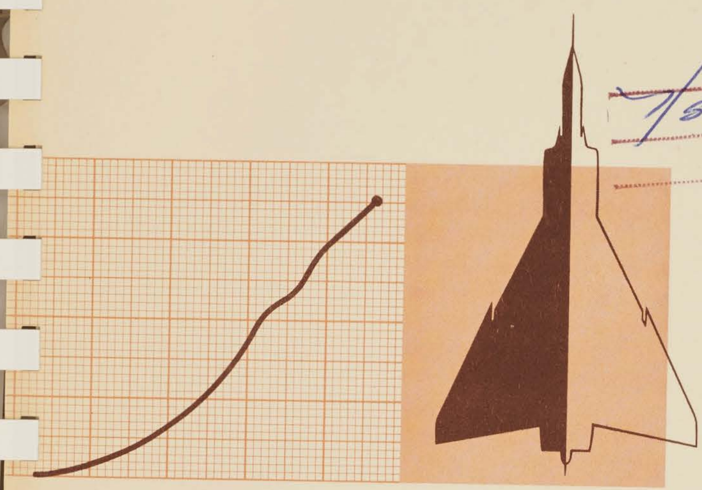

AVRO ARROW

quarterly technical report

FOR THE PERIOD ENDING

Sept. 30 1957

AVRO AIRCRAFT LIMITED

UNCLASSIFIED SECRET

ARROW

QUARTERLY TECHNICAL REPORT

70/ENG PUB/4

FOR PERIOD ENDING 30 SEPTEMBER 1957

Prepared By: PROJECT MANAGEMENT SERVICES

ENGINEERING DIVISION

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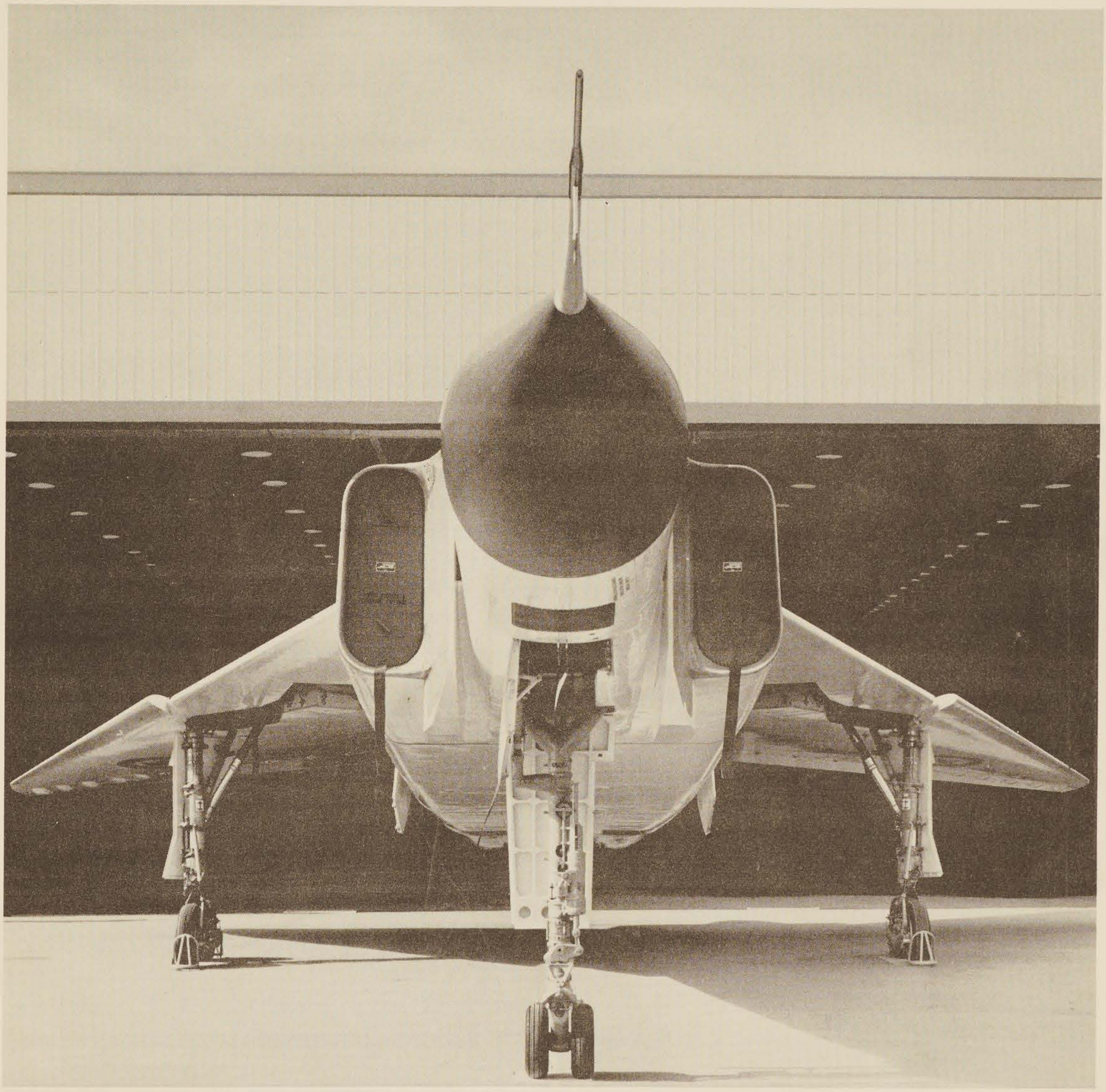

FIG. 1 AVRO ARROW 1

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1.0 INTRODUCTION

1.1 SCOPE OF REPORT

This Quarterly Technical Report on AVRO ARROW aircraft is the first in a series of such publications. It is compiled with the primary object of informing the Canadian Government of technical development of the project during the report period.

The report presents a description of work carried out and the ensuing results obtained in the design and development activities of the ARROW project. It summarizes technical progress, changes and problems in all phases of the program, during the report period. The text is divided into eight major sections which cover all aspects of design, testing and development.

Since this is the first ARROW Quarterly Technical Report to be issued, suggestions and recommendations for the improvement of the report, in format or content, will be welcomed.

1.2 SUMMARY

The ARROW is a high altitude, supersonic interceptor of advanced design, being developed by AVRO Aircraft Limited, at Malton, Ontario to RCAF Specification AIR 7-4 issue 3.

There are two versions of the ARROW, the ARROW 1 powered by two Pratt and Whitney J75 turbojets and the ARROW 2 powered by two Orenda Iriquois turbojets. The ARROW 1 is not armed and will fulfill the role of a development vehicle leading to production of the fully operational ARROW 2, which will incorporate Sparrow 2D air-to-air guided missiles and the ASTRA I electronic system. Both aircraft have essentially the same basic configuration, but the more powerful engines of the ARROW 2 give it superior performance.

The aircraft is designed to operate at altitudes up to 60,000 feet and at speeds in excess of Mach 1.5 with a minimum combat radius of action of 200 nautical miles and a time to 50,000 feet of approximately 5 minutes from engine start. It is characterized by its high wing, delta planform and general cleanness of design.

Production of the first ARROW 1 began early in 1955 and is now virtually complete. Manufacture of ARROW 2 details commenced in January of this year and is proceeding with increasing volume. A considerable amount of design and test work associated with the latter aircraft has, of course, been covered in connection with the ARROW 1.

1.3 DESCRIPTION OF ARROW

1.3.1 ARROW 1

The ARROW 1 is a development aircraft with the role of flight test vehicle leading to the production of the ARROW 2. It carries a crew of two, pilot and flight observer, in a pressurized and air conditioned cockpit with two split clam shell type canopies and automatic upward ejection seats.

The general appearance is of a low aspect ratio, high wing, delta planform aircraft with 4° anhedral and 0.75% negative camber. The wing leading edges are notched, extended in the outer wing and drooped. The tail unit consists of a vertical fin and rudder only, since no tailplane is required with this configuration. Both leading and trailing edges are swept back. Area rule theory is incorporated in the aerodynamic shape of the aircraft. The fuselage is of rounded cross-section from the nose probe to the engine air intakes where it evolves into a slab-sided, horizontally oblong cross-section, the front fuselage being slightly drooped to improve forward vision. Two side-by-side Pratt and Whitney J75 engines are installed in the fuselage with their air-intake fairings commencing immediately aft of the pilot's position. The engines are two-spool, axial flow turbojets with integral afterburners.

The airframe is an all-metal stressed-skin structure and consists of eleven major sections, the radar nose, front fuselage, centre fuselage, duct bay, engine bay, rear fuselage, inner and outer wings, elevators, ailerons, fin, rudder and dive brakes. The elevators and ailerons are hinged to the wing trailing edge forming part of the wing area. The landing gear is an electrically-controlled, hydraulically-actuated tricycle type, with the main gear retracting inward and forward into the inner wing and the steerable nose gear retracting forward into the front fuselage.

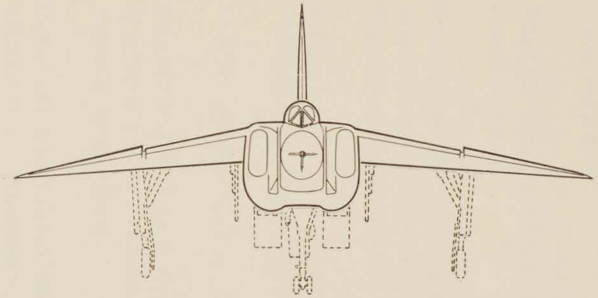
Space in the radar nose and weapon bay is utilized for test equipment and instrumentation to enable the aircraft to carry out its designated role as a flight test vehicle.

The landing gear, wheel brakes, nosewheel steering and speed brakes are actuated by a 4,000 psi utility hydraulic system. A hydraulic supply is available for the missile launching system, should it be required. Emergency air release of the landing gear is also available. The fully powered and irreversible flying control surfaces are operated by a separate 4,000 psi hydraulic system comprising two completely independent circuits.

Fuel is carried in integral wing tanks and bladder-cell type fuselage tanks.

An automatic cabin pressurization, ventilation and temperature control system is installed which is also used to cool electronic and electrical equipment in flight. The windshield, canopy, engine and engine air intakes and

Tank	Location	Capacity Imp.Gal.	Capacity Litres
1	Fuselage	252	1145
2	Fuselage	254	1155
3	Wing	151 each	686
4	Wing	90 each	409
5	Wing (collector)	146 each	664
6	Wing	154 each	700
7	Wing	279 each	1268
8	Wing	173 each	787
9	External Tank (Long Range)	500	2273



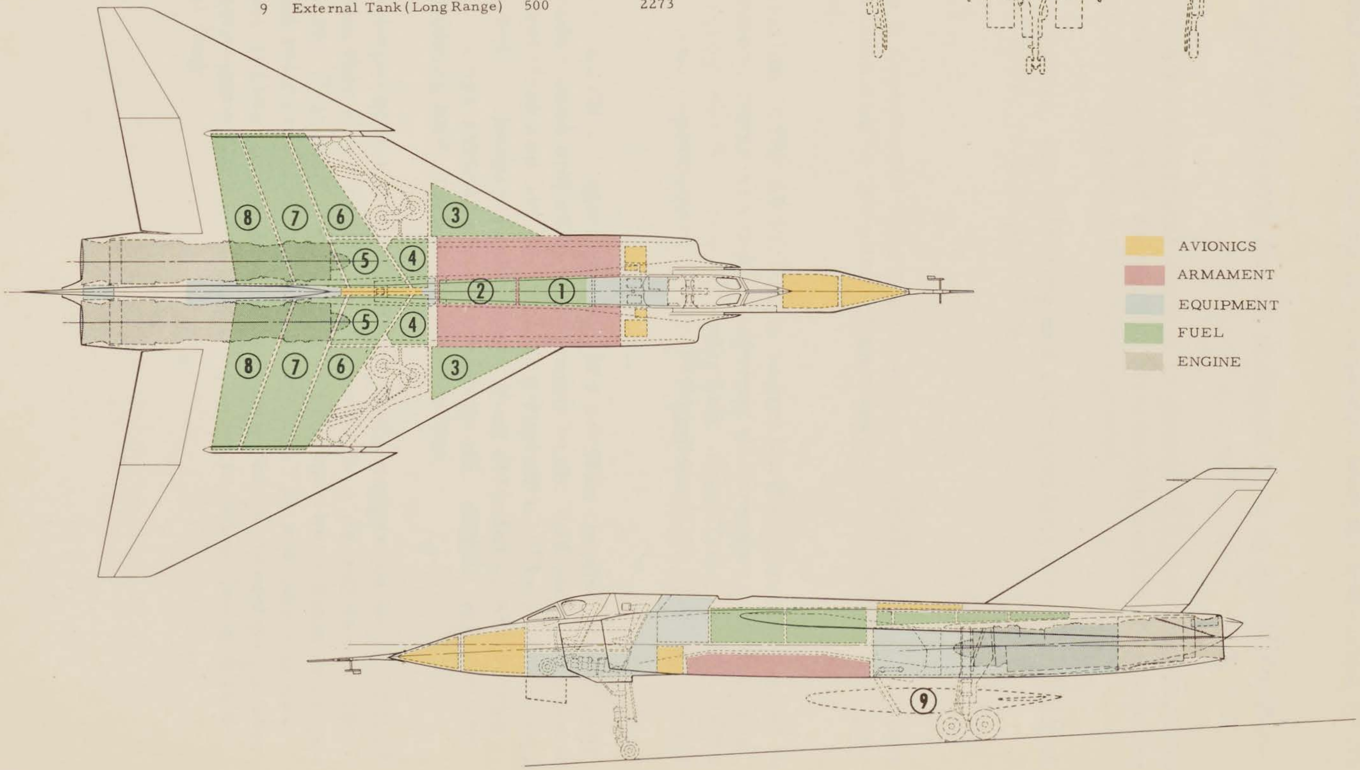


FIG. 3 EQUIPMENT ZONES

critical instrument reference areas are provided with an anti-icing or de-icing system.

