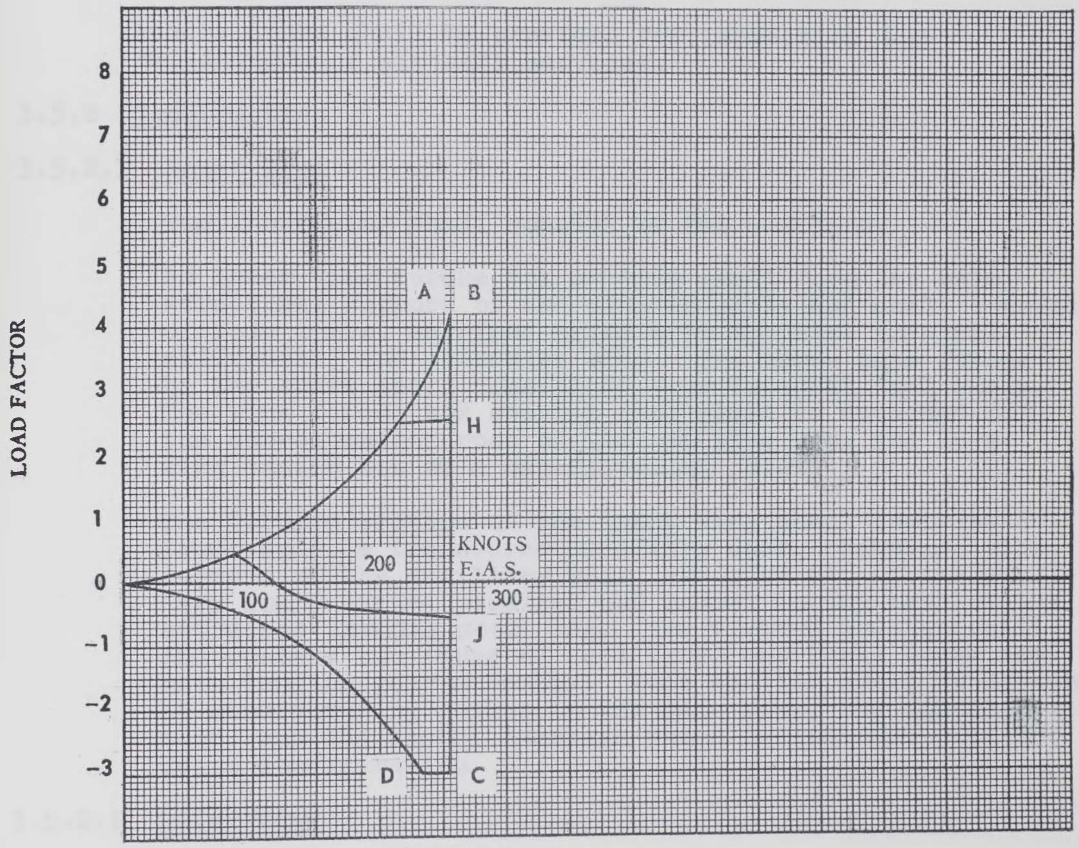
Structural Limit Maneuver
Load Factor for a Stressing
Weight of 47,000 lb.



CF105 FLIGHT ENVELOPE

70,000 FT.

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



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 and maintenance 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 landing gear 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 with loads transmitted to the inner

3.5.2.2 Outer Wing (Cont'd)

wing through the skin attachment, at a front spar joint, a rear spar joint, and three intermediate vertical shear joints. The outer 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 section shall house the elevator control unit.

A center trailing edge, forming an elevator control box, 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 section shall be bolted to the inner and outer wing panels and to the outer trailing edge.

An outer trailing edge, forming an aileron control box, 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 section 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 the aileron linkage in the outer trailing edge section of the outer wing. The aileron shall be hinged to the wing trailing edge by a piano hinge along the topside for the full span of the movable surface and shall be fully shrouded along the underside.

5

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 19.138 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 section. The elevator shall be hinged to the wing trailing edge by a piano hinge along the top side for the full span of the 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.

5

3.5.8 Elevator Tabs

Not applicable.

3.6 Tail Group

3.6.1 Description and Components

The tail group shall comprise a fin and rudder. Due to the aerodynamic configuration of the aircraft 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 Tabs

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 a 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. Pitot and static pressure heads shall be mounted on the upper portion of the fin leading edge. 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, 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 angular motion of the rudder shall be 30^{o} either way from the aircraft centre line. The design of the flying control system and its power operation shall obviate the necessity for aerodynamic and static balance.

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, 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 engine air intakes and cockpit where it shall evolve into a slab-sided horizontally oblong cross-section. A pilot's V-type windshield and a semicircular cockpit enclosure 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 an observer seated in tandem cockpits in the nose fuselage. The cockpits shall be separated by a bulkhead with a transparent access door installed on the bulkhead to provide vision and communication between the two crew stations. The cockpits shall be pressurized in accordance with paragraph 3.22.1 and suitable
70 insulation shall be installed to minimize heat transfer from the adjacent skin. The cockpit compartment shall be enclosed
71 by a fixed windshield and two split clamshell type canopies.

7 The pilot's windshield and canopy windows shall comprise optically flat panels of electrically heated tempered glass, and the rear canopy windows shall comprise curved panels of tempered glass. Each canopy shall be normally actuated electrically and locked by a manually operated latch. To

3.7.3 Crew Stations (Cont'd)

provide for single crew member operation, access to the latch
handle in the rear cockpit shall be gained through the door
in the cockpit separating bulkhead. A canopy actuating switch
shall be installed in each cockpit for the respective canopy.
A switch for each canopy shall be installed in a location
6 accessible from the aircraft exterior. Emergency opening of
the canopies shall be by means of gas pressure from an ex-
plosive cartridge. Internal and external hand operated
mechanical control for emergency opening shall be provided.
Safety interlocks shall be provided to prevent firing of the
ejection seats with the canopies in the closed position.

The line of vision from the pilot's cockpit shall be direct-
72 8 ly forward to a line $12\frac{1}{2}$ degrees below the horizon, aft to
120 degrees on both sides from a line directly forward, and
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 (Operative)

- Canopy Opening (Normal)
- Air Conditioning
- Pressurization Dump Valves
- Air Conditioning De-fog
- Rain Repellant
- High Altitude Light
- Fire Warning and Extinguishing
- Second Shot Fire Extinguishing
- Taxi and Landing Lights
- Navigation Lights
- Engine Starting
- Master Electrical
- Alternator
- Master Warning Test
- Master Warning Reset
- Artificial Horizon Erection
- Engine Relight
- Press-to-test Oxygen Pressure
- Speed Brake

Controls (Operative)

- Air Conditioning
- Cockpit Lighting ON-OFF and Intensity
- Landing Gear
- Power Control (Throttle)
- Anti-g Valve
- Emergency Oxygen Starting
- Drag 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
- UHF Radio AN/ARC-34
- Seat Adjustment
- Manual Harness Release

3.7.3.1 Pilot's Cockpit (Cont'd)

Switches (Operative)

Press-to-Transmit
Damping System ON-OFF
Damping System Engage
UHF Antenna Selector
RMI Function Selector
Nose Wheel Steering
Elevator and Aileron Trim
Rudder Trim
UHF/I.F.F. Emergency Test
Automatic Flying Control System Disconnect
Day and Night Lighting
DC Reset
Anti-Skid
Observer Bail-Out
Engine Fuel
Fuel Cross Feed

Controls (Operative)

Canopy Lock
Seat Ejection

Switches (Non-Operative)

Missile Firing Trigger

3.7.3.2 Observer's Cockpit

Instruments, warning indicators, and the following items of functional equipment shall be installed in the observer's cockpit.

Switches

Press-to-test Oxygen Pressure
Canopy Opening (Normal)
High Altitude Light
Press-to-Transmit
I.F.F./TACAN Antenna Selector
RMI Needle
Cockpit Lighting ON-OFF and
    Intensity

Controls

Manual Harness Release
Seat Ejection
Seat Adjustment
Emergency Oxygen Starting
Canopy Opening (Emergency)
Anti-g Valve
Canopy Lock
Radio Compass AN/ARN-6
Intercom AN/AIC-10
Cockpit Lighting ON-OFF
and Intensity

Controls (Non-Operative)

TACAN AN/ARN-21

3.7.4 Cargo Compartments

Not applicable.

3.7.5 Equipment Compartments

Compartments and bays as listed in the following subparagraphs shall be provided for the equipment and components of propulsion, armament, electronics, and services systems. 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 Electronics Compartments

The radar nose shall comprise two electronics 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 Compartment

82

The forward fuselage electronics compartment shall be located immediately forward of the armament pack bay. Conditioned air shall be provided to the compartment and installed equipment. Access shall be provided through hinged doors on the underside and sides of the fuselage.

3.7.5.5 Air Conditioning Equipment Bay

An air conditioning equipment bay shall be located immediately aft of the observer'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.

3.7.5.6 Oxygen Bay

The oxygen bay shall comprise the dorsal fairing area immediately aft of the observer'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.

3.7.5.7 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.

3.7.5.8 Service Bays

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

3.7.5.9 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 aluminum alloy and magnesium alloy box panel construction, 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

3.7.6 Speed Brakes (Cont'd)

actuated by a continuous positioning lever powered by the utility hydraulics system, and shall retract into a sealed well recessed into the underside of the fuselage.

3.7.7 Fuselage Power Plant Installation

Reference Section 3.10.

3.8 Landing Gear

3.8.1 Description

83 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 de-
9 signed to retract the gear, including door operation, in 5 seconds at -20°F and 30 seconds at -65°F. When completely retracted the landing gear shall be enclosed within the faired lines of the wing and front fuselage.

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 and telescopic spring strut shall be installed to position the gear during retraction.

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 3.14.1.1.3).

3.8.2.2 Wheels, Brakes and Brake Controls (Cont'd)

Metered and differential braking shall be obtainable by operation of toe pedals integral with the rudder pedals. It shall be possible to lock the brakes for parking by full depression of the toe pedals in conjunction with the positioning of a parking lever located at the left side of the pilot's cockpit. After engine shutdown the emergency hydraulic supply available from the accumulator shall permit three full applications of the brakes.

The main wheel brakes shall be applied automatically during the retraction cycle and released automatically during the extension cycle.

3.8.2.3 Tires

Tubeless tires (U.S.A.F. 29 x 7.7 Type VII E.H.P.) rated at 15,500 Lb. static load when inflated to 260 psi shall be installed.

3.8.2.4 Shock Absorbers

The shock absorbers shall be of liquid spring design and shall embody provision for topping up in situ. The shock absorber fluid shall be a blend of silicone oil and hydraulic oil, to Specification Dowcan 200.

3.8.2.5 Retracting, Extending, and Locking Systems

3.8.2.5.1 Retraction

Each main landing gear shall be retracted inward and forward by an hydraulic actuator until an uplock is engaged. During the retraction cycle, a mechanical linkage shall draw the shock absorber into the main strut and rotate the bogie, and a telescopic spring strut shall position the unloaded bogie in a front wheel down attitude to permit stowage of the retracted gear within a wing wheel well.

3.8.2.5.2 Extension

Each main landing gear shall be extended outward and aft by gravity and drag until a downlock is engaged.

During the extension cycle, the gear shall be lengthened and locked, and the bogie rotated to lie in a plane parallel to the aircraft center line.

3.8.2.5.3 Locking

The downlock and uplock for each main gear shall be designed to engage mechanically and to be released by hydraulic actuators. The shock absorber downlock in the gear main strut shall be designed to lock and unlock mechanically.

3.8.2.5.4 Controls

116 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 dual purpose warning light shall be installed in the knob of the selector to show either a steady light, indicating the landing gear is in motion and not locked up or down, or, a flashing light, warning that both engine throttle levers are retarded to 1/3 full throttle or less and the landing gear is retracted. A composite landing gear position indicator with a three-way indicator for each main gear shall be installed near the selector.

3.8.2.5.5 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.

3.8.2.6 Doors and Fairings (Cont'd)

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.

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 actuator.

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 pneumatic spring strut shall be installed to assist shock absorber extension during retraction.

An 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.
84 The nose wheels shall castor, or be steerable, up to 55° 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 absorber 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, to Specification Dowcan 200.

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 by an hydraulic actuator until an uplock is engaged. During the retraction cycle a telescopic pneumatic spring shall assist the extension of the unloaded liquid spring shock absorber assuring positive landing gear travel and positioning in the nose wheel well.

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. A three-way landing gear position indicator for the nose gear shall be installed as part of a composite landing gear position indicator. (Reference paragraph 3.8.2.5.4).

3.8.3.5.5 Emergency Extension

The emergency extension of the nose landing gear shall be affected by the means employed for the main landing gear (Reference paragraph 3.8.2.5.5.).

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

3.8.3.6 <u>Steering Control</u> (Cont'd)

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 an 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.

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 <u>Doors and Fairings</u>

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 an hydraulic actuator.

3.8.3.8 <u>Inspection and Maintenance</u>

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

3.8.4 <u>Drag Parachute</u>

3.8.4.1 <u>Description</u>

A "fist" type ribbon drag parachute pack, including a spring opened pilot parachute, shall be stowed in a compartment in the top of the stinger 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

69 The parachute shall be streamed or jettisoned as required 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

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

10 The primary flight controls shall be powered by two independent hydraulic power circuits with normal Manual Mode control 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 axes, sideslip minimization in maneuver, and spiral stability, in the Manual Mode of control; and yaw axis damping and stabilization only in the Emergency Mode of control. Structural provision shall be made in the wing and fin structure for the instllation of buzz damper units for each control surface.

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.

An ON-Off power supply control switch and a Manual Mode "engage" switch shall be installed on the pilot's console. An Emergency Mode selection switch shall be installed on the pilot's control column grip. The Emergency Mode of control shall be in an operable condition at all times when the surface control hydraulic system is charged, and either mode has been selected, and shall automatically become the effective mode of control in the event of failure of the Manual Mode.

