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MODEL SPECIFICATION FOR ARROW 2 INTERCEPTOR AIRCRAFT

AVRO AIRCRAFT LIMITED

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MODEL SPECIFICATION

FOR

ARROW 2

SUPERSONIC INTERCEPTOR AIRCRAFT

Prepared By

Model Specification Group
Project Management Services

Recommended for
Technical Approval

Recommended for
Contractual Approval

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G. Hake
Project Designer

J.R. Douglas
Contracts Administrator

R.N. Lindley
Chief Engineer

Contractual Approval

Technical Approval

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J.C. Floyd
Vice President,
Engineering

J.A. Morley
Vice President,
Sales and Service

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SECTION I

APPLICABLE SPECIFICATIONS AND PUBLICATIONS

1.1 Referenced Specifications and Publications

The following specifications and publications of the issue in effect on 23 April 1954 shall form a part of this model specification to the extent stated in this specification. The applicable paragraphs of this model 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, apply to this model specification, shall indicate the Contractor's intention to meet all such requirements even though no specific mention is made of the requirement in this model specification. Where the Contractor does not intend to comply with such requirements a deviation shall be raised.

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 CF-105
CAP 479	Manual of Aircraft Design Requirements for the Royal Canadian Air Force
ARDCM 80-1	Handbook of Instructions for Aircraft Designers
EMS-8	Orenda Engines Specification - Iroquois
RCAF Spec. C-28-96	Luminescent Material, Fluorescent - Radioactive
MIL-B-5087A	Bonding, Electrical (for Aircraft)
MIL-W-5088A	Wiring, Aircraft, Installation of
MIL-I-5099A	Indicator, Cabin Air Pressure, 1-7/8 Inch Dial, Type MA-1
MIL-H-5440A	Design, Installation and Tests of Aircraft Hydraulic Systems
MIL-F-5572A	Fuel, Aircraft Reciprocating Engine

1.1 <u>Referenced Specifications and Publications</u> (Cont'd)

MIL-O-5606 (2)	Oil, Hydraulic, Aircraft, Petroleum Base
MIL-F-5616	Fuel, Aircraft Engine, Grade JP-1
MIL-F-5624C	Fuel, Aircraft Turbine and Jet Engine, Grades JP-3, JP-4, and JP-5
MIL-S-5700	Stress Analysis Criteria
MIL-S-5702	Structural Criteria, Basic Flight Criteria
MIL-S-5703	Structural Criteria, Piloted Airplane Basic Ground Criteria
MIL-S-5711	Structural Criteria, Piloted Airplane 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-6181B	Interference Limits, Test and Design Requirements, Aircraft Electrical and Electronic Equipment
MIL-L-6503A	Lighting Equipment, Aircraft, General Specification for Installation Of
MIL-C-6818A	Clamp, Mounting, Aircraft Instruments
MIL-E-7080 (1)	Electrical Equipment, Installation Of Aircraft, General Specification
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
MIL-M-7911	Marking, Identification of Aeronautical Equipment, Assemblies and Parts

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1.1 Referenced Specifications and Publications (Cont'd)

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
MIL-R-25572	Regulator, Oxygen, Automatic Pressure Breathing High Altitude, General Specification for
USAF Spec. 1817	Flutter, Divergence and Reversal of Aircraft, Prevention of
US Radium R410AB	
CGSB 3-GP-22b	Aviation Turbine Fuel - Type II
CGSB 3-GP-23b	Fuel, Aviation Turbine, Type I
CGSB 3-GP-25c	Aviation Fuel
CGSB 3-GP-26A	Oil, Hydraulic, Petroleum Base
AN 3114	Receptacle, External Power
AN-L-1A	Luminescent Material, Fluorescent
CS-D-2	Protective Treatment Schedule Landplanes
Avrocan E-266	Environmental Testing, Aeronautical and Associated Equipment, General Specification for CF-105 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 GEN/STDS/4	Compliance with ABC Standards
Avro Report 7-0400-62	Weight and C.G. Summary - CF-105 with Iroquois Engines
Dowcan 200 (Issue 1)	Silicone Based Fluid (26 Oct. 1954)

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1.2 Precedence of Requirements

From the date of RCAF approval of this Model Specification the requirements of this specification shall take precedence over the requirements of all specifications listed herein. 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 - RCAF 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 RCAF 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".

SECTION 2

SCOPE

2.1 Aircraft

This Model Specification describes Avro Arrow 2 aircraft, designed to the requirements of RCAF Specification AIR 7-4 Issue 3, and such additional requirements as may be specified and agreed upon between the RCAF and the Company.

The first Arrow 2 aircraft, Serial Number 25206, 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 25206 are identified in the effectivity schedule of the Master Record Index referenced in Appendix 111A of this Specification.

The second and subsequent Arrow 2 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 Arrow 2 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 RCAF name and mark number - Arrow 2

2.1.1.2 RCAF 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 - Arrow 2

2.1.1.5 Number of engines - Two

2.1.1.6 RCAF name and mark number of engine - Iroquois

2.1.1.7 RCAF engine specification number -

2.1.1.8 Engine manufacturer's name - Orenda Engines Limited

2.1.1.9 Engine manufacturer's model designation - PS 13

2.1.1.10 Engine Specification number - EMS-8

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 RCAF Specification AIR 7-4.

2.3 Crew

The crew shall normally consist of a Pilot and an Observer/AI.

SECTION 3

REQUIREMENTS

3.1 Characteristics

3.1.1 Three-View Drawing

See Figure 1 Page 10

3.1.2 Interior Arrangement Drawing

See Figures 2 and 3, Pages 11 and 12.

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.

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

(c) Engine performance in accordance with Orenda Engine Specification EMS-8.

3.1.3.1 Tabulated Performance

	Estimated
Combat Load Factor at a Combat Speed of Mach 1.5, a Combat Altitude of 50,000 feet, and at a Combat Weight* of 52,170 lb.	
Maximum Level Speed of 50,000 feet and at a Combat Weight* of 52,170 lb.	
Combat Ceiling at a Combat Weight* of 52,170 lb.	
With aircraft at Normal Gross Weight* (60,070 lb), and positioned at end of runway; elapsed time from pushing first button to start first engine until aircraft becomes airborne.	

NOTE: *Weight is rough estimate only - corrected figure will be supplied when Iroquois performance becomes available.

3.1.3.1 Tabulated Performance (Cont'd)

Estimated

Elapsed time to reach a level flight Combat Speed of Mach 1.5 and a Combat Altitude of 50,000 feet from the time aircraft becomes airborne during take-off at Normal Gross Weight* (60,070 lb) under sea level conditions:

Take-off distance in still air at sea level, and standard summer temperature of 38°C, to clear 50 foot obstacle (Maximum Thrust with afterburning):

Maximum Gross Weight (67,956 lb)
Normal Gross Weight* (60,070 lb)

Landing distance from 50 foot obstacle in still air, at sea level (drag parachute utilized):

Maximum Landing Gross Weight (63,708 lb)
Normal Landing Gross Weight* (46,470 lb)

Touchdown Speed

Maximum Landing Gross Weight (63,708 lb)
Normal Landing Gross Weight* (46,470 lb)

3.1.4 Performance Curves

3.1.4.1 Maneuverability - 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

NOTE: *Weight is rough estimate only - corrected figure will be supplied when Iroquois performance becomes available.

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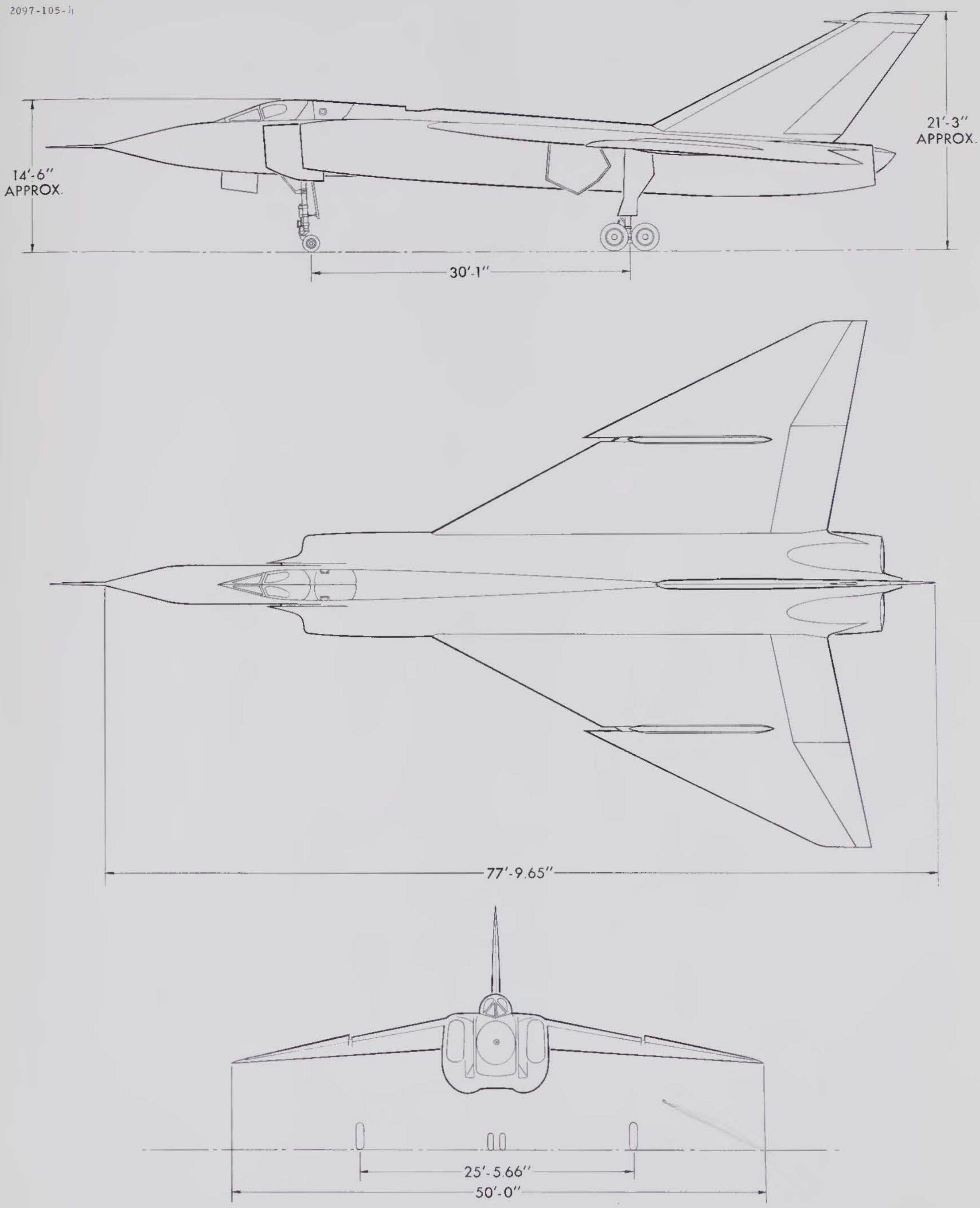

FIG. 1

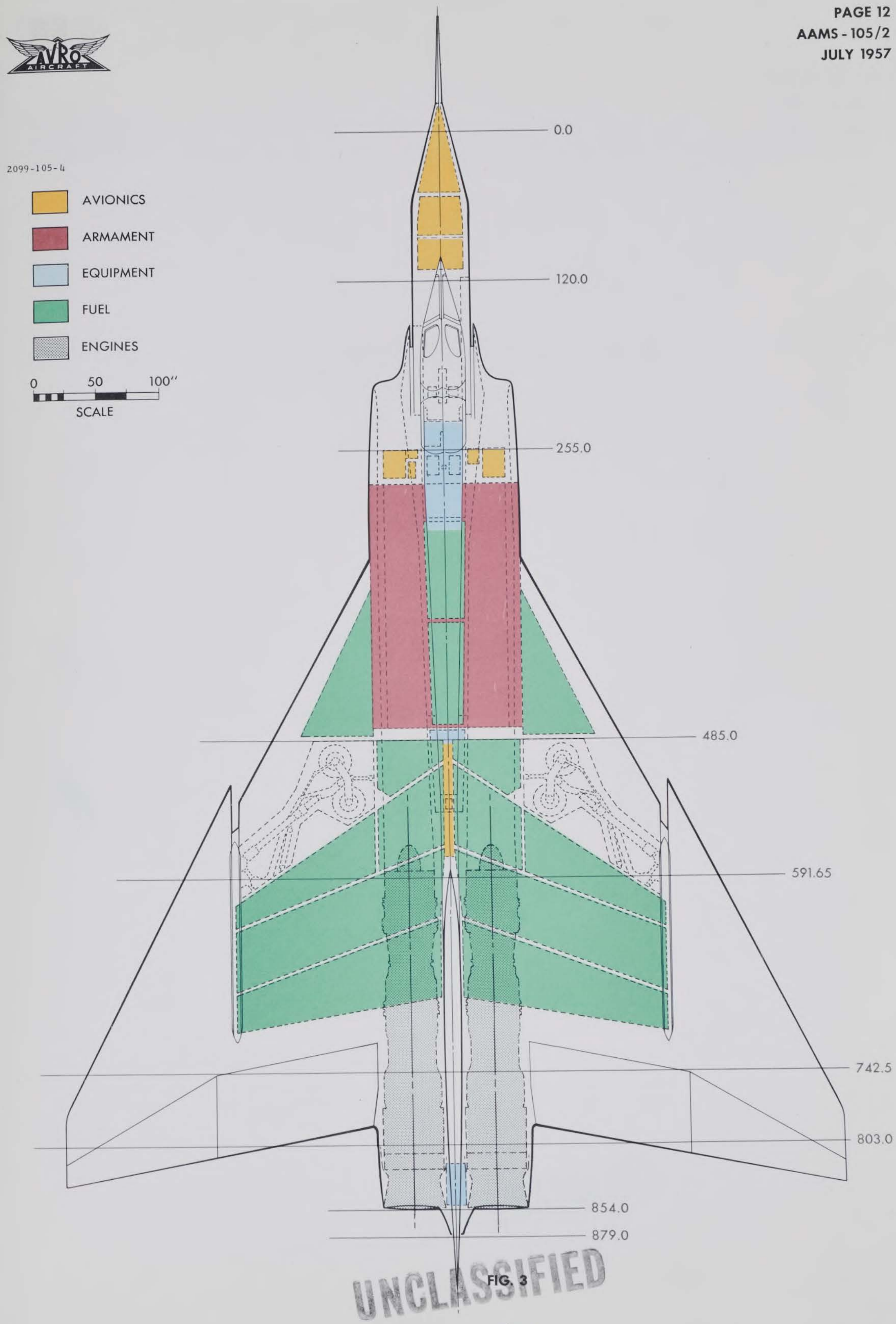

FIG. 3

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Performance Curves will be added when Iroquois engine performance becomes available.

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3.1.5 Weight and Balance

The estimated weight for the aircraft is as follows (For current weight status refer to latest issue of Avro Aircraft Report No. 7-0400-62).

3.1.5.1 Basic Weight

DESCRIPTION		WEIGHT lb	
STRUCTURE			18,678
Wing		10,029	
Fin & Rudder		1,024	
Fuselage	fwd. Sta. 255"	2,570	
	Sta. 255" - 485"	1,705	
	Sta. 485" - 591.65"	1,045	
	Sta. 591.65" - 742.5"	1,553	
	Sta. 742.5" Aft.	700	
	'Marry-up'	52	
LANDING GEAR			2,552
Main Landing Gear		1,902	
Main L/G Doors & Fairings		292	
Nose Landing Gear		334	
Nose L/G Door & Fairing		24	
POWER PLANT & SERVICES			10,628
Engines & Accessories		9,103	
Gear Box & Drives on Fuselage		282	
Engine Controls		32	
Gear Box, Starter & Drives on Engine		315	
Fire Extinguishing System		71	
Engine Mountings		132	
Fuel System		693	
FLYING CONTROLS GROUP			1,780
Mechanical Flying Controls		946	
Hydraulic Flying Controls		834	
EQUIPMENT GROUP			9,060
Instruments		46	
Probe		15	
Cockpit Pressure Sealing		5	
Oxygen System		24	
Ejector Seats		284	
Air Conditioning System		856	
Hydraulics Main System		643	
Cabin Insulation		14	
Brake Parachute		62	
Electrical System		1,259	
Low Pressure Pneumatics		56	
Surface Finish		100	
Intake De-icing Boots		52	

3.1.5.1 Basic Weight (Cont'd)

DESCRIPTION	WEIGHT lb	
EQUIPMENT GROUP (Cont'd)		
Radome Anti-icing	9	
Canopy Actuation	65	
Cabin Consoles	17	
Electronic Compartment Door Actuation	10	
Damping System	139	
Electronic System removable	559	
Electronic System fixed	621	
Emergency Ram Air Turbine	60	
Emergency Fire Protection	154	
Instrument Pack Structure	687	
Pack Instrumentation	1,750	
Flight Test Installations	1,323	
BASIC WEIGHT		42,448

3.1.5.2 Maximum Gross Weight

BASIC WEIGHT		42,448
Ballast		700
Operational Load (less fuel)		1,122
Crew	430	
Oil	139	
Alcohol for Radome De-icing	22	
Residual Fuel	218	
Oxygen Charge	13	
Engine Fire Extinguisher Fluid	25	
Water for Air-Conditioning	275	
OPERATIONAL WEIGHT EMPTY		44,270
Maximum Internal Fuel (2,492 gal @ 7.8 lb/gal.)		19,438
Maximum External Fuel (500 gal @ 7.8 lb/gal + drop tank)		4,248
MAXIMUM GROSS WEIGHT		67,956

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3.1.5.3 Combat Weight and Normal Gross Weight (Simulated)

DESCRIPTION	WEIGHT lb	
BASIC WEIGHT		42,448
Ballast		700
Operational Load		1,122
Crew	430	
Oil	139	
Alcohol for Radome De-icing	22	
Residual Fuel	218	
Oxygen Charge	13	
Engine Fire Extinguisher Fluid	25	
Water for Air Conditioning	275	
*Allowance for Fuel for Combat Mission		15,800
NORMAL GROSS WEIGHT		60,070
*Half Combat Mission Fuel (1,013 gals. @ 7.8 lb/gal.)		7,900
*COMBAT WEIGHT (Half Combat Mission Fuel)		52,170

*Rough estimate only - to be finalized on determination of Iroquois Performance.

3.1.5.4 Unit Weights

(a) Wing Group (Gross Area 1,225 sq. ft.) 8.187 lb/sq. ft.

(b) Vertical Tail (Gross Area 158.79 sq. ft.) 6.450 lb/sq. ft.

(c) Fuel System (Capacity 2,492 Imp. Gal.) 0.278 lb/Imp. Gal.

3.1.5.5 Balance

The C.G. limits of the aircraft are estimated to be: (Landing Gear Up or Down)

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

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

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3.1.6 Areas (Not to be used for inspection purposes)

Wing area (including ailerons, elevators, and 390.50 sq. ft. of fuselage, and not including 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)..........	14.37 sq. ft.

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

3.1.7.1 Wings (Reference Figure 14)

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 (Modified)
- 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.

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3.1.7.3 Vertical Tail

Span	12 ft. 10.50 in.
Chord - Root	19 ft. 0.00 in.
- Construction Tip	5 ft. 8.00 in.
Mean Aerodynamic Chord	13 ft. 6.41 in.
Airfoil Section	.0004-6-3.7 (Modified)
Sweep Back - Leading Edge	59.34 degrees
- Trailing Edge	33.08 degrees
- $\frac{1}{4}$ Chord	55.00 degrees
Aspect Ratio	1.04
Rudder - Span (average)	9 ft. 11.00 in.
- Chord (average percent vertical tail chord)	30.00%

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. 7.66 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

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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°	4.98 in.
Elevators: up	30°	aft6.63 in.
down	20°	fwd4.37 in.
Rudder: left and right	30°	fwd3.28 in.
		aft3.03 in.
Speed Brake	60°	-

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3.2 Construction

3.2.1 General Interior

A pressurized compartment for the accommodation of the crew and incorporating instruments, controls, and stowage 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 a pilot, and the rear cockpit shall be equipped to accommodate an Observer/AI. 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

The following order of precedence shall apply to materials used in construction of the airframe and contractor furnished equipment:

(a) Requirements issued by DND or approved by the RCAF as covered by CAP 479 Part 5; or

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

(c) RCAF approved Company specifications.

3.2.3 Standards

The following order of precedence shall apply to standard parts used in construction of the airframe:

(a) Standards issued or approved by the RCAF, as covered by CAP 479 Part 5; or

(b) ABC Standards to the extent agreed between the RCAF and the Company and as set forth in Avro Report GEN/STDS/4.

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

8 (d) RCAF 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 RCAF 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

In so far as practicable, the aircraft shall be free from objectionable vibration which might lead to malfunctioning of controls or equipment.

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.

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3.2.8 Processes

The following order of precedence shall apply to processes used in construction of the airframe and contractor furnished equipment incorporated in the various systems installed therein:

(a) Requirements issued by DND or approved by the RCAF as covered by CAP 479 Part 5; or

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

3.2.9 Finish

The finish on all parts and components shall be in accordance with RCAF approved Avro Aircraft Company Standard CS-D-2, or 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 Specification MIL-M-7911.