Oxygen for the use of the crew is stored in liquid form in quickly removable containers.

Power for the aircraft's electrical system is provided by two engine-driven alternators with constant speed drives for alternating current and two transformer-rectifiers for conversion to direct current.

A fire protection system, for detecting and extinguishing fires in the hydraulics and engine bays, is also provided.

1.3.2 ARROW 2

The ARROW 2 is a supersonic interceptor aircraft capable of all weather, day or night operation and is the production version of the ARROW 1 previously described.

External configuration of the ARROW 2 is basically the same as that of the ARROW 1. However, there are major internal differences, namely the weapon pack carrying four Sparrow 2D missiles, installation of the ASTRA I electronic system and replacement of the J75 engines with the Orenda Iroquois.

The introduction of ASTRA I has necessitated certain cockpit alterations, redesign of the radar nose and other equipment bays, and considerable modification to the electrical and air conditioning systems. The use of Iroquois engines has resulted in changes to the inner wing structure, air-intake geometry, rear fuselage structure, engine controls, engine bay structure and the air conditioning and pressurization system.

The ARROW 2 incorporates certain other changes suggested by the RCAF or found to be desirable from research and tests carried out during the ARROW 1 program. The windows in the rear cockpit have been enlarged and the windshield rain repelling system deleted. The mechanical proportioner type fuel system of the ARROW 1 has been replaced by an electrically controlled sequencing systemand provision is made for a jettisonable external fuel tank.

1.4 FIXED DIMENSIONS AND GENERAL DATA

CHARACTERISTICS:	ARROW 1 and ARROW 2
Length of aircraft (excluding probe) - Ref. line level	77 ft. 9.65 in
Height of aircraft - over highest portion of fin	21 ft. 3.0 in
Ground angle - Angle between aircraft ref. line and ground static line	4.55 degrees
Tread of main wheels	25 ft. 1.56 in
Wheel base	30 ft. 1.0 in

WINGS:

Wing area (including ailerons, elevators and 390.5 sq. ft. of fuselage and not including 28.63 sq. ft. of extended leading edge).	1,225.0 sq. ft.
Span	50 ft. 0.0 in
Chord - Root	45 ft. 0.0 in
- Construction tip	4 ft. 4.98 in
Mean Aerodynamic Chord	30 ft. 2.61 in
Airfoil section - Inner wing profile NACA - 0003.5-6-3.7 (Modified)	
- Outer wing profile NACA - 0003.5-6-3.7 (Modified)	
NACA - 0003.8-6-3.7 (Modified)	
Camber	.0075 (modified)
Incidence - At root	Zero degrees
- At construction tip	Zero degrees
Anhedral of chord plane	4.0 degrees
Aspect ratio	2.04
Taper ratio	0.0889
Thickness ratio - Parallel to ℄ of aircraft	3.5 and 3.8%
Sweepback at 25% chord	55 degrees

AILERONS:

Aileron area (aft of hinge line) - Total	66.55 sq. ft.
Span (each)	10 ft. 0.0 in.
Chord (average percent of wing chord) - Root	25.735
- Tip	35.0

ELEVATORS:

Elevator area (aft of hinge line) - Total	109.90 sq. ft.
Span (each)	10 ft. 2.0 in.
Chord (average percent of wing chord) - Root	14.109
- Tip	25.735

CHARACTERISTICS:	ARROW 1 and ARROW 2
Vertical tail area (including rudder)	158.79 sq. ft.
Span	12 ft. 10.5 in.
Chord - Root	19 ft. 0.9 in.
- Construction tip	5 ft. 8.0 in.
Mean Aerodynamic chord	13 ft. 6.41 in.
Airfoil section	NACA .0004-6-3.7 (modified)
Sweep Back - Leading edge	59.34 degrees
- Trailing edge	33.08 degrees
- 1/4 chord	55.0
Aspect ratio	1.04
Taper ratio	0.2982
Thickness ratio (parallel to aircraft datum)	4.0%
Rudder area (aft of hinge line)	38.17 sq. ft.
Rudder - Span (average)	9 ft. 11.0 in.
- Chord (average percent vertical fin chord)	30.0

SPEED BRAKES:

Speed brake area (2) - Projected	14.37 sq. ft.
Span (each)	2 ft. 1.08 in.
Chord	4 ft. 1.0 in.

1.5 CONTROL SURFACES AND CORRESPONDING CONTROL MOVEMENTS

CHARACTERISTICS:	ARROW 1 and ARROW 2	
	Surface Movement	Control Movement
Ailerons: Up and Down	19°	14.20°
Elevators: Up	30°	14.50°
Down	20°	9.67°
Rudder: Left and Right	30°	3.25 in.
Speed Brakes	60°	

PART 2

TECHNICAL DESIGN

2.0 WEIGHT AND CENTRE OF GRAVITY

2.1 ARROW WEIGHT HISTORY

Based on the monthly statements of weight and balance, a brief review of the airplane weight progress is presented in chart form. Significant weight changes are noted by appropriate explanations on the charts.

Figure 4 covers the period from the first recorded weight for the J75 powered aircraft to the time when the decision was made to distinguish between the J75 powered test vehicles and the Iroquois powered operational aircraft. The "Operational Weight Empty" is used in this chart since it is more representative of airplane weight growth. Figure 5 gives the weight history of the first J75 powered test vehicle. Figure 6 gives the weight history for the operational Iroquois powered aircraft. An elaboration of the reasons for the weight increases shown in Figure 6 is given in the following.

For the period February 1 to March 1 a weight increase of approximately 700 lb. is shown for the normal combat mission gross weight. This is due to an increase of normal combat mission fuel due to earlier revisions in the performance estimates for the aircraft.

During the period August 1 to September 1, the weight changes summarized below occurred:

1. Radome: + 70 lb.

Due to unsatisfactory electrical properties, the filled honeycomb radome construction has been replaced by heavier solid laminate construction.

2. Floating Duct: + 65 lb.

A part of the engine air intake system, the weight for this item was originally estimated from preliminary schemes and drawings. Weight is now based on detailed design drawings.

3. Rear Fuselage: + 170 lb.

Revised weight estimated for stinger and tailcone. Previous weight estimates were based on an all-titanium tailcone structure. The revised estimate assumes the use of N155 heat-resistant Co-Ni-Cr-Fe alloy with titanium outer skin.

4. Engines: + 160 lb.

Orenda's earlier estimate did not include inlet frames.

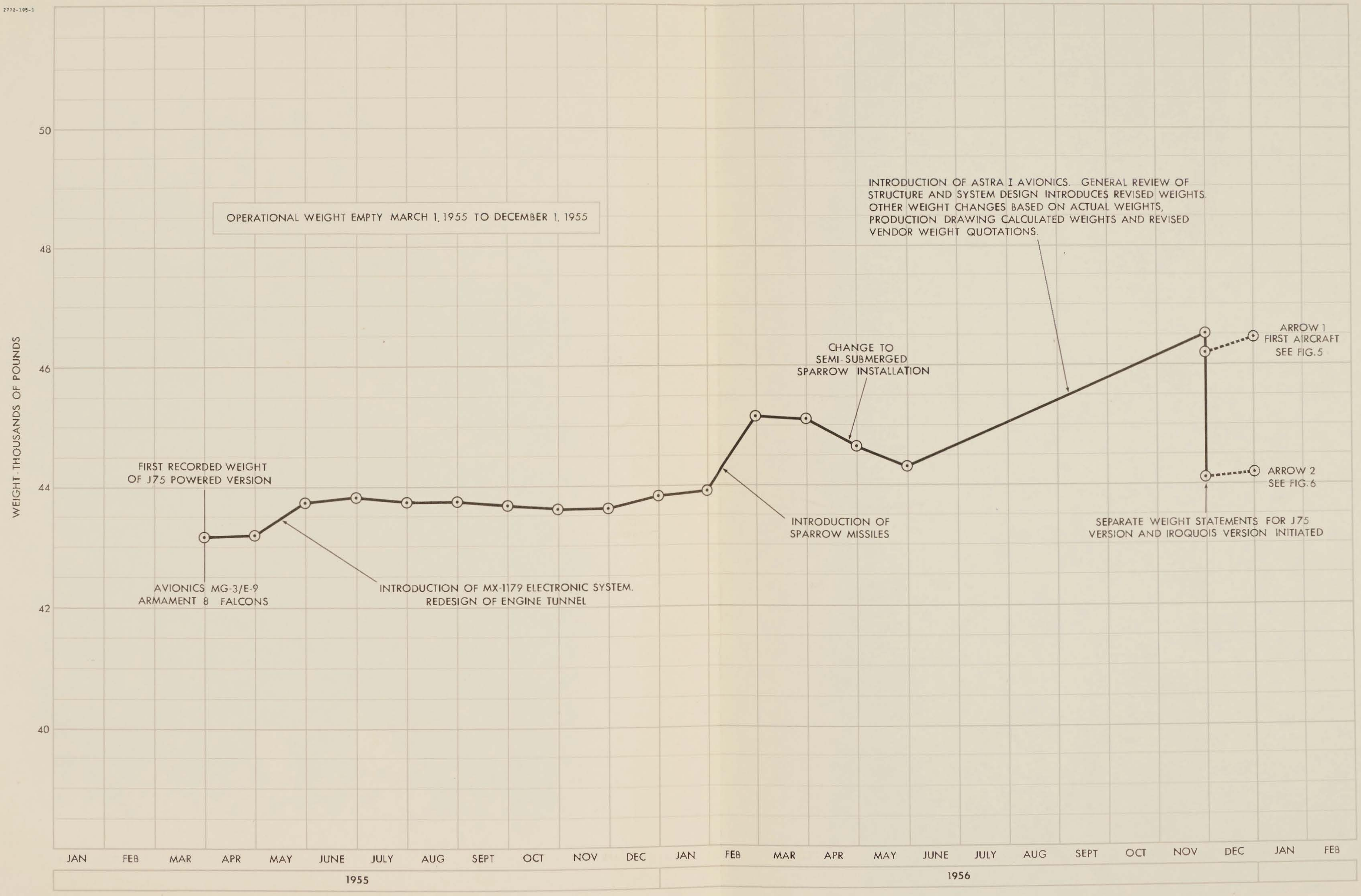


FIG. 4 ARROW WEIGHT HISTORY - OPERATIONAL VERSION WITH PRATT AND WHITNEY J75 ENGINES

5. Radar System: + 125 lb.

Revised weights from FCA have been incorporated. In addition the IR tracker system and radar homer antenna, which had previously been deleted, are now reinstated as basic requirements.

Increase in Operational Weight - empty sum of items 1 to 5 above = 590 lb.

6. Fuel: +110 lb.

The fuel requirement for the normal combat mission has been increased to cater for aircraft weight increases.

Increase in Normal Combat Mission Gross Weight 700 lb.

2.2 WEIGHT STATEMENT

A weight statement, based on the monthly Weight and Balance Reports (Ref. 1 and 2) is given in Table 'A' and Table 'B'. The definition of terms used in the table is given in para. 2.9.