Three warning lights shall be installed in the pilot's warning indicator panel, one to indicate that power "ON" has been selected but neither mode of control has been selected, one to indicate selection of Emergency mode of control, and one to indicate disengagement of either the roll or pitch axis of the damping system (Reference para. 3.9.4.2). Indication of flying control system hydraulic power failure shall be provided by two warning indicator lights and the Master warning indicator lights (Reference paragraph 3.14.1.2.1).

3.9.1.1 Elevators

13 14 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 control column. In the manual mode of control the transducers shall transmit the pilot's input stick forces as electrical signals through an amplifier to the elevator parallel (command) servo. Stick force command signals shall be limited such that "g" loads exceeding the structural integrity of the aircraft shall not be transmitted. 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
12 servo shall be in idling configuration, with the control column movement transmitted through the mechanical linkage direct to the actuator control valves.

3.9.1.2 Ailerons

13 The control column shall be linked to aileron actuator control valve 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 exceeding the structural integrity of the aircraft 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 in idling configuration, with the control column movement transmitted through the mechanical linkage direct to the actuator control valves.

3.9.1.3 Rudder

13 14 Co-ordinated rudder control in the manual mode of control shall be provided by the damping system (Reference paragraph 3.9.4.2.1). 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-oridination functions during maneuvers requiring unco-ordinated control.

3.9.1.4 Artificial Feel

3.9.1.4.1 Manual Mode Artificial Feel

Manual mode artificial feel for the elevators, shall be provided by feel springs earthed to the aircraft structure, the earthing point being monitored by the differential pressure across the piston of the parallel servo such that feel springs relieve the servo load. The stick

3.9.1.4.1 Manual Mode Artificial Feel (Cont'd)

force per "g" shall be essentially constant for all flight conditions.

Manual mode artificial feel for the ailerons shall be provided by the parallel servos which shall provide feel reaction against control column movement. Aileron stick force per unit rate of roll shall be approximately constant.

A rudder feel and trim unit shall be installed and shall incorporate an electrically driven adjustment linkage which shall automatically govern the rudder pedal load per unit of rudder surface deflection as a function of the compressible dynamic pressure.

3.9.1.4.2 Emergency Mode Artificial Feel

Elevator and aileron 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 elevator feel unit, providing additional feel in proportion to the "g" load in the pitching axis. The rudder feel shall be as described for the manual mode.

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

15 Vulnerability of the flying control system to anticipated types of aircraft damage shall be kept to the lowest degree 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 extended 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

3.9.2.2 Speed Brakes (Cont'd)

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 actuators through an 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 speed brake air loads (Reference paragraph 3.14.1.1.4).

68

3.9.3 Trim Control Systems

Aircraft trim for both the normal and emergency mode of control shall be effected by the actuation of an elevator and aileron trim selector button installed on the control column grip and a rudder trim switch installed on the pilot's left hand console. A control surface position indicator shall be installed on the pilot's left hand console.

Elevator and aileron trim adjustment for the manual mode of control shall be provided by an electrical trimming unit which shall provide a signal to substitute for the stick force transducer signals with a resultant zero stick force.

Elevator and aileron trim for the emergency mode of control shall be provided by trim units repositioning the earthing point of the feel units. Rudder trim for both the manual and the emergency mode shall be provided by the rudder feel and trim unit (Reference paragraph 3.9.1.4.1).

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. Pilot control of normal and emergency damping shall be effected in conjunction with Manual Mode and Emergency Mode selection, with roll and pitch axis cut-out re-engagement controlled through the Manual Mode "engage" switch (Reference paragraph 3.9.1).

3.9.4.2 Artificial Damping System(Cont'd)

Damping system warning indication shall be provided by the flying control system indicator lights (Reference paragraph 3.9.1).

3.9.4.2.1 Normal Damping

Normal operation shall provide automatic damping of short period oscillations about the three axes, control of spiral stability, and sideslip minimization in maneuver, in conjunction with the Manual Mode of Control.

Provision shall be made in the rudder damping circuit to permit the pilot to produce intentional sideslip. To safeguard the aircraft in the event of failure of the normal damping system, maneuver sensitive pitch and roll cut-out switches and a maneuver sensitive yaw axis transfer switch shall be installed to disengage the normal damping system and engage the emergency mode artificial feel and emergency yaw axis damping. The switches shall select the emergency function prior to the structural integrity limits being exceeded. It shall be possible to re-engage the normal damping by means of the "engage" switch in the pilot's cockpit (Reference paragraph 3.9.1). To prevent structural damage due to inertia-cross-coupling effects of high rolling velocities, the maximum command rate of roll shall be automatically decreased whenever sufficient sideslip occurs to exceed approximately half the allowable structural loads.

Air data from pitot and static systems, (Reference paragraph 3.13.2), 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 effectively comprise 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 system shall be through panels, provided where necessary.

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. Each support shall contain the ball and socket joint, which shall be 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 shall be 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

85 Access doors shall be provided for inspection, maintenance, removal and installation of engines and accessories. The fuselage tail cones and stinger shall be removable for engine change.

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. (Reference P.W.A. Specifications 2611 and 2605).

3.11.2 Engine Installation

86 Each engine and afterburner shall be contained in a shroud fabricated from aluminum alloy and titanium.

18 The accessory and compressor section shall be separated from the burner and tail pipe section by a transverse titanium firewall.

Forward of the transverse fire wall the upper segment of the engine shall be encased by a titanium can, the lower segment being sealed by longitudinal fire seals, thereby isolating the accessories compartment from the compressor section. The compressor section shall be defined as Zone 1.

19 17 A fireproof ceramically insulated stainless steel liner shall be installed in sections around the engine aft of the transverse firewall. The area between the engine and the liner shall be defined as Zone 2.

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

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

Engine installation and removal shall be carried out using 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 an air turbine 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.

3.11.4 Air Induction System

3.11.4.1 Description and Components

16 The air intakes shall be located outboard of the crew stations. They shall be approximately "D" shaped and external compression shall be achieved by a two dimensional 12° wedge shape ramp attached to the side of the fuselage. The duct from inlet to engine shall diverge 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 bled off through a porous strip on the ramp parallel to the duct intake face and discharged through a reverse scoop on the underside of the ramp. The air then entering the intake shall have the least possible turbulence, therefore the maximum relative speed (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 with an orifice which automatically increases 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 exhaust temperature sensing.

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 an afterburner 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).

98 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 shall be controlled by gills which shall be designed to 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

(98) A fuel cooled heat exchanger, comprising eight separate cooling segments, shall be installed in the centre service bay. Two segments shall be commoned and utilized to cool utility hydraulic oil, one segment shall be utilized to cool the hydraulic oil of each of the two flying control hydraulic systems, one segment shall be utilized to cool the constant speed drive and gear box oil of each engine, and two segments shall be utilized for cooling engine oil, supplementing the oil cooling system integral with each engine.

Two composite air cooled heat exchangers, a left hand installation comprising three separate units, and a right hand installation comprising four separate units, shall be located beneath the respective engine. The heat exchangers shall be cooled by air entering the lower segment of the peripheral intake. The left hand heat exchanger units shall be utilized to cool the hydraulic oil of one of the flying control hydraulic systems, constant speed drive and gear box oil, and engine oil. The right hand units shall provide cooling for the second flying control hydraulic system oil, constant speed drive and gear box oil, engine oil, and the utility hydraulic system oil.

3.11.7 Lubrication System

3.11.7.1 Description and Components

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

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

3.11.8 Fuel System

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

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

(77) Full fuel flow to the engines during inverted flight shall be provided for 15 seconds at sea level, or for approximately 45 seconds at 50,000 feet altitude.

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 wing auxiliary tanks of each sub-system to the respective main (collector) tank by pressurization (Reference paragraph 3.11.8.10). Fuel from the fuselage tanks shall be transferred to the respective main tank by tank pressurization assisted by electrically driven pumps. A 5-way flow proportioner shall maintain the flow from each tank at a predetermined proportion of the total flow in order to maintain the aircraft c.g. travel within specified limits.

(75) A booster pump submerged in each main tank, and driven by the airframe accessories gearbox, shall supply fuel to the engine feed manifolds. In the event of pump failure, tank pressurization shall provide fuel flow through a by-pass around the inoperative pump.

(78) A manually operated cross-feed system shall be installed to permit the transfer of fuel from either sub-system to either engine in order to maintain lateral balance in case of single engine failure.

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.

Switches shall be installed in the nose wheel well for ground control of the shut-off valves.

3.11.8.2 Fuel Specification and Grade

The fuel system shall be designed for the normal use of Aviation Turbine Fuel Type II-3GP-22b (MIL-F-5624C Grade JP4), limited operations and ferry mission use of Aviation Turbine Fuel Type I 3-GP-23b (MIL-F-5616 Grade JP1), and for limited ferry mission use of Aviation Fuel 3-GP-25c (MIL-F-5572A).

3.11.8.3 Fuel Tanks

(76) Twelve wing tanks and two fuselage tanks shall constitute the main fuel storage and shall be divided into a left-hand system comprising the left-hand wing tanks and aft fuselage tank, and a right-hand system comprising the right-hand wing tanks and forward fuselage tank.
(26) One wing tank in each system shall serve as a main (collector) tank.

(22) The wing tanks shall be fabricated as an integral part of the wing structure and the fuselage tanks shall consist of bladder type cells installed in aluminum alloy shells.

3.11.8.3.1 Tank Capacities

Tank design shall be based on the use of fuel with a specific gravity of .75, and shall provide for an expansion space of a minimum of 3% of the normal fuel capacity.

The tanks shall have the following capacities:

Tank No.	No. Of. Tanks	Gross Capacity of Tank Imp. Gal.	Useable Capacity of Tank Imp. Gal.	Total Useable Fuel Capacity Imp. Gal.
1(Fuselage)	1	308	277	277
2(Fuselage)	1	307	281	281
3	2	165	151	302
4	2	101	90	180
5(Collector)	2	170	146	292
6	2	176	154	308
7	2	322	279	558
8	2	207	173	346
				2544

3.11.8.4 External Fuel Tank

Not applicable.

3.11.8.5 Piping and Fittings

(100) 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.

3.11.8.6 Valves

Valves and other components of the fuel system shall be designed to withstand the appropriate environment and operating fluid, including air, fuel vapor, and the fuels designated in paragraph 3.11.8.2.

3.11.8.7 Strainers and Filters

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

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

3.11.8.8 Quantity Gauges, Flowmeters, and Indicators

(81) 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.

Three warning lights shall be installed in the front cockpit, one to indicate proportioning inoperative (bypass open, transfer pump inoperative or refueling access door open) in either sub-system, 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

(20) Not applicable.

3.11.8.10 Pressurization

The fuel system shall be pressurized, using air from the pneumatics system (Reference paragraph 3.15.1.2), to 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 main (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 main 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 the differential pressure relief valve installed in the main pressurization line. (Venting of the wing auxiliary tanks, which are

3.11.8.11 Vent System (Cont'd)

maintained at an absolute pressure, is unnecessary)

When the pressure in the main (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 main 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
80 24 with refueling nozzle Type D1 (MIL-N-5877A) shall be installed in each main landing gear well. The system shall permit refueling to full fuel load within ten minutes.

During refueling fuel shall pass from each adaptor to the respective 5-way proportioner and then along the fuel transfer lines to all tanks except main (collector) tanks. A separate line shall be installed to fill each main tank, and the transfer lines from the proportioners to the main tanks shall be closed during refueling. During a full refueling operation a by-pass on the proportioners shall be opened to provide a minimum restriction to filling. A dual shut-off valve, servo controlled by a duel level sensing unit, shall be installed in each tank. During
21 partial refueling the 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 auxiliary tanks, with venting of the main 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

(25) Combination condensate and drain valves shall be installed 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 main (collector) tanks, shall be transferred to the adaptors through the normal fuel transfer lines by pressurizing the tanks from a ground supply. Fuel shall be removed from the main 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 master switch (Reference paragraph 3.11.8.12).

3.11.8.16 Fuel Jettisoning

Not applicable.

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

(89) 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.

3.11.10.2 Engine Control System (Cont'd)

(67) Depression of the throttle levers in any position forward of two-thirds throttle shall operate micro-switches for selection of afterburning. With afterburners selected, variation of thrust shall be achieved by variation of engine power, with the afterburners operating at an automatically controlled constant temperature. Rearward movement of the throttle levers to two-third throttle or less shall operate the micro-switches to automatically cut out the afterburners. In the event of electrical failure further rearward movement of the throttles to approximately 1/2 throttle shall mechanically cut out the afterburners.

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 shall be provided with manual control. A warning light shall be installed in the front cockpit to indicate that the emergency (manual) fuel control system has been selected. Two warning lights, one left-hand and one right-hand, shall be installed in the front cockpit to indicate low pressure at the engine fuel inlet.

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 valves. Centrifugal switches shall be installed in each starter system to complete the circuits to the igniters when the engines reach light-up speed, and to break the circuits when the engines reach cut-out speed. Indicator lights adjacent to the starter switches shall be installed to signal the pilot to move the throttles from "off" to "idle" at light-up speed.

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.

3.13 Instruments and Navigational Equipment

Instrument arrangement shall be as agreed upon by the R.C.A.F. and the Company.

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 & 3.17.2.10)
Accelerometer

3.13.1.1.2 Navigational Instruments

Clock
Standby Magnetic Compass
R-Theta DR Repeater (Ref. Para. 3.13.3.2)

3.13.1.1.3 Engine Instruments

Exhaust Temperature Indicator (2) (Ref. Para. 3.11.5)
Pressure Ratio Indicator (2)
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 Quantity Gauge (Ref. Para. 3.21.1.4)
Landing Gear Position Indicator (Ref. Paras. 3.8.2.5.4 and 3.8.3.5.4)

3.13.1.2 Observer's Instruments

Pressure Altimeter
Ground Speed and Interception Computer
R-Theta Indicator
Radio Magnetic Indicator

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.

3.13.4 Installation

(27) (88) (87) 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.

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.

3.13.5 Test Instrumentation

Test instrumentation shall be installed in accordance with paragraph 4.3

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, emergency alternator drive, 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, and one including a sub-system for operation of a radar scanner drive.