3.2.11 Pipeline Identification

70 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 RCAF and the Company as set forth in Company Reports listed in Appendix III.

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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 Arrow 2 aircraft.

3.2.15 Equipment

The following order of precedence shall apply to contractor furnished equipment incorporated in the various systems installed in the aircraft:

(a) Requirements issued by DND or approved by the RCAF as covered by CAP 479 Part 5; or

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

(c) RCAF approved Company Specifications.

Government supplied equipment, specified as a mandatory requirement by the RCAF, shall be installed without modification or adjustment, except to the extent of normal calibration or minor 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. Upon 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 RCAF and recommend such modification for such equipment as it may deem necessary, but shall not undertake such a program of modification as 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 aircraft shall be designed to possess aerodynamic characteristics such as to permit the accomplishment of the primary role as defined in RCAF Specification AIR 7-4.

3.3.1.1 Special Characteristics

The wing leading edge shall be slotted, extended and drooped (as described in paragraphs 3.1.7 and 3.5.2.4) to alleviate transonic "pitch-up" at high lift coefficients and to produce favourable air-flow conditions.

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 elevator trim required at altitudes above 45,000 feet shall be effectively reduced by automatic upward deflection of the ailerons, resulting in a further reduction in 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.

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 for engine cooling.

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 USAF Specification 1817.

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3.4 Structural Design Criteria

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4
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. All weights shall be in accordance with the Definitions stated in Section 6, rounded to the next highest 500 lb. to provide a stabilized value.

3.4.1 Limit Flight Load Factors

Limit flight maneuver load factors shall be based on a positive factor of 7.33 and a negative factor of 3.00 at a Stressing Weight of 47,000 lb. At weights greater than 47000 lb., limit maneuver load factors shall be assumed to be the product of the stressing weight load factor and 47,000 lb divided by the greater weight.

3.4.1.1 Gross Weight for Stress Analysis 52,500 lb.

	Clean Configuration	Missile Lowering and Firing
Maneuver		
Positive	+6.56*	+4.00
Negative	-2.69	-1.00
Gust (55 fps)		
Positive	+4.91	**
Negative	-2.91	

3.4.1.2 Maximum Gross Weight 68,000 lb.

	Clean Configuration	Missile Lowering & Firing	Aux. Tank Installed
Maneuver			
Positive	+5.07*	+4.00	+4.50†
Negative	-2.07	-1.00	-1.50†
Gust (55 fps)			
Positive	+4.22	**	+3.65‡
Negative	-2.22		-1.65‡

* The limit load factor shall decrease due to the effect of skin temperature rise as shown on flight envelopes Figures 15 to 21 inclusive.

** Limit gust factors for Missile Lowering and Firing shall be a gust factor equal to the configuration limit maneuver factor, at an appropriately reduced gust velocity.

† Up to a Structural Speed Limitation of Mach .95.

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3.4.1.3 Normal Gross Weight 60,500 lb.

		Clean Configuration	Missile Lowering and Firing
Maneuver	Positive	+5.69*	+4.00
	Negative	-2.33	-1.00
Gust (55 fps)	Positive	+4.43	**
	Negative	-2.43	

3.4.1.4 Maximum Landing Gross Weight 64,000 lb.

		Clean Configuration	Missile Lowering and Firing
Maneuver	Positive	+5.38*	+4.00
	Negative	-2.20	-1.00
Gust (55 fps)	Positive	+4.40	**
	Negative	-2.40	

3.4.1.5 Normal Landing Gross Weight 46,500 lb.

		Clean Configuration	Missile Lowering and Firing
Maneuver	Positive	+7.33*	+4.00
	Negative	-3.00	-1.00
Gust (55 fps)	Positive	+5.30	**
	Negative	-3.30	

* The limit load factor shall decrease due to the effect of skin temperature rise as shown on flight envelopes Figures 15 to 21 inclusive.

** Limit gust factors for Missile Lowering and Firing shall be a gust factor equal to the configuration limit maneuver factor, at an appropriately reduced gust velocity.

3.4.1.6 Flight Envelopes

In addition to the above the limit flight load factors for the aircraft in the clean configuration shall be as shown in the following flight envelopes:

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3.4.1.6 Flight Envelopes (Cont'd)

Sea Level	Figure 15 Page 40
10,000 feet:	Figure 16 Page 41
20,000 feet:	Figure 17 Page 42
30,000 feet:	Figure 18 Page 43
40,000 feet:	Figure 19 Page 44
50,000 feet:	Figure 20 Page 45
60,000 feet:	Figure 21 Page 46

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
Line K	=	Max. Attainable Load Factor, 30° Up Elevator Deflection
Line L	=	Max. Attainable Load Factor 20° Down Elevator Deflection
Line M	=	Elevator Hinge Moment Limitation, Positive
Line N	=	Elevator Hinge Moment Limitation, Negative

3.4.1.7 Load Factors in Roll

2 The limit flight maneuver load factors in roll 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.

3.4.1.8 Load Factors in Spin

1 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 Factors

105

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3.4.2.1 Limit Take-Off Load Factors

Paragraph subject to completion of landing gear tests.

3.4.2.2 Limit Landing Load Factors

Paragraph subject to completion of landing gear tests.

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 Crash Criteria

3.4.4.1 Ditching Conditions

Not applicable.

3.4.4.2 Emergency Landing Conditions

Under emergency landing conditions as specified below permanent deformations shall be permissible provided there shall be no tearing loose of seats or other structural components which might cause injury to occupants, or provided that crew eggress is not prevented.

119 The seats, seat installations, canopy and canopy actuating mechanisms, and supporting structure for cockpit equipment shall be designed to withstand enertia loads corresponding to ultimate load factors of 25 'g' forward, 4 'g' laterally, or 20 'g' vertically, applied either separately or together.

3.4.5 Ultimate Loads

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

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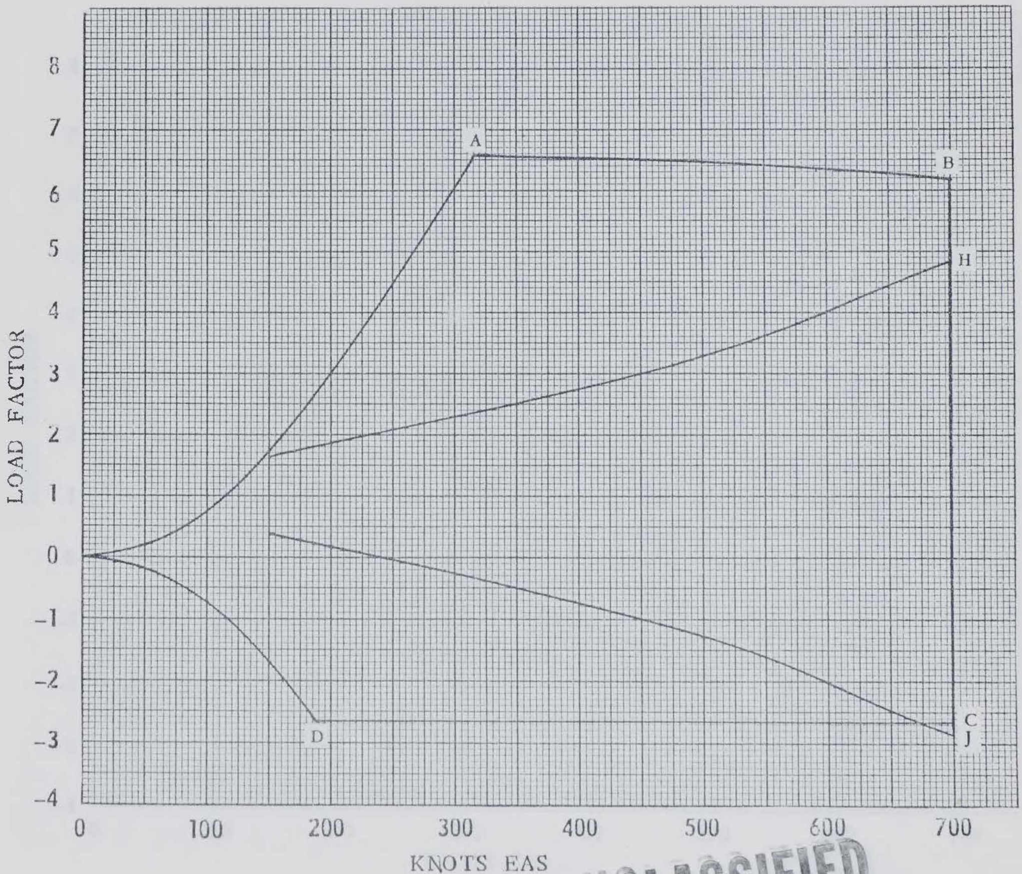
ARROW 2 FLIGHT ENVELOPE

SEA LEVEL

STRUCTURAL LIMIT MANEUVER LOAD FACTOR FOR A WEIGHT OF 52,500 LB. (BASED ON A STRESSING WEIGHT OF 47,000 LB.)



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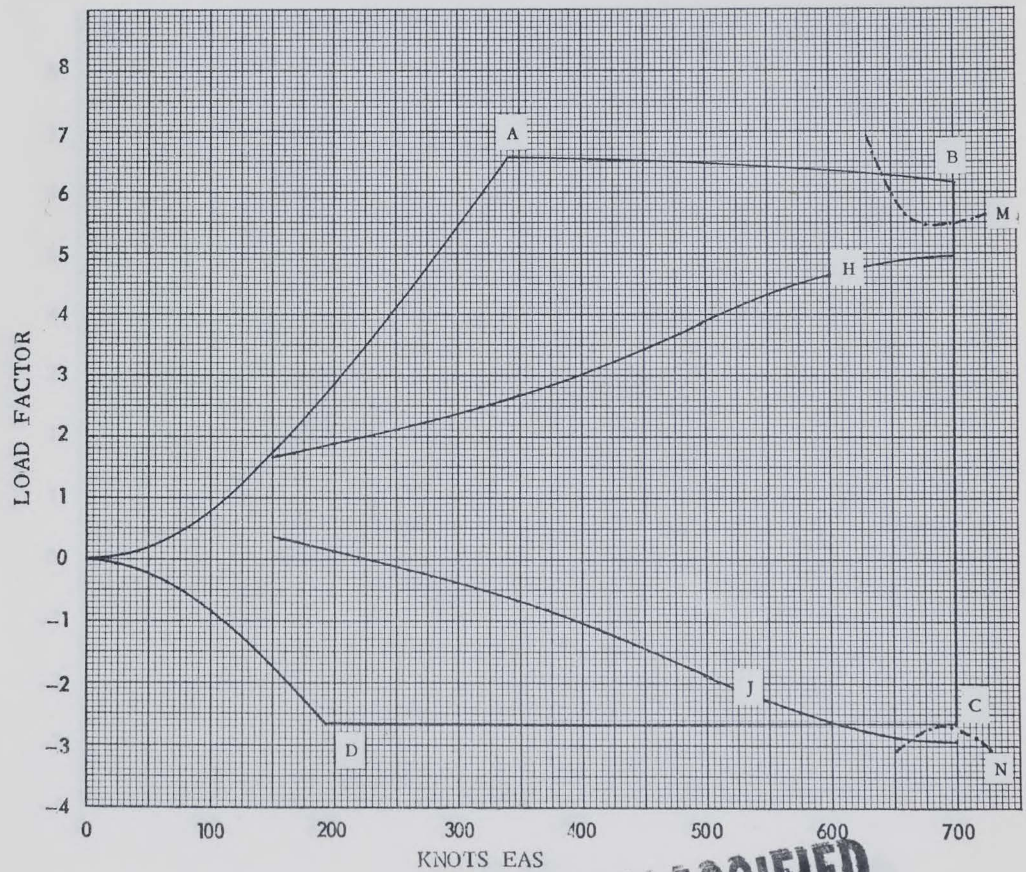
ARROW 2 FLIGHT ENVELOPE

10,000 FEET

STRUCTURAL LIMIT MANEUVER
LOAD FACTOR FOR A
WEIGHT OF 52,000 LB.
(BASED ON A STRESSING
WEIGHT OF 47,000 LB.)

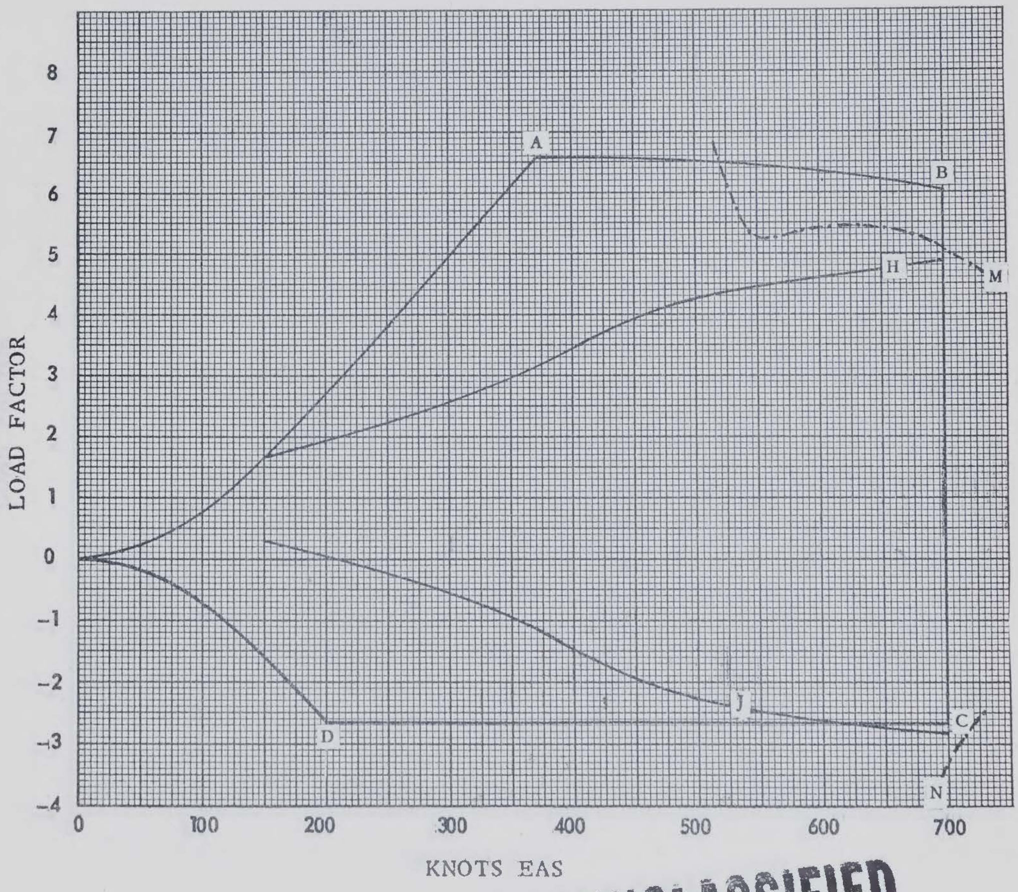


UNCLASSIFIED

ARROW 2 FLIGHT ENVELOPE

20,000 FEET

STRUCTURAL LIMIT MANEUVER
LOAD FACTOR FOR A
WEIGHT OF 52,500 LB.
(BASED ON A STRESSING
WEIGHT OF 47,000 LB.)

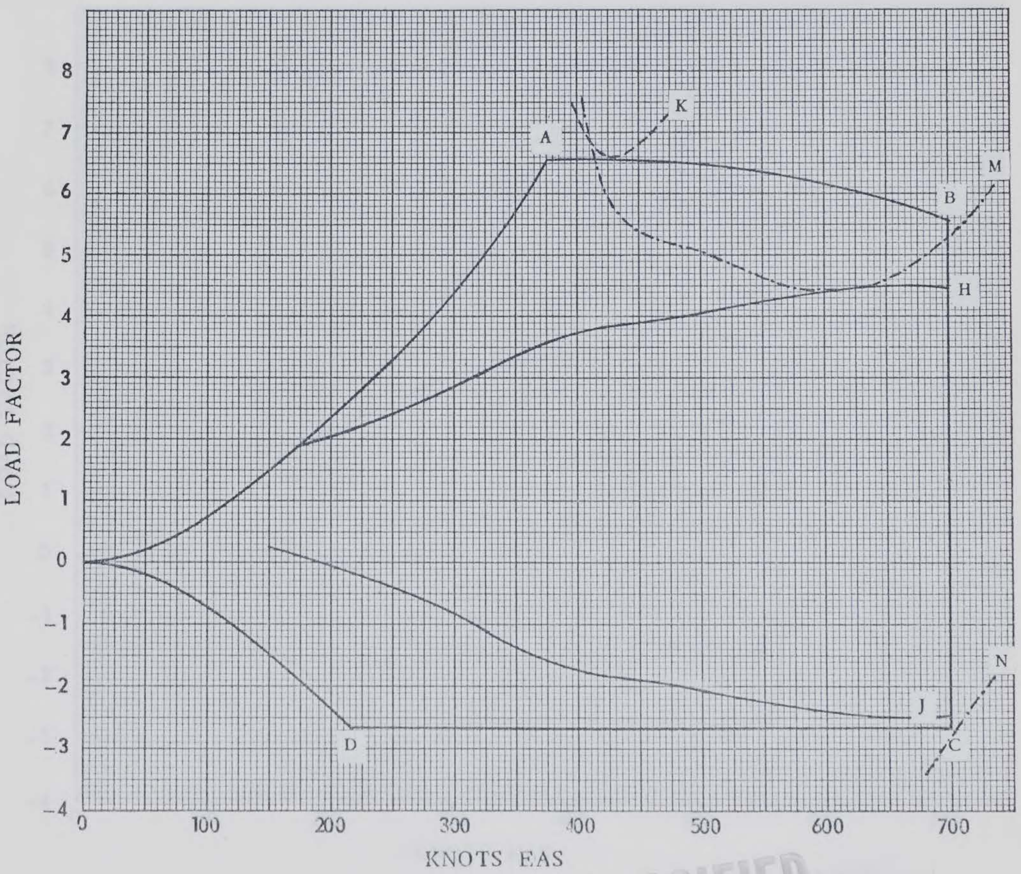


UNCLASSIFIED

ARROW 2 FLIGHT ENVELOPE

30,000 FEET

STRUCTURAL LIMIT MANEUVER LOAD FACTOR FOR A WEIGHT OF 52,500 LB. (BASED ON A STRESSING WEIGHT OF 47,000 LB.)

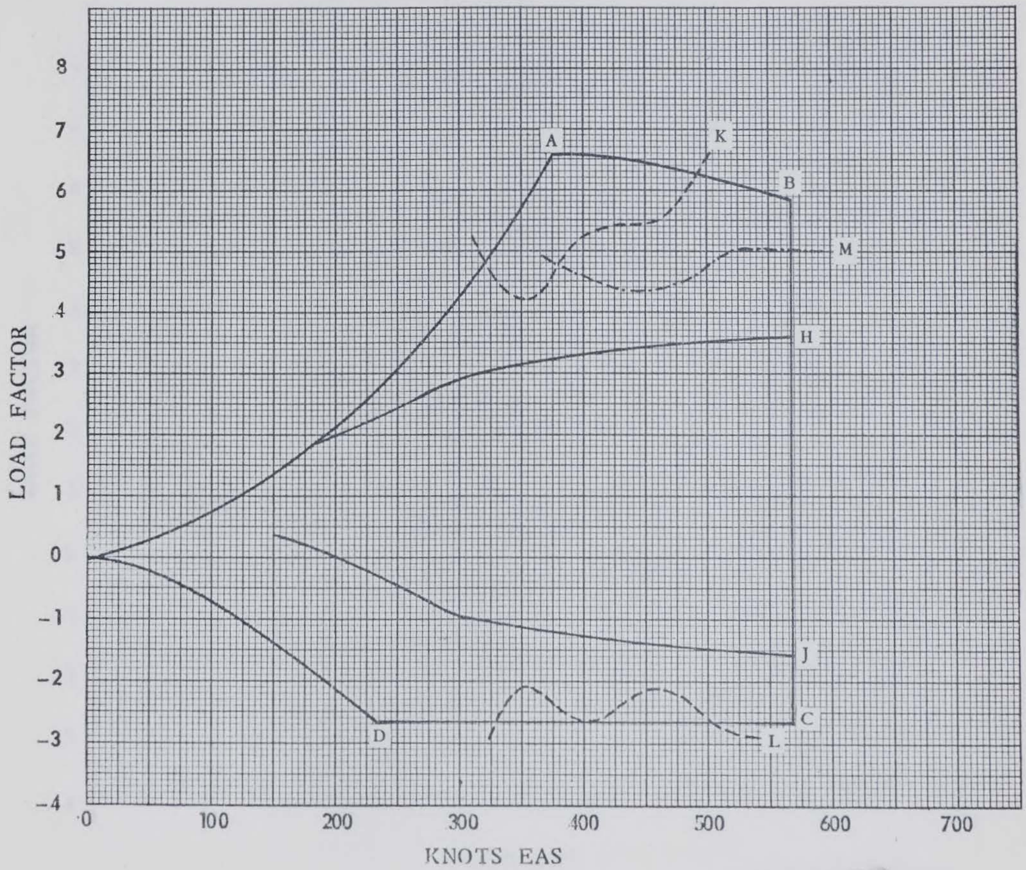


UNCLASSIFIED

ARROW 2 FLIGHT ENVELOPE

40,000 FEET

STRUCTURAL LIMIT MANEUVER LOAD FACTOR FOR A WEIGHT OF 52,500 LB. (BASED ON A STRESSING WEIGHT OF 47,000 LB.)