2.3 ARROW 1 WEIGHTS

TABLE A

ARROW 1 - 1st Aircraft - Weight Summary

Airframe	21141 pounds
Power Plant	13099 pounds
Primary Systems and Services	5623 pounds
Ancillary Systems, Equipment and Provisions	1084 pounds
Avionics	782 pounds
Flight Test Provisions	2011 pounds
Flight Test Instrumentation	3036 pounds
BASIC WEIGHT	46776 pounds
Useful Load	983 pounds
Ballast	959 pounds
OPERATIONAL WEIGHT EMPTY	48718 pounds

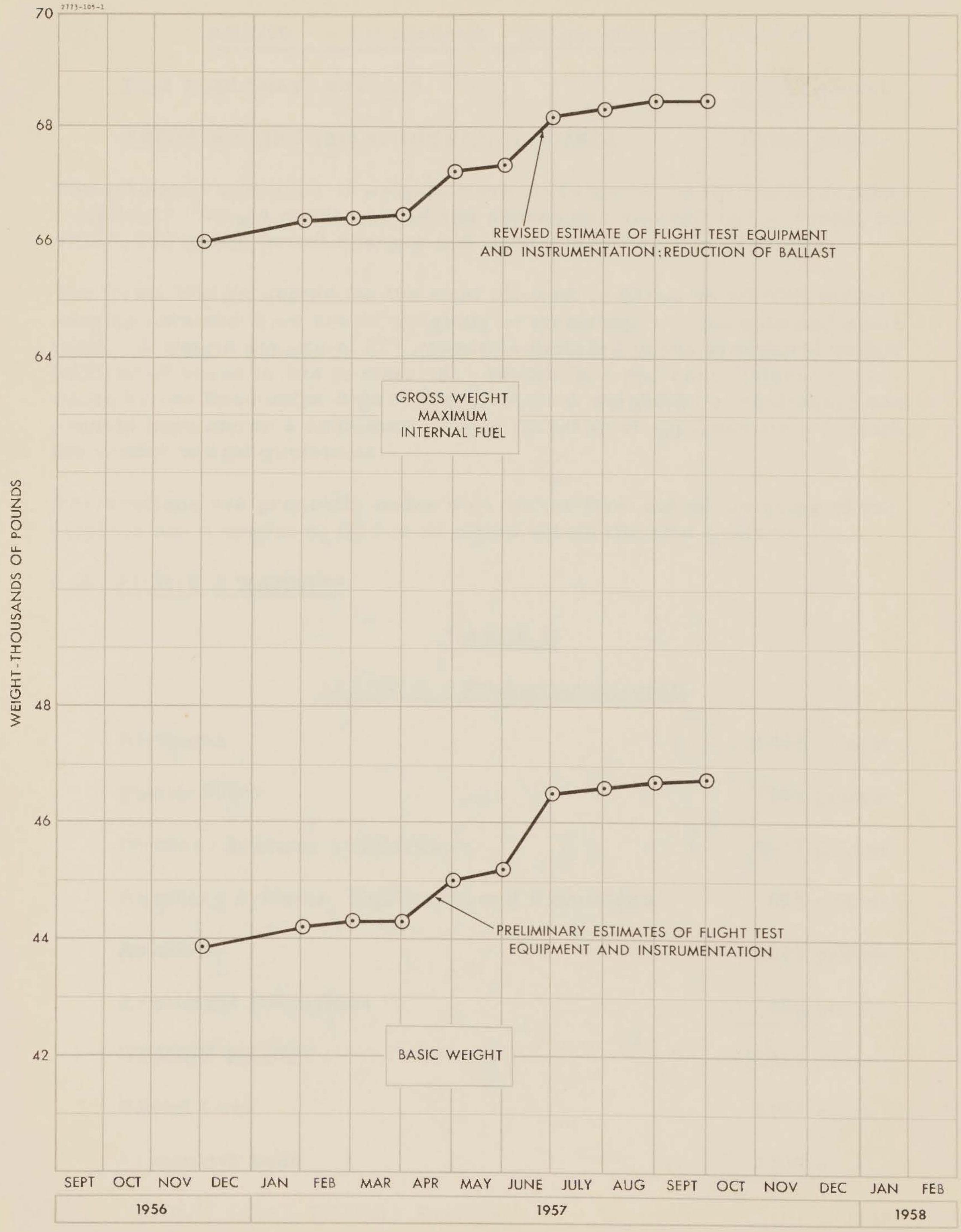


FIG. 5 WEIGHT HISTORY - ARROW 1, FIRST AIRCRAFT

ARROW 1 - 1st Aircraft - Weight Summary (Cont'd)

Fuel Load (Max. Internal)	19843 pounds
GROSS WEIGHT (MAX. INTERNAL FUEL)	68561 pounds

The tabulated summary of weights given in Table 'A' is for the first ARROW 1 aircraft. Weight of the second and subsequent aircraft in the series will differ, due to equipment changes and instrumentation installations.

The Basic Weight quoted for the first aircraft is based on 69.33% actual weights obtained from actual weighing of structural components and equipment. A weight penalty of 203 pounds is included in the structural weight (airframe) which is due to material substitutions and concessions introduced by the Production departments. Actual weighings of equipment components have shown a consistent weight increase of approximately 11% over the vendor weight quotations.

Preparations are presently under way for the first actual weighing of the airplane and a report on this is expected within the next quarter.

2.4 ARROW 2 WEIGHTS

TABLE B

ARROW 2 - Production Aircraft

Airframe	21448 pounds
Power Plant	9705 pounds
Primary Systems and Services	5575 pounds
Ancillary Systems, Equipment and Provisions	855 pounds
Avionics	2677 pounds
Armament Provisions	2050 pounds
WEIGHT EMPTY	42310 pounds
Useful Load	1123 pounds
Armament Load	1728 pounds
OPERATIONAL WEIGHT EMPTY	45161 pounds

ARROW 2 - Production Aircraft (Cont'd)

Fuel Load (Normal Combat Mission)	15940 pounds
GROSS WEIGHT (Normal Combat Mission)	61101 pounds

The tabulated summary of weights given in Table 'B' is for a fully operational ARROW 2 aircraft. It should be noted that the early ARROW 2 aircraft will be test vehicles, in which case this weight statement is not applicable.

The weights quoted here include the ARROW 1 weights which are applicable to the ARROW 2 aircraft. Hence, the weight quoted in this table is a combination of actual, calculated, estimated and vendor weights. The corresponding percentage breakdown is not presently available.

2.5 CENTRE OF GRAVITY

The centre of gravity of the aircraft for various flight conditions is given in graphical form in Figure 7. The centre of gravity data corresponds with the weights quoted in the weight statements of Table 'A' and Table 'B'.

2.6 ARROW 1 CENTRE OF GRAVITY

The ARROW 1 aircraft is ballasted to limit the extreme aft flight centre of gravity (C.G.) to 31% of the Mean Aerodynamic Chord (MAC). The C.G. envelope shown in Figure 7 is based on the use of fuel proportioners which maintain a constant fuel C.G., regardless of the quantity of fuel being carried. For the extreme aft C.G. condition, it has been assumed that the water for the air-conditioning system and the alcohol for the intake de-icing system have been completely consumed.

2.7 ARROW 2 CENTRE OF GRAVITY

A C.G. envelope for the ARROW 2 aircraft is not included here, since a fuel sequencing order has not as yet been established. Assuming the extreme forward C.G. occurs at the operational weight empty with the undercarriage retracted, and the extreme aft C.G. occurs at the operational weight empty, less missiles, with the undercarriage extended, the following is observed:

Extreme forward C.G.	27.82% MAC
Extreme aft C.G.	30.04% MAC

Since the C.G. of the maximum internal fuel load is approximately 30% of MAC, it is reasonable to assume that a fuel sequencing order which limits the airplane C.G. to the required limits of 28% MAC allowable forward limits of 28% MAC allowable forward limit to 31% MAC allowable aft limit is possible, without resorting to the use of ballast.

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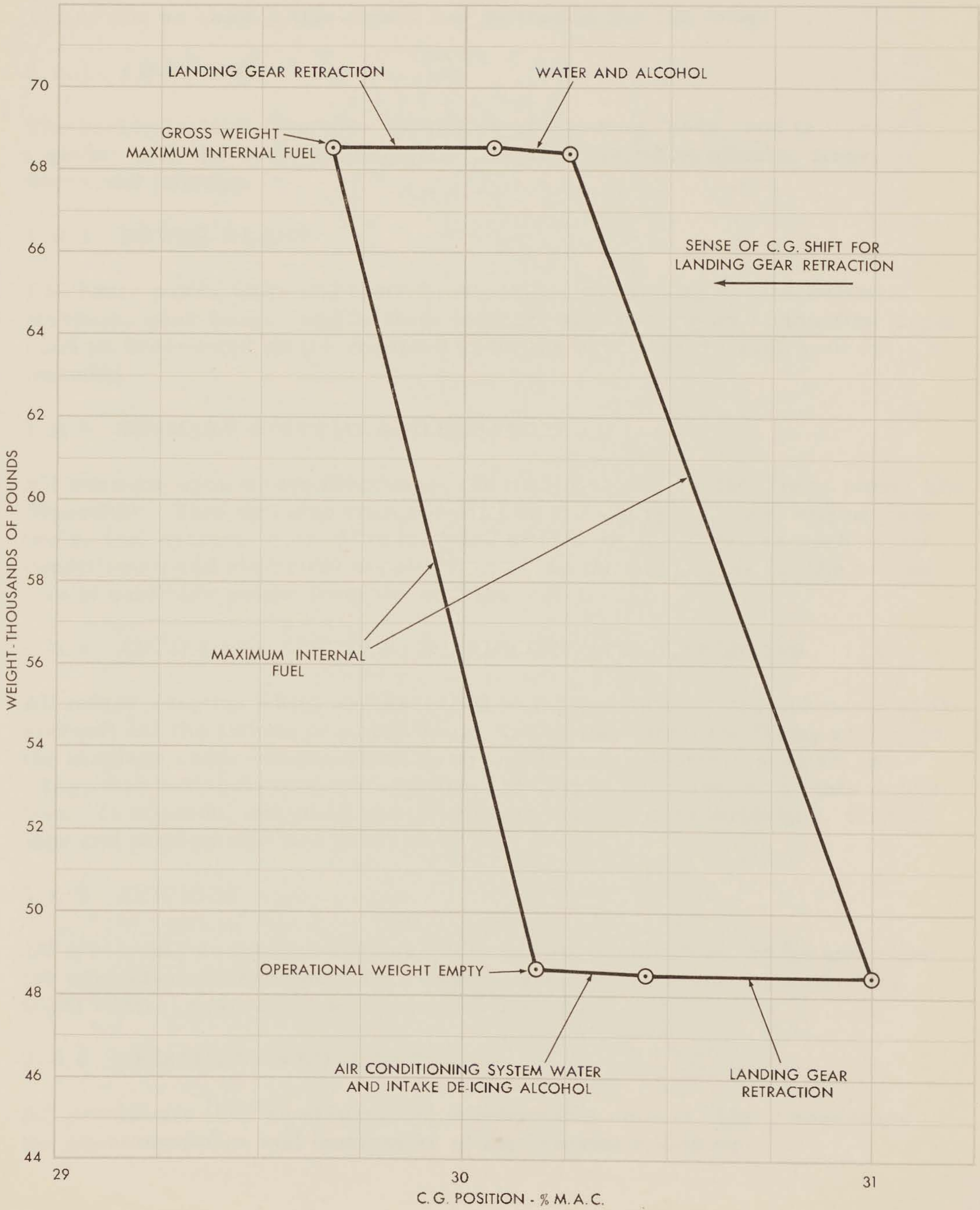

FIG. 7 ENVELOPE FOR IN-FLIGHT CENTRE OF GRAVITY POSITION, ARROW 1 - FIRST AIRCRAFT

2.8 WEIGHT AND CENTRE OF GRAVITY DEFINITIONS

The terms as used in this report are defined in the following:

2.8.1 AIRFRAME

The basic airplane structure consisting of the wing, body, and tail group, together with the basic landing gear structure including wheels, tires, doors and fairings.

2.8.2 POWER PLANT

The basic power units and their accessories, consisting of afterburners, starters, gear boxes, and fairings integral with the engine. Mounting items such as braces and struts required to tie the engine to the airframe are included.