The systems shall be designed in accordance with the requirements of Specification MIL-H-5440A 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, and pump inlet pressurizaton. Hydraulic power for emergency operation of the brakes, and pressurized nitrogen for emergency extension of the landing gear shall be provided.

3.14.1.1 Utility Services System

Two 4,000 psi constant delivery hydraulic pumps shall be installed, one on each aircraft accessories gear box, with the output of both pumps combined at a pressure regulating and check valve and utilized to power the utility services and charge two accumulators. The output of one accumulator shall be utilized to maintain power circuit pressure and the output of the second accumulator shall be reduced to 1,500 psi and utilized for the emergency brake supply and to pressurize the compensator in the return line of each hydraulic system.

Air cooled and fuel cooled heat exchangers shall be installed to limit the temperature of the hydraulic fluid at the pump inlets (Reference paragraph 3.11.6.2).

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 falls below 1,600 psi. in the accumulator utilized for emergency braking.

3.14.1.1.1 Landing Gear Sub-System

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

Emergency extension shall be by nitrogen from a 5000 psi storage bottle with release of the nitrogen controlled by a pneumatic release valve mechanically linked to the landing gear selector lever (Reference paragraphs 3.8.2.5.5 and 3.8.3.5.5).

3.14.1.1.1.1 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. With the landing gear in the retracted position, the landing gear sub-system shall be de-pressurized and vented to the utility services return line.

3.14.1.1.1.2 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.

3.14.1.1.1.3 Emergency Extension

Emergency down selection shall permit a supply of nitrogen from the emergency nitrogen storage bottle to enter an emergency extension circuit. The emergency circuit shall permit the compressed nitrogen 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.

3.14.1.1.2 Nose Wheel Steering Sub-System

A double ended hydraulic actuator shall be installed for nose wheel steering. A selector valve controlled

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

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 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).

3.14.1.1.3 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 4000 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.

29 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.

3.14.1.1.4 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 shall be installed within the selector valve to limit the degree of speed brake extension in relation to

3.14.1.1.4 Speed Brakes Sub-System (Cont'd)

68 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.

3.14.1.1.5 Emergency Alternator Drive

The emergency alternator drive shall comprise an hydraulic motor to drive an AC alternator to provide power for essential services during a double engine flame out. When the rotational speed of both engine low pressure compressors reaches a low windmilling rate the emergency alternator drive motor shall automatically become energized.

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 an 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

97 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 with the output of the "A" system also utilized for a radar scanner drive system.

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

Air cooled and fuel cooled heat exchangers shall be installed to limit the temperature of hydraulic fluid at the pump inlets to 225°F (approx.). (Reference paragraph 3.11.6.2).

A compensator designed to pressurize the return fluid at 90-100 psi and to separate air from the fluid, shall be installed in each system. Each compensator shall be pressurized by the utility hydraulic system accumulator (Reference paragraph 3.14.1.1.). 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.

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

A pressure and a return line self-sealing coupling for a hydraulic test stand shall be installed in each system to permit ground operation for system testing with the engines inoperative.

3.14.2 Hydraulic Fluid

The hydraulic systems shall be designed for the use of hydraulic fluid to Specification 3-GP-26a (MIL-O-5606).

3.14.3 Piping and Fittings

(28) High pressure lines shall be of stainless steel in accordance with Specification M-7-6 except lines subject to flexing which shall be in accordance with Specification MIL-T-6736A. Low pressure lines shall be of aluminum alloy to Specification MIL-T-7081.

3.15 Pneumatic System

The pneumatic system shall comprise several sub-systems which shall utilize pressurized air from the air conditioning system. Ground operation of the pneumatic services shall be possible by utilization of air supplied by ground service.

A ground chargeable storage bottle shall be installed in the aircraft to supply nitrogen 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) and 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 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.

3.15.1.1 Low Pressure Services Sub-System

Air at 85 ± 5 psi (max.) pressure from the air conditioning system heat exchanger shall be filtered and utilized to inflate the canopy seals and 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 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 to 20 psig pressure, to relieve seal pressure 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.

3.15.1.1.1 Canopy Seal Inflation (Cont'd)

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

3.15.1.1.2 Anti-G Suit Inflation

74 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.

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 paragraph 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 shall be of stainless steel. Couplings below 1 inch diameter shall be flareless type couplings to Avro

3.15.1.4 Piping and Fittings (Cont'd)

Aircraft Company Standards. Coupling of 1 inch and above diameter shall be band type couplings.

3.15.1.5 Inspection and Maintenance

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

3.15.2 Ground Operation

Provisitons shall be made for utilization of air supplied by pneumatic ground 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, automatic transfer of the primary load to the left alternator, and disconnection of the secondary AC services in the event of right alternator failure.

3.16.1.2 Emergency AC System

An emergency AC system shall be installed to supply alternating current in the event of failure of the normal supply due to double engine flame-out. Services essential to flight during flame-out conditions shall be transferred to an hydraulic motor driven emergency alternator.

3.16.1.3 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.

3.16.1.4 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.

3.16.1.4 Emergency DC System (Cont'd)

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.

3.16.1.5 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 30 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.

A 1.4 KVA 208/120 volt 3 phase, 400 cycle emergency alternator driven by an hydraulically operated motor shall be installed in the electrical equipment bay.

3.16.2.2 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 the engine starting power supply is connected and the DC requirements are supplied from the same source. Cooling air for the battery shall be supplied by the air conditioning system.

3.16.2.3 Voltage Regulators

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

3.16.2.4 Protective Devices

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

3.16.2.5 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.

3.16.4 Equipment Installation

65 The electrical equipment shall be installed in accordance with the requirements of specification MIL-E-7563, MIL-E-7080, and MIL-E-7614.

3.16.5 Wiring

34 35 115 The installation of all aircraft wiring shall be in accord-
103 to 110 ance with Specification MIL-W-5088A.

3.16.6 Bonding and Shielding

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

3.16.7 Controls

33 Rheostats, resistors, and switches shall be installed in accordance with the requirements of Specification MIL-E-7563 and/or MIL-E-7080.

3.16.7 Controls (Cont'd)

Circuit breakers shall be installed in accordance with the requirements of Specification MIL-E-7614. Damping system circuit breakers shall be located on the left-hand console in the front cockpit and all other circuit breakers shall be located on a panel in the nose wheel well.

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

113 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 accorcance with Specification CAP 479, MIL-P-7788, and MIL-L-6503A.

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. Two high altitude white flood lights shall be installed in each cockpit.

3.16.8.1.3 Interior Illumination Controls

Three continuously variable transformers (0-27.5 volts) shall be installed in the front cockpit to provide illumination control of post-type red flood lights, red edge lights, and the console flood lights respectively. Two continuously variable transformers shall be installed in the rear cockpit, one to provide illumination control of the post type flood lights and red edge lights, and one to provide control of the console flood lights. An ON-OFF switch shall be provided in each cockpit for the 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. Specification 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-6503A.

3.16.8.2.2 Landing and Taxi Lights

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

3.16.8.2.3 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.

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 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.

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 an automatic quick disconnect 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 automatic quick disconnect 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.

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

37 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.

3.16.11.2 Warning Indicator Panel

37 A panel with provision for 25 warning indicators shall be installed in the front cockpit to provide, in conjunction with the master warning lights, indication of specific system failure of system function.

The following warning lights shall be incorporated in the panel:

2 Fuel Low Level L.H. and R.H.
1 Fuel Proportioner Inoperative L.H. or R.H.
1 Engine Emergency Fuel Control Selected
2 Engine Fuel Pressure L.H. and R.H.
2 Oil Pressure L.H. and R.H.
2 Flying Control Hydraulic Pressure "A" and "B"
1 Utility Hydraulic Pressure
1 Emergency Brake Hydraulic Pressure
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 Flying Control System (and Damping) Disengaged
1 Flying Control System (and Damping) Emergency Selected
1 Damping System Roll and/or Pitch Axis Disengaged
1 Air Conditioning Overtemperature
1 Cabin Pressure
1 Ice Warning
1 No present warning function

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 lights, shall be incorporated in each of the fire extinguisher buttons.

3.16.11.4 Landing Gear Position and Warning Lights

A composite landing gear position indicator and a red warning light shall be provided as described in paragraphs 3.8.2.5.4 and 3.8.3.5.4.

3.16.11.5 Bail-Out Warning

A bail-out warning system shall be installed as described in paragraph 3.19.1.

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 electrical system where necessary. Interference limits and methods of measurement for all installations shall be to the requirements of MIL-I-6051 and MIL-I-6181B.

3.16.14 Inspection and Maintenance

32 Suitable provisions shall be made in the aircraft for the
36 inspection, maintenance, removal and re-installation of
96 electrical equipment.

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 and MIL-W-5088A, except as stated herein and in Appendix II. (Paragraph 3.16.5 shows wiring deviations against MIL-W-5088A)

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
Identification Equipment	AN/APX-6A

In addition to the above, complete provision shall be made for the installation of Distance Measuring Equipment AN/ARN 21 (TACAN) and a radar beacon type RBX-1, and space provision shall be made for the installation of automatic ground speed and track indicating equipment (Doppler) including antenna window and wiring, and for the installation of a radar homer operating on S and L bands.

111 Temperature and pressure within the electronics compartments shall be limited by the air conditioning system (Reference Section 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 microswitch incorporated in each seat installation shall provide a means for switching the UHF communication transmitter and the IFF transponder to transmit on the distress frequency 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, any twenty of which may be preset. This equipment incorporates a guard channel receiver sub-assembly tuned to a preset guard frequency with the main receiver selected to any other frequency. Provision shall be made in the system for utilization of the AN/ARA-25 UHF homing adaptor (Reference paragraph 3.17.2.6).

3.17.1.1 Command Set (Cont'd)

A UHF remote control unit shall be installed in the front cockpit.

Circuits shall be provided to operate the transmitter tone modulated on a designated distress frequency. These circuits shall be operated by the seat micro-switches (Reference Section 3.17).

The pilot's and the observer'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 dual purpose antenna incorporating a fan shaped UHF 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 flush mounted in the skin of the forward fuselage electronics compartment. A selector switch located on the front cockpit right hand console shall provide means for connection of either antenna to the set.

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 to provide a means of selection and audio signal level control of the aircraft's communication and navigational radio facilities. The interphone system 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.

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

39 Radio filters additional to those integral with electronic units shall not be required in the electronic system. 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 and MIL-I-6181B.

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 shall be installed in the dorsal electronics compartment and a flush type directional loop shall be installed in the centre door of the forward fuselage electronics compartment.

3.17.2.2 Radar Altimeter

Not applicable.

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.

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 of the UHF communication antennas or the homer directional antenna.

3.17.2.7 VHF Navigation Receiver

Not applicable.

3.17.2.8 Distance Measuring Equipment

Not applicable.

3.17.2.9 Arbitary Course Computer

Reference paragraph 3.13.3.2.

3.17.2.10 L-Band Navigation Equipment

Complete provision shall be made for Type AN/ARN-21 (TACAN) air navigational equipment. When installed this equipment shall provide continuous indication of distance of the aircraft from a selected TACAN ground station, and with RMI needle selection shall display the bearing of the TACAN station on the radio magnetic indicators. Antenna requirements shall be furnished by the L-Band antenna installation (Reference paragraph 3.17.3.4).

3.17.3 Radar

3.17.3.1 Search Equipment

Not applicable.

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-6A, 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. Circuits shall be provided to operate the identification set automatically on emergency mode when the crew seats are ejected. These circuits shall be operated by the seat micro-switches (Reference Section 3.17).

L-Band Antenna requirements shall be furnished by a fan-shaped vertical radiator mounted under a fiberglass fairing at the top of the fin, and a downward facing annular slot antenna mounted in the skin of the access panel forward of the forward fuselage electronics compartment. An L-band antenna transfer switch in the rear cockpit shall permit connection of either antenna to the set. The system shall obtain DC power from the 27.5 volt DC emergency bus and 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 compartment before flight. A dual horn antenna, waveguide coupled to the transmitter-receiver unit, shall be installed in the fin.

112

3.17.4 Electronic Countermeasures

3.17.4.1 Search Equipment

Not applicable.

3.17.4.2 Analyzing Equipment

Not applicable.

3.17.4.3 Panoramic Receiving Equipment

Not applicable.

3.17.4.4 Panoramic Adaptor

Not applicable.

3.17.4.5 Direction Finding Equipment

Reference paragraph 3.17.2.6 (U.H.F. Homer)

3.17.4.6 Transmitting Equipment

Not applicable.

3.17.4.7 Radar Homer

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.

3.18 Armament

3.18.1 Weight and space provision shall be made for the installation of an integrated electronic system in accordance with the requirements of Specification AIR 7-5 and for the carrying of Sparrow 2 - Model D air-to-air guided missiles. The weight and space provision allocated shall be utilized for the installation of test instrumentation required to carry out the intended role of the aircraft as a flight test vehicle (Reference paragraph 4.3). A fairing shall be installed to close the opening designed to provide accommodation of a missile package.

3.19 Equipment and Furnishings

44

45 Equipment and furnishings appropriate to the role of the aircraft shall be installed in accordance with the requirements of R.C.A.F. Specifications AIR 7-4 and CAP 479 except

64 as stated in Appendix II (Deviations).

3.19.1 Personnel Accommodation

A fully automatic ejection seat with an ejection velocity of 80 feet per second shall be installed for each member of the crew. Each seat shall incorporate a single harness to serve as a combined seat and parachute harness for the occupant. The harness shall be attached to a reel, controlled by a lever on the right-hand side of the seat, to provide freedom of movement for the seat occupant. Normal firing of the seat shall be by means of a face blind, and an alternative firing handle shall be provided on the seat pan. Action to fire the seat shall also clear the ejection path. Ejection of the seat shall initiate distress signals (Referenece Section 3.17).