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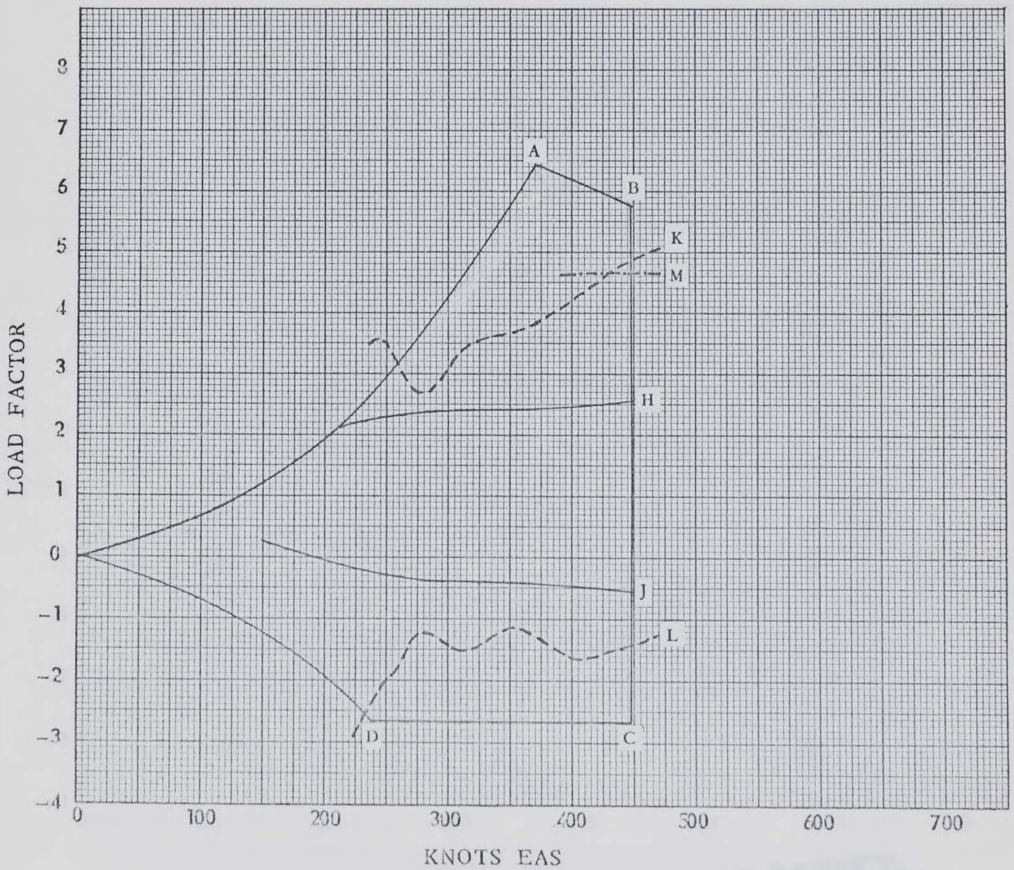
ARROW 2 FLIGHT ENVELOPE

50,000 FEET

STRUCTURAL LIMIT MANEUVER
LOAD FACTOR FOR A
WEIGHT OF 52,500 LB.
(BASED ON A STRESSING
WEIGHT OF 47,000 LB.)



UNCLASSIFIED

ARROW 2 FLIGHT ENVELOPE

60,000 FEET

STRUCTURAL LIMIT MANEUVER LOAD FACTOR FOR A WEIGHT OF 52,500 LB. (BASED ON A STRESSING WEIGHT OF 47,000 LB.)

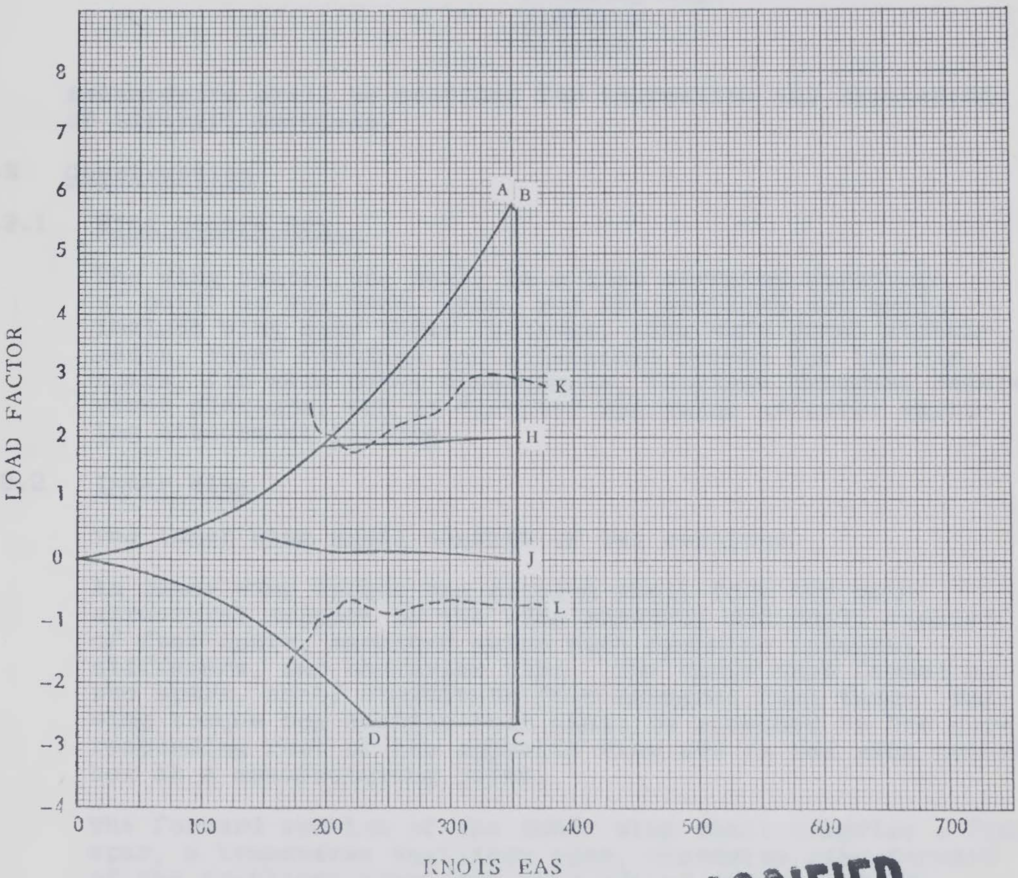


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FIGURE 21

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. The wing shall comprise a wing centre box, and six main sections on each side of the aircraft centre 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 Wing Centre Box

The wing centre box shall be a load carrying structure of built up box beam, beam, and rib sections and shall include wing spar shear fittings, wing skin panel attachments, shear fittings and attachment points for the fin spars, fin skin panel attachments, fuselage internal centre strut pick-ups, and an elevator hydraulic actuator earthing attachment fitting.

3.5.2.2 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 the wing centre 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

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3.5.2.2 Inner Wing (Cont'd)

attached at manufacturing joints to the main spar, the opposite wing root, and the centre fuselage which is indented into the delta configuration.

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

3.5.2.4 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 $6\frac{1}{2}$" 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.

3.5.2.5 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 centre 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 centre box, and the centre trailing edge. The inner trailing edge section shall house the elevator control unit.

A centre 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 centre trailing edge section shall be bolted to the inner and outer wing panels and to the outer trailing edge.

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3.5.2.5 Trailing Edge (Cont'd)

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 the ribs 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

5 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.

The angular motion of the aileron shall be 19° up and 19° down from the aileron neutral position. The centroid of the aileron area shall be 19.138 feet 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.

3.5.4 Aileron Tabs

Not applicable.

3.5.5 Lift and Drag Increasing Devices (Flaps)

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 centre 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

3.5.7 Elevator (Cont'd)

and shall be fully shrouded along the underside. The angular motion of the elevator shall be 30° up and 20° down from the elevators neutral position. The design of the flying control system and its power operation shall obviate the necessity for aerodynamic and static balance.

3.5.8 Elevator Tabs

Not applicable

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3.6 Tail Group

3.6.1 Description and Components

The tail group shall comprise a fin and rudder. Due to the 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 are installed on the trailing edge of the inner wing (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.

The main structural assembly shall comprise five spars, spanwise compression ribs, and ribs running normal to the rudder hinge line. Loads shall be transmitted to the wing centre box where the fin is attached at a manufacturing joint. A rudder control unit shall be installed in the fin forward of the main structural assembly rear spar. A detachable fin tip of fibrous material shall be installed to house radio antennas. A pitot static pressure head shall be mounted on the upper portion of the fin leading edge.

The rudder control linkage box shall be a built up assembly comprising machined skins, a rudder hinge spar, a compression spar, and companion rib and support fittings for the rudder control linkage. The control linkage box shall be bolted to the fin rear spar.

Access doors shall be provided for inspection and maintenance of the rudder control unit and aircraft services within the fin.

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3.6.6 Rudder

The rudder shall be of stressed skin construction and shall comprise a hingespar, an intermediate spar, and ribs running normal to the hinge spar line. The rudder shall be supported

3.6.6 Rudder (Cont'd)

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° 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.

3.6.7 Rudder Tab

Not applicable.

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3.7 Fuselage

3.7.1 Description

The fuselage shall be arranged below and extend forward of the wing and shall be designed to house two turbo-jet engines, armament, two crew members, 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 horizontally oblong cross-section. A pilot's V-type wind-shield 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, centre 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 bulkheads, frames and longitudinal stringers in the radar nose, nose fuselage and centre 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 centre struts between the fuselage main frames and inner wing spars, and through piano hinged underwing skin to fuselage skin joints.

Complete provision shall be made in the centre fuselage for the installation of a removable interchangeable armament pack structure comprising longitudinal beams, transverse beams, two box edge members, a partial inner skin, and an outer skin incorporating cut-outs to accommodate four Sparrow II missiles.

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3.7.3 Crew Stations

(28) The crew stations shall provide for a pilot and an observer/AI 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 insulation shall be installed to minimize heat transfer from the adjacent skin. The cockpit compartment shall be enclosed by a fixed windshield and two split clamshell type canopies.

(30) The pilot's windshield and canopy windows shall comprise optically flat panels of glazing, and the observer's canopy windows shall comprise curved panels of glazing. All glazing shall be of tempered glass incorporating transparent electrical heating elements (Reference paragraph 3.23.4).

A manually operated latch shall be installed in each cockpit to secure the respective canopy in the closed position. With the rear cockpit unoccupied, access for securing the rear cockpit latch shall be gained through the door in the cockpit separating bulkhead.

Each canopy shall be normally actuated by an electrically powered screw jack controlled by two OPEN-OFF-CLOSE switches. One switch for each canopy shall be located on the left hand console of the respective cockpit, and one switch for each canopy shall be located on the aircraft's exterior left hand side between the two canopy openings. A micro-switch shall be installed in the canopy securing latch to prevent electrical canopy actuation with the latch in the secured position.

(27) Emergency opening of the canopies shall be by means of gas pressure from a gas generating cartridge. Interior cockpit canopy emergency opening shall be initiated by means of:

(a) a handle in each cockpit, cable connected to the sear of the respective canopy actuating cartridge.

(b) action to initiate seat ejection.

Selection of emergency opening by means of either (a) or (b) above shall also release the canopy securing latch of the respective cockpit. Exterior emergency opening of the canopies shall be initiated by means of a cable connected to

3.7.3 Crew Stations (Cont'd)

the sears of both canopy actuating cartridges. The exterior end of the cable shall be stowed in an enclosed recess on the fuselage right hand side forward of the engine intake ramp.

25 The line of vision from the pilot's cockpit shall be directly forward to a line 12 1/2 degrees below the horizontal, aft to
26 120 degrees on both sides from a line directly forward, and with reasonable pilot movement, vertical vision on each side
29 to a line 30 degrees below the horizontal. 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)

Air Conditioning (Normal/ Emergency Air Supply)
Air Conditioning De-fog
Engine Air Bleed Shut-Off
Pressurization Dump
Canopy Opening (Normal)
Observer Bail-Out Warning
Engine Starting or Motoring
Engine Relight
Fire Warning and Extinguishing
Second Shot Fire Extinguishing
Engine Fuel (Normal/Emergency)
Low Pressure Fuel Cocks
Fuel Cross Feed
Fuel Transfer Forward
External Tank Jettison
Master Electrical
AC Generator
DC Reset
Taxi and Landing Lights
Navigation Lights
High Altitude Light
Day and Night Lighting
Master Warning Test
Master Warning Reset
Speed Brakes

Controls (Operative)

Air Conditioning
Press-to-Test Oxygen Pressure
Canopy Opening (Emergency)
Canopy Lock
Seat Firing
Seat Firing (Alternative)
Seat Adjustment
Manual Harness Release
Harness Reel
Leg Restraint Disconnect
Anti-g Valve
Emergency Oxygen Starting
Power Control (Throttles)
External Tank Manual Jettison
Cockpit Lighting ON-OFF and Intensity
Rudder Pedal Adjustment
Landing Gear
Drag Parachute
Parking Brake
UHF Radio AN/ARC-52
Intercom AN/AIC-10A
Radio Compass AN/ARN-6
IFF AN/APX-25A
J-4 Compass

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3.7.3.1 Pilot's Cockpit (Cont'd)

Switches (Operative)

Rudder Trim
Elevator and Aileron Trim
Elevator Trim Disengage
Flight Control:
  Damping System ON-OFF
  Damping System Engage
  Emergency Damping Engage
Nose Wheel Steering
Press-to-transmit
UHF Antenna Selector
UHF Transfer
UHF/IFF Emergency Test

Switches (Non-operative)

Anti-skid
Stores Jettison
Flight Control:
  Automatic Mode Disengage

Switches (Provision for)

Flight Control:
  Automatic Mode Selector
  Automatic Mode Function Selector
  Automatic Mode Mach/Altitude Hold
Armament:
  Missile Firing Trigger
  Missile Safe Arm
  Missile Mode Selector
  Missile Selector
  Launcher Retract/Optical Extend
De-ice or Missile Power Supply

Controls (Provision for)

Flight Indicator (Controls)

3.7.3.2 Observer/AI Cockpit

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

Switches (Operative)

Canopy Opening (Normal)
High Altitude Light
Press-to-transmit
Muting (Communication)
UHF Transfer
UHF/L Band Antenna Selector

Controls (Operative)

Press-to-Test Oxygen Pressure
Canopy Opening (Emergency)
Canopy Lock
Seat Firing
Seat Firing (Alternate)
Seat Adjustment

3.7.3.2 Observer/AI Cockpit (Cont'd)

Controls (Operative)

Manual Harness Release
Harness Reel
Leg Restraint Disconnect
Anti-g Valve
Emergency Oxygen Starting
Cockpit Lighting ON-OFF
and Intensity
UHF Radio AN/ARC-52
Radio Compass AN/ARN-6
Intercom AN/AIC-10A

Controls (Provision for)

Data Link AN/ARR-48
ECM Homer AN/ARD-501
IFF AN/APX-26 and 27
Doppler
Dead Reckoning
Position Data
Target Data
Radar
Radar Auxiliary
Radar In-flight-check
Radar Hand Control
Wind Data
Boresight

3.7.4 Cargo Compartments

Not applicable

3.7.5 Equipment Compartments

Compartments and bays listed in the following sub-paragraphs 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

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The radar nose shall comprise three compartments for electronic equipment. The forward compartment shall comprise a detachable radome constructed of organic material, and the forward portion of the fuselage structure. Access to the forward compartment shall be provided by removal of the radome, and by access doors at the aft end of the forward compartment. The two aft compartments shall comprise the

3.7.5.1 Nose Electronics Compartments (Cont'd)

fuselage structural area forward of the pilot's cockpit. Access to the aft compartments 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 Forward Fuselage Electronics Compartment

6

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

3.7.5.4 Air Conditioning Equipment Bay

An air conditioning equipment bay shall be located aft of the oxygen equipment bay and forward services bay. The bay shall be supplied with conditioned air. Access to the bay shall be provided by removal of a section of the dorsal fairing, an air outlet duct, and a shear panel.

3.7.5.5 Oxygen Equipment Bay

The oxygen equipment bay shall comprise the dorsal fairing area immediately aft of the rear cockpit. Conditioned air shall be supplied to the installed equipment. Access shall be gained through a door in the dorsal fairing.

3.7.5.6 Armament Pack Bay

The armament pack bay shall comprise a recess in the underside of the fuselage designed to permit the installation of interchangeable armament packs or instrumentation packs. Conditioned air shall be provided to the armament pack bay.

3.7.5.7 Aircraft Services Bays

The fuselage area aft of the nose wheel well and the cockpit rear bulkhead shall comprise a forward service

3.7.5.7 Aircraft Services Bays (Cont'd)

bay. Access shall be provided through a panel on the underside of the fuselage.

The fuselage area between the left and right hand air intake floating ducts and engines shall comprise a service bay. The forward region of the bay shall primarily house electrical equipment, and the aft region shall primarily house hydraulic equipment, airframe accessories gearboxes, and fire extinguisher bottles.

Access doors and panels for the bay shall be installed on the underside of the fuselage. Sections of the engine shroud shall be removable to provide additional access with engines removed.

3.7.5.8 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. Access to the compartment shall be provided by removal of a section of the dorsal fairing.

3.7.6 Speed Brakes

Two 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. Each brake shall be retracted and extended by an actuator powered by the utility hydraulics system. The brakes shall retract into sealed wells recessed into the underside of the fuselage.

3.7.7 Fuselage Power Plant Installation

Reference Section 3.10.

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3.8 Landing Gear

3.8.1 Description

(7) 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
(69) line at 50° to the aircraft centre line. The nose gear shall be steerable and shall retract forward into the nose fuselage. The hydraulic actuating system shall be designed to retract the gear, including door operation, in 5 seconds at -20°F and 30 seconds at -65°F. When completely retracted the 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 RCAF 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 strut shall comprise an integral transverse shaft pivoted at the front and main spars near the outer end of the inner wing. The main strut shall be braced by a drag strut and a telescopic downlock strut.

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. The hydraulic pressure available for normal brake operation shall be a maximum of 2500 psi, and for emergency operation a nominal 1500 psi (Reference paragraph 3.14.1.1.3).

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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 an accumulator shall permit three full applications of the brakes.

The main wheel brakes shall be applied automatically during the retraction cycle and released automatically when the landing gear is in the locked up position.

3.8.2.3 Tires

Tubeless tires (USAF 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 a hydraulic actuator until an uplock is engaged. During the retraction cycle, a mechanical linkage shall draw the shock absorber into the main landing gear 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 parallel to the aircraft longitudinal axis.

3.8.2.5.3 Locking

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

3.8.2.5.4 Controls and Indicators

68 A pilot operated landing gear retraction and extension selector lever 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 actuated by full extension of the shock absorbers. The actuation of the main gears and the main gear locks shall be 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 red 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, at altitudes below 10,000 ft. a flashing light, warning that both engine throttle levers are retarded to 1/3 full throttle or less, and the landing gear is retracted.

107 An indicator with one green and one red light for each landing gear unit shall be installed. The associated electrical circuit shall be designed to furnish indication as follows:-

(a) An individual green light indication for each landing gear unit when the unit is locked down (tri-light green indication when all three units are locked down).

(b) An individual red light indication for each landing gear unit when the unit is not locked up or down (tri-light red indication when all three units are unlocked.)

(c) Corresponding lights for each unit extinguished when each unit is locked in the up position (no indication when all units are locked up).

3.8.2.5.4 Controls and Indicators (Cont'd)

The indicator shall embody a duplicate set of three filament lamps for the green lights with a change-over switch on the indicator face for selection of either set of filament lamps.

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 nitrogen 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 axis end of the main landing gear. The main door shall be hinged parallel to the aircraft centre line and hydraulically actuated. The outer door shall be hinged parallel to the main gear axis line and actuated by a linkage to the main landing gear leg.

The main door shall be locked in, and unlocked from, the down position by a lock within the door actuator. The main door shall be locked in the up position by mechanically engaged locks which shall be releasable by hydraulic actuators.

The main door and door lock actuation shall be 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

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3.8.3.1 Description (Cont'd)

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 hinge on shafts supported by fittings projecting forward from the rear cockpit aft bulkhead. The strut shall be braced by a folding, lockable drag strut. A pneumatic spring strut shall be installed to assist shock absorber extension during gear retraction.

A hydraulic self-centering actuator shall be installed on the gear main strut and linked to the nose wheel suspension lever to provide for castoring with self-centering of the wheels, or for steering when steering is selected. The nose wheels shall castor, or be steer-
104 able, up to 55° on either side of the aircraft centre line. Shimmy damping restrictor valves shall be installed in the steering actuator hydraulic circuit.