2.8.3 PRIMARY SYSTEMS AND SERVICES

All systems upon whose functioning the airframe and/or the power plant, are dependent. This includes such systems as the flying controls, engine controls, fuel system, etc. Also included are the major services such as air conditioning and electrical supply, as well as the provisions for the extraction of auxiliary power from the engines.

2.8.4 ANCILLARY SYSTEMS, EQUIPMENT AND PROVISIONS

All minor systems which are essential to the operational requirements of the aircraft but the failure of which do not in any way affect the flying ability of the airplane under most normal flying conditions. Systems such as anti-icing, fire extinguishing, pneumatics and oxygen are covered by this definition. In addition, services and provisions such as pressurization, furnishings and instruments are covered by this group.

2.8.5 AVIONICS

All electronic equipment such as radio for communication, radio and radar for navigation, search and track radar, fire control systems, automatic flight control systems, and integrated systems.

2.8.6 ARMAMENT PROVISIONS

All provisions such as structures, mechanisms, power, etc., necessary to the accommodation and functioning of the armament system.

2.8.7 WEIGHT EMPTY

The sum of all the groups defined above, that is items 2.8.1 to 2.8.6 inclusive. This definition of weight empty corresponds to the definition given in CAP 479, Part 3, Chapter 30.

2.8.8 USEFUL LOAD

Crew and crew gear, residual fuel, and expendable fluids such as engine oil, de-icing fluids, fire extinguishing fluids, and oxygen system charge.

Useable fuel and expendable armament are not included in this definition.

2.8.9 ARMAMENT LOAD

The expendable portion of the armament system; the missiles in this particular case.

2.8.10 OPERATIONAL WEIGHT EMPTY

The sum of the weight empty, operating load and armament load.

2.8.11 FUEL LOAD

The useable fuel available for a specified mission. The fuel load is usually qualified, for example: normal combat mission fuel, maximum internal fuel, etc.

2.8.12 GROSS WEIGHT

The sum of the operational weight empty and the fuel load. Gross weight should be qualified in the same manner as fuel load.

2.8.13 FLIGHT TEST PROVISIONS

The fixed installations for flight test instrumentation, and the instrument pack structure.

2.8.14 FLIGHT TEST INSTRUMENTATION

The special instrumentation within the removable instrument pack.

2.8.15 BASIC WEIGHT

The gross weight less the useable fuel and the operating load. This definition corresponds to the definitions given in the model specifications for ARROW 1 and ARROW 2 aircraft.

3.0 PERFORMANCE

Performance characteristics, for both the ARROW 1 and ARROW 2, are at present being revised. This revision, scheduled for issue in November 1957, is a result of new propulsion data now available to AVRO on the J75 and Iroquois engines. (See para. 3.1). Since the present performance characteristics released in December 1956 are now out of date they will not be included in this issue. The next ARROW Quarterly Technical Report will, however, present a complete discussion of the new performance characteristics.

3.1 PROPULSION

(a) ARROW 1 - J75 Engines

Installed engine performance characteristics are at present being revised.

Ejector characteristics determined experimentally at the Orenda Nobel Test Facility have necessitated a detailed re-estimate of installed thrust. These characteristics confirm the performance prediction of pumping and thrust properties for both fixed cylindrical and divergent ejector geometries as finalized for the J75 and Iroquois engine installations. This work is being done for a 39-inch divergent ejector, to a limited extent, and a 45-inch divergent ejector. No figures have been published up to time of printing.

(b) ARROW 2 - Iroquois Series 2 Engines

As described above for the ARROW 1, the installed engine performance characteristics are at present being revised. This will be the first detailed thrust estimate for the Iroquois Series 2 using Orenda non-dimensional performance data. These figures will not be available until after printing of this report.

3.2 TACTICAL EVALUATION

A tactical evaluation group has recently been organized within the Technical Design department of AVRO's Engineering Division. The objectives of this group are the evaluation of the ARROW as a weapon system and the initiation and examination of proposals for its development and improvement as such. Where improvements in systems outside company control are considered, investigations are carried out with a view to examining their influence on the development of the aircraft. At the present time work is being done in three sections: operation in the midcourse phase, evaluation studies of the terminal phase and studies of area defence capabilities.

3.3 MIDCOURSE PHASE STUDIES

A specification is being written for the complete simulation of the aircraft's operation in the SAGE (Semi-Automatic Ground Environment) system. In addition to the aircraft performance other factors such as radar errors and noise, blip/scan ratio, tracking and data smoothing delays will be included for as accurate simulation as possible. The actual sequence of operation of the SAGE system will be represented as nearly as possible. Programming for parts of the simulation, to be carried out on the IBM 704 computer, has started and some results should be available by the end of the year.

3.4 TERMINAL PHASE STUDIES

Mechanizations are currently being set up for both analog and digital computer solutions of the fire control problem and the evaluation of positioning probabilities in the terminal phase of combat. To obtain an appreciation of such quantities as radar range requirements, optimum aspect of attack and duration in the firing zone, several small scale studies have been carried out using desk calculation methods. These methods are also being used to investigate a combined lead-collision/lead-pursuit mode of attack. A study of fuel requirements in the combat phase of an interception has been made for the purpose of isolating the parameters of fuel management; however, results have been somewhat inconclusive.

3.5 AREA DEFENCE STUDIES

An investigation has been started to determine the capabilities of the ARROW in the defence of North America. At present only operation within the SAGE system is being examined; but, various suggestions for the improvement of the close control radar facilities are being considered. Studies will shortly be extended to include operation on broadcast control, and operation on A.I. (airborne interception) search, with a minimum of ground support.

4.0 STABILITY AND CONTROL

Stability augmentation was shown to be necessary following the analysis of wind tunnel tests made during early design stages. These tests indicated that natural damping alone was not capable of providing flying characteristics compatible with the role of the aircraft. For this reason, electronic equipment has been developed and integrated with the flying controls to provide the "damper system". This system has been developed, for the most part, by the Minneapolis-Honeywell Regulator Company. The damper system block diagrams and gain schedules have been established and finalized. The greatest part of the remaining work lies in the testing and dynamic analysis of the system.

Rudder monitor trouble has been experienced since a configuration to cope with all conditions of flight had not previously been finalized. Certain modifications have now been made and further development is in progress. The final unit will provide protection against malfunction of the system by switching over to emergency whenever transverse acceleration or sideslip approaches structural integrity limits. The switching mechanism will operate on all three axes e.g. whenever normal mode in the rudder axis fails the monitor will switch not only the rudder axis to emergency but the remaining two as well. An interim version of the monitor will be installed in the first aircraft if available.

The idea of the rudder monitor operation is based on either one of two transverse accelerometers and a sideslip switch. In the two accelerometer configuration one accelerometer (designated A_{13}) is located at 13 feet forward from centre of gravity; the other (termed A_{40}) is 40 feet forward from centre of gravity. The accelerometers are set to open at .25 g and .40 'g' respectively. The circuit connections are as shown in Figure 8.

Switching to emergency will occur only when both accelerometers register accelerations larger or equal to values shown above, providing that both act in the same direction. Such a combination of accelerations is possible only in the case of rudder runaway, because in co-ordinated rolling manoeuvres accelerations will have opposite signs. In fact A_{13} detects directions of sideslip while A_{40} provides an indication of yawing acceleration. If yawing acceleration is such that it helps to reduce sideslip the cut-out will occur at relatively large values of acceleration (about .6 - .7 g at centre of gravity), but when it acts in the opposite sense the cut-out will occur at relatively low values (.2 - .3 g at centre of gravity). This complicated monitor configuration results from cross coupling considerations. In some manoeuvers relatively large values of transverse and yawing accelerations are needed and the system must be able to pass these and cut out only when failures occur. Accelerometers alone will not produce desired functions at low speeds since a small acceleration may correspond to a rather high sideslip angle. Since the aircraft has undesirable characteristics above angle of sideslip of approximately 12 - 14 degrees, an additional switch is provided when angle of sideslip exceeds 10 degrees. (Ref. 3).

Recently a somewhat simpler version of the rudder monitor has come under consideration. The new proposal contains one accelerometer and a switch. The mode of operation is similar to that of the version using two accelerometers, but details are not yet available.

The g limiting system, applicable to ARROW 2 aircraft only, has also undergone some modifications. The finalized version is as described below. A g limiter is provided to disengage the system in case of malfunction of any of the components. The limiter operates from two accelerometers (one forward and one aft from the centre of gravity of the aircraft) and two servo positions pick-offs (parallel and differential). It is necessary to combine these signals in an amplifier through filtering networks to obtain enough anticipation and ensure proper functioning of the limiter under all flight conditions. A differential servo pick-off was added to provide protection against differential servo hard-over failure. A separate pick-off is provided on each differential servo. Parallel servo pick-off protects against servo runaway type of failure. The g limiter will automatically disengage the normal pitch axis at such a level of normal acceleration that resulting overshoot will not exceed structural limits of the aircraft in all flight conditions. The rudder monitor signal is also connected to the g limiter to disengage the normal mode whenever the rudder axis fails. The disengagement is obtained by a contact on the parallel servo which releases the pressure and another contact which recentres the differential servo. The differential servo will recentre immediately only if this action is going to reduce the resulting load factor, otherwise the differential servo is centred by normal damper action and then disengaged. When the disengagement is completed the aircraft will be in the emergency mode of control and the pilot is then responsible for not exceeding structural limits of the airframe - (Ref 4).

Block diagrams of the various modes of the Automatic Flight Control System (AFCS) have been obtained from Minneapolis-Honeywell and detailed analysis is being carried out. Compatability of the AFCS with the damping system and with the requirements for tactical application have yet to be established. Gain schedule mechanization has yet to be established. All these problems are being resolved in co-operation with Minneapolis-Honeywell and RCA.

There has been some electrical trouble in the stick-force transducer and associated loop, causing stick-force characteristics objectionable to the pilot This problem is being investigated as a top priority item. The main control valves are also a source of undesirable stick-force characteristics. This problem has been solved in part, but investigation is being continued.

4.1 WIND TUNNEL TESTING

Wind tunnel results from NAE and NACA combined with free flight model results (for which the analysis is now completed) have been sent to Minneapolis-Honeywell where they are being investigated by computer to ensure that damper system parameters are not affected.

The relative wind sensor (α - β vane) will be undergoing dynamic response and calibration testing at the Cornell Aeronautical Laboratory transonic tunnel. These tests should be completed by the end of October 1957. The pitot static system, located in the same unit, will be tested for icing at NAE during October.

A preliminary investigation is being made into the post stall gyrations of the ARROW. The equipment and model requirements, necessary to complete a program of testing are being determined.

Tests to determine subsonic spin and recovery characteristics are nearing completion at NAE. At the present time, spin recovery parachutes are being tested. Test results indicate that spins are oscillatory with fairly high rates of roll and yaw developed and that the aircraft recovers with control motions typical for a delta configuration but which may be objectionable to the pilot.

Canopy hinge moment tests have been carried out at the Cornell Aeronautical Laboratories and data correlated with AVRO test results on the canopy opening mechanism. Wind tunnel tests on the release of the pilot's seat are scheduled for December of this year at NAE. The path of the seat with respect to the aircraft and the stability of the seat are to be investigated. Detailed test cases have been evaluated for a proposed sled program.

Missile jettison tests are now completed and are reported in P/Wind Tunnel/139. As a conclusion, the jettison characteristics are acceptable and no sequencing is required other than in the case of forward missile jettisoning with rear missiles extended. In this particular case, the rear missiles must either be jettisoned first or retracted before the release of the forward missiles.

Dynamic analysis of captive flight is being carried out to ensure that launching deflections will not exceed the limits of launching accuracy required. Wind tunnel testing has been used to investigate the initial trajectories of the missiles, and the aerodynamic forces on the missiles in the vicinity of the aircraft. The object of thest tests is to ensure that a safe clearance exists between the missile and the aircraft with due allowance for the effects of tolerance on missile wing setting and thrust misalignment. The wind tunnel results were correlated with theory to obtain a full flow pattern in the vicinity of the aircraft. This analysis shows that no sequencing is required for firing the missiles and that sufficient clearance does exist for all flight cases within the flight envelope of the missile launch.

4.2 DAMPER SIMULATOR AND COCKPIT RIG TEST

The damping system simulator has been operated in conjunction with an analog computer and the cockpit rig. The combination called the aircraft simulator, has been used to assess the operation and performance characteristics of the damper alone and to establish low speed handling characteristics of the aircraft. At the present time the aircraft simulator is undergoing modification to be made compatible with the finalized version of the damper. New and more complete instrumentation, hydraulics and hardware are being installed in the cockpit rig. Additional equipment, which had to be manufactured by AVRO, is being added in the analog computer. Extensive wiring changes are also being made between the basic units of the aircraft simulator. Final simulation will be in seven degrees of freedom. The remaining phases of the simulator test program will be started in November 1957.