A bail-out warning system comprising a green light in the front cockpit, and a red light and a warning horn in the rear cockpit shall be installed. Operation of a switch installed in the front cockpit shall illuminate both lights and sound the horn. When the rear seat is ejected from the aircraft the green light in the front cockpit shall be automatically switched off.

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

- 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

42 Not applicable.

3.19.2.2 Crew Relief Provisions

41 Not applicable.

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

118 Not applicable.

3.19.2.5 Map Stowage

46 Not applicable.

3.19.3 Windshield Wipers

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

3.19.4 Furnishings

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

3.19.5 Emergency Equipment

Accommodiation 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.

3.20 Fire Protection

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

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 triple outlet fire extinguisher bottles, each containing 12 pounds of freon, shall be installed in the
47 hydraulics bay. The bottles shall be interconnected to provide a single charge to any two of the protected regions, or two charges to any one region. The extinguishing agent shall be discharged initially through stainless steel pipes and finally 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 extinguishing 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 bay shall be similar but without the time delay and fuel shut-off features.

A toggle switch, which shall remain dead until the 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.

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 all three protected compartments.

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.

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

63 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).

The individual emergency oxygen supply shall feed into the normal system through a trip valve installed on each ejection seat. The trip valve shall provide for manual selection, or automatic selection 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.

3.21.1.4 Indicators

A capacitance type oxygen quantity gauge, incorporating an electrical "power off" warning flag, shall be installed in the pilot's cockpit to indicate the quantity of liquid oxygen contained in the converter.

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. Low pressure and high pressure piping shall be to Specification CS-T-148 and CS-T-135 respectively.

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 on 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 conditions of air temperature and/or pressure, as specified herein, 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 pressurized air for the system through a quick disconnect.

Air from the air conditioning system shall be utilized during flight to supply the requirements of the pneumatic services (Reference Section 3.15).

52 53 The air conditioning and pressurizing system shall be designed to meet the requirements of R.C.A.F. Specification 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

3.22.1 In-Flight Air Conditioning (Cont'd)

to automatically pressurize the cockpits above an ambient pressure altitude of 10,000 feet and shall incorporate limited manual control of cockpit temperature.

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, and to select a cockpit temperature of +90°F to disperse cockpit fog.

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 of the installed system shall not exceed 42.5 cu. ft./min. with a differential pressure of 4.75 psi at a temperature of 60°F at sea level.

A warning light shall be installed in the front cockpit to indicate that cabin pressure has fallen below 31,000 feet.

62 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-5099A, and the air supply selection control shall be installed in the pilot's cockpit.

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 the compartments and the air surrounding the equipment below +140°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.

3.22.1.2.2 Secondary Conditioning Disbribution

The exhaust air from te 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 utlized to maintain a cooling air supply, at a temperature below 150°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 pressure reducing and check valves, set at 85 psi, and ducted to a heat exchanger. A pressure switch downstream of each pressure reducing valve shall shut off the air supply at the respective engine bleed should the downstream pressure exceed 120 psi. Thermostats at each duct joint shall sense overtemperature (due to leakage)

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

and shut-off the supply at the respective engine bleed. System design shall prevent simultaneous shut-off of the supply from both engine bleeds.

Ducting shall be installed to convey cooling (ram) air to the heat exchanger from the boundary layer air bleed between each engine air intake and the fuselage wall. The cooling air supply 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

73 The turbine shall cool the conditioned air by expansion and shall power a fan to draw ram air through the air-to-air heat exchanger (Reference paragraph 3.22.1.3.1). A warning light installed on the pilot's warning indicator panel shall indicate a turbine outlet temperature in excess of 80°F.

3.22.1.4 Ram Air Ventilation

55 Selection of ram air shall permit air from the ram air 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

54 An automatic disconnect ground air conditioning coupling embodying a check valve, shall be installed to utilize ground service air at a pressure of 4.5 psi and at a temperature within the range of 60 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.

3.22.3 Inspection and Maintenance

Fittings shall be installed to provide for the connection of cockpit leakage test equipment. Access doors and detachable panels shall be installed to facilitate inspection and maintenance.

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 utilized to protect the air bleed area of each ramp. 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 icing detectors installed at the top of each air intake, in conjunction with a de-icing controller which actuates separate shedding distributors for the left hand and right hand intakes. Icing indication shall be provided by a warning light installed on the pilot's warning indicator panel. 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 energized. The boots shall be protected from overheating by thermostats and temperature control relays which override the de-icing controller signals.

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 be controlled by the icing detectors on the engine air intakes (Reference paragraph 3.23.3.1).

3.23.4 Cockpit Transparencies

56 Anti-icing and anti-misting of the windshield and canopy windows in the forward cockpit only shall be accomplished 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 Antenna Masts

Not applicable.

3.23.7 Pressure Heads

Pitot pressure head protruding forward from the fin leading edge and a front fuselage nose mounted air data sensing boom, housing alpha and beta angle sensors, shall be protected from the formation of ice by means of built-in electrical heaters.

3.23.7 Pressure Heads (Cont'd)

The circuits shall be of the constant heat type protected by overheat thermostats.

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 (alcohol), 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 fluid shall be sprayed from a distributor mounted near the base of the nose boom. Pneumatic system air (Reference paragraph 3.15.1.1.3) shall provide power to pressurize the tank and operate the distributor. An ice detector system, operating from the 27.5 volt DC supply, and located on the underside of the radome, shall be designed to satisfactorily control operation of the distribution system. A red indicating flag shall be incorporated in an annunciator box, located in the nose wheel well, to provide ground indication of de-icing fluid useage.

A scissors switch shall be installed on the main landing gear to prevent operation of the system on the ground.

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 radome de-icing system, shall be installed on the panel adjacent to the left hand speed brake.

3.25 Auxiliary Gear

3.25.1 Towing Provisions

Towing provisions shall conform to the requirements of Specification MIL-T-7935. Special type fittings shall be provided at the nose wheel pivot for attachment of a tow bar. The turning angle limitation shall be 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 paragraphs 3.16.10.1 and 3.17.1.3). The intercom system shall provide a warning signal to the tractor driver when the maximum turning angle is approached, and a warning signal to the cockpit occupant if the tow bar shear pin fails. During towing, power to operate the interphone system shall be supplied by the towing vehicle.

102 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
58 at three points, with the use of a removable jack pad at each jacking point, one on the airplane center line aft of the nose landing gear, and one inboard of each outer wing root.

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

3.25.3 Mooring Provisions

59 60 Provision shall be made for the attachment of mooring fittings to the main and nose landing gear.

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

3.25.4 Hoisting Provisions (Cont'd)

nose center fuselage joint, and one on each inner wing panel adjacent to the outer wing root.

3.25.5 Leveling

61 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° nose up attitude of the longitudinal axis.

SECTION 4

TESTS

4.1 Ground Tests

Functional ground tests shall be conducted under a program established by the Company and approved by the R.C.A.F. to prove ground functioning of the aircraft systems and installed equipment.

Structural integrity tests to demonstrate the structural design criteria as stated in paragraph 3.4 shall be conducted under a static test program conforming to the technical requirements of Specification MIL-S-5710, except that individual deviations from, or additions to the requirements of Specification MIL-S-5710 shall be negotiated between the RCAF and the Company at the time of establishment of the program.

4.2 Flight Tests

After aircraft acceptance, as set forth in paragraph 5.1, functional flight tests shall be conducted under a program established by the Company and approved by the R.C.A.F. to prove in-flight functioning of the aircraft systems and installed equipment.

After aircraft acceptance, as set forth in paragraph 5.1, structural flight tests to demonstrate the structural design criteria as stated in paragraph 3.4 shall be conducted under a flight test program conforming to the technical requirements of Specification MIL-S-5711, except that individual deviations from, or additions to the requirements of Specification MIL-S-5711 shall be negotiated between the R.C.A.F. and the Company at the time of establishment of the program.

4.3 Test Instrumentation

Test instrumentation necessary for conducting the tests of paragraphs 4.1 and 4.2 shall be installed in the aircraft.

SECTION 5

PREPARATION FOR DELIVERY

5.1 Acceptance Procedure

On completion of the first flight the aircraft shall be officially transferred to the RCAF and immediately returned to the Company on an indefinite loan basis to carry out the various phases of a Flight Test Development Program.

SECTION 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

SECTION 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

6.2.2.1 Deviation (Cont'd)

thereto), and the airplane design as defined by this Model Specification.

6.2.2.2 Interchangeability

Interchangeability assemblies, components, and parts shall be capable of being readily installed, removed, or replaced without alteration, misalignment, or damage to parts being installed or to adjoining parts. No fabricating operations such as cutting, filing, drilling, reaming, hammering, bending, prying, or forcing shall be required. Only those tools generally available to aircraft mechanics shall be required for installation procedure. This is not intended to preclude the use of special tools, fixtures, and other shop aids during original assembly of the parts into the article.

6.2.2.3 Replaceability

Replaceability applies to parts, the installation of which may require work or operations addition to the application of the attaching means. In general, such operations include drilling, reaming, cutting, filing, trimming, shimming, or other means normally associated with original assembly into the aircraft or guided missile. Many instances may require match drilling or reaming from the original part or portion of the item. Replaceable parts shall be designed to permit replacement under field maintenance conditions.

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 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 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 simulated combat mission fuel.

6.2.4.2 Maximum Gross Weight

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

6.2.4.3 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).

6.2.4.4 Basic Weight

Basic weight is the weight of an aircraft with fixed and removable equipment installed for the purpose of performing a specific role. The term "Basis 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 included airframe, power plant, accessories, trapped fuel and oil, and non-expendable fluid systems (hydraulic, coolant) filled to capacity, but without expendable items.

6.2.4.5 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

6.2.6.3 Normal Rated Thrust (Cont'd)

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.

ALPHABETICAL INDEX

APPENDIX I

EQUIPMENT CATALOGUE

Issued under Separate Cover.

APPENDIX II

DEVIATIONS

1. Detail Design, Castings

Requirement: ARDCM 80-1, paragraph 3.260

In the design of magnesium alloy castings, wall thickness shall not be less than 5/32 (0.15625) inch, ----------.

Deviation: Minimum wall thickness of magnesium castings may be taken to 0.13 inch in certain limited areas of the castings.

Reason for Deviation and Remarks: To save weight in the more lightly stressed portions of castings.

2. Limit Ground Loads

Requirement: Bulletin ANC-2, Table 4-1

The design requirements specified by ANC-2 call for a 12,000 pound load straight ahead and a 6,000 pound load inclined at 45° to the aircraft longitudinal axis.

Deviation: The aircraft is designed for a 10,000 pound load straight ahead and 6,000 pounds at 45°.

Reason for Deviation and Remarks: These loads were established at the 17th meeting of the Co-ordinating Committee, 2 March 1955, Item No. 9, and confirmed by letter S1038-105-11 (ACE-1) dated 22 August 1955.

3. Yaw Velocity in Flat Spins

Requirement: Specification MIL-S-5702, paragraph 4.3.2.1

Flat Spins: The yawing velocity (in this condition) shall be 5.0 radians per second for fighters and pilot trainers -------------.

Deviation: The yawing velocity in a flat spin shall be taken as 3.5 radians per second.

Reason for Deviation and Remarks: On the basis of all available data it is not considered that a yaw velocity of 3.5 radians per second can be exceeded.

7. Windshield Angle

Requirement: Specification ARDCM 80-1, paragraph 6.21

Flat panels in those areas used for vision in taking-off, flying ---------- should be placed at an angle of incidence no greater than 55° ----------------.

Deviation: The angle of incidence of the windshield shall be 65°.

Reason for Deviation and Remarks: Aerodynamic requirement.

8. Visibility of Wing Tips to Pilot

Requirement: Specification CAP 479, paragraph 20.22

The pilot should be able to see both wing tips in fighters---for formation flying.

Deviation: Wing tips not visible to pilot.

Reason for Deviation and Remarks: Impossible to achieve with accepted aircraft configuration and limitations on pilot movement imposed by required accoutrements.

9. Landing Gear Retraction Time

Requirement: Specification ARDCM 80-1, paragraph 7.60

The time of operation of the landing gear at temperatures between -65°F to -20°F shall not exceed a value which is double the fastest time selected for the -20°F to +120°F range.

Deviation: Design based on a retraction time of 5 seconds at -20°F and 30 seconds at -65°F.

Note: ARDCM 80-1 requirement for retraction time:- 10 seconds (Reference paragraph 7.601)

Reason for Deviation and Remarks: The above criteria adopted as basis for design to save weight imposed by larger piping (Refer to item 4 of the Minutes of the CF105 Development Co-ordinating Committee's 20th Meeting, 22nd June 1955).

13. Cable Guards

Requirement: Specification ARDCM 80-1, paragraph 8.315.1

All pulleys and quadrants shall be provided with stationary guards fitting close to the points of tangency of the control cables.

Deviation: Tension regulating quadrants are equipped with cable guards attached to the quadrants themselves.

Reason for Deviation and Remarks: The above guards move with their respective quadrants and are much simpler and lighter than normal fixed guards. The moveable guards provide ample protection against cables jumping the cable grooves on the quadrants.

14. Control Cable Duplication

Requirement: Specification ARDCM 80-1, paragraph 9.210(a)and (b)

(a) ----- the direct (elevator control) system shall be duplicated from the base of the ---------- control column to the elevator spars.

(b) Where cables are used for the rudder control on aircraft equipped with a single rudder, duplicate cables shall be provided from each rudder pedal to the rudder mast.

Deviation: Single mechanical control linkages are installed between all control surface actuator valves and the pilot's controls.

Reason for Deviation and Remarks: Complexity and space reasons.

15. Control Cable Spacing

Requirement: Specification ARDCM 80-1, paragraph 9.207

Cables of any one control surface shall be separated by at least three inches, preferably more.

Deviation: In a few places, notably where the two cables for a particular control surface change direction at pulleys, the cables are not spaced according to the above requirement.

Reason for Deviation and Remarks: Space restriction. Fairleads or guide tubes are installed where necessary.

16. Engine Air Intake Screens

Requirement: Specification ARDCM 80-1, paragraph 16.625

Where retractable inlet screens are not provided with axial flow engines, the airframe manufacturer shall mount a retractable screen in the inlet duct of the aircraft.