3.8.3.2 Wheels

The wheels shall be demountable and shall be retained on a splined live axle by lockable axle nuts.

3.8.3.3 Tires

A tubeless tire (USAF 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 a 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 thus assuring positive landing gear positioning in the nose wheel well.

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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 sequenced with the nose landing gear door (Reference Section 3.14).

3.8.3.5.4 Controls and Indicators

The nose landing gear shall be controlled in conjunction with the main gear. The nose landing gear position and warning system shall be as described for the main gear (Reference paragraph 3.8.2.5.4).

3.8.3.5.5 Emergency Extension

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

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 shall be installed on the nose gear suspension lever, to prevent selection of steering unless the nose wheels are in a loaded attitude. The rudder pedals shall be mechanically linked to the steering control valve through a hydraulically operated clutch integral with the valve. Synchronization of the rudder pedals with 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 hydraulic system pressure.

3.8.3.7 Doors and Fairings

The retracted nose landing gear shall be enclosed within the 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. The door shall be locked in the up position by mechanically engaged locks which shall be releasable by a hydraulic actuator.

3.8.3.8 Inspection and Maintenance

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

3.8.4 Drag Parachute

3.8.4.1 Description

A FIST Ribbon Canopy drag parachute, complete with deployment bag and pilot chute, shall be stowed in a compartment in the top of the fuselage stinger between the two engine jet pipe fairings. The parachute pack shall be retained by two spring loaded doors which shall maintain the skin line when in the closed position. The doors shall retract to a position inside the adjacent skin surface when parachute deployment is selected.

An indicator flag shall be installed on the underside of the rear fuselage to provide parachute non-stowage indication.

3.8.4.2 Release Gear

36 Deployment and jettison of the drag parachute shall be controlled through a selector lever installed in the front cockpit. Selection of "Stream" shall mechanically release the spring loaded parachute retaining doors, and selection of jettison shall disconnect the drag parachute attachment cable.

The drag parachute attachment cable shall be secured to the aircraft structure by a shear pin which shall permit breakaway at a predetermined load.

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3.9 Surface Control System

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

(42) 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. Complete provision shall be made for an Automatic Mode of control utilizing manual, navigation, or weapon fire control input command signals. Pilot artificial feel systems shall be provided for both the Manual and Emergency Modes of control.

A system shall be installed to provide artificial damping. In the Manual Mode of control, the system shall provide damping about all three axes, stability augmentation in yaw, and sideslip minimization in maneuver. In the Emergency Mode of control the system shall provide yaw axis damping and stability augmentation in yaw. Space provision shall be made in the wing and fin structure for the installation of buzz damper units for each control surface.

Speed brakes powered by the Utility Hydraulic System shall be installed.

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 in the Manual and Emergency Modes of control.

Mode and function selection controls shall be in accordance with Radio Corporation of America Specification ...., and shall comprise an ON-OFF power supply control switch and a Manual Mode "engage" switch installed on the pilot's console. Complete provision shall be made for an Automatic Mode Selector switch, an ATTACK-NAVIGATION function selector, and a three position ALTITUDE-OFF-MACH hold switch. An Emergency Mode selection switch and an Automatic Mode disengage 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 suitably pressurized, and either mode has been selected. The Emergency Mode shall automatically become the effective mode of control in the event of failure of the Manual Mode.

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3.9.1 Primary Flight Control System (Cont'd)

Three warning lights shall be installed in the pilot's warning indicator panel, one to indicate that no mode of control has been selected, one to indicate selection of Emergency Mode of control, and one to indicate disengagement of the roll and/or pitch axes of the damping system, (Reference paragraph 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

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. The transducers shall transmit the pilot's input stick forces when in Manual Mode as electrical command signals through an amplifier to the elevator parallel (command) servo. The command signals shall be limited such that "g" load commands exceeding the structural integrity of the
38 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 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

The control column shall be linked to aileron actuator control valves by bell cranks, quadrants, cables and push rods; with stick force transducers installed in the linkage. The transducers shall transmit the pilot's input stick forces when in Manual Mode as electrical command signals through an amplifier to the aileron parallel (command) servo. The command signals shall be limited such that roll rates exceeding the structural integrity of the aircract 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. Upward deflection of both ailerons at altitudes above approximately 45,000 ft. shall be provided by electrically actuated bias of the aileron control quadrant bell crank linkage controlled by an aneroid switch.

3.9.1.3 Rudder

Co-ordinated rudder control in the Manual Mode and Emergency Mode of control shall be provided by the damping system (Reference paragraph 3.9.4.2).

3.9.1.3 Rudder (Cont'd)

A mechanical linkage comprising bell cranks, quadrants, cables, and push rods shall connect the rudder pedals to the rudder hydraulic actuator control valves to permit the pilot to over-ride the damping system rudder co-ordination function during maneuvers requiring unco-ordinated control.

3.9.1.4 Artificial Feel

Artificial feel shall be provided for the Manual Mode and the Emergency Mode of control. The Manual Mode system shall provide normal, and low speed (landing gear down) feel, with selection of landing gear down automatically engaging low speed feel.

With Manual Mode normal feel the stick force to trim the elevators in level flight shall be essentially zero, and the elevator stick force per "g" and the aileron stick force per unit rate of roll shall be essentially constant, for all flight conditions.

With Manual Mode low speed feel and Emergency Mode feel, the feel characteristics shall be such that the stick forces shall be proportional to control surface deflection, resulting in variable stick forces to trim the elevators in level flight, variable elevator stick forces per "g", and variable aileron stick force per unit rate of roll.

3.9.1.4.1 Manual Mode Artificial Feel

Manual mode normal and low speed artificial feel for the elevators shall be provided by feel springs and parallel servo reactions. The feel spring earthing point shall be monitored by the differential pressure across the piston of the parallel servo operating a trim motor such that the feel springs relieve the servo load.

Manual Mode normal and low speed artificial feel for the ailerons shall be provided by the parallel servos which shall provide feel reaction against control column movement.

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.

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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 pitch axis. The rudder feel shall be as described for the Manual Mode.

3.9.1.5 Cable Tensioning Devices

40 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

37 Vulnerability of the flying control system to anticipated types of aircraft damage shall be kept to the lowest degree
39 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 (Flaps)

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 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 a hydraulic selector valve. (Reference paragraph 3.14.1.1.4).

3.9.3 Trim Control Systems

Aircraft trim for both the Manual and Emergency Mode of control shall be effected by the actuation of an elevator and aileron trim selector button installed on the control

3.9.3 Trim Control Systems (Cont'd)

column grip, and a rudder trim switch and an elevator trim disengage 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 Mode of Control

108 Complete provision shall be made for the installation of an automatic flight control system conforming to the requirements of Radio Corporation of America Specification .... When installed, the system shall provide an Automatic Mode of control operable in conjunction with the normal damping system. The automatic mode shall utilize components of the Manual Mode of control. The system shall provide for heading hold, attitude pitch and roll hold, and for pilot selection of either Mach hold or Altitude hold. Pilot selection of either Mach hold or Altitude hold shall disengage the pitch attitude hold. The Automatic Mode shall accept command inputs from the following:

(a) Dead Reckoning System (Reference paragraph 3.17.2.10) and/or UHF Data link AN/ARR-48 (Reference paragraph 3.17.5.1).

(b) Weapon Fire Control System (Reference paragraph 3.18.3.1).

3.9.4.2 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.

3.9.4.2 Damping System (Cont'd)

A flying control system failure in the yaw axis shall automatically transfer the damping system to emergency operation.

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

Electrical power for operation of the damping system shall be provided by the main aircraft power supply (Reference paragraph 3.16.1.1). In the event of a double AC generator failure, electrical power for emergency damping shall be provided by the emergency AC generator.

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 maneuvers, in conjunction with the Manual Mode of control.

Provision shall be made in the rudder damping circuit to permit the pilot to produce limited intentional sideslip at high speed and up to 10 degrees sideslip in the low speed (landing gear down) configuration.

To safeguard the aircraft in the event of failure of the normal damping system, maneuver sensitive pitch and roll axis cut-out switches ('g' limiter and roll rate limiter respectively) shall be installed, and a maneuver sensitive yaw axis transfer switch to disengage normal damping and engage the yaw axis emergency damping shall be installed. The switches shall be set to function prior to the structural integrity limits of the aircraft being exceeded. It shall be possible to re-engage 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 at 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 3.13.2.1), shall be utilized with data from the damping system flight sensing instruments (gyros, accelerometers) for scheduling of aileron, elevator, and rudder control signals. These scheduled signals shall be continuously transmitted by magnetic amplifier to the appropriate servos operating the control surfaces. In response

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3.9.4.2.1 Normal Damping (Cont'd)

to the applied signals the servos shall operate the hydraulic valves which control the surface hydraulic jacks resulting in adjustment of the control surfaces according to the sensed aircraft stability requirements, and pilot's commands.

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 and rear fuselage shall form an integral part of the fuselage structure and shall house two power plants.

3.10.2 Construction

Construction shall be in accordance with paragraph 3.7.2.

3.10.3 Engine Mounts

The engine mounts of each engine shall provide two planes of attachment for securing the engine to the aircraft structure. Two mounts in a forward plane and three mounts in the rear plane shall be designed to carry all engine loads.

The mounts in the forward mounting plane of each engine shall comprise an upper centre spigot designed to accept longitudinal and side loads, and an inboard suspended side support strut designed to carry vertical loads. The upper centre spigot, housed in the wing structure, shall engage a fitting on the engine vertical centre line. The inboard side support strut shall be suspended from an attachment on the wing structure and shall mate with a fitting on the engine horizontal centre line.

The mounts in the rear mounting plane of each engine shall comprise one upper centre lateral strut designed to accept side loads, and two suspended side support struts designed to carry vertical loads. One end of the lateral strut shall attach to a supporting bracket on the wing centre box and the other end shall mate with an attachment on the engine vertical centre line. Each side support strut shall be suspended from fittings located between two beams in the centre trailing edge region and the lower ends of the struts shall pick up on the engine adjacent to the engine horizontal centre line.

The mounting provision for each engine shall be designed to permit all adjustments necessary for correct engine installation and alignment. The mounts shall be designed to permit linear expansion between the front and rear engine mounting planes.

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3.10.3 Engine Mounts (Cont'd)

Access openings shall be provided on the wing upper surface to facilitate insertion and removal of all engine mounting struts except the inboard forward mounting strut. The inboard forward mounting strut shall be accessible through an access door on the underside of the fuselage.

3.10.4 Vibration Isolation

Not applicable.

3.10.5 Fire Walls

48 45 Each power plant installation shall be enclosed by a titanium shroud providing isolation of the hot zone (Zone 2) of the Iroquois engine from the fuselage (Reference paragraph 3.11.2). The wet zone (Zone 1) of the Iroquois engine shall be isolated from the hot zone (Zone 2) by a titanium shroud integral with the engine.

3.10.6 Cowling and Cowl Flaps

Not applicable.

3.10.7 Inspection and Maintenance

47 Access doors shall be provided for inspection, maintenance, removal and installation of engines and accessories. Additionally the nacelle tail cones and the stinger shall be removable for engine change.

3.11 Propulsion

The propulsion system shall be designed to the requirements of RCAF 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 Iroquois turbo-jet engines, each engine having the following thrust ratings in accordance with Orenda Engine Specification EMS-8: a static sea level military thrust rating of 20,000 pounds and a maximum thrust rating, with afterburners, of 25,000 pounds.

3.11.2 Engine Installation

Titanium shrouds supported by aluminum formers shall form a tunnel for each engine.

All services shall enter the engine tunnel in the region of the access doors in the underside of the engine bay, and shall be quickly detachable to facilitate engine removal.

Engine mounts shall be in accordance with paragraph 3.10.3.

49 Engine installation and removal shall be carried out using
43 an engine stand. Each engine shall be guided from the engine stand into the mounting position by means of an engine roller which runs on a longitudinal rail installed on the outboard side in the engine tunnel. Securing of the engine mounts shall lift the engine sufficiently to clear the rail and permit removal of the roller.

3 11.3 Engine Driven Accessories

3.11.3.1 Description

An AC generator, with a constant speed drive, shall be mounted on the accessories pad at each engine inlet face. An oil system for a constant speed drive fluid coupling and for generator cooling shall be integrated with the engine oil system. Cooling shall be provided for the system oil by heat exchangers (Reference paragraph 3.11.6.2)

The power take-off located on the underside of each engine shall be utilized to power a remote gear box through a drive shaft. The drive shaft to each gear box shall incorporate a quick disconnect at each end.

3.11.3.2 Remote Gear Boxes and Drives

Two aircraft accessories gear boxes shall be installed in the service bay between the engines. Each gear box shall be shaft driven from the adjacent engine power take-off. The two gear boxes shall each drive three hydraulic pumps and provide a power take-off to drive a shaft to a fuel booster pump. Cooling shall be provided for gear box oil by heat exchangers (Reference paragraph 3.11.6.2).

3.11.4 Air Induction System

3.11.4.1 Description and Components

44

An air intake shall be located outboard on each side of the crew stations. Each shall be approximately "D" shaped and external compression shall be achieved by a 12° wedge shape ramp attached to the side of the fuselage. Each duct shall diverge from 6 square feet at the minimum throat area (approx. 7.5 inches aft of the inlet face) to 7.06 square feet at a station 10.12 feet aft of the minimum throat area, then hold a constant diameter circular section back to the engine 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.

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. At speeds below Mach 0.5 (approx.) the annular area between the end of the afterburner exhaust nozzle and the engine shroud shall serve to eject cooling air.

3.11.5 Exhaust System (Cont'd)

At speeds above Mach 0.5 (approx.), the annular area around the afterburner nozzle shall serve as an exit for ram cooling air.

3.11.6 Cooling System

3.11.6.1 Engine Cooling

At speeds less than Mach 0.5 (approx.) ventilating and cooling air shall be drawn into the engine bay through spring loaded, inwardly opening doors by the ejector action of the exhaust system annular nozzle (Reference paragraph 3.11.5). The doors shall be installed in the sides and undersides of the fuselage adjacent to the engine compressor and turbine regions. At speeds in excess of Mach 0.5 (approx.), gills installed in the tunnel upstream from the forward periphery of the engine shall be opened by the differential pressure between the intake tunnel and the air intake duct to admit ram cooling air. The ram cooling air shall flow around the engine and accessories and leave the engine bay via the exhaust system annular nozzle.

3.11.6.2 Heat Exchangers

A fuel cooled heat exchanger, comprising eight separate cooling segments, shall be installed in the service bay. Two segments shall be commoned and utilized to cool the 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 oil of each of the two constant speed drive and generator cooling oil systems, and one segment shall be utilized for cooling the oil of each of the two aircraft accesories gear boxes.

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. At speeds above Mach 0.5 (approx.) the heat exchangers shall be cooled by air directed through spring loaded doors installed on the lower surface of each engine intake duct forward of the engine face. The left hand heat exchanger units shall be utilized to cool the hydraulic oil of one of the flying control hydraulic systems, constant speed

3.11.6.2 Heat Exchangers (Cont'd)

drive and generator cooling oil, and aircraft accessories gear box oil. The right hand units shall provide cooling for the second flying control hydraulic system oil, constant speed drive and generator cooling oil, aircraft accessories gear box 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. Low oil pressure warning lights shall be installed in the pilot's warning indicator panel.

For further details of engine lubrication, see Orenda Engine Specification EMS-8.

3.11.8 Fuel System

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

60 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.

61 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. One tank in each sub-system shall be utilized as a main (collector) tank.
58 Fuel shall be transferred from auxiliary tanks to main tanks by means of tank pressurization, and from the main tanks to the engine feed manifolds by an engine driven booster pump submerged in each main tank. In the event of pump failure, tank pressurization shall provide fuel flow through a by-pass around the inoperative pump.

An electrical sequencing control system for each fuel sub-system shall comprise a control unit, detector relays, liquid level sensors, and fuel-no-air valve override solenoids which shall be installed to sequence fuel transfer from the auxiliary tanks to the main tank.

3.11.8.1 Description and Components (Cont'd)

The control unit shall be utilized to establish and maintain the sequence of tank usage. The fuel-no-air valve override solenoids shall be energized by the control unit to maintain the valves in the closed position, with the exception of the fuel-no-air valve override solenoid of the first auxiliary tank in the sequence which shall remain de-energized. The fuel-no-air valve override solenoid of the following auxiliary tanks in sequence shall be de-energized, in turn, by accepting signals from the liquid level sensors, through the detector relay, of the preceding tank in the sequence.

The fuel content of the collector tank shall be monitored such that if the content is below 90% of the collector tank capacity the solenoid override of the fuel-no-air valves of the next auxiliary tank in sequence shall be de-energized, permitting fuel flow from two tanks simultaneously. If the content of the collector tank is below 70% of capacity, fuel flow shall be provided simultaneously from three consecutive tanks in the sequence.

A switch shall be installed on a master refueling panel (Reference paragraph 3.11.8.12.1) to provide for ground selection of either a Normal or an Alternative sequence.

In the event of power input failure, all fuel-no-air valve override solenoids shall be de-energized permitting fuel transfer from all auxiliary tanks.

59 A manually selected cross-feed shall be installed between the engine feed lines to permit transfer of fuel from either sub-system to both engines, or in the event of single engine operation, from either sub-system to either engine.

Fuel shut-off valves shall be installed adjacent to the engine fire walls to provide for isolation of each engine. Switches shall be installed in the front cockpit for control of the firewall fuel shut-off valves.

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3.11.8.2 Fuel Specification and Grade

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

3.11.8.3 Fuel Tanks

3.11.8.3.1 Internal Tanks

57 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 fuse-
64 lage tank. One wing tank in each system shall serve as a main (collector) tank.

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

3.11.8.3.2 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.	Number Of Tanks	Gross Capacity Of Tank Imp. Gal	Useable Capacity Of Tank Imp. Gal.	Total Useable Fuel Capacity Imp. Gal
1 (Fuselage)	1	283	252	252
2 (Fuselage)	1	280	254	254
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
				2492
External (Fus.)	1	525	500	500
				2992

3.11.8.4 External Fuel Tank

Complete provision shall be made for the carriage of an external drop tank. When installed, the tank shall be hung from the fuselage lower surface by a forward bomb slip release attachment and an aft release linkage. (Reference paragraph 3.11.8.16). The drop tank shall be braced by two floating compression struts, one attached to the upper arc of each side of the tank. Fuel shall be transferred from the drop tank to the main (collector) tanks through a dry disconnect valve and a spring loaded coupling with fuel transfer effected by pressurized air from the fuel pressurization system.

Transfer of fuel shall be controlled by the sequencing control system (Reference paragraph 3.11.8.1), with the drop tank emptying first in sequence. When the drop tank is installed, the sequencing control unit shall close the fuel-no-air valves of the first auxiliary tanks in sequence by energizing the fuel-no-air valve override solenoids. A liquid level sensor shall be installed in the external drop tank to; de-energize the fuel-no-air valve override solenoids of the first auxiliary tanks permitting sequence continuation, close the pressurization supply air shut-off valve, and provide cockpit indication when the drop tank is empty.

A fuel filler cap incorporating a manual air release valve, designed to prevent cap removal until tank pressure has been released, shall be installed on the tank forward upper surface.

3.11.8.5 Piping and Fittings

55

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 vapour, and fuels to the specifications designated in paragraph 3.11.8.2.

3.11.8.7 Strainers and Filters

An eight mesh strainer shall be installed at the inlets of each booster pump. A 200 mesh filter shall be installed in the feed line to each engine.

3.11.8.7 Strainers and Filters (Cont'd)

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

3.11.8.8 Quantity Gauges, Flowmeters, and Indicators

53

A capacitance type fuel quantity gauging system which utilizes sensors separate from the aircraft fuel sequencing system, shall be installed. Two quantity gauges, indicating in pounds the quantity of fuel in each subsystem, shall be installed in the front cockpit.

Six warning lights shall be installed in the front cockpit to provide indication of the following:

(a) Low fuel in right-hand main (collector) tank
(b) Low fuel in left-hand main (collector) tank
(c) Fuel system sequencing or transfer failure
(d) Engine fuel low pressure (Reference para. 3.11.10.2)
(e) Drop tank empty
(f) Drop tank jettison failure

3.11.8.9 Purging and Explosion Suppression System

54

Not applicable

3.11.8.10 Pressurization

The fuel system shall be pressurized utilizing air from the pneumatics system (Reference paragraph 3.15.1.2) to transfer fuel, to forestall fuel boiling at altitude and to provide pressure for defueling.

All fuel tanks except the main (collector) tanks shall be pressurized at 19 psia (nominal). Pressure regulating valves shall be installed to control tank pressure. A low pressure air sink shall be installed in each subsystem to provide a means of extracting air from the associated main tank under low altitude high rate transfer conditions. Engine bleed air shall be passed through a venturi to produce the low pressure sink.

Pressure relief valves which vent to atmosphere shall be installed in the main pressurization lines to the fuel tanks to prevent over-pressurization in the event of a pressure regulating valve failure. Flow limiters shall be installed in the main pressurization lines to the fuel tanks to limit the flow of air to the air pressure regulators to within the capabilities of the corresponding air pressure relief valve.