The flying controls (B-1) test rig is being prepared for testing the flying controls with the damper simulator, in three axes, roll, pitch and yaw, separately and in combination. The cockpit rig will not be required for these tests, as was originally planned. A small instrument panel, to replace the cockpit rig, will be produced to facilitate the "flying" of the rig. Testing will start in October.

The first aircraft will be connected to the computer, and the damper system of the aircraft tested for several days prior to flight. At the present time the wiring and switching is being laid out and recording equipment set up.

Specifications are being prepared for a flight test data reduction process to be performed on the IBM 704. They will indicate the data to be reduced, the equations to be used, how they are to be used and the method of presenting the results. Because of the vast number of digits that will make up the results, it will be necessary to have these results automatically plotted. Additional machinery for this plotting process will be required. There are approximately one hundred channels of information available for stability and control analysis, including forty channels of damping system items.

A flight test program is at present being prepared which will describe the flight conditions to be tested, how they are to be tested and the sequence of testing.

A report on compliance with Specification MIL-F-8785, Flying Qualities of Piloted Aircraft, is being prepared.

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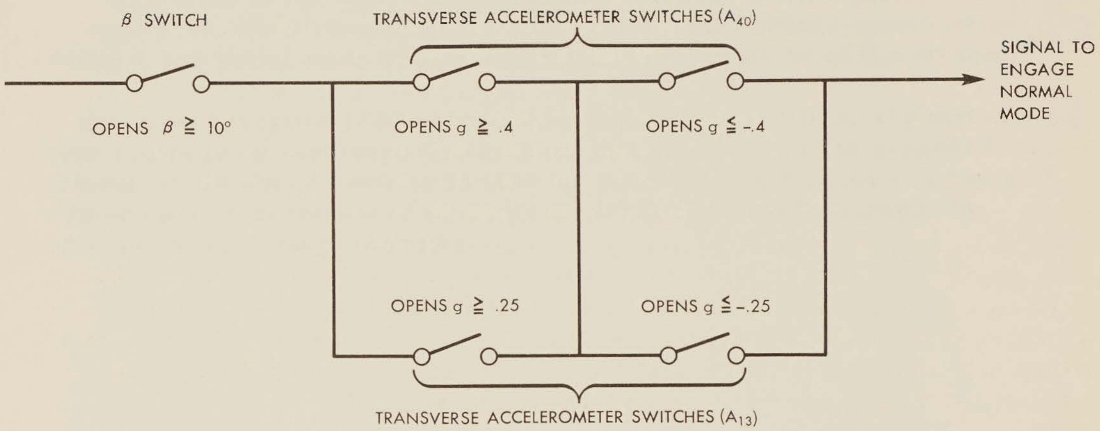

FIG. 8 RUDDER MONITOR SWITCH DIAGRAM

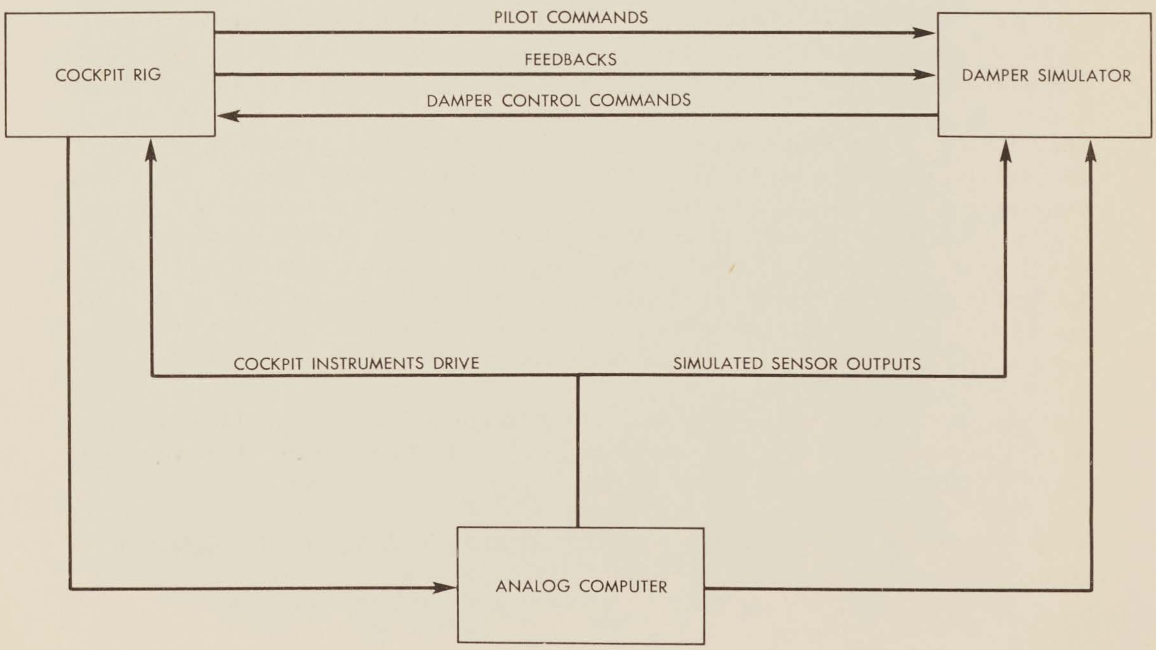

FIG. 9 DAMPER SIMULATOR AND COCKPIT RIG TEST BLOCK DIAGRAM

5.0 AIRLOADS

The calculation of subsonic and supersonic airloads on the ARROW, has, for the most part, been completed for some time. Only minor modifications and additions are being made to airload data on various parts of the aircraft.

To clear the operation of the air brakes in supersonic flight, the airloads on these components have been determined analytically. At the present time, airloads on the fin and rudder for the asymmetric flight cases are being revised, based on results of dynamic analysis of aircraft manoeuvres obtained on the analog computer.

6.0 THERMODYNAMICS

6.1 HEATING EFFECT OF EXHAUST GASES

Following the redesign of the rear fuselage of the ARROW 1, a reinvestigation was necessary to determine the heating effect of radiation from the hot exhaust gases on the surrounding structure. As a result of the publication of more complete data on heat radiation from jet gases a more thorough investigation has been carried out than was done initially. This data has indicated that the heating of the structure by gas radiation is a minor effect. It is by the process of convection that the greatest amount of heat is transferred to the structure. This process depends largely on the mixing which takes place between the cool bypass air and the hot efflux gases. The mixing process is, in turn, a function of the relative velocity between the two gas streams and the shock wave pattern at the nozzle exit. At present there is no analytical method which makes it possible to predict the mixing pattern in any given flight case to enable an estimation of heating effect on structure due to convection. Since test data is also not available at the present time, general information on the mixing of subsonic jet gases has been used, with variations, to determine a conservative estimate of the heating effect of these hot gases. In the final analysis, flight test data will be essential to complete a successful design of the rear structure exposed to high temperatures. Other manufacturers faced with a similar problem have followed this practice. A report is being written on this work and will be available at a later date.

6.2 REAR FUSELAGE TEMPERATURE ESTIMATION

Temperature estimates on the rear fuselage and nacelles of the ARROW 2 have been made. This structure has the same temperature limitations as the ARROW 1.

6.3 TEMPERATURE VARIATIONS IN FLIGHT

Several flight conditions have been investigated to study variations in the temperature of oil, air and fuel in the ARROW 2. These have shown that reducing power to idle may overheat the fuel temporarily because the fuel-oil heat exchangers are designed to do a specific amount of cooling under the design condition. Under this one particular transient condition when the fuel flow is low but the oil temperature higher than the design condition, heat exchangers cannot be prevented from absorbing more heat than necessary, unless they are by passed. (This also applies to the ARROW 1). This overheating has not been considered critical as the situation is transient during operational flying. During testing, however, when this situation may be repeated many times while on the ground, supplementary cooling may become necessary. The length of time the transient may last is a function of the heat capacity of the system. Ground and flight testing is expected to provide data from which it will be possible to estimate the over-temperature time in flight. Until this data becomes available, no realistic estimate is possible. A report will be available in due course.

6.4 CENTRE REAR ENGINE MOUNT

The design of the centre rear engine mount of the ARROW 2 has been altered as a result of testing which indicated that fusing of parts under maximum conditions of load and temperature, was probable. In brief, metals of low thermal conductivity have been used to provide a poor thermal path for heat to travel from the engine to the aircraft structure, thus reducing temperature maximums to an allowable level (Ref 5).

6.5 BRAKE PARACHUTE SECTION

The insulation blanket in the parachute section, which was one-half inch thick in the ARROW 1, has been reduced to one quarter inch thickness and the density doubled. This has been done to allow more space between the blanket and shroud for better ventilation. The result is a decrease in the insulation value of the blanket but the overall effect is to give the aluminum alloy skin in this area more protection from the hot engine shroud. The installation has been designed so that the skin temperature can be measured with or without the blanket. Calculations indicate that the best cooling of the structure may result from removing the insulation blanket entirely and relying only on the improved ventilation. The foil covering, used to reflect heat away from the parachute, has been changed from steel to aluminum for greater efficiency.

6.6 ENGINE PERFORMANCE INDICATOR

The thermodynamics group has been largely responsible for the development of the basic design of a new engine instrument referred to as the Engine Performance Indicator. This indicator, which will be used in the ARROW 2 to replace the ARROW 1 pressure ratio indicator, is still in a state of development. The instrument has recently been altered and simplified and now depends on a relationship between throttle angle, pressure ratio and total inlet temperature. Instrument reading will no longer have a direct relationship to the thrust (which, at altitudes is difficult to interpret) but will have a one-to-one relationship to the throttle angle. This relationship will enable the pilot to check the "health" of the engine either on the ground or in flight at any power setting regardless of ambient temperature. To allow for small differences between engines, an adjustment for ground calibration will be incorporated in the design. This will afford a means of detecting deterioration of engine performance. A report is currently in preparation.

7.0 AEROELASTICITY

Aeroelastic problems on the ARROW have been under investigation since the latter part of 1953. These problems were, for the most part, the determination of critical speeds for wing divergence, wing and control surface flutter and aileron reversal. Aerodynamic margins, based on the aircraft flight envelope, were then established for these critical speeds. During this report period progress has consisted of completing and extending the work done on these problems as described below.

7.1 LOW SPEED FLUTTER MODEL PROGRAM

This program was initiated to investigate the adequacy of the theoretical low speed flutter work. Testing was started some time ago, and recently completed, using a 1/10 scale model with a speed ratio of 6.5, in the National Aeronautical Establishment low speed wind tunnel at Ottawa. The testing program was delayed when the model was badly damaged in November 1956. The model was rebuilt and testing continued until it was completed in September 1957 after a total tunnel time of nine weeks. The wing and fin were tested individually in addition to tests of the complete free model. The control surfaces were tested at varying degrees of stiffness. The complete model was also tested with several wing fuel distributions. The results of all tests showed that the original analysis in general gave conservative results, and somewhat wider speed margins over the flight envelope were obtained. A report, with a motion picture documentary, is in preparation.

7.2 STRUCTURAL FEEDBACK EFFECTS ON DAMPING SYSTEM

This work predicts the spurious responses of the damper system to aircraft structural distortion or vibration, resulting from high airloads. The extension of this work to cover a wider frequency range and its revision to include correlations with model tests, as a means of cross checking, is one of the main efforts of the section at this time.

7.3 REVISION OF FIN FLUTTER CHARACTERISTICS

This revision is due to the inclusion of infra-red seeker equipment mounted near the top of the fin. No drastic effect is expected as a result of this modification and work is proceeding at a low priority.