Deviation: Screens not provided.

Reason for Deviation and Remarks: Penalty to performance and weight does not justify complexity required for very doubtful protection. High location of air inlets is considered adequate protection.

17. Engine Isolation

Requirement: Specification ARDCM 80-1, paragraph 15.620

All engines of (multi-engine) aircraft, which are located adjacent to one another in the fuselage or in nacelles shall be isolated from one another by a stainless steel firewall. This firewall shall be as liquid and gas-tight as possible.

Deviation: A composite structure has been used to fulfill the conditions quoted. An aluminum shroud effectively forms an air tight barrier between Zone 2 and the fuselage bay. Insulation blankets are attached to the aluminum shroud to provide flame resistance.

Reason for Deviation and Remarks: Weight.

18. Engine Isolation

Requirement: Specification ARDCM 80-1, paragraph 15.620

Each engine installation of all aircraft, regardless of the number or relative position of the engines, shall incorporate a stainless steel diaphragm that separates the burner and tail pipe section from the accessory and compressor section.

Deviation: Titanium is used to separate the burner and tail pipe section from the accessory and compressor section.

Reason for Deviation and Remarks: Weight.

19. Firewall and Shut-Off Valves

Requirement: Specification CAP 479, paragraph 23.22

Firewall shut-off valves shall be incorporated in fuel, oil, and hydraulic fluid lines which pass through the firewall, in all twin and multi-engine aircraft. The shut-off valves shall be located as near as possible to the firewall and yet still be in a location not liable to be swept by a nacelle fire. Valves already provided in these systems can be used to perform the functions of firewall shut-off valves if the controls are convenient to the pilot, second pilot or flight engineer in an emergency, or are automatically closed by operation of the fire fighting controls.

Deviation: Shut-Off cocks not installed for engine and accessories oil systems. Hydraulic system does not enter engine compartment - i.e. does not pass through firewall.

Reason for Deviation and Remarks: Shut-off cocks are not provided in engine oil and accessories oil systems since both are high rate of flow systems with small total capacity. If rupture of either feed or return lines should occur, almost the whole system would be drained before the fault could be detected and shut-off valves operated. Design is based on ARDCM 80-1 requirements which are considered to be more realistic.

20. Purging of Fuel Tanks

Requirement: Specification ARDCM 80-1, paragraph 16.400

------------. A purging system shall be provided for all combat aircraft.--------.

Deviation: A purging system is not provided.

Reason for Deviation and Remarks: Requirements for purging deleted from AIR 7-4 at Issue 2, implying not required. This was agreed at 7th Co-ordinating Meeting, 14 July 1954, Item 39.

21. Tank Selection - Refueling

Requirement: Specification ARDCM 80-1, paragraph 14.323 (j)

It shall be possible to select any tanks for filling, and conversely, to avoid filling any tanks. This is necessary for either c.g. control, selective fuel loading or to avoid the filling of battle damaged tanks or tanks with inoperative fuel booster pumps.

Deviation: Selective tank filling is not provided.

24. Refueling Connection (Cont'd)

Reason for Deviation and Remarks: To permit refueling within the specified time. Agreed at 18th Meeting of CF105 Development Co-ordinating Committee.

25. Fuel Drain Valves

Requirement: Specification ARDCM 80-1, paragraph 15.432

The sump shall be provided with ------------ an approved (Specification 28208) self-locking drain valve.

Deviation: Combination service and condensate drain valves, to Company Specification E-368 are installed.

Reason for Deviation and Remarks: Drain valves to Specification 28208 will not meet temperature requirements.

26. Collector Tank Outlets

Requirement: Specification ARDCM 80-1, paragraph 15.431

The fuel outlet fitting from all tanks --------- shall be of a booster pump flange conforming with AN 4135, AN 4130-10, or AN 4128 ------------.

Deviation: Booster pump mounting does not conform to the above requirement.

Reason for Deviation and Remarks: Structural strength requires minimum size mounting holes in the tank base.

27. Instrument Mounting

Requirement: Specification ARDCM 80-1, paragraph 19.00

A minimum clearance of 10 inches shall be provided behind the instrument board to accommodate the instruments and connections when installed.

Deviation: The clearance at the top corners of the instrument panel is less than 10 inches.

Reason for Deviation and Remarks: The clearance at the top corners of the instrument panel is reduced by the inboard slope of the wind-screen panel.

28. Hydraulic Fittings

Requirement: Specification ARDCM 80-1, paragraph 10.21

Standard, approved, hydraulic components, as indexed in specification MIL-H-5440, related specifications and ANA Bulletins, shall always be used where applicable.

Deviation: (1) All connections shall be flareless type in accordance with Company standards.
(2) The hydraulic connecting pipes for the flying control parallel servos and control surface actuators will be of 4130 seamless steel tubing (spring flexing) to Specification MIL-T-6736.

Reason for Deviation and Remarks: (1) Flarelss type connections to Company Standards incorporate better sealing and strength features.
(2) Approved flexible tubing designed for 4,000 psi is not available and use of swivel type fittings is precluded in some applications by space and weight considerations.

29. Emergency Wheel Brakes

Requirement: Specification MIL-H-5440A, paragraph 3.10.1

All hydraulically operated services which are essential to safety in flight or landing, except types I and IV brake systems, shall be provided with emergency devices ---------. The emergency system shall be completely independent of the main systems up to, but not necessarily including the shuttle valve, the actuating cylinder or the motor.

Deviation: (1) The emergency braking system makes use of the normal braking anti-skid return lines and is physically interconnected in brake valve, transfer valve, and anti-skid valve.
(2) The emergency braking system is powered by an accumulator charged by the Utility Services main hydraulic power circuit.

Reason for Deviation and Remarks: (1) The above deviation obviates the necessity of installing another hydraulic pipe run on the main landing gear main struts.
(2) Failure of the normal brake pressure piping in this design will not prevent operation of brakes from the emergency supply, however failure of theemergency pressure piping up to the brake control valve will de-pressurize the entire Utility Hydraulic System (This is similar in principle to the CF-100. Investigation on use of fuses is being carried out).

30. Moisture Traps - Hydraulic System

Requirement: Specification CAP 479, paragraph 24.45

Traps shall be provided to collect and drain off moisture from the ---------- hydraulic systems.

Deviation: No traps are provided in the hydraulic systems.

33. Switches - Space Provisions (Cont'd)

Deviation: No space provided for spare switches.

Reason for Deviation and Remarks: Space limitations on switch panels prevent installation of additional switches.

34. Cable Routing

Requirement: Specification MIL-W-5088A, paragraph 3.7.3.5

Wires and cables to each equipment which must operate to maintain flight control of the aircraft under normal or emergency conditions shall be separately routed from other wires and cables.

Deviation: Cables essential to maintain flight under normal and emergency conditions are not separated from other cables.

Reason for Deviation and Remarks: Space limitations prevent separate routing of cables essential to maintain flight.

Main A.C. power cables are separate from emergency A.C.
Main A.C. power cables mainly isolated from remaining cables.
Emergency A.C. cables are light wiring and run with distribution cabling.
Main D.C. cables mainly isolated from all others.
Emergency D.C. cables to services run with main D.C. lines.

Normal and emergency wiring must come together at transfer point.

For normal or emergency control lines must run together to selector switch.

35. Cable Grouping

Requirement: Specification MIL-W-5088A, paragraph 3.7.3.4

Unprotected wires and cables of the primary electrical power system shall not be bundled or grouped with distribution circuit wires and cables.

Deviation: Power source cables are bundled with distribution cables in some instances.

Reason for Deviation and Remarks: Space limitations prevent segregation of cables in certain locations. Although the runs are separated for the greater portion of their length the main and distribution cables pass through a large conduit in the nacelle to get from the armament bay to the nose landing gear compartment. Also, to get through the bulkhead at station 485 the cables run through a hole in the bulkhead.

See also deviation number 34.

36. Batteries Disconnect

Requirement: Specification CAP 479, paragraph 70.05 (3)

Disconnect - A quick disconnect device shall be provided at the battery terminals to disconnect the battery from the electrical system.

Deviation: No quick disconnect device shall be provided at the battery terminals. Nut-type terminals are used with a cover provided for terminal insulation, corona barrier, and high creepage protection.

Reason for Deviation and Remarks: Weight saving factor and reliability of a hermetically sealed nickel cadmium battery preclude the necessity for quick disconnect devices.

37. Warning Lights

Requirement: Specification ARDCM 80-1, paragraph 6A.172 (b)

The caution indicator system shall consist of a master indicator light and an indicator panel. The master light shall be red in color and shall be labelled "Master Caution"----------.

-----. The caution indicator panel ------ shall provide a suitable visual indication, red in color ---------.

Deviation: One master warning light and the caution indicators are amber in colour.

Reason for Deviation and Remarks: The warning light system proposed by Avro was approved by the R.C.A.F. Reference letter S1038-105-4 (ACE-1) dated 23 August 1955.

38. Circuit Breakers

Requirement: Specification CAP 479, paragraph 21.62 (1)

In single or tandem pilot aircraft, the circuit breakers shall be located forward on the inboard face of the right console.

Deviation: Circuit breakers, other than those in damping system circuits, are located on a circuit breaker panel in the nose wheel bay. Damping system circuit breakers are located on the left-hand console in the front cockpit.

Reason for Deviation and Remarks: Limitation of space precludes the installation of all circuit breakers in the cockpit and necessitates location of damping system circuit breakers in left-hand console. (Reference R.C.A.F. letter S-1038-105-10 (ACE) 9 Nov. 1956)

Circuit breakers are used for protection only and not as combination protection and switch. Trip free breakers are used which cannot be closed when a fault in the circuit exists.

39. Interference Limits and Methods of Measurement

Requirement: Specification MIL-I-6051, paragraph 4.2.3.5

----------- accomplished. Where in an electronic system any receiver output is normally fed into a radio interphone amplifier, the headset and output meter shall be connected in the amplifier output circuit. The controls for the radio-interphone amplifier shall be adjusted for the conditions of normal system operation.

Deviation: The controls for radio interphone system AN/AIC-10 shall not be set as required for normal operation, but as required for the emergency mode.

Reason for Deviation and Remarks: When the output is measured with the interphone system selected for normal operation, the gain and the inherent noise of the AN/AIC-10 amplifier will give an incorrect measure of the noise content of the particular system under test due to the high gain of the AN/AIC-10 compared to the equipment around which MIL-I-6051 was written. When the interphone system is set to emergency mode of operation the input to the amplifier is directly connected to the output circuit resulting in no noise being introduced or amplified by the interphone system.

40. Quick Disconnects - Crew Services

Requirement: Specification CAP 479, paragraph 21.83

The quick disconnect assembly receptacle, which incorporates the oxygen connection, micro-telephone lead, anti "g" connector, etc., shall be located on the left-hand side of the seat.

Deviation: The quick disconnect assembly is located on the right-hand side of the seat.

Reason for Deviation and Remarks: R.C.A.F. letter S1038CF105-16 (ACE) dated 9 December 1954 requires mounting on right-hand side of seat.

41. Crew Relief Provisions

Requirement: Specification CAP 479, paragraph 42.30

Relief horns shall be installed in all aircraft having an endurance of more than three hours.

Deviation: Relief horns are not installed.

Reason for Deviation and Remarks: This requirement arises only as a result of a secondary role, and as weight prejudices primary role performance, relief horns are not installed.

44. Baggage and Tool Compartment (Cont'd)

Reason for Deviation and Remarks: The R.C.A.F. has no requirement at the present time for any ground handling or servicing equipment to be stowed aboard the aircraft. Reference letter S1032-105-11 (ACE-1), dated 26 July 1955.

45. Stowage(s) in Radar Operator's Cockpit

Requirements: Specification CAP 479, paragraph 20.62

A convenient stowage shall be provided for writing pads, logbook, maintenance manuals, spare fuses and tools.

Deviation: The above stowage(s) is not provided.

Reason for Deviation and Remarks: Not compatible with operational role of the aircraft.

46. Map and Flashlight Stowage in Cockpits

Requirement: Specification CAP 479, paragraph 20.24

The map stowage shall be located on the right hand side of the cockpit in single and tandem seat aircraft. Map stowage shall include provision for stowage of a flashlight.

Deviation: Map and flash light stowage not provided in Mark 1 aircraft.

Reason for Deviation and Remarks: Space at a premium

47. Fire Extinguishing System

Requirement: Specification CAP 479, paragraph 23.72.

Separate extinguishing systems shall be provided for each power plant.

Deviation: The power plant extinguishing systems are not separate.

Reason for Deviation and Remarks: To comply with this requirement would involve an increase from two to three bottles with a consequent increase in weight.

48. Fire Axe

Requirement: Specification CAP 479, paragraph 23.100

Stowage shall be provided for a fire axe in all cabin type aircraft.

Deviation: Fire axe is not installed.

Reason for Deviation and Remarks: Twenty first meeting of Co-ordinating Committee, 20 July 1955, Item XV, Minute 42 (j) states: "Axes are not required in either cockpit".

49. Standard Atmosphere

Requirement: Specification AIR 7-4 paragraph 3.1.1

The aircraft shall meet the performance requirements detailed hereinafter under NACA Standard Atmosphere conditions, except where otherwise specified.

Deviation: ICAO Standard Atmosphere conditions utilized for performance calculations.

Reason for Deviation and Remarks: To conform to international standards.

50. Hand Fire Extinguisher

Requirement: Specification CAP 479, paragraph 23.75

All aircraft, except single seat types, shall have at least one hand fire extinguisher in each crew compartment.

Deviation: Hand fire extinguishers are not installed.

Reason for Deviation and Remarks: Seventeenth meeting of Co-ordinating Committee, 2 March 1955, Item 19, cancels requirement for cockpit fire extinguishers.

51. Overheat Detection - Turbojet Engine Installations

Requirement: Specification CAP 479, paragraphs 23.60 and 23.61

FIRE WARNING SYSTEM - A fire warning system of approved type shall be installed in all aircraft, to indicate fires in all ppotential fire zones.