3.11.8.10 Pressurization (Cont'd)

The pressurization lines to individual tanks shall be appropriately sized to forestall excess spillage of air in the event of tank damage.

A negative "g" and low level air admission valve shall be installed at each main tank inlet to permit the entry of pressurization system air during final emptying of the tanks, and also to permit entry of air during periods of negative "g".

3.11.8.11 Vent System

Outward venting for the auxiliary tanks, and the external drop tank when installed, shall be through the main pressurization line relief valve (Reference paragraph 3.11.8.10).

A fuel-level-sensitive air release valve shall be installed in each main (collector) tank to vent accumulated air admitted by the required function of negative "g" and low level air admission valves (Reference paragraph 3.11.8.10).

It shall be possible to connect ground equipment at the overboard vent orifice to provide for vapor removal during refueling.

3.11.8.12 Refueling System (Ground)

3.11.8.12.1 Internal Tanks

(51) The refueling system shall provide for pressure refueling (and defueling) of internal tanks. Two adaptors, one installed in each main landing gear well
(52) shall mate with refueling nozzles Type D1 (MIL-N-5877A). The adptor in each main landing gear well shall connect to the normal fuel transfer lines in each sub-system. The system shall permit ten minute refueling to the Combat Mission fuel load specified
(114) in paragraph 3.1.5.3.

Controlsand indicators installed on a panel adjacent to the left-hand speed brake and a switch located adjacent to each refueling adaptor, shall provide for selection and indication of the refueling or defueling operation.

(65) Control selection shall be provided for full refuel, partial refuel 1, and partial refuel 2. With the selection of "full refuel", shut-off valves shall open to permit filling each sub-system to capacity through the respective adaptor. With the selection

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3.11.8.12.1 Internal Tanks (Cont'd)

of "partial refuel 1", shut-off valves shall open to permit filling of all tanks except right-hand tank number 1 and left-hand tank number 7. With the selector switch set at "partial refuel 2" shut-off valves shall open to permit filling of all tanks except right-hand tanks number 1 and 3 and left hand tanks number 7 and 8.

3.11.8.12.2 External Tanks

A refueling point shall be provided on the drop tank. Provision shall be made for automatic release of tank pressure prior to removal of the tank filler cap (Reference paragraph 3.11.8.4).

3.11.8.13 Refueling System (In Flight)

Not applicable.

3.11.8.14 Drainage

50 Combination condensate and drain valves shall be installed at the low point in each wing tank, except tank number 4, to permit ground draining of fuel and ground purging of water from each tank. Wing tank number 4 and the fuselage tanks shall be provided with condensate drain valves only. A drain plug shall be provided in the external dropable tank. Each booster pump shall be provided with a seal drain connected to the vent system.

3.11.8.15 Defueling Provisions (Ground)

3.11.8.15.1 Internal Tanks

Defueling shall be accomplished through the two fuel servicing adaptors. Fuel from the internal tanks, in the selected usage sequence shall be transferred to the adaptors through the normal fuel transfer lines by pressurizing the tanks from a ground supply. After defueling of the auxiliary tanks is complete, fuel shall be removed from the main tanks by suction of the ground service unit. All valves shall be appropriately positioned by selection of "defuel" on the control panel (Reference paragraph 3.11.8.12.1).

3.11.8.15.2 External Tanks

Defueling shall be accomplished through the drain outlet on the bottom of the tank (Reference paragraph 3.11.8.14).

3.11.8.16 Fuel Jettisoning

3.11.8.16.1 Internal Tanks

Not applicable.

3.11.8.16.2 External Tank

An external tank jettison switch shall be installed in the front cockpit to electrically release the forward bomb slip attachment, permitting the forward end of the tank to drop. The rear attachment linkage shall be automatically released by downward movement of the tank.

A mechanical release, operated by a control in the front cockpit, shall be installed for release of the bomb slip attachment in the event of failure of the electric release.

The tank shall be automatically jettisoned:

(a) by depression of the stores jettison switch; or

(b) by depression of the missile firing trigger during attack.

Cockpit indication of tank jettison failure shall be provided.

3.11.8.17 Inspection and Maintenance

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 controls for each power plant shall comprise an engine starting switch, an engine relight button, and a throttle lever.

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3.11.10.2 Engine and Afterburner Control

(24)

The throttle levers shall be mounted on a quadrant on the left-hand console and shall provide for selection of a full range of powers with positions for "off", "ground idle", "flight idle", "military", and "maximum" thrust.

Initial movement of the throttle levers shall open the high pressure fuel cocks and complete a ground circuit for the engine starting control relay. Forward movement of the throttle lever through a spring gate at the "military" position shall be required for selection of afterburning. Rearward movement of the levers through a spring gate at approximately two-thirds throttle shall be required to cut out afterburning. Rearward movement the throttle levers through a detent, from the "flight idle" position, shall be necessary to permit selection of "ground idle".

Variation in afterburner thrust with full engine thrust shall be possible between the "military" and "maximum" throttle positions. Variation of engine and afterburner thrust, with afterburning selected, shall be possible between the two-third throttle spring gate and the "military" position. With afterburner selected unrestricted movement shall be permitted between the two-third throttle spring gate and the maximum position.

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

A normal/emergency switch shall be installed in the front cockpit to provide the pilot with emergency (manual) control of the engine fuel metering in the event of failure of the engine automatic fuel metering controls. A warning light shall be installed in the front cockpit to indicate that the emergency (manual) fuel control valve has been actuated to the emergency position. 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

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Two START-OFF-MOTOR switches shall be installed on the right-hand console in the front cockpit. With the throttle

3.11.10.4 Starter Controls (Cont'd)

levers positioned (Reference paragraph 3.11.10.2), selection of "start" shall permit the electrical power supply circuits to the starting external air supply control valves and to the engine igniter systems to be energized. 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. Selection of "motor" shall permit use of the ground starting unit, and rotation of the engine without ignition, for ground test.

A relight button shall be installed in each throttle lever to permit relighting the engines in flight within the relight flight envelope.

3.11.10.5 Propeller Controls

Not applicable.

3.11.10.6 Cooling Air Control

(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 installed on each engine. The starters shall be powered from a ground source and shall be capable of meeting the starting time limit specified in the scrammble requirements of paragraph 3.1.3.1. Automatic auick disconnects shall be provided for the ground air supply.

3.11.12 Propeller

Not applicable.

3.11.13 Rocket Propulsion System

Not applicable.

3.12 Auxiliary Power Units

3.12.1 Ram Air Turbine

A ram air turbine power pack shall be installed in a well below the left engine air intake duct. The turbine shall be equipped with a constant speed propeller and shall be utilized, at speeds above 140 kts. EAS, to power a flying control hydraulic system emergency pump and an emergency AC generator. The unit installation shall be contained within the faired lines of the fuselage, and shall be extended horizontally into the slipstream by an actuator powered by the utility hydraulic system. Retraction of the unit shall be possible by ground operation only.

3.13 Instruments and Navigational Equipment

Instrument arrangement shall be as agreed upon by the RCAF, the Radio Corporation of America, and the Company.

3.13.1 Instruments

3.13.1.1 Pilot's Instruments

3.13.1.1.1 Flight Instruments

Mach Meter (Limit, Command, Actual)
Air Speed Indicator
Rate of Climb Indicator
Turn and Bank Indicator
Pressure Altimeter
Accelerometer
Flight Director Attitude Indicator

3.13.1.1.2 Navigation Instruments

Clock
Standby Magnetic Compass
Integrated Destination Indicator

3.13.1.1.3 Tactical Navigation Instruments

Flight Indicator (Radar Scope) (Provision only)
Command and Target Altitude Indicator (Provision only)

3.13.1.1.4 Engine Instruments

115 Combined Percentage Engine Thrust, Percentage Afterburner Thrust, and Exhaust Temperature, Indicator (2)
Fuel Contents Indicator (2) (Ref. Para. 3.11.8.8)

3.13.1.1.5 Miscellaneous Instruments

Cabin Altimeter (Ref. Para. 3.22.1.1.1)
Oxygen Quantity Gauge (Ref. 3.21.1.4)
Landing Gear Position Indicator (Ref. Paras. 3.8.2.5.4, and 3.8.3.5.4)

UHF Channel Selection Indicator

3.13.1.2 Observer's Cockpit

3.13.1.2.1 Navigation and Tactical Navigation Instruments

True Airspeed Indicator (Provision only)
Altitude Data Counters (Provision only)
Integrated Destination Indicator
Clock
Total Fuel Indicator (Provision only)
Oxygen Quantity Gauge

3.13.1.2.1 Navigation and Tactical Navigation Instruments (Cont'd)

ECM Homer Indicator (Provision only)
Flight Indicator (Radar Scope) (Provision only)
Range and Range Rate Meter (Provision only)
Position Data Indicator (Provision only)
Target Data Indicator (Provision only)
Wind Data Indicator (Provision only)

3.13.2 Air Data System

An air data system comprising pitot-static, relative wind sensing, total temperature sensing, and air data computation facilities shall be installed to provide air data information for a fire control system, an Automatic Mode of control, the damping system, and for cockpit presentation.

A probe to provide for the installation of a pitot-static head, an 'Alpha' (pitch) vane, and a 'Beta' (yaw) vane shall be installed on the radar nose. A pitot-static probe shall be installed on the fin upper leading edge.

3.13.2.1 Pitot-Static System

A pitot-static system comprising a nose boom providing one source of pitot pressure and two sources of static pressure, and a fin probe providing pitot and static pressure shall be installed.

Pitot pressure shall be supplied to the air data computer, indicated airspeed indicator, and normal damping system. One nose static pressure source shall supply the air data computer, front cockpit altimeter, and normal damping system. The second nose static pressure source shall supply the rate of climb indicator, indicated airspeed indicator, rear cockpit altitude data computer, and cockpit pressure regulators.

Pitot and static pressure from the fin probe shall be supplied to the emergency damping system.

3.13.2.2 Relative Wind Sensors

Two relative wind sensors shall be installed on the nose boom probe, an 'Alpha' (pitch) vane sensor to provide angle of attack information to a central air data computer, and a 'Beta' (yaw) sensor vane to provide yaw information to the pilot's sideslip indicator, the damping system, and for a fire control system. A dummy vane pedestal shall be mounted horizontally on the nose probe to provide for a greater degree of symmetrical flow about the 'Beta' vane.

3.13.2.3 Air Temperature Sensor

A total temperature probe, containing an air temperature sensor, shall be installed externally on the underside of the nose fuselage. Total temperature shall be converted to electric signals to be fed into the central air data computer.

3.13.2.4 Central Air Data Computer

A central air data computer shall be installed to provide functions of total and static pressure to an Automatic Mode of control system; true angle of attack, altitude, static pressure, and true airspeed, to a fire control system; and air data to an indicated airspeed indicator, a flight director/attitude indicator, engine performance indicator, Mach meter, and true airspeed indicator.

The computer shall receive data from the pitot static system, the 'Alpha' wind sensor, and from the air temperature sensor.

3.13.3 Navigational Equipment

The navigational radio and radar aids shall be as described in paragraph 3.17.

3.13.3.1 J-4 Compass

A roll stabilized J-4 type compass system shall be installed to provide indication of the magnetic heading of the aircraft on an Integrated Destination Indicator in each cockpit. A controller incorporating the system. switches and controls shall be installed in the pilot's cockpit.

The system shall operate on 200/115 volt AC power obtained from the main electronic supply system, and 27.5 volt DC power obtained from the emergency DC bus. In the event of failure of the main AC supply, AC power shall be supplied by the emergency generator.

3.13.4 Installation

(22) (23) (35) The instruments and main instrument panels shall be installed in accordance with the requirements of specifications AIR 7-4, 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.

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3.13.4 Installation (Cont'd)

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. The main instrument panel shall be designed to permit quick removal for inspection and maintenance.

3.13.5 Test Instrumentation

Test instrumentation shall be installed in accordance with paragraph 4.3.

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3.14 Hydraulic Systems

3.14.1 Description and Components

Three separate 4000 psi main 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 ram air turbine extension gear, and missile extension gear of a removable armament pack.

- Two flying control systems, each capable of providing sufficient power for limited control of the aircraft in event of failure of the other. One flying control system shall include a line to a hydraulic motor-pump combination for operation of a radar antenna drive sub-system.

The system 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 circuit, and two in each of the two flying control system power circuits. Three compensators, one for each system, shall provide fluid reserve and pump inlet pressurization.

Emergency hydraulic power for one flying control system shall be provided by a ram air turbine driven hydraulic pump.

Hydraulic power shall be provided for emergency operation of the brakes. Pressurized nitrogen shall be provided for emergency extension of the landing gear.

3.14.1.1 Utility Services System

Two constant delivery hydraulic pumps shall be installed, one driven by each of the two aircraft accessories gear boxes (Reference paragraph 3.11.3.2). The output from the pumps shall be combined at a pressure regulating and check valve and shall be 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 three

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3.14.1.1 Utility Services System (Cont'd).

compensators, one in the utility hydraulic system return line, and one in each flying control system hydraulic return line.

The compensator of the utility hydraulic system shall be designed to pressurize the return fluid at 90 to 100 psi and to separate air from the fluid. It shall be possible to manually ground bleed the separated air from the compensator.

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 in 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 sequenced during retraction and normal extension. Sequencing valves, operated by landing gear and door movement during retraction and extension, shall be installed in the hydraulic pressure lines to the landing gear and door actuators. Normal actuation shall be controlled by a solenoid operated selector valve signalled from a manually operated selector lever installed in the front cockpit.

A pneumatic release valve, mechanically linked to the landing gear selector lever, shall be installed to release nitrogen from a 5000 psi storage bottle to effect landing gear emergency extension. (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 uplocks engage mechanically. In the last stages of the engagement of each gear uplock, a rider on the landing gear leg shall mechanically open a sequence valve permitting 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 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 downlocks are engaged. As the door is locked down a rider in the door mechanism shall open a sequence valve permitting hydraulic pressure to release the gear uplock and operate a transfer valve. The transfer valve shall release the hydraulic pressure from the landing gear actuator permitting the landing gear to fall by gravity and drag forces until a mechanical downlock is engaged.

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 a landing gear emergency extension circuit. The emergency circuit shall permit the compressed nitrogen to simultaneously release all gear and door uplocks, operate the door actuators, and operate the landing gear transfer valves. 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 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.

Shimmy damping restrictor valves shall be installed to permit a restricted runaround 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 maximum of 2500 psi reduced from the 4000 psi utility hydraulic system. Pressure available for emergency brake application shall be a nominal 1500 psi reduced from a 4000 psi accumulator.

3.14.1.1.3 Wheel Brakes Sub-System (Cont'd)

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 of the brakes for parking shall be controlled from the front cockpit (Reference paragraph 3.8.2.2).

72 In the event of normal supply pressure failure, the emergency brake pressure shall be routed to the shuttle valve and to 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. The actuation mechanism shall be designed to limit the degree of speed brake extension in relation to the speed brake design limit hinge moments. A relief valve shall be installed within the selector valve to permit blow back of the speed brakes as a function of limiting hinge moments. A check valve shall be incorporated in the pressure line to prevent excessive back pressures from entering the pressure lines of the utility system.

3.14.1.1.5 Ram Air Turbine Sub-system

A hydraulic actuator shall be installed for extension of a ram air turbine emergency power pack into the slipstream (Reference para. 3.12.1). Extension shall be controlled by a solenoid operated valve signalled to open on a combined failure signal from both main AC generating systems.

3.14.1.1.6 Armament Pack Sub-System

Complete provision shall be made for the installation of a removable interchangeable armament pack. A hydraulic pressure line and return line terminating in self-sealing half-couplings shall be installed to supply the hydraulic requirements of an installed pack. When installed the armament pack shall contain all other hydraulic components of the sub-system as an integral part of the pack.

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3.14.1.1.6 Armament Pack Sub-System (Cont'd)

Two hydraulic actuators shall be installed in the pack to effect retraction and extension of each of four missile launchers (Reference paragraph 3.18.3.1.7) and to hydraulically hold the launchers in the down position. A hydraulically actuated uplock shall be installed for each launcher. A hydraulic actuator shall be installed for actuation of each pair of missile wing doors, each pair of missile fin doors, and each extension gear drag link door, of the armament pack.

The launchers and doors shall be sequenced during extension and retraction by solenoid operated control valves, operated by launcher and door movement.

3.14.1.2 Flying Control Systems

3.14.1.2.1 Flying Control System Power Circuits

66

The two flying control hydraulic systems shall comprise an "A" system and a "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 (Reference paragraph 3.11.3.2). The output of the two pumps for each system shall be combined and utilized to power control surface actuators and servo units, the output of the "A" system also being utilized for a radar antenna drive sub-system.

The flying control hydraulic "A" system shall be powered in emergency by a 10 gal/min. 500 psi constant delivery pump. The pump shall be installed in a ram air turbine driven emergency power unit. (Reference para. 3.12.1). A check valve shall be installed in the pressure line from the emergency pump to prevent excessive back pressure on the pump in the event of continued normal system operation or system re-activation with the ram air turbine unit operative.

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

3.14.1.2.2 Control Actuators and Servo Units (Cont'd)

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 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 to 100 psi and to separate air from the fluid shall be installed in the return circuit of each of the two flying control systems. The compensators 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. It shall be possible to manually ground bleed the separated air from each compensator.

In each hydraulic system return circuit, an accumulator shall be installed to damp out surges set up by the continual operation of the control actuators.

3.14.1.2.4 Radar Antenna Drive Sub-System

Complete provision shall be made for the installation of a closed sub-system for radar antenna drive.

A pressure and a return line from the flying control hydraulic "A" system, an accumulator in the pressure line, and an air to oil heat exchanger (Reference paragraph 3.22.1.3.4) shall be installed.

Quick disconnects shall be installed in the pressure and return lines of the sub-system to facilitate installation and removal of an antenna installation.

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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.

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.

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

Pressure and return line self-sealing couplings for a hydraulic test stand shall be installed in each system to permit system testing with the engines inoperative.

Separate ground testing of the armament pack, when not installed, shall be possible by use of the pressure and return line half couplings installed in the armament pack. (Reference paragraph 3.14.1.1.6).

3.14.2 **Hydraulic Fluid**

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

3.14.3 **Piping and Fittings**

(71) High pressure lines shall be of stainless steel except lines subject to flexing which shall be chrome molybdenum steel. Low pressure lines shall be of aluminum alloy.

3.15 Pneumatic System

94 The low pressure pneumatic system shall comprise two 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 a ground service.

A ground chargeable high pressure 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 low pressure pneumatic system shall comprise the following sub-systems:

(1) A services sub-system for:

(a) Canopy seal inflation
(b) Anti-G suit inflation
(c) Radome anti-ice system air supply
(d) Nose Radar pressurization air supply
(e) Radar antenna drive hydraulic reservoir pressurization.
(f) Armament pack seal inflation

(2) Fuel tank pressurization and fin waveguide pressurization air supply.

3.15.1.1 Services Sub-System

The services sub-system shall utilize air at 65 psi (maximum) tapped from the downstream side of the air conditioning system air cooling water evaporator (Reference paragraph 3.22.1.3.2). A filter, which incorporates a drainable moisture trap, shall be installed in the sub-system supply line.

3.15.1.1.1 Canopy Seal Inflation

Air for canopy seal inflation shall be ducted from the services sub-system filter to a solenoid operated valve. With the solenoid in the energized position, the valve shall be designed to act as a 20 psig pressure regulator, as a pressure relief valve at pressures in excess of 25 psig, and as a check valve to prevent back

3.15.1.1.1 Canopy Seal Inflation (Cont'd)

flow of air from the seals. With the solenoid in the de-energized position, the valve shall vent seal pressures. The control solenoid shall be electrically linked to the canopy latches of both cockpits.

3.15.1.1.2 Anti-G Suit Inflation

21 A branch duct shall convey air from the services sub-system filter to an anti-G suit inflation control valve located on the right hand side in each cockpit. Each of the two anti-G valves shall automatically control anti-G suit inflation for the occupant of the respective cockpit.

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 10 psi pressure reducing valve. The reduced pressure air shall be utilized to:

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

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

3.15.1.1.4 Radar Pressurization Air Supply

A branch from the radome anti-ice system air supply ducting shall convey air to a nose radar pressurization system integral with the electronics system installation.

3.15.1.1.5 Radar Antenna Drive Hydraulic Reservoir Pressurization Supply

A branch duct shall convey air from the services sub-system filter to the radar antenna drive hydraulic sub-system reservoir.

3.15.1.1.6 Armament Pack Seal Inflation

Air for armament pack seal inflation shall be ducted from the services sub-system filter to a solenoid operated valve. With the solenoid in the energized position, the valve shall be designed to act as a 20 psig pressure

3.15.1.1.6 Armament Pack Seal Inflation (Cont'd)

regulator, as a pressure relief valve at pressures in excess of 25 psig, and as a check valve to prevent back flow of air from the seals. With the solenoid in the de-energized position, the valve shall vent seal pressures.

The control solenoid shall be energized by a micro-switch installed on the interchangeable pack. The micro-switch shall be actuated by closing an access door for the armament pack test panel.