7.4 PREPARATIONS FOR GROUND RESONANCE TESTS

A program is being prepared, and theoretical predictions of resonance properties are being made of the results to be obtained from ground testing of the first aircraft. These calculations are necessary as a guide to the correct location of vibration exciters, the best location for the low frequency supports (in the event that the aircraft landing gear is not suitable) and to

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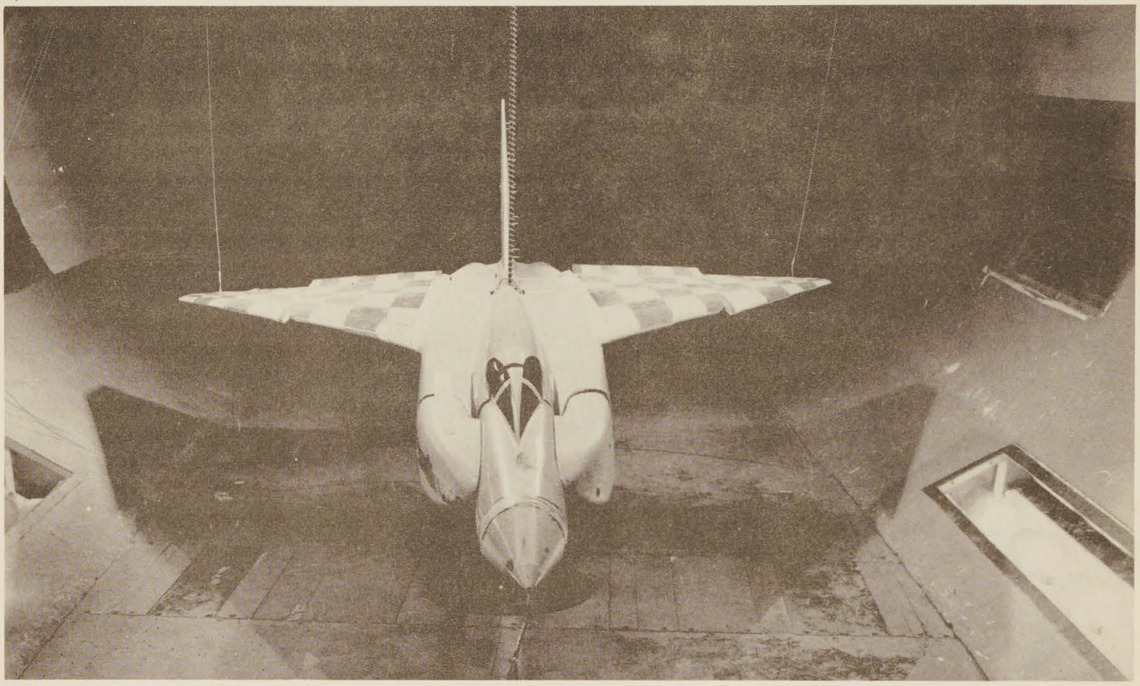


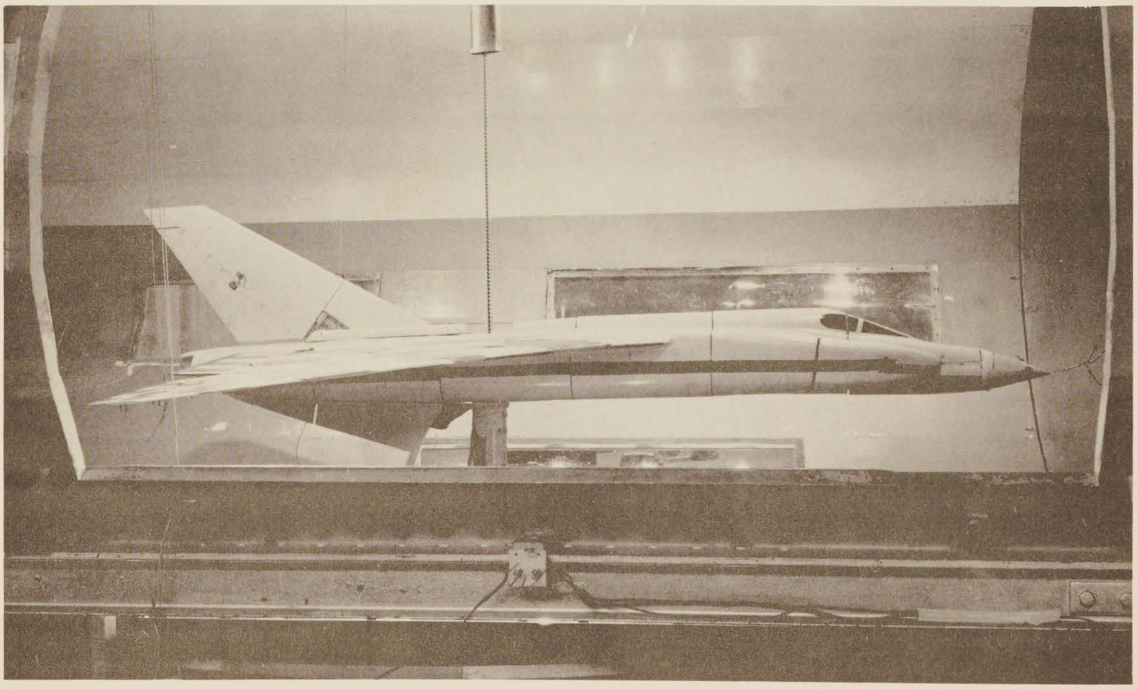


FIG. 10 1/10TH SCALE FLUTTER MODEL

provide a nodal plot of the aircraft as a means of confirming results. The data to be obtained in these tests, which are a check on dynamic calculations of resonance conditions, consist of

(a) Natural frequencies of the aircraft, both vertical and lateral.

(b) Modes of deformation and nodal lines for each natural frequency. Calculations have had to be revised since the decision to test the ARROW on its landing gear (with partly deflated tires) was made. Calculations had been made earlier with landing gear in the up position and the aircraft mounted on low frequency supports.

7.5 HIGH SPEED FLUTTER MODEL

With the availability of the transonic facility for flutter tests at the Massachusetts Institute of Technology, a high speed model program was begun in May 1957. These tests will provide a further check on aeroelastic problems in the air compressibility speed ranges. The first model was completed and preliminary tests were run at the end of September. The complete tests are scheduled for November and December.

8.0 ELECTRONIC SYSTEM

The ARROW electronic system will be considered from those aspects of its development which are the prime responsibility of Avro Aircraft Limited. Technical progress of the ASTRA I system is detailed in RCA quarterly progress reports.

A considerable amount of engineering effort during the report period has been expended in liaison with RCA studying findings and recommendations, and approving acceptable RCA reports. Progress of the ASTRA 1 system has been carefully monitored to ensure compatibility with AVRO requirements.

8.1 ARROW 2 MOCK-UP - ELECTRONICS

Space models of the majority of ASTRA I equipment have been received from RCA and installed in the ARROW 2 mock-up, although some of the models are in an estimated form rather than the final configuration. Recent RCAF evaluation at the ARROW 2 mock-up conference has resulted in a number of change requests, on which action will be according to priority. Mock-up conference brochures describing the electronic system and its functions were produced and issued to the RCAF prior to the conference. (Ref. 6 and 7).

Mock-up information is not yet available to AVRO for the Doppler and IR installations.

8.2 ARROW I ELECTRONICS - INSTALLATION DESIGN

Electronic installation design for ARROW 1 is complete, including mounting of equipment and wiring. All installation drawings for damping system wiring have been completed and the latest changes requested by the damping system manufacturer are being incorporated.

8.3 AIRCRAFT 4 AND 5 ELECTRONICS - INSTALLATION DESIGN

The fourth and fifth ARROW 1 aircraft will be converted for use as ASTRA I development vehicles. Preliminary design of structural alterations to convert these aircraft is progressing but insufficient information is available from RCA to enable the completion of wiring diagrams at this stage. Requirements for wiring are expected to be similar to those for ARROW 2.

The proposed AVRO program for the fourth and fifth aircraft is as follows:

1. Change radar nose and radome to ARROW 2 version.

2. Change both cockpits to accommodate ASTRA I controls and instruments.

3. Augment and redistribute air conditioning.

4. Install ASTRA 1 wiring.

5. Alter hydraulic, pneumatic and electrical systems for ASTRA I.

6. Alter structure to mount ASTRA I.

7. Install ASTRA I mountings and sufficient telecom and navigation for ferrying.

8. Provide pylon mounting for one Sparrow 2D seeker.

9. Install built-in instrumentation (similar to aircraft 1, 2 and 3). ASTRA instrumentation wiring is to be included in ASTRA system wiring. Responsibility for special instrumentation packs is not yet decided. RCA may sub-contract this to AVRO.

10. Fly to RCA's Flight Test Facility at New Castle, Deleware, for installation of remainder of ASTRA I, ground check-out and de-bugging (estimated 3 months).

11. Return to Malton to carry out ASTRA 1 development flying.

8.4 ARROW 2 ELECTRONICS - INSTALLATION DESIGN

Mountings for ASTRA I equipment in the ARROW 2 are being schemed and the layout of cable runs has now commenced, although full wiring requirements have yet to be received from RCA.

8.5 ANTENNA DESIGN AND DEVELOPMENT

Certain aspects of the work involved in design and development of ARROW telecommunications antennas are being conducted on behalf of AVRO by Sinclair Radio Laboratories Limited. This work, in support of antenna evaluation being carried out by AVRO, will be activated as required by the progress of the antenna program. The overall program may be summarized as follows:

(a) Investigations of the effects of an irdome installation in the vicinity of the UHF/L-band fin cap antenna. Antenna pattern studies have been carried out for the installation, with and without pitot tubes, prior to the introduction of the irdome. Relocation studies for the antenna will be made if necessary when the configuration of the IR pod is decided upon.

(b) Preliminary UHF pattern studies on a CF-100 in support of the antenna evaluation program for the ARROW.

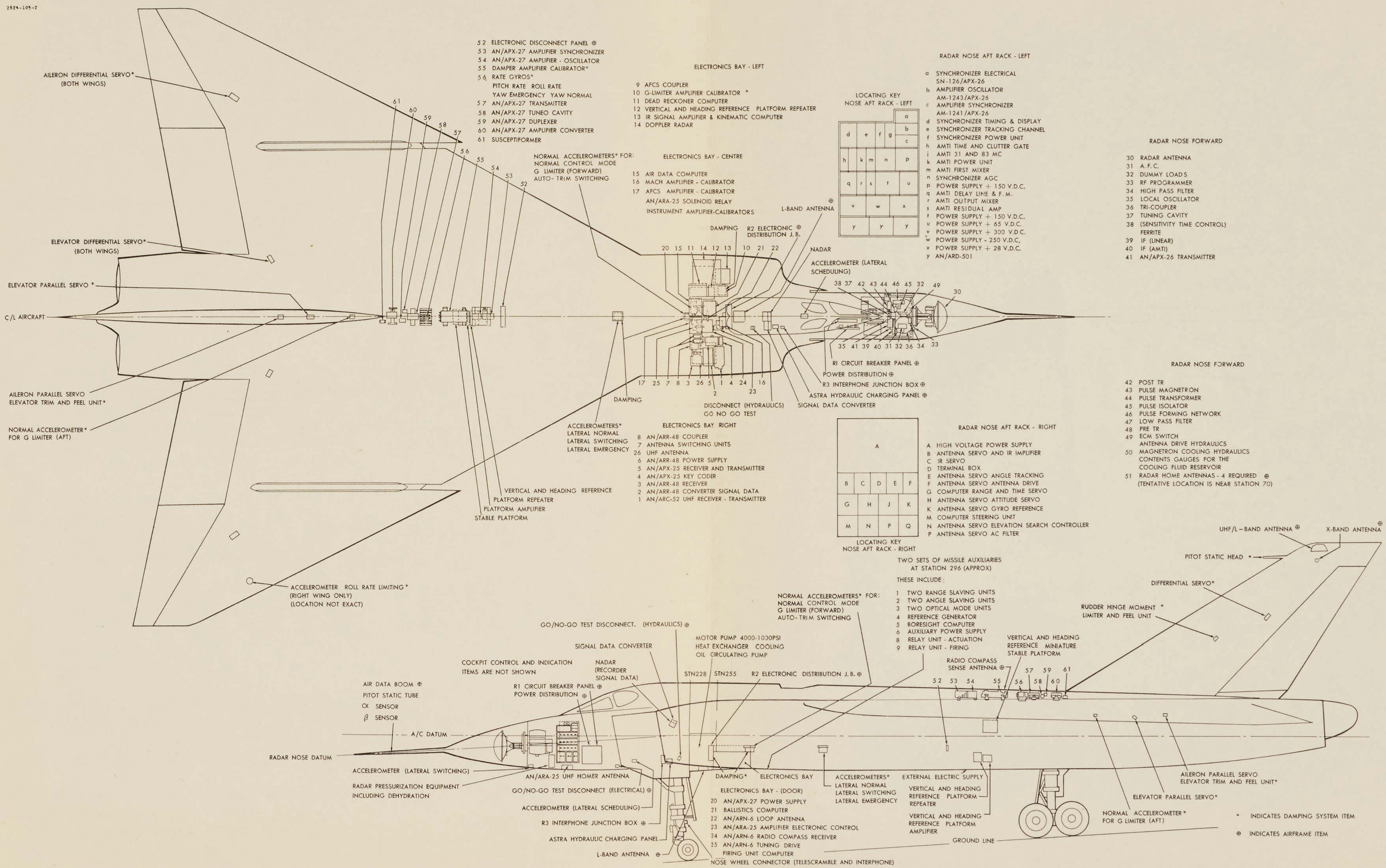


FIG. 11 INSTALLATION OF ELECTRONIC EQUIPMENT IN THE ARROW 2

(c) UHF pattern studies on three frequencies for both fin and belly antennas in support of the antenna evaluation program. Although patterns have been measured at the low and high ends of the UHF band, measurements are now required for the exact frequencies at which airborne trials will be performed.