51. Overheat Detection - Turbojet Engine Installation (Cont'd)

Requirement: (Cont'd)

OVERHEAT DETECTION SYSTEM - An overheat detection system of approved type shall be installed in all turbojet, turbine, propeller and rocket propelled aircraft.

Deviation: No specific overheat detection system is installed.

Reason for Deviation and Remarks: Fire warning system is based on overheat, and additional overheat protection would, therefore, be duplication.

52. Duct Pressure Drop

Requirement: Specification ARDCM 80-1, paragraph 12.443

Total duct pressure drop, including bends and elbows, shall not exceed 3 in. Hg. from engine or cabin supercharger air manifold ---------- to cabin pressure level.

Deviation: Total duct pressure drop will exceed the above requirement.

Reason for Deviation and Remarks: System design is predicated on a large pressure drop through the system (Volume and cooling).

53. Ducting Alignment

Requirement: Specification ARDCM 80-1, paragraph 12.444

At least 6 in. of flexible duct shall be provided immediately adjacent to each fitting on one fitting side only in order to provide for rapid alignment of the tubing during fitting connections. At least 6 in. of flexible ducting shall also be provided in the turbine discharge fitting of the cabin cooling unit, to minimize the effect of aircraft and duct vibration upon turbine wheel vibration characteristics.

Deviation:

(1) Not complied with at some connections.
(2) The expansion cooling turbine and outlet ducting will constitute a firm assembly which will be rigidly installed.

Reason for Deviation and Remarks: Non-compliance only where impracticable or a different design approach is rendered necessary by the basic design of the aircraft as a whole.

57. Jack Pad Installation

Requirement: Specification MIL-J-8711, paragraph 3.3.4.2

Axle jack pads installed on main and nose alighting gear must be integral with or permanently attached to the alighting gear, unless deviation is specifically granted by the procuring activity.

Deviation: The nose gear axle jack pads are not integral with or permanently attached to the nose gear. A special bar is required.

Reason for Deviation and Remarks: Configuration of nose landing gear precludes use of integral jack pad.

Confirmed at 13th Meeting of CF105 Co-ordinating Committee, 1st December 1954, Item 22, Minute 49a.

58. Jack Pads - Stowage

Requirement: Specification MIL-J-8711, paragraph 3.5

Provision shall be made to stow all removable jack pads within the aircraft.

Deviation: No provision made for stowing jack pads.

Reason for Deviation and Remarks: The R.C.A.F. has no requirement at the present time for any ground handling or servicing equipment to be stowed aboard the aircraft. Reference letter S1032-105-11 (ACE-1) dated 26 July 1955.

59. Mooring Fittings

Requirement: Specification ARDCM 80-1, paragraph 8.521

When detachable fittings are furnished, they shall be securely fastened in the baggage or tool compartment.

Deviation: No provision made for stowing mooring fittings.

Reason for Deviation and Remarks: The R.C.A.F. has no requirement for any ground handling or servicing equipment to be stowed aboard the aircraft. Reference letter S1032-105-11 (ACE-1) dated 26 July 1955.

54. Ground Air Disconnects

Requirement: Specification ARDCM 80-1, paragraph 8.5.2

Connections shall be provided on the aircraft, at applicable stations for air conditioning on the ground. These connections shall have a nominal diameter of either 5 in. or 8 in. and shall be in accordance with NAS 400 or NAS 401.

Deviation: One 3 in. quick disconnect air conditioning system coupling shall be installed.

Reason for Deviation and Remarks: The type and size of the coupling installed are compatable with the duct sizes. Confirmed by R.C.A.F. letter S1038-105-11 (ACE-1) dated 22 August 1955.

55. Air Conditioning

Requirement: Specification ARDCM 80-1, paragraph 12.442

A ram air duct shall be installed so as to provide for ingress of ram ventilating air into the cabin when air from the pressure source is not used.

Deviation: Ram air is not provided for ventilating the cockpit.

Reason for Deviation and Remarks: The pressure suits worn by the crew render the supply of ventilation air unnecessary. The ram air supply can be most advantageously used to provide some degree of cooling for equipment vitally necessary for flight, at moderate speeds only.

56. De-frosting - Transparent Areas

Requirement: Specification CAP 479, paragraph 26.06

Means shall also be provided for preventing the fogging and frosting of all transparent areas provided for the use of the crew.

Deviation: No such provision for rear cockpit windows on Mk. 1.

Reason for Deviation and Remarks: Assessed as being not necessary for compliance with the role of the aircraft (Reference R.C.A.F. letter S-1038-105 (ACE) 2 May 1956).

60. Mooring Points

Requirement: Specification ARDCM 80-1, paragraph 8.521.

A mooring fitting shall be provided ------ near the (aircraft) tail. In the case of a nosewheel installation, an additional fitting shall be provided near the nose wheel ----- two wing mooring points on each side of the plane of symmetry shall be providéd.

Deviation: Three mooring points, one on each landing gear, are provided.

Reason for Deviation and Remarks: (1) The configuration and weight distribution in the aircraft make the provision of three mooring points, one on each landing gear, a most practicable arrangement.

(2) The distance between the landing gears coupled with the small amount of weight outside the triangle formed by the landing gears will furnish good stability when the aircraft is moored.

61. Leveling Provisions

Requirement: Specification ARDCM 80-1, paragraph 8.53.

Provisions for measuring and leveling shall be in accordance with Specification MIL-M-6756.

Deviation: A special fixture is used for harmonizing armament and "leveling" the aircraft in a 4° nose up attitude.

Reason for Deviation and Remarks: The method used is considered to be more suitable for the CF105.

62. Air Conditioning, Controls, Interconnection

Requirement: Specification ARDCM 80-1, paragraph 12.442

A valve in the ram air line shall be mechanically or electrically linked with both the emergency pressurized air shut-off valve (in the cabin air duct) and the cabin air dump valve. The linkage shall provide for positive operation of the three valves when operating personnel desire to operate any one of the three.

62. Air Conditioning, Controls, Interconnection (Cont'd)

Deviation: (1) No emergency pressurized air shut-off valve will be installed in the cabin air duct.

(2) The ram air shut-off valve is not linked to the dump valve.

Reason for Deviation and Remarks: (1) A normal system ON-OFF valve which will shut off the flow of conditioning air from the heat exchanger to all conditioned compartments is fitted.

(2) Individual control of the dump valve and the ram air valve will permit control more suited to the system.

(3) Air conditioning system approved in principle at 15th Co-ordinating Committee Meeting, 7 January 1955.

63. Oxygen Regulator

Requirement: Specification CAP 479, paragraph 21.80

In single pilot aircraft the oxygen regulator, oxygen pressure gauge, and oxygen flow indicator shall be located forward on the left or right hand console, readily visible and accessible to the pilot with his shoulder harness locked.

Deviation: Separate pressure demand regulators are mounted on the pilot's and observer's ejection seats.

Reason for Deviation and Remarks: The above requirement cannot be met on aircraft equipped with pressure demand, high altitude, bail out oxygen equipment in conjunction with ejector seats. Reference CF105 Oxygen System Sub-Panel Meeting I.A.M., 23 September 1954, Item 3, paragraph 7(a).

64. Pilot's Operating Instructions, Stowage

Requirement: Specification CAP 479, paragraph 41.03 (1)

A stowage shall be provided in all aircraft for the pilot's operating instructions, within reach of the pilot with his shoulder harness locked.

Deviation: Storage provision for pilot's operating instructions at present not intended.

Reason for Deviation and Remarks: Requirement not compatible with the role of the aircraft.

65. Isolation of Electrical Equipment

Requirement: Specification ARDCM 80-1, paragraph 13.615

Electrical equipment and fuel should be isolated to prevent ignition of the fuel by arcing of broken electrical and fuel lines resulting from battle damage, accidental breaking or normal arcing. ------Fuel, oil and hydraulic lines and equipment shall never be located in a position where leaking fluid will come in contact with electrical equipment through either the effect of gravity, air flow or battle damage, and hydraulic lines will be routed below electrical equipment and wires whenever they cross paths, persuant to Specification MIL-E-7563.

Deviation: Fuel and hydraulic lines and electrical cables are located in close proximity in the fuselage under the wing and aft of station 485.

Reason for Deviation and Remarks: The fuel tanks in the wing sections are integral with the wing structure and space limitations in other sections of the airplane preclude possibilities for wider separation of electrical components and cables from fuel and hydraulic lines. Where necessary, explosion proof type components and connectors and fuel-submersible wiring are installed to avoid possible arcing and fire hazards.

The following electrically operated fuel system components and associated electrical cables are located inside the fuel tanks: Tank capacitance units, fuel-no-air valves, tank shut-off valve switches, level sensing valves and level sensors.

66. Circuit Breaker - Space Provisions

Requirement: Specification CAP 479, paragraph 70.24 (6)

Space shall be provided on the circuit breaker panels for the installation of at least one additional circuit breaker for each group of six breakers.

Deviation: Space is not provided for an additional circuit breaker for each group of six circuit breakers.

Reason for Deviation and Remarks: Space limitation on the panel prevents the fulfillment of the requirement.

67. Afterburner Controls

Requirement: Specification CAP 479, paragraph 21.591 (3)

The afterburner control shall be actuated by movement of the power control lever through a detent or gate in the direction of increased thrust.

67. Afterburner Controls (Cont'd)

Deviation: The afterburners are switched on by depression of the power control lever knobs.

Reason for Deviation and Remarks: On the J75 Model JT4A-23 engine the afterburner is operable at constant power over a range of engine power to give a power range between the Military and Maximum Thrust Ratings. It is therefore, necessary to "bring in" the afterburner by micro-switch operation and use part of the power lever movement for variation of the augmented engine power. Approved by R.C.A.F., Reference letter S1038-105-19 (ACE-1) 25 October 1955.

68. Speed Brakes

Requirement: Specification AIR 7-4, paragraph 4.4.1

Actuation of the speed brakes shall have a minimum effect on the trim or attitude of the aircraft throughout the speed range of the aircraft.

Deviation: A pitch-up condition will occur when the speed brakes are opened at speeds in excess of Mach 1.0.

Reason for Deviation and Remarks: Use of the speed brakes at speeds in excess of Mach 1.0 will not be required to fulfill the intended role of the aircraft.

In the manual mode of control the damping system will automatically counteract the pitch-up condition, and in the emergency mode the change of trim required will be within the trim range.

The pitch-up condition may be used to advantage in dive recovery.

69. Brake Parachute Control

Requirement: Specification CAP 479, paragraph 21.32

The (Brake Parchute Control) actuating motion shall be to pull backward or downward----to deploy the parchute and upward or forward to jettison the parachute.

Deviation: Motion is downward to deploy, and inboard and down to jettison.

Reason for Deviation and Remarks: Design of control motion dictated by the nature of the release mechanism in the rear fuselage. Cockpit approved at 15th Meeting of Co-ordinating Committee, 19th January 1955, Item XVI, paragraph 33.

70. Cockpit Head Room

Requirement: Specification CAP 479, paragraph 20.21

No part of the canopy roof or canopy shall be within $8\frac{1}{2}$" of the pilot's eye-line, within a distance extending forward 21 inches from the intersection of the eye line and the seat back line, or the forward face of the pilot's headrest.

Deviation: The clearance at the pilot's eye-line, 21 inches ahead of the forward face of the pilot's headrest is $6\frac{1}{2}$ inches (approximately).

Reason for Deviation and Remarks: Aerodynamic canopy contour requirement. Cockpit approved at 15th Meeting of Co-ordinating Committee, 19 January 1955, Item XVI, paragraph 33.

71. Canopy Structure

Requirement: Specification CAP 479, paragraph 20.21

There should be no rigid member immediately above the pilot's head in any position in which the cabin roof can be locked.

Deviation: The canopy hatches incorporate rigid structure over the pilot's head when in the closed and locked position.

Reason for Deviation and Remarks: Rigid structure required to strengthen canopy hatches. Cockpit approved at 15th Meeting of Co-ordinating Committee, 19 January 1955, Item XVI, paragraph 33 .

72. Vision

Requirement: Specification CAP 479, paragraph 20.22

The view downward and directly forward shall not be less than 15 degrees below the horizontal.

Deviation: The view downward and directly forward shall be $12\frac{1}{2}$ degrees below the horizontal.

Reason for Deviation and Remarks: Windscreen configuration dictated by performance requirements. Cockpit approved 15th meeting of Co-ordinating Committee, 19 January 1955, Item XVI, paragraph 33.

73. Air Conditioning - Water Separator

Requirement: Specification ARDCM 80-1, paragraph 12.445

When an expansion turbine is used for cooling air, a water separator shall be provided to remove condensed moisture.

Deviation: Water separator is not provided.

Reason for Deviation and Remarks: (1) Weight and space penalty. (2) Effective water separator not available. (Provision is made for pilot to select 90°F inlet temperature to disperse cockpit fog). (3) Air conditioning system approved in principle at 15th Co-ordinating Committee Meeting, 19 January 1955.

74. Anti-G Suit Control Valves

Requirement: Specification CAP 479, paragraph 21.82

The anti-G suit control shall be located on the left hand side of the cockpit adjacent to the seat.

Deviation: In each cockpit the valve shall be installed on the right side of the seat.

Reason for Deviation and Remarks: The seat adjustment handle for each crew seat is on the left side, leaving little space for other equipment. Cockpit approved at 15th Meeting of Co-ordinating Committee, 19 January 1955, Item XVI, paragraph 33.

75. Booster Pump Inlets

Requirement: Specification ARDCM 80-1, paragraph 16.331 (b)

Booster Pumps

------. There shall be no obstructions (not even short lengths of lines) between the tank and the pump inlet.

Deviation: Each booster pump has two large diameter pipes.

Reason for Deviation and Remarks: Inlet pipes are required to insure flow under extreme aircraft attitudes, such as inverted flight.