3.15.1.2 Fuel System Pressurization and Fin Wave-Guide Pressurization Supply

3.15.1.2.1 Fuel System Pressurization Supply

Air at 65 psi (normal maximum) shall be ducted from the downstream side of the air conditioning system heat exchanger to the pressure regulating valves of the fuel tank pressurizing system (Reference paragraph 3.11.8.10).

3.15.1.2.2 Fin Wave-Guide Pressurization Supply

A branch from the fuel system pressurization ducting shall convey air to a 10 psi pressure regulating valve. Air at the regulated pressure shall be utilized to pressurize the wave-guide installed in the fin.

3.15.1.3 Ground Operation

Air at a pressure of approximately 4.5 psi from a ground air conditioning air supply unit shall be necessary for ground operation of the pneumatic services sub-system with the engines not operating.

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

Air at a pressure of approximately 45 psi from a ground supply shall be necessary for ground pressurization of the fuel tanks with the engines not operating. The ground air connection shall be made at the air conditioning system heat exchanger.

3.15.1.4 Inspection and Maintenance

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

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

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3.16 Electrical System

The electrical system prime source of electric power shall be two engine driven 200/115 volt, 3 phase, 400 cycle AC generators. The aircraft 27.5 volt DC power shall be supplied by means of two AC to DC transformer-rectifier units.

The electrical system shall be designed to the requirements of RCAF 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 ground system.

Power drawn from one AC generator shall be utilized for electrical services in the aircraft, and power drawn from the other AC generator shall supply the power requirements of the aircraft electronic systems. Provision shall be made for cockpit indication of power failure, automatic transfer of the primary loads to the operative AC generator, and disconnection of secondary AC services in the event of failure of one AC generator.

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. During a double AC generator system failure or during double engine flame out conditions, services essential to flight shall be transferred to a ram-air turbine driven emergency AC generator (Reference para. 3.12.1)

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 or single transformer-rectifier unit failure.

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3.16.1.4 Emergency DC System

A battery shall be installed to supply emergency DC power. Distribution of power to the emergency services shall be through the emergency and battery buses. Provision shall be made in the system for the isolation of the battery and emergency buses from the main DC bus in the event of failure of both transformer-rectifier units. With the master switch in the off position, (Reference paragraph 3.16.2.5) the supply of power to the emergency bus shall be discontinued, and only those services connected to the battery bus shall be supplied with power.

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 AC Generators

Two 40 kva, 200/115 volt, 3 phase, 400 cycle AC generators shall be installed, one in the nose bullet of each engine. Each generator shall be engine driven through a constant speed drive fluid coupling.

Cooling for each AC generator shall be provided by oil from the associated constant speed drive oil system (Reference paragraph 3.11.3.1).

A 200/115 volt, 3 phase, 400 cycle emergency AC generator of sufficient capacity to supply 1.4 kva shall be installed in, and driven by, a ram air turbine power pack (Reference para. 3.12.1).

3.16.2.2 Battery

A 24 volt, 15 amp hour, nickel cadmium, hermetically sealed storage battery shall be installed in a battery compartment located in the nose wheel well. The battery shall normally be connected to the battery bus. When engine starting power supply is connected, the

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3.16.2.2 Battery (Cont'd)

battery circuit shall automatically open permitting the aircraft DC services to draw current from the ground source only. 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 aircraft services bay to provide voltage regulation of each AC generator output and each transformer rectifier output. Cooling for the control/transformer rectifier panels shall be provided by the air conditioning system.

3.16.2.4 Protective Devices

AC overvoltage, undervoltage, and overcurrent; and DC overvoltage 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 AC generator failure warning lights, and two TRIP-RESET switches to permit individual control of each generator 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 4.5 Kw 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

73 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

74 to 80, 83 86 87 The installation of all aircraft wiring shall be in accordance with Specification MIL-W-5088A.

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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

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

32 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. All other circuit breakers shall be located on a panel in the nose wheel well.

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

106 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, instrument panel, console panel, and map lighting, shall be installed in accordance with Specification CAP 479, MIL-P-7788, MIL-L-6503A, and MIL-L-25467A.

3.16.8.1.1 Instrument Panel Lighting

Instruments shall be internally lit or shall be illuminated by post type red lights.

3.16.8.1.2 Console Panel Lighting

Console panel lighting shall consist of plastic plate type red lighting and hooded type red flood lights. Two high altitude white flood lights shall be installed in each cockpit.

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3.16.8.1.3 Interior Illumination Controls

Three continuously variable transformers (0 to 27.5 volts) shall be installed in the front cockpit to provide illumination control.

One transformer shall control the console flood lights, one shall control the plastic plate lighting, and one shall control the post type red lights and internally lit instruments.

Two continuously variable transformers (0 to 27.5 volts) shall be installed in the rear cockpit. One transformer shall provide illumination control for the plastic plate type red lighting, the post type red lights, and internally lit instruments; the other transformer shall provide illumination control for 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

A combination red and white flood lamp with an integral intensity control shall be installed in each cockpit to provide illumination for map reading. The map lights shall be connected directly to the emergency DC bus for use as emergency lighting.

3.16.8.2 Exterior Lighting

The exterior lighting, comprising navigation lights, taxi light, and landing light, shall be installed in accordance with RCAF Specifications AIR 7-4 and CAP 479.

3.16.8.2.1 Navigation Lights

Navigation lights shall comprise a left wing tip red light, a right wing tip green light, and one red and one white light in fin trailing edge. A flasher unit shall be installed in accordance with Specification MIL-L-6503A.

3.16.8.2.2 Taxi Light

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

3.16.8.2.3 Landing Light

A landing light shall be installed on the nose landing gear assembly.

3.16.8.2.4 Exterior Lighting Controls

Controls for exterior lighting shall be located in the front cockpit. A STEADY-OFF-FLASHING switch shall be installed for navigation light selection. A LANDING-OFF-TAXI switch shall be installed for landing light and taxi light selection. On selection of "landing" both the landing and the taxi light circuits shall be energized.

3.16.9 Ignition System

The engine ignition system shall be in accordance with Orenda Specification EMS-8. Switches located in the front cockpit shall provide for selection and control of engine starting.

3.16.9.1 Engine Starting

Two START-OFF-MOTOR switches shall be installed on the right hand console in the front cockpit to provide control of DC power to the relevant engine igniter system and to an 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 Power Receptacles

An external power receptacle for ground supply of AC power shall be installed in accordance with Specification MIL-E-7563. The receptacle shall conform to the outline of AN 3114, and shall be suitable for mating with an automatic quick disconnect plug.

An external receptacle suitable for mating with an automatic quick disconnect connector shall be installed for ground supply of DC power for engine starting, for engine starting control, and for 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 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 jack shall be installed adjacent to the main AC external supply plug to facilitate attachment of a pull-away quick disconnect grounding cable.

3.16.11 Indicators

3.16.11.1 Master Warning Lights

(34) One red and one amber master warning light shall be installed on 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

(34) A panel with provision for 38 warning indicators shall be installed in the front cockpit to provide, in conjunction with the master warning lights, indication of specific system failure or system malfunction.

3.16.11.2 Warning Indicator Panel (Cont'd)

The following warning lights shall be incorporated in the panel:

2	Fuel Low Level L.H. and 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	AC Generator Failure L.H. and R.H.
1	Battery in Use
1	DC Failure L.H. or R.H.
2	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
2	Engine Air Bleed Failure
1	Air Conditioning Overtemperature
1	Cabin Pressure
1	Ice Warning
1	Fuel Transfer Off
1	External Tank Empty
1	Tank Jettison Failure
1	Missile Hang Fire
1	Radar Overheat
8	Spare

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 warning light in the landing gear selector lever and a composite landing gear position indicator 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 Specifications 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

81 82 Provision shall be made in the aircraft for the inspection, maintenance, removal, and re-installation of electrical equipment.

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3.17 Electronics

108 Telecommunication and navigation sub-system components of the Astra 1 integrated electronic system shall be as defined by Radio Corporation of America Specification .... (Reference para. 3.18 for Astra 1 Fire Control sub-system, para. 3.9.4.1 for Automatic Flight Control sub-system, and para. 3.13 for cockpit presentation and instruments sub-system).

The following Astra 1 telecommunication and navigation equipment shall be installed in the aircraft:

Command Set	AN/ARC-52
Interphone	AN/AIC-10A
Radio Compass	AN/ARN-6
Identification Equipment	AN/APX-25A
Air Data Computer	(Ref. para. 3.13.2)

Complete provision except wiring shall be made for the installation of the following equipment:

Homing Adaptor	AN/ARA-25
UHF Data Link	AN/ARR-48
Identification Equipment	AN/APX-27
Interrogation Equipment	AN/APX-26
Dead Reckoning System	
Doppler	
Vertical Heading and Reference System	
NADAR	

Space provision shall be made for the installation of ECM Homer type AN/ARD-501.

A J-4 Compass system shall be installed (Reference paragraph 3.13.3.1).

The electronics system shall be installed in accordance with 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 Specification MIL-W-5088A).

Conditioned air for pressurization and cooling within the electronic compartments shall be supplied 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 micro-switch incorporated in each seat installation shall provide a means for switching the UHF communication transmitter to transmit on the emergency frequency and the air-to-ground IFF transponder to transmit in the emergency mode

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3.17 Electronics (Cont'd)

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-52 type UHF transceiver shall be installed to provide air-to-air and air-to-ground communication facilities on 1750 channels, any eighteen of which may be preset. The equipment incorporates a guard channel receiver tuned to a preset guard frequency with the main receiver selected to any other frequency. Provisions shall be made in the system for utilization of the AN/ARA-25 UHF homing adaptor (Reference paragraph 3.17.2.6).

A UHF remote control unit for the main transceiver shall be installed in each cockpit to permit selection of any of the eighteen preset channels, selection of the guard channel, or selection of a manually set channel. The control unit in the rear cockpit shall also permit manual setting of 1,750 channels. An indicator shall be installed on the pilot's instrument panel to provide indication of the selected preset channel.

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

A press-to-transmit button shall be installed on the inboard throttle grip in the front cockpit. A foot operated press-to-transmit switch and a foot operated muting switch shall be installed in the rear cockpit.

Two antennas to provide omni-directional coverage shall be installed for use with this equipment. A dual purpose antenna incorporating a fan shaped UHF vertical radiator shall be mounted under a fibre-glass 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 normally operate on 200/115 volt AC power obtained from the main electronic [illegible] and 27.5 volt DC power obtained from [illegible] bus. In the event of failure of the main AC supply, AC power shall be supplied by the emergency generator.

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3.17.1.2 Liaison Set

Not applicable

3.17.1.3 Interphone

A type AN/AIC-10A 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 navigation 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 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 microphone and headset cable connections between the crew and the ejection seat, and between the ejection seat and the aircraft. (Reference paragraph 3.19.1.)

3.17.1.5 Filters

85

Radio interference filters of the electronics system shall be integral with electronic units. Radio interference caused by the operation of the electronic equipment installed in the aircraft shall not exceed the limits defined in Specifications MIL-I-6051 and MIL-I-6181B.

3.17.1.6 Recording Equipment

Complete provision shall be made for the installation of NADAR type RO-14/APH display recorder equipment.

3.17.2 Navigation Equipment

3.17.2.1 Radio Compass

A type AN/ARN-6 LF-MF radio compass system shall be installed to provide visual bearing indication of a selected radio station on indicators located in the front and rear cockpit instrument panels. Control facilities, which provide for servo-tuning of the receiver, shall be installed in each cockpit.

A non-directional sense antenna shall be installed in the dorsal electronic compartment. A flush type directional loop shall be installed in the centre door of the forward fuselage electronics compartment. Matching devices shall be installed between the sense antenna and the receiver to compensate for the length of the sense antenna cable.

The system shall operate on 200/115 volt AC power obtained from the main electronic supply system and 27.5 volt DC power obtained from the main DC bus.

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

Complete provision shall be made for the installation of a type AN/ARA-25 UHF homing adaptor to be used in conjunction with the UHF communication receiver to provide a continuous visual indication on two indicators of the direction of a selected UHF signal source (Reference paragraph 3.17.1.1).

3.17.2.7 VHF Navigation Receiver

Not applicable

3.17.2.8 Distance Measuring Equipment

Not applicable

3.17.2.9 Arbitrary Course Computer

Not applicable

3.17.2.10 Dead Reckoning System

Complete provision shall be made for the installation of a dead reckoning system to provide aircraft position with respect to:

(a) base
(b) an arbitrary reference point
(c) a destination
(d) a target
(e) a target offset point

3.17.2.11 Vertical and Heading Reference System

Complete provision shall be made for the installation of a vertical and heading reference system to establish earth reference in three planes from which the aircraft heading, pitch angle, and roll angle is derived for supply to the automatic flying control system, the fire control system, and dead reckoning system, and to indicators in the front and rear cockpits.

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

Complete provision shall be made for the installation of Doppler true ground speed and ground track measuring equipment.

3.17.3.4 Identification Equipment.

A radar identification set type AN/APX-25A shall be installed to permit the aircraft to identify itself automatically when interrogated by ground or airborne L-band radars.

A control box for the set shall be installed in the front cockpit. Circuits operated by the seat micro-switches shall be provided to operate the set automatically on emergency mode when the crew seats are ejected (Reference 3.17).

Antenna requirements shall be furnished by a fan-shaped vertical radiator mounted under a fibre-glass fairing at the top of the fin, and a downward facing annular slot antenna mounted in the skin of an access panel forward of the forward fuselage electronics compartment.

The system shall operate on 200/115 volt AC power obtained from the main electronic supply system and 27.5 volt DC power obtained from the emergency DC bus. In the event of failure of the main AC supply, AC power shall be supplied by the emergency generator.

84 Complete provision shall be made for the installation of an identification set type AN/APX-27 for response to airborne L-band interrogation. A dual-horn antenna and waveguide for the AN/APX-27 shall be installed in the fin.

3.17.3.5 Interrogation Equipment

Complete provision shall be made for the installation of an interrogation set, type AN/APX-26 to interrogate any target illuminated by airborne interception radar (Reference paragraph 3.18.3.1.2).

3.17.3.6 Radar Beacon

Not applicable

3.17.4 Electronic Countermeasures

3.17.4.1 Search Equipment

Not applicable

3.17.4.2 Analyzing Equipment

Not applicable.

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3.17.4 Electronic Countermeasures (Cont'd)

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 (UHF Homer)

3.17.4.6 Transmitting Equipment

Not applicable

3.17.4.7 Radar Homer

Space provision shall be made for the installation of type AN/ARD-501 DF equipment to provide bearing and elevation of airborne enemy L & S band ECM stations. Space provision shall be made for AN/ARD-501 antennas. 1

3.17.5 Electronic Guidance System

3.17.5.1 Guide Links and System

Complete provision shall be made for the installation of a type AN/ARR-48 UHF data link to receive coded information of target position and velocity from ground controlled interception.

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.

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3.18 Armament

3.18.1 Description

Complete provision shall be made for the installation of an ASTRA I air-to-air fire control system as defined in Radio Corporation of America Specification ...., and a missile armament system providing for the semi-submerged carriage, in an interchangeable armament pack, of four Sparrow 2 Model D missiles suitable modified for carriage, launch, and effective operation, under conditions compatible with Arrow 2 performance. 2

The system installation shall be designed in accordance with the requirements of Specification AIR 7-4 except as stated herein, and in Appendix II (Deviations).

3.18.2 Fixed Guns

Not applicable

3.18.3 Missiles

3.18.3.1 Fire Control System

Complete provision shall be made for a fire control system to provide for: radar search, radar automatic or manual tracking of targets forward of the aircraft, computation of an attack course for missile launching, preparation and automatic or manual firing of the missiles, or jettisoning of the missiles, attack course pull-out, ground map, and beacon interrogation.

3.18.3.1.1 Fire Control

To provide for system control, complete provision shall be made for the installation of:

(a) A three position mode selector switch to provide for selection of missile firing in one of three modes; AI-AUTO, AI-MANUAL, or OPTICAL.

(b) An intervalometer to generate missile firing pulses.

(c) A three position FRONT-REAR-ALL rotary firing selector switch to provide for selection of ripple firing in pairs, or ripple firing of all four missiles.

(d) A trigger switch to provide for firing the missiles in any selected mode of operation.

3.18.3.1.1 Fire Control (Cont'd)

(e) An armament SAFE-ARM switch to control the electrical circuits utilized in firing the missiles in any selected mode of operation.

(f) Safety interlocks to prevent missile firing until the nosewheel is in the locked up position and to prevent missile jettisoning until the weight of the aircraft is off the landing gear wheels.

A stores jettison switch (panic button) shall be installed in the front cockpit to initiate jettison of all attached missiles and the external fuel tank. (Reference paragraph 3.11.8.16).

3.18.3.1.2 Radar

Complete provision shall be made for the installation of radar to provide automatic or manual search, and automatic or manual tracking of targets forward of the aircraft.

Complete provision shall be made for the installation of a 32 inch paraboloidal reflecting antenna to provide for automatic and manually controlled scanning within a forward search zone.

3.18.3.1.3 Computer

Complete provision shall be made for the installation of a computer in the fire control system to compute lead angles and allowable lead angle errors, to provide steering signals and present maximum and minimum missile launching range limits.

3.18.3.1.4 Displays

Complete provision shall be made in the front and rear cockpits for the installation of a flight indicator.

3.18.3.1.5 Missile Auxiliaries

Complete provision shall be made in an interchangeable armament pack for the installation of missile auxiliaries, comprising a power control unit, and range and angle slaving units.

3.18.3.2 Missile Installation

116 Complete provision shall be made for the installation of a removable interchangeable armament pack incorporating four hydraulically actuated launchers to provide for the carriage of four Sparrow 2 Model D missiles (Reference paragraphs 3.7.2 and 3.14.1.1.6). The launchers shall be arranged in a spanwise row of two staggered pairs. The two forward launchers shall be hydraulically interconnected for simultaneous operation, and the two rear launchers shall be hydraulically interconnected for simultaneous operation.

With the missiles in the retracted position, the missile centre-line shall be located approximately on the aircraft skin-line. After missile firing the launchers shall be partially retracted to close the space in the aircraft skin formerly occupied by the missiles.

118 Hydraulically actuated wing and fin doors for each missile, and hydraulically actuated drag link doors for the aft launchers only, shall permit extension of the launchers and missiles (Reference paragraph 3.14.1.1.7).

3.18.3.3 Pressurization and Conditioning

Pressurization and conditioning air shall be supplied by the pneumatics system (Reference paragraph 3.1.5.1) and the air conditioning system (Reference paragraph 3.22).

3.19 Equipment and Furnishings

17 Equipment and furnishings appropriate to the role of the aircraft shall be installed in accordance with the require-
18 ments of RCAF Specifications AIR 7-4 and CAP 479 except as stated in Appendix II (Deviations).
19

3.19.1 Personnel Accommodation

A fully automatic Martin Baker Mark C5 ejection seat with an ejection velocity of 80 feet per second shall be installed for each member of the crew. Each seat shall incorporate leg restraints, and 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 left hand side of the seat, to provide freedom of movement for the seat occupant. A leg restraint disconnect shall be located on the right hand side of the seat.

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. Provision shall be made in the seat firing mechanism to permit insertion of a safety pin to prevent inadvertent seat firing on the ground.

Action to fire the seat shall also clear the ejection path (Reference paragraph 3.7.3). Ejection of the seat shall initiate distress signals (Reference section 3.17).

A bail-out warning system comprising a green light in the front cockpit, and a red light and 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.

16 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 de-misting
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

15 Not applicable.

3.19.2.2 Crew Relief Provisions

14 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

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

3.19.2.5 Map Stowage

33 A map stowage shall be located on the right hand side of the rear cockpit.

3.19.3 Windshield Wipers

Not applicable.

3.19.4 Furnishings

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

3.19.5 Emergency Equipment

A special pack parachute shall be installed in each ejection seat. Accommodation shall be provided in each ejection seat for a survival kit, packaged such as to be compatible with the seat design.

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3.20 Fire Protection

11 The fire protection system shall provide for detecting and
extinguishing fires in the aircraft services bay and each
12 power plant compartment. The detection system and the ex-
tinguishing system shall be in accordance with Specifications
89 AIR 7-4 and CAP 479 respectively, except as stated herein and
Appendix II (Deviations).

3.20.1 Description and Components

3.20.1.1 Detection System

Continuous wire type fire detector circuits shall be installed for each power plant and in the aircraft services bay. The detector circuits shall be connected to both a master warning light on the pilot's main instrument panel, and to three warning lights, each combined with an extinguishing switch, one for each protected region. 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 con-
88 taining 12 pounds of freon, shall be installed in the
aircraft services 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 wet and hot zones of each power plant installation, and in the aircraft services bay.

3.20.1.3 Fire Protection System Controls

Actuation of the extinguishing switches shall initiate discharge of the extinguishing agent to the associated region. 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 Fire Protection System Controls (Cont'd)

An inertia switch shall be installed to complete a circuit from the battery to automatically discharge the extinguishing agent to all three protected compartments in the event of a crash landing.