(d) L-band pattern studies on two frequencies and for two antennas. These will assist RCA multiplexing studies as well as AVRO antenna evaluation. Patterns for the three principal planes have been measured previously but full spherical coverage is now required.

(e) Final reports on fin tip UHF/L-band antenna and ventral UHF antenna.

(f) Preliminary development and pattern studies of the antenna for the AN/ARD-501 homer. An initial pattern study has been carried out but this work is suspended until completion of Canadian Westinghouse and RCA design studies.

(g) Development of the UHF (annular slot type) belly antenna for the ARROW 2, including electrical and mechanical development and construction of prototypes. The ARROW 2 model of this antenna will have identical overall dimensions to the ARROW 1, but the matching section will be redesigned to lower the voltage standing wave ratio. This will increase the range of the antenna by approximately 15%, and will eliminate the necessity for structural alterations due to antenna dimensional changes which had previously been considered.

The ARROW 1 version of the UHF belly antenna has been developed and has passed engineering tests in respect of mechanical strength, environment and electrical properties. A VSWR of less than 3 has been obtained over the required frequency range.

Preliminary ARROW 2 studies to reduce the VSWR from 3 to 2.5 or less have commenced, and a modification is under investigation to strengthen the antenna feed line, which is liable to distort under high g. Experiments are also being conducted with the matching section, using printed circuits and capacitors for tuning.

(h) Pattern investigation of the government-furnished Doppler antenna to ensure that the proposed location is satisfactory.

8.6 UHF AND L-BAND ANTENNA EVALUATION PROGRAM

To optimize the performance of the UHF and L-band antenna installation it will be necessary to evaluate antenna coverage by means of model range studies and flight tests. In order to reduce the required flight time to a minimum it is proposed to compare range patterns in a single plane of 360°

coverage established by both model and flight tests. Close correlation of the respective patterns will indicate that model range patterns for complete coverage can be considered representative of the in-flight performance of the antenna installation.

Field strength measurements will be recorded at a ground station for a CF-100 aircraft in level flight, at constant altitude, over a range of headings at a fixed position. This will establish the technique and produce results which can be compared with model range studies to verify antenna coverage diagrams to within 1 db.

Airborne equipment for UHF transmission, to enable ground station measurements to be recorded, will be the AN/ARC-34. A suitable monitor will be used to record the output power of the equipment. An AN/APX-6, modified to increase the pulse width, with a suitable pulse generator connected to the "BM Trigger" input, will be used for L-band transmission. A block diagram of the airborne system is shown in Figure 12.

UHF and L-band field strength measurements will be recorded at the ground station on chart-type recorders. The AN/GRT-3 receiver will provide UHF field strength measurements. L-band field strength measurements will be obtained using an AN/APX-6, suitably modified, as the receiver at the ground station. A diagram of the ground station system is shown in Figure 13.

To ensure that the ground station antennas provide the required coverage to the aircraft, it will be necessary to adjust the height of the isotropic UHF antenna for each frequency used for evaluation. (The L-band antenna height will remain constant, as it is a directional antenna.) As the UHF antenna is isotropic, its performance is affected by direct rays from the aircraft and also by rays reflected from the earth's surface. It is necessary to optimize the antenna height in order to minimize out-of-phase destructive interference. Reference to Figure 14 will show that for a specified altitude and range there is an optimum antenna height h_2 for each frequency used. By selecting values of d_1 and h_1 it is possible to calculate d_2 and h_2 for maximum antenna gain, taking into account reflection coefficients of the reflecting medium.

The earth gain factor, which is the ratio of the resultant field to the free space field, can also be calculated and provides a measure of the antenna gain, taking into account multi-path propogation. Curves of surface range vs. earth gain factor can then be plotted and from these it is possible to obtain the gain factor at any desired range. Figure 15 shows a typical curve.

The ground station will be calibrated prior to commencing evaluation and, when the antenna height has been satisfactorily adjusted to give optimum