76. Fuel Tank Locations

Requirement: Specification ARDCM 80-1, paragraph 15.421(a)

---------. No fuel tanks shall be located in or over the engine compartment or over the tail pipe or afterburner section.

Deviation: Tanks No. 5, 7, and 8, R and L, are located partly over the engines.

Reason for Deviation and Remarks: The aircraft layout makes the present fuel tank locations a necessity.

77. Inverted Flight Fuel Supply

Requirement: Specification ARDCM 80-1, paragraph 16.311

-------- design shall be such as to provide for full continuous fuel flow from the tank to the engine for at least 1 minute during inverted flight for jet fighter(aircraft)---------.

Deviation: Provision is made for 15 seconds inverted flight at sea level and approximately 45 seconds at combat altitude with maximum power.

Reason for Deviation and Remarks: It is not possible to provide sufficient inverted flight capacity for 1 minute at all engine and afterburner fuel flows without installing a prohibitively large main tank. Requirement not compatible with the performance of the aircraft at maximum power.

78. Engine Fuel Feed

Requirement: Specification ARDCM 80-1, paragraph 16.320

--------, the fuel system must be designed so that fuel from each tank can be made directly available to the engine(s) in case of boost pump failure or in the event of a damaged main tank.

Deviation: Fuel from each tank will not be directly available to either engine.

Reason for Deviation and Remarks: To provide fuel from each tank directly to the engine(s) would involve considerable penalties in weight and system complexity.

79. Fuel System - Strike Loss

Requirement: Specification AIR 7-4, paragraph 6.4.2

The fuel system shall be designed --------- and shall be such that in the event of a single strike the maximum amount of fuel is retained but in any case not more than 20% of the fuel in the tanks shall be lost or made unavailable to the engine.

Deviation: 50% of the fuel will be lost, if a strike is made on a main (collector) tank.

Reason for Deviation and Remarks: A direct feed is not employed. Reference deviation No. 78.

80. Location of Refueling Adaptors

Requirement: Specification ARDCM 80-1, paragraph 16.323 (14.323q)

The ground servicing adaptor shall be located such that servicing personnel shall require no ladders, supports or elevating devices to insert the nozzle.

Deviation: Elevating devices are required to couple refueling nozzles to the two adaptors.

Reason for Deviation and Remarks: (1) R.C.A.F. requested two refueling points and accessibility to them with the aircraft resting on the bottom of the fuselage.

(2) Location away from the bottom of the fuselage is an overriding requirement for simultaneous re-arming and other system checks during turn-around time. Reference Minutes of 18th Meeting of Co-ordinating Committee, Item 20c

81. Fuel Flow Meters

Requirement: Specification ARDCM 80-1, paragraph 19.241

Flowmeters are required on all jet propelled aircraft ----------.

Deviation: Fuel flowmeters are not installed.

Reason for Deviation and Remarks: The specific requirement was deleted when Specification AIR 7-4 was raised from issue 1 to issue 2.

82. Hinged Doors

Requirement: Specification ARDCM 80-1, paragraph 8.6.2

If hinged doors are used, the hinges shall be located so that the air stream tends to keep them closed, ---------.

Deviation: The forward fuselage electronics compartment door is hinged along its aft edge.

Reason for Deviation and Remarks: If hinged along the forward edge this door could not be opened with the nose jack in position.

83. Tail Skid

Requirement: Specification ARDCM 80-1, paragraph 7.10

Any aircraft equipped with a tricycle landing gear shall be provided with a tail skid or buffer which will adequately protect the control surfaces and the rear portion of the structure from damage and which will provide clearance between the ground and all parts of the structure in the event of a tail down landing.

Deviation: Neither a tail skid nor buffer is installed.

Reason for Deviation and Remarks: Requirement waived by Co-ordinating Committee in the interests of weight saving.

84. Turning Radius

Requirement: Specification ARDCM 80-1, paragraph 7.300

The nose wheel shall swivel through an angle which will permit turns to be made about one wheel as a pivot.

Deviation: Nose wheel swivel will be limited to 55° each way to limit the inner bogie to a described circle of approximately 8.5 ft. radius at maximum turn rate.

Reason for Deviation and Remarks: Minimum safe turning circle of the landing gear bogies is estimated to be 8.5 ft. radius and is accepted by the Co-ordinating Committee.

85. Removal and Replacement of Fuel Nozzles

Requirement: Specification ARDCM 80-1, paragraph 15.241

The following (components) shall be readily removable and replaceable without removing the engine, tanks, or important parts of the aircraft structure:-

Fuel Nozzles

(Note: This requirement refers to engine fuel nozzles.)

Deviation: The fuel nozzles are not accessible for removal, or replacement, with the engines installed.

Reason for Deviation and Remarks: Prohibitive weight penalties do not justify the provision of access.

86. Interchangeability of Power Plants

Requirement: Specification ARDCM 80-1, paragraph 15.25

The power plant installations of multi-engine aircraft shall be identical, permitting complete interchangeability.

Deviation: The complete power plants are not interchangeable as the following are handed:

(1) Front and rear engine mount attachments
(2) Heat exchanger duct
(3) Starter motor and accessories gearbox take-off
(4) Compressor bleed valve outlets
(5) High pressure air take-off

Reason for Deviation and Remarks: Engine design dictates the necessity for handing items (1) to (4).

Item (5) is handed from weight considerations.

87. Instrument Installation

Requirement: Specification AIR 7-4, paragraph 8.2.4

All air lines and electrical leads shall be flexible and fitted with quick disconnects and shall be of sufficient length to allow easy instrument removal.

Deviation: Air lines are not fitted with quick disconnects.

87. Instrument Installation (Cont'd)

Reason for Deviation and Remarks: Space and weight limitations prevent installation of quick disconnects on air lines.

88. Panel Space Provision

Requirement: Specification AIR 7-4, paragraph 8.2.3

Panel space shall be provided for 5 x 5¼ inches case size for the directional indicator and the artificial horizon.

Deviation: Space for 5 x 5¼ inches case size for artificial horizon and directional indicator shall not be provided.

Reason for Deviation and Remarks: Interim R.C.A.F. requirements for artificial horizon and directional indicator override this requirement.

89. Power Plant Controls Identification

Requirement: Specification ARDCM 80-1, paragraphs 6A.14 and 6A.140

(1) Power plant control for each engine shall be located and identified in accordance with MIL-STD-203.

(2) All power plant controls shall be clearly marked in accordance with Specification 98-24105.

Deviation: Power plant controls (throttles) are not identified.

Reason for Deviation and Remarks: Because of location and orientation, it is impossible to confuse the throttles with other controls.

90. Limit Flight Loads

Requirement: Specification AIR 7-4, paragraph 5.2.2.1

At the gross weight for stress analysis, the limit load factor as defined in Specification MIL-S-5700 shall not be less than +7.33 and -3.0.

Deviation: The positive limit load factor decreases from 7.33 as skin temperature increases.

Reason for Deviation and Remarks: Weakening of structure due to temperature rise.

91. Weight for Stress Analysis

Requirement: Specification AIR 7-4, paragraph 5.2.1.2

The gross weight for stress analysis shall not be less than the normal gross weight less 50% of the combat mission fuel.

Deviation: The gross weight for stress analysis is 47,000 lb.

Reason for Deviation and Remarks: Of necessity, the weights to be used for stressing were established in the early stages of design. The weight cannot be related to actual weights which will be obtained when operating the aircraft as a flight test vehicle.

92. Normal Gross Weight

Requirement: Specification AIR 7-4, paragraph 5.2.1.1

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.

Deviation: A normal take-off weight of 55,000 pounds used for stress analysis.

Reason for Deviation and Remarks: Of necessity, the weights to be used for stressing were established in the early stages of design. The weight cannot be related to the actual take-off weight which will be utilized when operating the aircraft as a flight test vehicle.

93. Maximum Gross Weight

Requirement: Specification AIR 7-4, paragraph 5.2.13

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

Deviation: A maximum take-off weight of 65,000 pounds used for stress analysis.

Reason for Deviation and Remarks: Of necessity, the weights to be used for stressing were established in the early stages of design. The weight cannot be related to the actual take-off weight which will be utilized when operating the aircraft as a flight test vehicle.

94. Landing Weights

Requirement: Specification MIL-S-5701, paragraphs 3.2.2.10 and 3.2.2.11

---------- the normal design landing weight shall not be less than the applicable take-off weight less the following items; 75% of fuel (internal and external) carried in the basic mission for fighters (and) bombs, rockets, missiles and ammunition.

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

Deviation: A normal landing gross weight of 45,000 lb. used for stress analysis. A maximum landing gross weight of 55,000 lb. used for stress analysis.

Reason for Deviation and Remarks: Of necessity, the weights to be used for stressing were established in the early stages of design. The weights cannot be related to the actual landing weights which will be utilized when operating the aircraft as a flight test vehicle.

95. Fuel Strainers

Requirement: Specification ARDCM 80-1, paragraph 16.333

The aircraft fuel system shall incorporate the necessary strainers or filters to ensure that the particle size of contaminants in the fuel delivered to the engine(s) is within the limits set forth in the applicable engine Model Specifications.

Since engine specification MIL-E-5007 and MIL-E-8593 require engines to be capable of satisfactory performance on fuel strained to 200 mesh, strainers shall be utilized and shall be the responsibility of the aircraft manufacturer.

Deviation: A 200 mesh strainer is not fitted.

Reason for Deviation and Remarks: The size of 200 mesh strainers capable of handling high performance engine and afterburner fuel flow requirements with low pressure loss is prohibitive. There are no manual fuel filler openings for ingress of foreign matter. Filtered fuel is supplied from pressure refueling ground equipment and the tank pressurization air is strained by 200 mesh air filters.

96. Reverse Current Cut-Outs - Accessibility

Requirement: Specification CAP 479, paragraph 70.26 (1)

The reverse current cut-out(s) shall be accessible for unhampered inspection and maintenance while the engines are running with the aircraft on the ground.

Deviation: The reverse current cut-outs are not accessible for unhampered inspection and maintenance when installed.

Reason for Deviation and Remarks: The reverse current protection devices require an air conditioned location and are therefore installed in the transformer rectifier unit and alternator controls box. These protection devices are accessible only when the transformer rectifier unit and alternator controls box is removed from the aircraft.

97. Flying Controls Hydraulic Circuits

Requirement: Specification AIR 7-4, paragraph 4.7.3.2

(1) The aircraft shall be capable of meeting the scramble requirement of paragraph 3.4.1 under all climatic conditions when housed in a readiness hangar (32°F inside at -40° outside).

Specification AIR 7-4, paragraph 4.7.3.3

(2) The aircraft shall be capable of meeting the scramble requirement of paragraph 3.4.1, with a delay of not more than one minute, when dispersed in the open. Details of the environmental conditions involved in this case will be provided by the Department.

Specification ARDCM 80-1, paragraph 10.02

(3) Flight controls system shall be designed for operation at temperatures between +160°F and -65°F. After the initial breakaway, the increase in force required to operate the control system at -65°F shall not exceed 150% of the force required at +70°F.

Deviation:

(1) Conformity cannot be guaranteed below -40°F ambient (See (1) and (2) below).

97. Flying Controls Hydraulic Circuits (Cont'd)

Deviation: (Cont'd)

(2) & (3) The design of the Flying Control Hydraulic System will permit full operation from 0°F to 250°F. Adequate control with limited maneuverability will be available down to -20°F. At environmental temperatures below -20°F, a delay of over 1 minute will be required for the necessary control exercising to warm the system up to -20°F. At -65°F this will require a delay of about 5 minutes (estimated).

Reason for Deviation and Remarks:

(1) Temperature conditions within readiness hangars are not available for ambient temperatures below -40°.

(2) & (3) It is necessary to cater for temperatures as high as 250°F present during flights. The weight penalty for installing piping of adequate size to permit full control operation down to -65°F would not be justified since the hydraulic fluid would be above 0°F with the aircraft airborne. The fluid must be at a minimum temperature of -20°F before take-off.

98. Oil Cooler Air Flaps

Requirement: Specification ARDCM 80-1, paragraph 15.530

Oil cooler air exit flaps shall be employed whenever an air-to-oil cooler is utilized in a turbine installation. The flaps shall be thermostatically controlled and shall control the engine oil inlet temperature so as not to exceed the value specified in the engine model specification.

Deviation: Oil cooler inlet flaps are used, with two positions only.

Reason for Deviation and Remarks: Inlet flaps are used to maintain high engine inlet efficiency. This air to oil cooler is supplementary to the main cooling system at high speed and high altitude.

99. Reserved

100. Piping Connections - Fuel System

Requirement: Specification ARDCM 80-1, paragraph 13.322

All fittings --------- shall conform to Air Force-Navy or U.S. Air Force Standards.

Deviation:

(1) Flexible couplings to Company Specification are used.
(2) Flareless type fittings to Company Specifications are used.

Reason for Deviation and Remarks:

(1) Special flexible couplings are required to give in-flight and installation flexibility.
(2) Flareless type fittings are used in accordance with latest design practice to give a higher vibration life than may be achieved with AN type flared fittings.

101. Anodizing

Requirements: Specification MIL-A-8625A, paragraph 3.3.5

(Type 1 - Chromic Acid Coating)

Type 1 coatings shall not be applied to alloys with nominal copper content in excess of 5.0 precent or when the total content of alloying elements exceeds 7.5 percent.

97. Flying Controls Hydraulic Circuits (Cont'd)

Deviation: (Cont'd)

(2) & (3) The design of the Flying Control Hydraulic System will permit full operation from 0°F to 250°F. Adequate control with limited maneuverability will be available down to -20°F. At environmental temperatures below -20°F, a delay of over 1 minute will be required for the necessary control exercising to warm the system up to -20°F. At -65°F this will require a delay of about 5 minutes (estimated).

Reason for Deviation and Remarks:

(1) Temperature conditions within readiness hangars are not available for ambient temperatures below -40°.

(2) & (3) It is necessary to cater for temperatures as high as 250°F present during flights. The weight penalty for installing piping of adequate size to permit full control operation down to -65°F would not be justified since the hydraulic fluid would be above 0°F with the aircraft airborne. The fluid must be at a minimum temperature of -20°F before take-off.