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

Externally accessible test switches shall provide for detector circuit ground testing. 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

120 The normal oxygen supply shall be stored in a 5 litre, portable type, 70 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

13 A high altitude, automatic pressure demand, dual outlet oxygen regulator to Specification MIL-R-25572 shall be installed on each ejection seat. Oxygen shall be supplied to the regulator through a 70 psi reducing valve and 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 close 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 each 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.

3.21.2 Ground Service

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

The emergency oxygen bottles shall be rechargeable through a quick disconnect charging valve installed on each ejection seat.

3.21.3 Inspection and Maintenance

Suitable provision 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 by "Secondary Conditioning" in the table below.

Primary Conditioned	exhausting to	Secondary Conditioned
Cockpits Missile Auxiliaries Compartment		Armament Pack
Nose Electronics Equipment Battery Windshield Transformer		Nose Wheel Well (Forward end)
Oxygen Equipment Forward Fuselage Electronics		Air Conditioning Bay
Dorsal Radar Stable Platform Transformer Rectifier Units		

Ground air conditioning shall be possible by utilizing conditioned air from a ground service unit connected to the system through two quick disconnects.

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

92 The air conditioning and pressurizing system shall be designed to the requirements of RCAF Specification AIR 7-4 and
95 ARDCM 80-1 except as stated in Appendix II (Deviations) and additionally herein.

3.22.1 In Flight Air Conditioning

Air cooled by successive stages of the system shall be mixed with thermostatically metered hot engine bleed air, to provide conditioned air to the cockpits, equipment compartments, and equipment.

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3.22.1 In Flight Air Conditioning (Cont'd)

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

Three warning lights shall be installed in the front cockpit; one to indicate air temperatures greater than 80°F in the duct immediately downstream from the turbine, and two, one left and one right hand to indicate pressures exceeding 75 psi downstream of the engine bleed pressure reducing valves, or to indicate leakage sensed by overtemperature at duct joints. A three position selector switch shall be installed in the front cockpit to permit either engine bleed shut off valve to be closed.

An air supply selection control shall be installed to permit manual selection of either conditioned air or ram ventilating air.

3.22.1.1 Occupied Compartments

3.22.1.1.1 Cockpits

110 111 112 Primary conditioned air shall be supplied to the front and rear cockpits in seat back and floor regions. A manual temperature control shall be installed in the front cockpit to provide for selection of cockpit temperatures within the range of 40°F to 80°F. A de-fog switch shall be installed in the front cockpit to provide selection of inlet air at 90°F for fog dispersion.

96 A cockpit pressure regulator shall be installed to provide the following pressure schedule:*

(a) A cockpit to atmosphere pressure differential of zero from sea level to 10,000 feet.

(b) A linear increase in cockpit to atmosphere pressure differential from zero at 10,000 feet, to 4.25 ± 0.25 psi, at 50,000 feet.

(c) A constant cockpit to atmosphere pressure differential of 4.25 ± 0.25 psi, at altitudes above 50,000 feet.

A secondary pressure regulator shall be installed for safety operation in the event of failure of the normal regulator. The secondary regulator shall be designed to maintain the following cockpit pressure schedule:*

* Assuming NACA Standard Atmosphere

3.22.1.1.1 Cockpits (Cont'd)

(a) A cockpit to atmosphere pressure differential of one plus zero minus 0.25 psi from sea level to 10,000 feet.

(b) A linear increase in cockpit to atmosphere pressure differential from one psi at 10,000 feet to 5.30, plus or minus 0.20 psi, at 28,000 feet.

(c) A constant cockpit to atmosphere pressure differential of 5.30, plus or minus 0.20 psi, at altitude above 28,000 feet.

93 A solenoid operated dump valve, controlled from a switch in the front cockpit, shall be installed on the secondary pressure regulator. A cockpit air negative pressure relief valve shall be installed.

A warning light shall be installed in the front cockpit to indicate when the cockpit altitude has exceeded 31,000 feet. A cockpit pressure altimeter conforming to Specification MIL-I-5099A shall be installed.

The cockpit leak rate of the installed system shall not exceed 42.5 cubic feet per minute under the following conditions: sea level, 4.75 psi pressure differential, and 60°F temperature.

3.22.1.2 Unoccupied Compartments

3.22.1.2.1 Primary Conditioning Distribution

A system of ducts shall convey conditioned air at 70°F ± 2.5°F to all the primary conditioned equipment and equipment compartments. The air flow shall be sufficient to maintain the internal air temperature of the compartments and the air surrounding the equipment below 140°F. An air valve, controlled by the ram air selector switch, shall be installed to stop the flow of conditioned air to the nose radar compartments on the selection of ram air.

3.22.1.2.2 Secondary Conditioned Distribution

Air from the cockpits and missile auxiliaries shall be exhausted to the armament pack bay to maintain the internal air temperature, of an installed pack between 0° and 130°F.

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3.22.1.2.2 Secondary Conditioned Distribution (Cont'd)

Exhaust air from the radar nose, and the battery compartment shall be utilized to maintain a cooling air supply at a temperature below 160°F for equipment in the forward end of the nose wheel well. Exhaust air from the forward fuselage electronics compartment and the oxygen equipment bay shall be utilized to maintain the air in the air conditioning bay below 250°F.

3.22.1.3 Conditioning Air Supply System

97 Air for conditioning shall be bled from a 10th stage bleed port of each engine compressor. The air shall be passed through 60 psi pressure reducing valves and check valves, then ducted to a heat exchanger. Hot bleed air from the input side of the heat exchanger shall by-pass the system cooling components and shall be utilized for temperature control.

3.22.1.3.1 Air to Air Heat Exchanger

An air to air heat exchanger shall be installed to provide initial cooling of the engine bleed air. On the bleed air output side of the heat exchanger, the major portion of the air shall be ducted to an air cooling water evaporator. The remaining portion of the heat exchanger bleed air output shall be utilized for fuel tank pressurization (Reference paragraph 3.11.8.10).

Cooling air for the heat exchanger shall be ducted from ram air intakes located on each side of the fuselage immediately inboard from the engine intake ramp (Reference paragraph 3.11.4.1). Cooling air, after passing through the heat exchanger, shall be vented to atmosphere.

During conditions of negative pressure in the engine intake tunnel (taxi, engine run-up, and take-off), the negative pressure shall be utilized to draw cooling air in reverse direction through the air to air heat exchanger. A three-way reverse flow valve shall be installed between the ram air duct and engine intake tunnel on each side of the aircraft to block air from the forward section of the ram duct and open a passage into the engine intake tunnel, to provide a path for the reverse flow cooling air.

3.22.1.3.2 Air Cooling Water Evaporator

The air cooling water evaporator shall have a usable capacity of 260 pounds of water and shall be designed to withstand freezing and thawing of its contents. Steam shall be vented to the cooling air duct to the air to air heat exchanger. The vent shall be designed to prevent loss of water during inverted flight or under negative "g" loads of 3g maximum.

The major portion of the air output of the cooling evaporator shall be ducted to an expansion turbine. The reamining portion of the air output shall be utilized for a low pressure pneumatic services sub-system (Reference paragraph 3.15.1).

3.22.1.3.3 Air Cooling Expansion Turbine

91

An expansion turbine and compressor, together with associated equipment, shall form an air-cycle refrigeration unit.

The expansion turbine shall dispel heat energy from the engine bleed air by utilizing the turbine's power output to drive a compressor which shall draw air from the engine intake tunnels and exhaust it overboard through a restrictor. The inlet to the expansion turbine shall comprise a variable area nozzle to control air flow in the conditioning system.

3.22.1.3.4 Air to Oil Heat Exchanger

An air to oil heat exchanger shall be installed in the main conditioned air duct to the equipment and equipment compartments. The heat exchanger shall be utilized to cool the radar magnetron coolant fluid and radar antenna drive hydraulic fluid.

3.22.1.4 Ram Air Ventilation

Ram air from the left hand ram air duct (Reference paragraph 3.22.1.3.1) shall be utilized for ventilation of the cockpit and all conditioned compartments and equipment except the nose electronics. On selection of ram air, a ram air shut off valve shall open, a reverse flow valve shall open shutting of ram air to the air to air heat exchanger, the expansion turbine unit shall shut down, and a shut off valve to the nose electronics shall close. A thermostat shall be installed to override ram air selection when the ram air temperature exceeds 100°F.

3.22.2 Ground Air Conditioning

(90) Ground air conditioning shall be possible by utilizing 4.5 psi, 60°F, conditioned air from a ground service unit connected to the system through two automatic disconnect self-sealing couplings. Ducts shall be installed leading from the automatic disconnect coupling to connections into the system downstream from the expansion turbine outlet. From the system connections, ground conditioned air shall flow downstream through the system ducts to 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
Air Data Sensing Heads
Radome

A power supply selection switch shall be installed in the front cockpit to permit the power supply to the engine and air intake de-icing systems to be discontinued during missile firing.

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

3.23.1 Propeller De-Icing

Not applicable.

3.23.2 Engine Anti-Icing

Each engine, as supplied by the engine manufacturer, shall include an integral hot air return anti-icing system as defined in Orenda Specification EMS-8.

Automatic selection of anti-icing air flow for both engines shall be controlled by icing detectors on the engine air intakes (Reference paragraph 3.23.3.1).

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. 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 115/200 volt AC system.

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3.23.3.1 Engine Air Intakes (Cont'd)

The de-icing cycle shall be automatically controlled by icing detectors, installed at the top of the engine air intakes, 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 in the front cockpit. These components shall operate from the 27.5 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.

3.23.4 Cockpit Transparencies

Anti-icing and anti-misting of the windshield and canopy windows in the front and rear cockpits 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 Air Data Sensing Heads

The pitot-static head, the 'alpha' vane, the 'beta' vane, the vane pedestals, and the adjacent areas of the nose boom, 3.23.7 Air Data Sensing Heads (Cont'd)

and the pitot-static head on the fin shall be anti-iced by integral electric heaters. The heaters shall be powered by the aircraft main AC power supply system, and shall be energized by "on" selection of the master electrical switch. The heaters shall be automatically controlled to prevent overheating.

3.23.8 Landing Gear

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 (isopropyl-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 controlled by the ice detector shall be incorporated in an annunciator box located in the nose wheel well, to provide ground indication of possible de-icing fluid usage.

A scissors switch installed on the main landing gear shall be utilized 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 installed on the panel adjacent to the left-hand speed brake shall permit override of the scissors switch of the radome de-icing circuit.

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, thus 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.

103 Towing lugs shall be provided on each main landing gear unit for forward or rearward towing of the aircraft 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.

102 Provision shall be made for jacking the complete aircraft at three points, with the use of a removable jack pad at each jacking point, one on the aircraft centre line aft of the nose landing gear, and one inboard of each outer wing root.

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

3.25.3 Mooring Provisions

99 Provision shall be made for the attachment of mooring fittings to the main and nose landing gear.
100

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3.25.4 Hoisting Provisions

Provision shall be made for hoisting the entire aircraft from four points, two on the nose centre fuselage joint, and one on each inner wing panel adjacent to the outer wing root. The wing hoisting points are intended for emergency use only and require removal of structure.

3.25.5 Leveling

98 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 a four degrees 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.

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.

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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, electrical wiring, hydraulic lines, etc. will not be required. Space provision does not imply that adequate attaching structure is provided unless otherwise stated.

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6.2.2 Statements

6.2.2.1 Deviation

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

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

6.2.3.3 Combat Altitude

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

6.2.3.4 Combat Climb and Acceleration Time

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

6.2.3.5 Combat Ceiling

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

6.2.4 Weights

6.2.4.1 Combat Weight

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

6.2.4.2 Normal Gross Weight

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

6.2.4.3 Gross Weight for Stress Analysis

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

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6.2.4.4 Maximum Gross Weight

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

6.2 4.5 Maximum Landing Gross Weight

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

6.2.4.6 Normal Landing Gross Weight

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

6.2 4.7 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 "Basic Weight" shall be qualified as to role when referred to an aircraft in which various items of removable equipment may be installed for different roles. It includes airframe, power plant, accessories, trapped fuel and oil, and non-expendable fluid systems (hydraulic, coolant) filled to capacity, but without expendable items.

6.2.4.8 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 without augmentation with the power lever in the "military" position.

6.2.6.3 Normal Rated Thrust

Normal rated thrust is the maximum thrust which the contractor specifies the engine will deliver at standard sea level static conditions for continuous operation. In flight, normal thrust will be the maximum continuous thrust developed without augmentation while maintaining turbine exit temperature within prescribed limits.

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 "flight idle" position.

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APPENDIX I

EQUIPMENT CATALOGUE

Issued under separate cover.

APPENDIX II

DEVIATIONS

1. 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: Stress analysis has been conducted on the basis that a yaw velocity of 3.5 radians per second will not be exceeded.

2. Load Factors in Rolling Pull-Out

Requirement: Specification MIL-S-5702, paragraph 4.2.1.1.

Rolling Pull-Out: For this condition, all points within the positive Vn diagram up to and including a load factor of $1 + 2/3\Delta n$ shall be considered ------- .

Arrow requirement where $n = 7.33$ $(\Delta n = n-1)$
$$1 + 2/3 \times (7.33-1) = 5.22$$

Deviation: The positive load factors to be considered for a rolling pull-out will be based on $2/3\ n_1$.

Arrow consideration where $n_1 = 7.33$
$$2/3 \times 7.33 = 4.89$$

Reason for Deviation and Remarks: Requirement of superseded specification 1803 (original contractual specification) used; prior to the introduction of MIL-S-5702.

3. 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 when otherwise specified.

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3. Standard Atmosphere (Cont'd)

Deviation: ICAO Standard Atmosphere conditions utilized for performance calculations.

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

4. Landing Gear Door Loads

Requirement: Specification MIL-S-5705, paragraph 4.4.

The doors, mechanisms and supporting structure shall be designed for opening at all speeds up to the maximum speed at which the landing gear is to be lowered and for the load factors obtainable at that speed from both a gust and manoeuvre standpoint. (Arrow Sea Level case; n pos. 5.1, n neg. 3.0 at 250 KT. EAS at 47,000 lb. stressing weight).

Deviation: The doors, mechanisms and supporting structure are designed for operation under acceleration normal to the flight path (up to L.G. design speed) over the range of n pos. 1.3 to n pos. 0.8, and in assymmetric flight up to 10° of yaw, and with the fuselage datum up to 50° above the horizontal.

Reason for Deviation and Remarks: The specified requirement is considered unrealistic for an Arrow type aircraft. The above criteria was assessed as being adequate for the design.

5. Piano Hinge Pins

Requirement: Specification ARDCM 80-1, paragraph 5.451

Where the removal of the control surface is accomplished by removal of the hinge pin (piano type hinge) the continuous length of pin shall not exceed 48 inches.

Deviation: The aileron hinge pins exceed this length and are intended to be removable.

Reason for Deviation and Remarks: Established assembly and removal practice permits handling of the long hinge pins.

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6. 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.

7. 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.

8. 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.

9. Anodizing

Requirement: Specification MIL-A-8625A, paragraph 3.3.5.

(Type 1 - Chromic Acid Coating).

9. Anodizing (Cont'd)

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

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.

10. Thermal Radiation Protection

Requirement: Specification ARDCM 80-1, paragraph 23.137

All combat fighter, bomber and reconnaissance aircraft shall provide stowable hoods, curtains, or other devices incorporating 14.77 ounce bleached white cotton duck fabric conforming to Specification MIL-D-10861, Type II, for protection of the following items from thermal radiation caused by the explosion of nuclear weapons:

(a) All aircrew members
(b) Crew member's personal equipment
(c) Exposed wiring.

These devices shall preclude any light rays originating outside the aircraft from striking any of the above items in the aircraft when the devices are in the unstowed or protecting position. The pilot's protective device shall be operable and stowable in 20 seconds or less. Protective devices for other members of the aircrew must be operable and stowable in 4 minutes or less.

Deviation: Protection from thermal radiation is not installed.

Reason for Deviation and Remarks: Design is under study.

11. 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.

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11. Hand Fire Extinguisher (Cont'd)

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.

12. 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".

13. 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 CF-105 Oxygen System Sub-Panel Meeting I.A.M., 23 September 1954, Item 3, paragraph 7 (a).

14. 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.

14. Crew Relief Provisions (Cont'd)

Deviation: Relief horns are not installed.

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

15. Drinking Liquid Containers

Requirement: Specification CAP 479, paragraph 42.20

Aircraft having an endurance of more than three hours shall have installed, insulating drinking liquid containers of sufficient capacity to provide one pint of liquid per occupant.

Deviation: Drinking liquid containers are not installed.

Reason for Deviation and Remarks: The requirement arises only as a result of a secondary role, and as weight prejudices primary role performance, drinking liquid containers are not installed.

16. 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: RCAF letter S1038CF105-16 (ACE) dated 9 December 1954, required mounting on right-hand side of seat.

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17. 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: Stowage provision for pilot's operating instructions at present not intended.

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

18. Stowage(s) in Rear Cockpit

Requirement: 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.

19. Baggage and Tool Compartment

Requirement: Specification CAP 479, paragraph 41.04

All aircraft shall be equipped with a baggage and tool compartment or locker, provided with suitable door locks.

Deviation: Provision of a baggage and tool compartment at present not intended.

Reason for Deviation and Remarks: The RCAF 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.

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20. Switches - Space Provisions

Requirement: Specification MIL-E-7080, paragraph 3.4.1.4

Space shall be provided on each switch panel containing four or more switches, for subsequent installation of one spare switch conforming to drawing AN 3022 and one switch conforming to drawing AN 3023.

Deviation: No space provided for spare switches.

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

21. 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.

22. 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.

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

23. 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 windscreen panel.

24. 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.

25. 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 12½ degrees below the horizontal.

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

26. 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..

27. Canopy Jettison

Requirement: Specification CAP 479, paragraph 22.30 (1)

Canopies in single and tandem cockpit aircraft --- and shall be jettisonable in flight -------- .

Deviation: The canopies shall be openable, but not jettisonable in flight.

Reason for Deviation and Remarks: A non-jettisonable canopy provides enhanced crew safety, elimination of possible damage to airframe structure, and ability to ground test.

28. 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 incorporates 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. Cockpit approved at 15th Meeting of Co-ordinating Committee, 19 January 1955, Item XVI, paragraph 33.

29. Cockpit Head Room

Requirement: Specification CAP 479, paragraph 20.21

No part of the canopy roof or canopy shall be within 8½" 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 6½ 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.

30. 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 indicence no greater than 55° ---------- .

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

Reason for Deviation and Remarks: Aerodynamic requirement.

31. Number reserved.

32. 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 circuit breaker panels in the nose wheel bay. Damping system circuit breakers are located on the left-hand console in the front cockpit.

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32. Circuit Breakers (Cont'd)

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 RCAF 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.

33. 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: Flashlight stowage not provided in Arrow aircraft.

Reason for Deviation and Remarks: Space at a premium

34. 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: Two master indicator lights are installed, one red and one amber. The caution indicators are amber in color.

Reason for Deviation and Remarks: The warning light system proposed by Avro was approved by the RCAF. Reference letter S1038-105-4 (ACE-1) dated 13 August 1955.

35. Panel Space Provision

Requirement: Specification AIR 7-4, paragraph 8.2.3

Panel space shall be provided for 5 x $5\frac{1}{4}$ inches case size for the directional indicator and the artificial horizon.

Deviation: Space for 5 x $5\frac{1}{4}$ inches case size for directional indicator is not provided. A smaller instrument is used.

Reason for Deviation and Remarks: RCAF requirements for directional indicator override this requirement.

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36. Brake Parachute Control

Requirement: Specification CAP 479, paragraph 21.32

The (Brake Parachute Control) actuating motion shall be to pull backward or downward --- to deploy the parachute 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 rease mechanism in the rear fuselage. RCAF approval of parabrake control lever action or movement is contained in letter S1038-105-16 (ACE-1) dated 11 Mar., 57.

37. 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.

38. Elevator Interconnection

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

Elevators shall be rigidly interconnected or consist of a continuous structure.

Deviation: Each elevator is linked to a separate corresponding actuator and is not connected to the other elevator.

Reason for Deviation and Remarks: Space requirements dictate use of two actuators. This requirement is not met due to the difficulty of achieving the necessary degree of synchronication between two actuators when connected to a single surface and used in a stability augmented system.

39. Control Cable Duplication

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

39. Control Cable Duplication (Cont'd)

(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.

40. 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.

41. Flying Controls Rigidity and Balance

Requirement: Specification ARDCM 80-1, paragraph 9.206

When ------- power control systems are used, the rigidity and balance of the control surfaces shall be such as to preclude flutter or undesirable oscillations if the actuator or any one of the actuators used is disconnected for any reason, including battle damage.

Deviation: The rigidity of each control surface is dependent on multiple connections to the control tube. The control surfaces are not balanced.

Reason for Deviation and Remarks: This requirement is not compatible with the design aims of a fully powered, irreversible flying control system and, if it were met, it would involve prohibitive weight penalties.

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39. Control Cable Duplication (Cont'd)

(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.

40. 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.

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When ------- power control systems are used, the rigidity and balance of the control surfaces shall be such as to preclude flutter or undesirable oscillations if the actuator or any one of the actuators used is disconnected for any reason, including battle damage.