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Deviation: Oil cooler inlet flaps are used, with two positions only.

Reason for Deviation and Remarks: Inlet flaps are used to maintain high engine inlet efficiency. This air to oil cooler is supplementary to the main cooling system at high speed and high altitude.

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All fittings --------- shall conform to Air Force-Navy or U.S. Air Force Standards.

Deviation:

(1) Flexible couplings to Company Specification are used.
(2) Flareless type fittings to Company Specifications are used.

Reason for Deviation and Remarks:

(1) Special flexible couplings are required to give in-flight and installation flexibility.
(2) Flareless type fittings are used in accordance with latest design practice to give a higher vibration life than may be achieved with AN type flared fittings.

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(Type 1 - Chromic Acid Coating)

Type 1 coatings shall not be applied to alloys with nominal copper content in excess of 5.0 precent or when the total content of alloying elements exceeds 7.5 percent.

101. Anodizing (Cont'd)

Deviation: Chromic acid anodizing processes are used on 24S and 75S Aluminum Alloys having alloying element contents above the maximum stated in the requirement.

Reason for Deviation and Remarks: Satisfactory results have been obtained for approximately ten years of processing using chromic acid anodizing.

102. Tow Rings, Main Gear

Requirement: Specification MIL-T-7935, paragraph 4.1.3.1.2

Main gear tow rings or other suitable fittings, for attaching the tow bar, shall have a clear opening of pi (II) square inches area, with the minor axis of the opening being not less than 1 inch.

Deviation: The tow rings on the main gear are not suitable for attaching the tow bar, and have a minor axis of .75 inch.

Reason for Deviation and Remarks: Space does not permit a larger ring. It is not intended to tow the aircraft from the main gear utilizing a tow bar. A towing bridle will be used (Normal towing is from the nose gear).

103. Wires and Cables - Under 600 Volts

Requirement: Specification MIL-W-5088A, paragraph 3.5.1.1

For applications under 600 volts, wires and cables shall be in accordance with Specification MIL-W-5086, MIL-W-7072 and MIL-C-7078.

Deviation: (1) Wire to Specification MIL-W-8777 is used in place of wire to specification MIL-W-5086.
(2) Cable to Specification Avrocan M-11-9 is used in place of cable to Specification MIL-C-7078.

Reason for Deviation and Remarks: (1) MIL-W-8777 exceeds MIL-W-5086 with regard to maximum ambient working temperature.
(2) M-11-9 exceeds MIL-C-7078 with regard to maximum ambient working temperature.
(Note: Aluminum wire is not used and Specification MIL-W-7072 is therefore not applicable.)

104. Shielded Wires

Requirement: Specification MIL-W-5088A, paragraph 3.4.1

Unless otherwise specified by the detail installation specification for the equipment involved, shielded wires shall have the shields grounded at each end of the aircraft structure.

Deviation: Shielded wires grounded at one point only.

Reason for Deviation and Remarks: To avoid electrical interference by the formation of ground loops, as for example as specified for the installation of the AN/AIC-10 equipment.

105. Wires for High Temperature

Requirement: Specification MIL-W-5088A, paragraph 3.5.1.3

In areas where the wire temperature exceeds 212°F but does not exceed 400°F, wires in accordance with Specification MIL-W-7139 shall be installed.

Deviation: Wire to MIL-W-8777 is used.

Reason for Deviation and Remarks: Wire to MIL-W-8777 will satisfy temperature requirements up to 250°F and is less expensive and easier to print than Teflon insulated wire to MIL-W-7139.

(Note:- Wire to MIL-W-8777 is not suitable for immersion in fuel and wire to Avrocan Specification M-11-8 is used for these applications.)

106. Coaxial Cables

Requirement: Specification MIL-W-5088A, paragraph 3.5.1.4

Coaxial cables suitable for the application shall conform to Specification JAN-C-17 ---------.

Deviation: Coaxial cables to JAN-C-17 supplemented by cables to Avrocan Specification M-11-10.

Reason for Deviation and Remarks: The cable run up the fin is of considerable length. If used, conventional cables of reasonable diameter and weight would give excessive attenuation.

107. Terminal Block Identification

Requirement: Specification MIL-W-5088A, paragraph 3.6.5.1

The identification shall be of a "permanent" nature affixed to the aircraft, -------------.

Deviation: Identification strip attached to terminal block.

Reason for Deviation and Remarks: See Company letter 5737/02B/J dated 1 December 1955 (paragraphs 33 and 34).

Identification is visible with wiring run couplers in place.

108. Co-axial Connectors

Requirement: Specification MIL-W-5088A, paragraph 3.6.6.5

Connectors for co-axial cables shall be in accordance with Specification MIL-C-71, MIL-C-3607, MIL-C-3608, MIL-C-3650, or MIL-C-3655; unless otherwise specified by the procuring activity.

Deviation: Co-axial connectors to MIL Specifications are used except on aluminum sheath cable to Avrocan Specification M-11-10 (Avro drawing CS-C-162) where co-axial connectors to Avrocan Specification E-413 are used.

Reason for Deviation and Remarks: Connectors must be specially constructed for use with this cable (Reference Deviation 106).

109. Conduit Fittings

Requirement: Specification MIL-W-5088A, paragraph 3.10.1.3

The use of conduit fittings shall be in accordance with Drawing AND 10380, or other applicable drawings.

Deviation: Bulkhead fittings and joints are replaced by flanges welded on conduit.

Reason for Deviation and Remarks: Standard fittings are not suitable for this application.

110. Pressurized Connectors

Requirement: Specification MIL-W-5088A, paragraph 3.6.6.6

Pressurized connectors shall be installed with the flange on the high pressure side.

110. Pressurized Connectors (Cont'd)

Deviation: Pressurized connectors in cockpit are installed with flange on low pressure side.

Reason for Deviation and Remarks: Wiring installation is such that compliance with the above requirement could only have been achieved by the use of an additional connector.

111. Electronic Equipment - Installation

Requirement: Specification MIL-I-8700, paragraph 3.2.1

---------, and located so as not to be subject to conditions exceeding the limits specified in the applicable equipment specifications ---------.

Deviation: Government furnished equipment to MIL-E-5400 is qualified only to 50,000 feet, whereas maximum pressure altitude in cooled equipment areas is approximately 55,000 feet, and in non-conditioned areas is 60,000 feet.

Reason for Deviation and Remarks: AIR 7-5 which is the contractual document covering design and installation of electronic equipment requires, in paragraph 3.1.3 that the equipment is to be compatible with the operational environment of the airplane.

112. Waveguides

Requirement: Specification MIL-W-9053, paragraph 3.6

Rigid waveguid lengths in increments of 6 inches should be used up to and including 72 inches.

Deviation: Waveguid in straight lengths exceeding 72 inches is used.

Reason for Deviation and Remarks: The X-band fin antenna is more than 72 inches away from the equipment and losses must be kept to a minimum, consequently the number of joints is kept to a minimum. Also, space is at a premium and precludes the use of joints in many locations.

113. Fuses

Requirement: Specification CAP 479, paragraph 70.24 (1)

Fuses----------------shall only be used where specifically approved by the R.C.A.F.

113. Fuses (Cont'd)

Deviation: Fuses are used in cockpit lighting circuits, and in instrumentation circuits where current demand is less than 5 amperes.

Reason for Deviation and Remarks: Circuit breakers are not available for 5 amperes or less.

114. Spot Welding - Primary Structure

Requirement: Specification ARDCM 80-1, paragraph 3.220

Primary Structure

-----------. Spot welding shall not be used for the following:

(h) On each side of a joggle or wherever there is a possibility of a tension load component, unless "stop" rivets are used.

Deviation: Spot welding will be used without employing "stop" rivets on the engine supported intake adaptor ring.

Reason for Deviation and Remarks: Hazard introduced by having a riveted structure attached to the engine nose is considered greater than that of utilizing spot welded structure, due to probability of rivets entering the engine.

115. Wire Coding

Requirement: Specification MIL-W-5088A, paragraph 3.1.51(e)

Wire Number: The wire number consisting of one or more digits is used to differentiate between wires in a circuit.

(1) -----------. A different number shall be used for wire not having a common terminal or connection.

Deviation: A different number was not used in the case of the A.C. power cables. Each of the three phase lines begin with "XIA" and thence carries through the wiring of the three phase A.C. power lines.

Reason for Deviation and Remarks: The wire coding was established in its present form due to a misinterpretation of the Specification. As the complete code of each cable is different, in that the phase letter follows the cable size, i.e. XIA4A
XIA4B
XIA4C

118. Check List Holder

Requirement: Specification CAP 479, paragraph 40.41

All aircraft shall be provided with a holder for the Pilot's Check List, mounted convenient to the pilot and second pilot.

Deviation: Check list holder not provided in Mark 1 aircraft.

Reason for Deviation and Remarks: Space at a premium

119. Marking of Pipe Lines and Conduit

Requirement: Specification CAP 479, paragraph 6.02

All pipe lines and electrical conduit in aircraft shall be marked in accordance with RCAF Engineering Order 05-1-2Y, Pipe Line Identification, together with such additional markings as may be required by the specification governing each system.

Deviation: Permanently installed pipe line and conduit is only marked at connecting points.

Reason for Deviation and Remarks: Marking of permanently installed pipe lines and conduit at other points would serve no useful purpose.

120. Anodizing

Requirement: Specification CAP 479, paragraph 51.40(1) and 51.41(1)

Except as indicated in this chapter, or otherwise approved by the RCAF, all aluminum and aluminum alloy parts shall be protected against corrosion by an approved anodic oxidation treatment.

Chemical Films. Approved chemical films may be used on landplanes as an alternative to anodic oxidation, in applications specifically approved by the RCAF.

Deviation: Alodine treatment has been used for certain applications (e.g. - machined wing skins).

Reason for Deviation and Remarks: Use of this treatment is based on specification MIL-S-5002 which permits the use of approved chemical films in accordance with specification AN-C-170, (superseded by MIL-C-5541). The Alodine process meets MIL-C-5541.

Appendix III

Appendix IIIA Drawings

Detail design documents are listed in the following Master Record Index:

MRI-CA-C105/1

Appendix IIIB Reports

Reports covering engineering data shall be listed in a report index and shall be supplied as agreed between the RCAF and the Company.

Appendix IIIC Additional Publications

Not applicable.

Appendix IIID Novel Features

To be added.

APPENDIX IV

AMENDMENTS TO MODEL SPECIFICATION

AVRO AIRCRAFT

MODEL SPECIFICATION AMENDMENT

AIRCRAFT TYPE: ARROW 1	CONTRACT: B69-12-44 Serial 2-B-5-309 SO-4877	AMENDMENT NO. 1 Page 1 of 3
SUBJECT: Applicable Specifications and Publications - Contractual Clarification		E.C.P. Nil
		MOD. NO. Nil
REASON FOR CHANGE: RCAF request; ref letter S36-38-105-9 (ACE-1) dated 2 August, 1957		EFFECTIVITY 25201
		RETROFIT Nil

EFFECT ON PERFORMANCE Nil	
WEIGHT CHANGE Nil	EFFECT ON BALANCE Nil

AMENDMENT
PARAGRAPH 1.1 PAGE 1

Delete

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 in this specification. At the discretion of the Company, subsequently dated issues may be used.

Add

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 in this specification. The applicable paragraphs of this specification shall, in each case, state the extent to which the aircraft design complies with the following specifications. Failure to list non-compliance with the requirements of the specifications listed in this paragraph, which, by a reasonable engineering interpretation should apply to this specification shall indicate the Contractor's intention to meet all such requirements even though no specific mention is made of the requirement in this specification. Where the Contractor does not intend to comply with such requirements a deviation shall be raised.

Engineering Approval	Contract Approval	
[signature] Date 25th Sept '57	[signature] Date 26 Sept /57	

MODEL SPECIFICATION AMENDMENT

AIRCRAFT TYPE	CONTRACT	AMENDMENT NO. 1 Page 2 of 3
ARROW 1		E.C.P.
SUBJECT Applicable Specifications and Publications - Contractual Clarification		MOD. NO.
REASON FOR CHANGE		EFFECTIVITY
		RETROFIT

EFFECT ON PERFORMANCE	
WEIGHT CHANGE	EFFECT ON BALANCE

AMENDMENT

PARAGRAPH 1.1 PAGE 1 (Cont'd)

Add (Cont'd)

Contractor specifications and publications shall be approved by the RCAF prior to forming a part of this specification.

At the discretion of the Company subsequently dated RCAF approved issues may be used.

Delete

AIR 31-2 (Issue 6) - Acceptance of New and Newly Erected Aeroplanes of an Approved Type
(This specification listed in error)

PARAGRAPH 1.4 PAGE 4

Delete

Deviations are set forth in Appendix II to this document and are indicated throughout the text by the appropriate deviation number encircled in the left-hand margin. A definition of "Deviation" appears in paragraph 6.2. From the date of approval by the R.C.A.F. of the Model Specification, additional deviations from the requirements of the specifications listed in paragraph 1.1 shall be submitted in the form of Specification Amendments.

MODEL SPECIFICATION AMENDMENT

AIRCRAFT TYPE	CONTRACT	AMENDMENT NO. 1 Page 3 of 3
ARROW 1		E.C.P
SUBJECT Applicable Specifications and Publications - Contractual Clarification		MOD. NO.
REASON FOR CHANGE		EFFECTIVITY
		RETROFIT

EFFECT ON PERFORMANCE	
WEIGHT CHANGE	EFFECT ON BALANCE

AMENDMENT
PARAGRAPH 1.4 PAGE 4 (cont'd)

Add

Deviations are set forth in Appendix II to this document and are indicated throughout the text by the appropriate deviation number encircled in the left-hand margin. A definition of "Deviation" appears in paragraph 6.2. From the date of approval by the RCAF of the Model Specification, required additional deviations from the requirements of the specifications listed in paragraph 1.1 shall be submitted in the form of Specification Amendments.