Deviation: The rigidity of each control surface is dependent on multiple connections to the control tube. The control surfaces are not balanced.

Reason for Deviation and Remarks: This requirement is not compatible with the design aims of a fully powered, irreversible flying control system and, if it were met, it would involve prohibitive weight penalties.

42. Emergency Flying Controls

Requirement: Specification ARDCM 80-1, paragraph 9.205

Where power boost or power control systems are employed, an emergency manual or power means shall be provided -----------------.

Deviation: Two separate hydraulic power circuits are used. Both are normally in use, but either system alone will provide sufficient power for adequate control of the aircraft.

Reason for Deviation and Remarks: The two separate hydraulic power circuits, with each being capable of automatically carrying on when the other has failed, provide a better emergency means of operation than a specific emergency source of power.

As a precautionary measure until system proving is complete, a hydraulic pump powered by a ram air turbine provides a source of power for the hydraulic system in the event of a double emergency (e.g. double engine flame-out).

43. Power Plant Change

Requirement: Specification ARDCM 80-1, paragraph 15.261 (a)

No special tools, other than a power-unit sling, shall be required for power plant change.

Deviation: A special tool is required for attaching and detaching the forward top mount on the Iroquois engine installation.

Reason for Deviation and Remarks: To simplify and speed up installation and removal procedures.

44. 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.

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45. **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 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, 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.

46. Number reserved.

47. **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 Nozzle

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

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47. Removal and Replacement of Fuel Nozzles (Cont'd)

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.

48. 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: Titanium used in place of stainless steel.

Reason for Deviation and Remarks: Weight

49. 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) Gear Box Assembly (Positioning only)
(4) Percentage Thrust Transducers (Positioning only)

Reason for Deviation and Remarks: Engine design dictates the necessity for handing the above items to permit an economical structure design.

50. 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.

50. Fuel Drain Valves (Cont'd)

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.

51. Refueling Connection

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

Refueling shall be accomplished through the use of a single adaptor unless otherwise specified --------.

Deviation: Two refueling adaptors are installed.

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

52. 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) RCAF 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.

53. Fuel Flow Meters

Requirement: Specification ARDCM 80-1, paragraph 19.241

Flow meters are required on all jet propelled aircraft -------.

53. Fuel Flow Meters (Cont'd)

Deviation: Fuel flow meters 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.

54. 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.

55. 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.

56. Installation or Removal of Fuel Tanks

Requirement: Specification ARDCM 80-1, paragraph 13.421 (c)

It shall be possible to remove tanks without removing any other part of the aircraft, except cowling or access panels. No dis-assembly of structural parts shall be required.

56. Installation or Removal of Fuel Tanks (Cont'd)

Deviation: Structural tie tubes must be removed to remove and install fuselage fuel cells.

Reason for Deviation and Remarks: Permits an "economical" fuselage structure.

RCAF approved at Co-ordinating Committee Meeting - 14 December 1955.

57. 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, right and left are located partly over the engines.

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

58. 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 ensure flow under extreme aircraft attitudes, such as inverted flight.

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59. **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.

60. **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. 59.

61. **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.

62. Fuel System Component Identification

Requirement: Specification MIL-F-8615, paragraph 3.5

Each fuel system component shall be marked by a red color in conformance with Army-Navy Aircraft Color Standard Code No. 509.

Deviation: No color marking will be made on fuel system components.

Reason for Deviation and Remarks: Authorized by CF105 Co-ordinating Committee at the 23rd Meeting held 23rd November 1955.

63. Number reserved

64. Collector Tank Outlets

Requirement: Specification ARDCM 80-1, paragraph 15.431

------- a removable cover plate containing both the booster pump flange and the fuel tank sump or an inspection plate arrangement containing all units and fittings of the tank as described ----- .

Deviation: The booster pumps are not mounted on removable cover plates.

Reason for Deviation and Remarks: Aircraft configuration dictates that the mountings, fuel outlets, and drives be on the bottom of the tanks. Structural strength requires a minimum size mounting hole. An inspection hand hole is located on the top surface of each tank.

65. Tank Selection - Refueling

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

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

Deviation: Two alternative groups of tanks can be filled, but selective individual tank filling is not provided.

Reason for Deviation and Remarks: Automatic cg control is provided by means of emptying tanks in a set sequence. Use of selective tank filling would act directly against the object of providing automatic control of cg position. (It is possible to avoid filling a battle damaged tank by disconnecting the electrical lead to the appropriate refueling servo control valve.).

66. Flying Controls Hydraulic Circuits

Requirement: Specification AIR 7-4, paragraph 4.7.3.2

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

(2) & (3) The design of the Flying Control Hydraulic System will permit full operation from 0°F to 250°F. Adequate control with limited maneouvreability 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°F.

(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.

67. 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.

Reason for Deviation and Remarks: The hydraulic systems are of the airless type and are sealed from contact with the atmosphere which would otherwise be the main cause of moisture entering the systems.

68. Emergency Retraction

Requirement: Specification AIR 7-4, paragraph 4.3.2.2

A control shall be installed to override the device for prevention of inadvertent retraction, thus allowing emergency retraction of the landing gear when the wheels are on the ground.

Deviation: No such control is installed.

Reason for Deviation and Remarks: The design and geometry of the landing gear makes this requirements undesirable.

69. 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 requirements 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.

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70. Marking of Pipe Lines and Conduits

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, Pipeline Identification, together with such additional markings as may be required by the Specification governing each system.

Deviation: Fuel immersed pipes and conduits are not identified

Reason for Deviation and Remarks: No satisfactory fuel immersible tapes are available.

RCAF authorization to omit identification markings on fuel lines immersed in fuel is contained in letter S1038-105 (ACE-1) dated 15 February 57.

71. 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 spring flexing seamless steel tubing.

Reason for Deviation and Remarks: (1) Flareless 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.

72. 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.

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72. Emergency Wheel Brakes (Cont'd)

Deviation: The emergency braking system is powered by an accumulator charged by the utility services main hydraulic power circuit.

Reason for Deviation and Remarks: Failure of the normal brake pressure piping in this design will not prevent operation of brakes from the emergency supply; however, failure of the emergency pressure piping up to the brake control valve will depressurize the entire utility hydraulic system. (This is similar in principle to the CF100.)

73. 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.

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74. 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 75.

75. 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 AC power cables are separate from emergency AC. Main AC power cables mainly isolated from remaining cables. Emergency AC cables are light wiring and run with distribution cabling.

Main DC cables mainly isolated from all others.
Emergency DC cables to services run with main DC lines.

Normal and emergency wiring must come together at transfer point.

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

76. 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.

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76. 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.

77. 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.

78. 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.)

79. 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 to the aircraft structure.

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79. Shielded Wires (Cont'd)

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.

80. 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.

81. Alternators and Drives Accessibility

Requirement: Specification MIL-E-7614, paragraphs 3.5.2 and 3.6.1

The constant speed drive and flexible shaft shall be accessible for inspection and servicing while installed and for removal for servicing without requiring the removal of other accessories except the generator.

Deviation: The constant speed drive shall be accessible for inspection, servicing and/or removal only when the engines are removed.

Reason for Deviation and Remarks: This is precluded by the engine installation.

82. 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.

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82. Reverse Current Cut-Outs - Accessibility (Cont'd)

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.

83. 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 Specifications 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.

84. Waveguides

Requirement: Specification MIL-W-9053, paragraph 3.6

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

Deviation: Waveguide 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.

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85. 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-10A 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-10A 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-10A 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.

86. Co-axial Connectors

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

Connectors for co-axial cables shall be in accordance with Specifications 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 87).

87. Co-axial Cables

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

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

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87. Co-Axial Cables (Cont'd)

Deviation: Co-axial 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 attentuation.

88. 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 from the service bay extinguishing system.

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.

89. Overheat Detection - Turbojet Engine Installation

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 potential fire zones.

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.

90. 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.

90. Ground Air Disconnects (Cont'd)

Deviation: Two 3-1/4 in. quick disconnect air conditioning system couplings shall be installed.

Reason for Deviation and Remarks: The type and size of the coupling installed is compatible with the duct sizes. Confirmed by RCAF letter S1038-105-11 (ACE-1) dated 22 August 1955.

91. 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, space, and system performance 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.

92. 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: Design of the system is predicated on a large pressure drop across the refrigeration turbine.

93. 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.

93. 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.

94. Moisture Elimination - Pneumatic System

Requirement: Specification CAP 479, paragraph 24.32 and 24.45

Pneumatic systems shall incoproate a dehydrating device. Traps shall be provided to collect and drain off moisture from the pneumatic ---------- systems.

Deviation: No dehydrating device is fitted. A trap is provided only in the low pressure services sub-systems.

Reason for Deviation and Remarks: The air supplied to the sub-systems is too hot for effective dehydration except in the case of the low pressure services sub-system. A filter which incorporates a drainable moisture trap is fitted to the low pressure sub-system.

95. Ducting Alignment

Requirement: Specification ARDCM 80-1, paragraph 12.444

(1) 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.
(2) 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.

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95. Ducting Alignment (Cont'd)

Deviation:

(1) Not complied with at some connections.
(2) The expansion cooling turbine and outlet ducting will constitute a firm assembly, suitable allowance being made for vibration and misalignment in ducting leading to and from the turbine.

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.

Flex ducting not considered suitable because of high pressure loss characteristic and sharp bends required.

96. Cabin Air Safety Valves

Requirement: Specification ARDCM 80-1, paragraph 12.460

A combined pressure relief and dump valve in conformance with Specification MIL-V-5379 shall be installed to perform the triple function defined herein.

(a) Positive pressure relief.....
(b) Negative pressure relief.....
(c) Emergency pressure relief....

Deviation: A combined valve to MIL-V-5379 is not fitted.

Reason for Deviation and Remarks: An arrangement utilizing two cabin pressure regulators and a separate inward relief valve gives the required protection and permits a lower blow-off pressure, thus reducing the design pressurization loads at low altitude.

97. Air Bleeds

Requirement: Specification ARDCM 80-1, paragraph 12.43

In order to regulate the quantity of air bleed from the high pressure air source so as to ensure that the quantity of air bleed is not so great as to result in loss of engine power -----, a flow limiting nozzle substantially in accordance with Drawing 44D20311 shall be installed........

Deviation: Flow limiter 44D20311 is not installed.

97. Air Bleeds (Cont'd)

Reason for Deviation and Remarks: The limiting nozzle as recommended would introduce a pressure drop which would adversely affect system performance.

Flow is limited by a variable nozzle on the refrigerating turbine, by pressure reducing valves, and by a leak detector system coupled to a shut-off valve.

98. Leveling Provisions

Requirement: Specification ARDCM 80-1, paragraph 8.53

Provision 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 Arrow.

99. 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 provided.

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

Reason for Deviation and Remarks: (1) The configuration and weight distributionin 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.

100. 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 RCAF 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.

101. 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.

102. 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 RCAF 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.

103. 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).

104. 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.

105. 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.

106. Fuses

Requirement: Specification CAP 479, paragraph 70.24 (1)

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

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

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

107. Landing Gear Indication System

Requirement: Specification MIL-E-7614, paragraph 3.19

A landing gear indication system shall be installed in accordance with Specification MIL-I-6339.

MIL-I-6399, paragraph 3.2

The landing gear position indicator shall conform to Drawing AN 5839.

Deviation: "Dowty" indicator type Q 1416 is installed.

Reason for Deviation and Remarks: "Dowty" indicator is installed in accordance with RCAF instructions (Reference letter S22-1-4 (ACE-1) dated 29 January 57).

108. Integrated Electronics Installation

Requirement: Specification AIR 7-4, paragraphs 7.1.1 and 7.2.4

An Integrated Electronic System ---- constructed to the requirements of AIR 7-5------- shall be installed to perform the following functions:--------.

The aircraft Contractor shall integrate and make compatible the automatic flight control sub-system defined in RCAF Specification INST 92-1 with the damping system defined in RCAF Specification INST 92-4.

108. Integrated Electronics Installation (Cont'd)

Deviation: Aircraft installations shall provide for (a) an Integrated Electronic System, including telecommunications, navigation, fire control, and automatic flight control sub-systems; and (b) a damping sub-system. Each of the above is designed to the requirements of its respective specification: (a) the RCA Astra I model specification and, (b) Avrocan Specification E-276.

Reason for Deviation and Remarks: Since the equipment is not designed to the requirements of AIR 7-5, INST 92-1, INST 92-4, Avro cannot comply with these specifications.

109. Limit Manoeuvre Load Factors

Requirement: Specification AIR 7-4, paragraph 5.2.2.1

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

Deviation:

(a) At a gross weight for stress analysis of 52,500 lb. (Reference paragraph 3.4) the positive limit manoeuvre load factor is 6.56 and the negative limit manoeuvre load factor is 2.69.

(b) Additionally, the positive limit load factor decreases from 6.56 as skin temperature increases.

Reason for Deviation and Remarks:

(a) Limit load factors of plus 7.33 and minus 3.00 were established for a stressing weight of 47,000 lb. at an early stage of the design. Consequent aircraft weight growth has therefore resulted in reduced load factors at the higher weight.

(b) Weakening of structure due to temperature rise.

110. Safety Limits of Pressurization

Requirement: Specification ARDCM 80-1, paragraph 12.22

The relative expansion of integral gases (RGE)......

$$RGE\ (max) = 17.0t + 2.1$$

where t is the time of decompression in seconds.

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110. Safety Limits of Pressurization (Cont'd)

Requirement: Specification ARDCM 80-1, paragraph 12.22 (Cont'd)

For all operational conditions in combat areas, RGE (Calc) must never be greater than RGE (Max).

Deviation: The cockpit pressure schedule does not comply with the limits of RGE (Max).

Reason for Deviation and Remarks: The above requirement is not compatible with Institute of Aviation Medicine recommendations for cockpit pressure altitudes.

111. Cockpit Pressurization:

Requirement: Specification AIR 7-4, paragraph 4.2.1

The cabin pressure differential shall be .... reaching a value of 4.5 plus 0.5, minus 0 psi at an ambient altitude of 60,000 feet. Above this point a constant pressure differential of 4.5 plus 0.5, minus 0 psi shall be maintained.

Deviation: The cockpit to atmosphere pressure differential at 50,000 feet and above is 4.25 ± 0.25 psi.

Reason for Deviation and Remarks: To provide for most effective use of system cooling while maintaining Institute of Aviation Medicine recommendations for cockpit altitudes. Approved by 34th meeting of Co-ordinating Committee, 19 June 57, Item 38 (b) (ii) (a).

112. Temperature Distribution - Cockpits

Requirement: Specification ARDCM 80-1, paragraph 12.474

The difference in temperature between any two points in a compartment shall not be greater than 10°F.

Deviation: The cockpit temperature difference between air inlet and outlet does not meet this requirement (actual difference to be determined by test).

Reason for Deviation and Remarks: To maintain a 10°F differential between cockpit inlet and outlet is not practicable due to the high air flows which would be required to maintain this differential at high altitudes.

113. Engineering Data

Requirement: Specification AIR 7-4, paragraph 14.1

--------- reports shall be submitted by the Contractor as detailed in CAP 479.

CAP 479, paragraph 2.02

-------. The following data are required:

(b) Wind Tunnel Test Reports;
(d) Performance Calculations;
(e) Stress Analysis;
(f) Structural Strength Test Reports;
(p) Aircraft Ground and Flight Test Reports;
(s) Functional Type Test Reports;
(t) Vendor's report;

Deviation:

(b) Wind Tunnel Test Reports: Data without explanatory text will be submitted if it is needed to fill a specific need.
(d) Performance Calculations: Requests for performance calculations will be complied with by the issue of reports termed Periodic Reports which will be drawn up to contain the specific data requested.
(e) Stress Analysis: An index to preliminary stress reports will be submitted. Stress reports requested will be submitted in preliminary form, not certified as to accuracy.
(f) Structural Strength Test Reports: Structural strength test reports compiled in support of stress analysis and/or as proof of complete airworthiness of structure will be available upon request.
(p) Aircraft Ground and Flight Test Reports: Flight test data will be included with the Performance Calculations in Periodic Reports on specific request.
(s) Functional Type Test Reports: On satisfactory completion of tests to Avrocan equipment specifications, Approval Statement will be issued to the RCAF. Supporting data will be available in Avro Engineering central files.
(t) Vendor's Reports: Vendor's reports will be available in Avro Engineering central files.

Reason for Deviation and Remarks: The procedures outlined appear to have adequately met RCAF needs in the past and provide for the most economical use of manpower and funds.

114. Turn Around Time

Requirement: Specification AIR 7-4, paragraph 6.4.6

Complete pressure refueling of the internal fuel system to the combat radius of action fuel load detailed in paragraph 3.6.1.1 shall be accomplished within the 5 minutes turn-around time of paragraph 4.7.2.2.

Deviation: The system shall permit 10 minute refueling to the combat mission fuel load.

Reason for Deviation and Remarks: Minutes of the 25th meeting of Co-ordinating Committee, item 12 (b), state; "DOR have confirmed that the turn-around time required for the CF105 has been increased from 5 minutes to 10 minutes".

115. Engine Instruments

Requirement: Specification AIR 7-4, paragraph 8.3.1

All engine instruments shall be of the electrical remote single indicating type and 2 inch case size in accordance with U.S. Drawing AND 10412 and shall be clamp mounted in accordance with MIL-C-6818.

Deviation: The combined percentage engine thrust, percentage afterburner thrust, and exhaust temperature indicators fitted are triple indicating, 3 inch case size instruments.

Reason for Deviation and Remarks: Compatibility with cockpit layout.

116. Missile Launching

Requirement: Specification AIR 7-4, paragraph 10.2

The Sparrow 2 missiles shall be ripple fired with the following firing combinations possible:

(a) Operation of the two (2) outboard launcher mechanisms.

Deviation: Provision for missile firing in pairs shall utilize the two forward launchers first.

Reason for Deviation and Remarks: This is dictated by the missile installation configuration.

117. Provision for Falcon Missiles

Requirement: Specification AIR 7-4, paragraph 10.3

Complete provision shall be made to carry eight (8) Falcon GAR 1A and/or GAR 1C missiles.

Deviation: No provision is made for the carriage of Falcon missiles.

Reason for Deviation and Remarks: Requirements deleted by RCAF letter Reference S1038-105-7-6 (Armt E), dated 19 January 1956.

118. Missile Wing and Fin Doors

Requirement: Specification AIR 7-4, paragraph 10.11

The doors shall open downward and be attached to the package by piano type hinges.

Deviation: Wing and fin doors open inward and are not attached by piano hinges.

Reason for Deviation and Remarks: Present door configuration presents the most economical structure.

119. Emergency Landing Loads

Requirement: Specification MIL-S-5705, paragraph 4.5.1.1.2

(Emergency landing loads) ---------- shall not be less than those listed below

Fighter ------------------ 32g Forward ---------------

Deviation: Design has been based on an ultimate forward load factor of 25g.

Reason for Deviation and Remarks: Recommended as design case by RCAF (Reference letter S1038-105-16 (ACE) dated 25 Jan 1955).

120. Oxygen Supply Pressure

Requirement: Specification AIR 7-4, paragraph 11.1.2

The oxygen system ---------- operating at 300 psi -----------

Deviation: A 70 psi system is installed.

Reason for Deviation and Remarks: A 300 psi system providing adequate operating characteristics is not presently available.

Appendix III

Engineering Data

Appendix IIIA Drawings

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

MRI-CA-C105/2

Appendix IIIB Reports

113 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

The following descriptive reports have been produced as Mock-Up Brochures and are listed herein as reference documents only:

The ASTRA 1 system	72/System 13/7
Engine Installation	72/AIREQ 25/1
Electrical System	72/System 11/27
Air Conditioning System	72/System 22/48
Low Pressure Pneumatics System	72/System 18/29
Fire Protection System	72/System 23/31
Protection against Ice	72/System 20/51
Fuel System	72/System 16/21
Hydraulics - Utility	72/System 19/26
- F/Controls	72/System 32/25
- Armament	72/System 19/40
Flying Controls System	72/System 15/28
Oxygen System	72/System 21/30
Armament System	72/System 26/8
Escape System	72/Eng Pub/2

Appendix IIID Novel Features

To be added.

APPENDIX IV

AMENDMENTS

MODEL SPECIFICATION AMENDMENT

AIRCRAFT TYPE	CONTRACT	AMENDMENT NO. 1 Page 1of 1
Arrow 2	B69-12-44 Serial 2-B-5-309 SO-4877	E.C.P. Nil
SUBJECT Radar Homer - AN/ARD-501		MOD. NO. Nil
REASON FOR CHANGE Typographical error		EFFECTIVITY 25206 RETROFIT Nil

EFFECT ON PERFORMANCE Nil	
WEIGHT CHANGE Nil	EFFECT ON BALANCE Nil

AMENDMENT

PARAGRAPH 3.17.4.7 **PAGE** 120

Delete

Complete provision shall be made for the installation of type AN/ARD-501 DF equipment to provide bearing and elevation of airborne enemy L & S band ECM stations.

Add

Space provision shall be made for the installation of type AN/ARD-501 DF equipment to provide bearing and elevation of airborne enemy L & S band ECM stations.

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