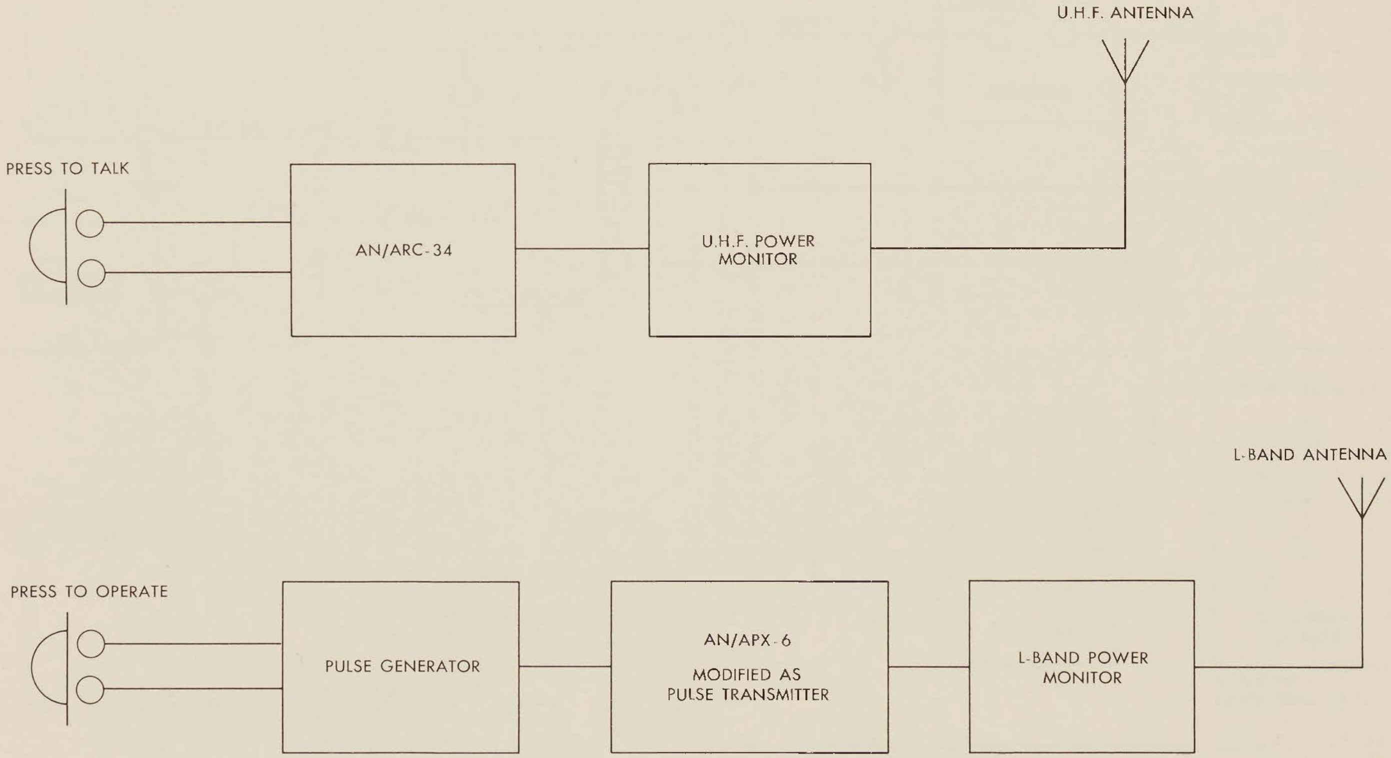


FIG. 12 AIRBORNE SYSTEM - ANTENNA TESTING

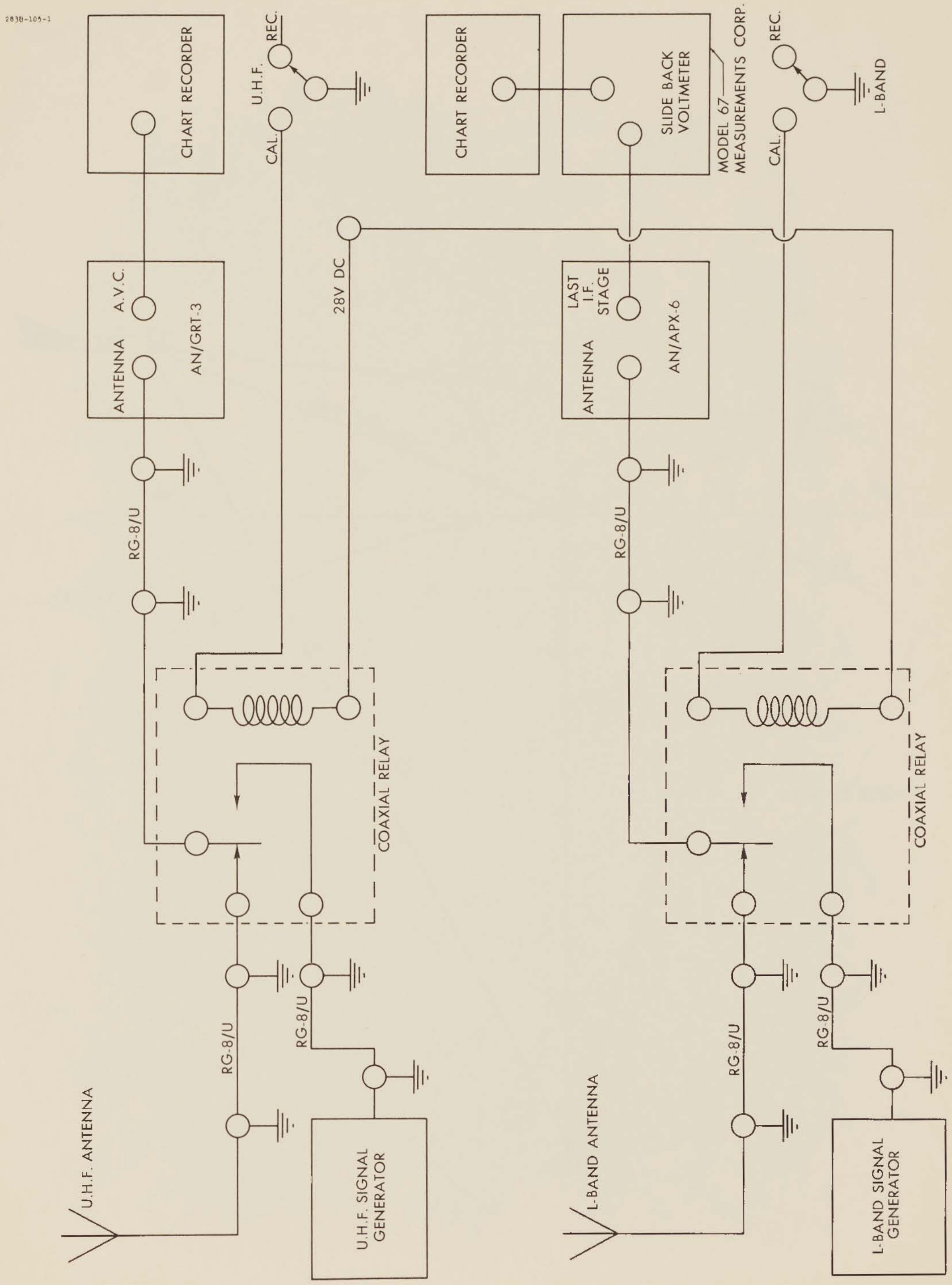


FIG. 13 GROUND STATION SYSTEM - ANTENNA TESTING

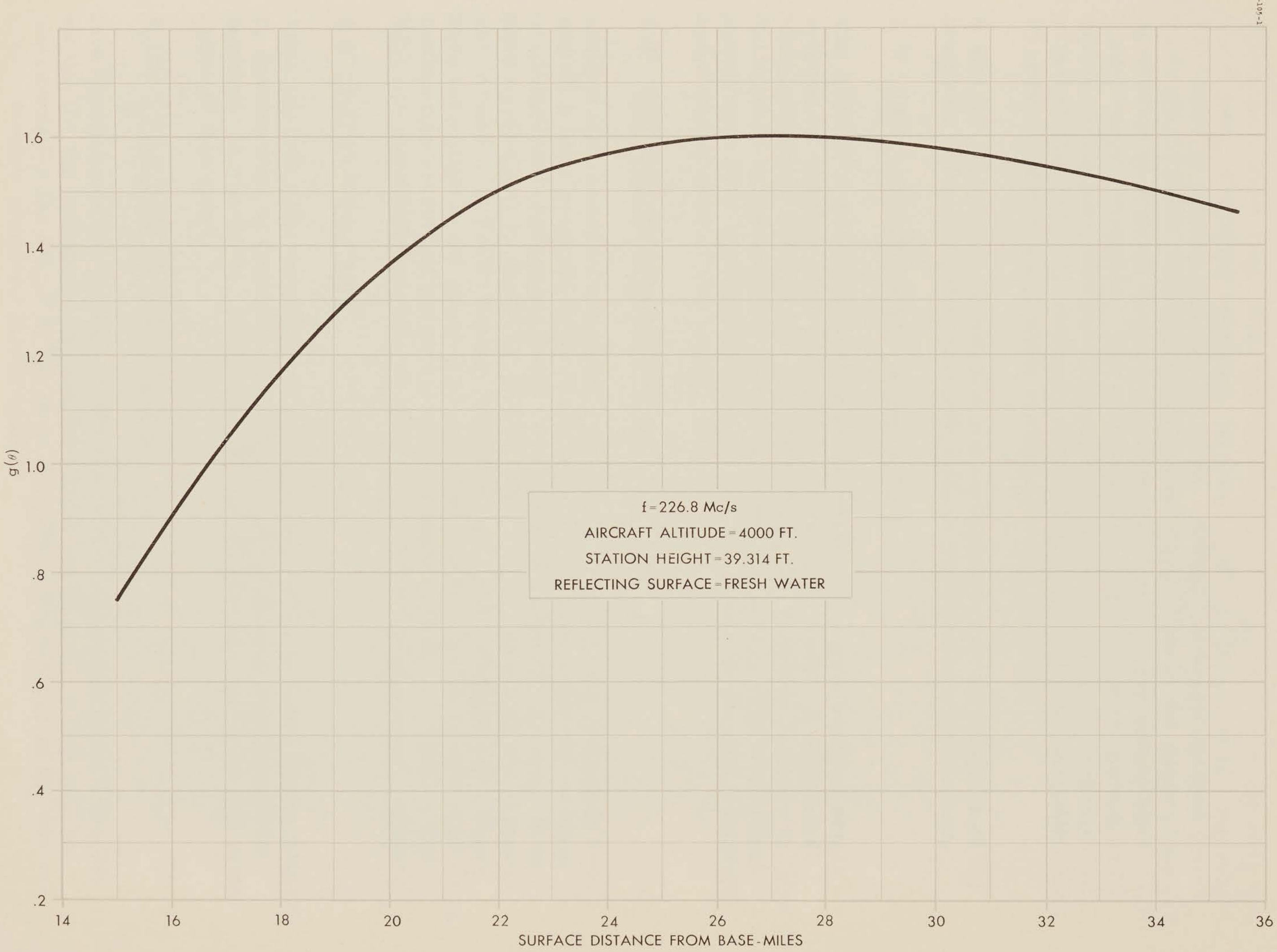


FIG. 15 GROUND STATION ANTENNA GAIN

range/gain conditions, field strength measurements will be made over the check point at varying headings, on three UHF and two L-band frequencies. Repeatability checks will also be made. Antenna patterns derived from these tests will be checked against model range data in the same plane and agreement will result in the entire antenna pattern being determined by model range techniques. The advantages of this method of antenna evaluation over others that have been considered are that flight test time is substantially reduced, automatic data reduction will not be necessary and instrumentation requirements are considerably reduced.

A site for the ground station has been selected and further details of the progress of the evaluation program will be provided at a later stage.

8.7 RADIO COMPASS SENSE ANTENNA

The ARROW's radio compass sense antenna is a copper sheet of semi-cylindrical configuration mounted in the dorsal fairing. A simulated installation in a CF-100 test vehicle is being flight tested in order to trim the antenna to the required electrical characterisitcs. These requirements have been determined as an effective antenna height of 0.10 metre and a capacitance of 100 mmf ± 10%.

8.8 UHF COMMUNICATION CONTROL SYSTEM

Liaison and recent meetings with Collins Radio Company of Canada Ltd. and RCA have resulted in the formulation of proposals for the methods of control and changeover of the UHF communication equipment. It has been agreed that it should be possible for the pilot or observer to transfer control to himself, but not away from himself. Each cockpit would have indication of the channel in use at all times. AVRO is now conducting installation studies of the equipment and wiring involved. Collins and RCA are negotiating separately with the RCAF regarding requirements.

8.9 DOPPLER RADAR

Although it is anticipated that Doppler radar will be included in the ARROW 2, development at Canadian Marconi Company is not proceeding at the moment. Accordingly, AVRO is only able to allocate space for the installation at this atage.

8.10 "WAR AND PEACE" SWITCH

Investigations of the various implications of a "War and Peace" facility switch for the armament system have been carried out, but it will be necessary for the RCAF to lay down the required functions of the system before further progress can be made.

8.11 ANTENNA SWITCHING

AVRO is considering the use of automatic antenna selector units for all aircraft prior to the introduction of data link. Their function would be to select the UHF and L-band fin or belly antenna providing maximum signal strength.

An initial investigation of the circuit requirements has been carried out (Figure 16) but further study will be necessary before a finalized system is arrived at. The problem at present is that an audio signal is not available from the L-band (IFF) equipment.

Prior to the introduction of automatic switching a manual antenna selector switch will be incorporated.

8.12 PRODUCTION ANTENNA TESTING PROCEDURES

Production testing procedures have been prepared for antenna system and intercom tests. These will provide for post-installation checks on the UHF and L-band antenna system. UHF antenna selection will be tested using aural transmissions and operating the manual antenna selector switch.

L-band IFF antennas will be tested using a simulated ground interrogator to trigger aircraft responses. If the equipment functions satisfactorily it will indicate that the antennas are performing correctly. Attenuation and VSWR tests will be done to test the efficiency of the X-band antenna, together with pressure checks on the waveguide.

8.13 ELECTRONIC SYSTEM POWER DISTRIBUTION

Analysis of electronic equipment power loading is currently in work. The results of this study will be fed into the electrical load analysis for the complete aircraft. Data from RCA on this subject is being checked as it is received. Correct wire sizes and breaker requirements have been determined. Attempts have been made to arrive at balanced three phase AC loads, and recent work in this respect was devoted to the emergency power system.

8.14 MISSILE FIRING CIRCUITRY

Design studies for a missile firing circuit are being carried out to assist in monitoring RCA development of the final system, and to enable intelligent appraisal of the system to be made. The conditions and requirements for system operation have been laid down and agreed between RCA and AVRO

The design of firing circuits for the missile test armament package is also progressing, this being the responsibility of AVRO.

8.15 RADOME

A working agreement for development of the ARROW radome has been entered into between Avro Aircraft Limited and the Brunswick-Balke-Collender Company of Canada Limited (B-B-C). The requirements for the radome are outlined in Avrocan Specification E-411 which will be revised and interpreted by AVRO as necessary to ensure production of radomes with adequate performance within allotted delivery dates. Liaison channels are being instituted between Avro Aircraft Limited, RCA and B-B-C to co-ordinate the progress of design and development of the radome. Regular meetings to discuss progress have been arranged and a system of progress reporting is in operation.

Basic objectives in the development of the radome are:

(a) A maximum effort to reduce transmission losses to an absolute minimum, especially in regions near the nose.

(b) A rate of change of boresight shift, both on "waterline" and circumferential cuts, sufficiently low that the tracking behaviour of the radar will not be adversely affected.

(c) Retention of the basis external shape. Small changes in the vicinity of the nose boom will be considered when necessary in the interest of improved radome performance.

The B-B-C program for development of the radome is as follows:

(a) Design of the radome.

(b) Manufacture of production tooling when design becomes stable.

(c) Construction of two qualification radomes for electrical correction.

(d) Performance of qualification tests, including provision of necessary tooling.

(e) Manufacture of production radomes.

AVRO's program in support of the development program takes the form of:

(a) A ray study in order to gain information on ray study techniques so that intelligent compromises on radome specification requirements can be made when necessary.

(b) Programming of the AVRO computer for simulation studies in close collaboration with similar work being performed by RCA.

RCA is assisting B-B-C and AVRO in their design studies and is carrying out a three-dimensional simulation, using radar, autopilot and aircraft transfer functions. The boresight error rate for the top and bottom of the radome was shown to be less critical than that for the sides, 0.02 degrees (or less) per degree of angular scan being considered acceptable.

A preliminary ray analysis has now been completed by B-B-C. From the data obtained, recommendations as to the shape of the radome and boom attachment configuration have been submitted to AVRO for evaluation and aerodynamic consideration. These recommendations were based on a reduction in the range of ray incidence angles, using a modified radome shape, from 0°-75° to 0°-68°. However, this is regarded by AVRO as presenting insufficient justification at this stage for a change of shape involving aerodynamic considerations and the associated wind tunnel and other aerodynamic testing.

AVRO is presently carrying out a digital program to determine the comparative electrical performance of a range of radome shapes. The requirements of the radome have been simplified by the deletion of Sparrow 3 which eliminated the need for meeting performance specifications over the 10 to 10.25 KMC band.

8.16 INFRA-RED INSTALLATION

An investigation into the airframe aspects of the IR installation is now being conducted on the basis of preliminary information recently received from RCA. The fin has been agreed upon as presenting the most promising location for the seeker head. This position offers optimum performance of the seeker with minor airframe and aerodynamic penalties. Earlier proposals for location of the seeker in the radome nose have been dropped.

Initial studies of the configuration of a pod forming an integral part of the upper portion of the fin are being conducted, but the aerodynamic shape is not yet finalized. The pod is to be designed to accept a 10 inch diameter seeker head and the associated IR system equipment. A preliminary scheme for the supply of cooling air to the equipment has also been formulated.

The effects of the pod, which is situated in the region of most severe buffeting, upon aircraft performance is being evaluated in the following respects:

(a) Assessment of drag effects and investigation of optimum shape in this respect, with consideration of area rule theory, to determine the effect on performance.

(b) Structural problems involved, with regard to g and aerodynamic loading.

(c) Possible effects upon flutter characteristics and aeroelasticity.

(d) Effects upon stability and control. Stabilization problems may be encountered due to relative motion between the IR seeker and the radar antenna, although tracking is normally synchronized.

(e) Installation space problems and their effect upon airframe design.

(f) Possible relocation of the UHF/L-band fin antenna. Pattern studies for this problem will be done by Sinclair Radio Laboratories when design of the IR pod is finalized.

Analytical study and evaluation of the IR system is also being performed in order that AVRO can effectively monitor and supplement the work of RCA in this field, as it affects the ARROW.

8.17 FLYING CONTROLS SIMULATION ON A CF-100

During the latter part of 1956, arrangements were made for development and mechanization in the pitch axis of a stick force, electrical, manual mode control system on a CF-100. Flight simulation tests in conjunction with the analog computer were planned, using a system which would in many respects be the equivalent of the ARROW damping system, thereby providing a valuable medium for development and evaluation of the control stick steering system.

Design of the system was centred around the equipment available at that time, namely a stick force transducer, an accelerometer, a parallel servo and a hydraulic servo amplifier. Analytical studies raised a requirement for the addition of an integrator and, since a DC system was being used and gain compensation was required, a modulator-demodulator scheme was incorporated. Inadequacies of the integrator also dictated the need for a phase compensator.

The DC system finally evolved provided the optimum compromise with respect to aircraft response, damping for gusts, stick force per g, trim and stick motion which it was possible to obtain at that stage.

Although the DC system was adopted as the most suitable for the components available at the outset, it was decided early in 1957 that an AC system would be desirable, the final object being the development of a flightworthy system. Since the parallel servo amplifier, integrator and stick force transducer were basically AC items and considerable amplification was involved, it was decided to adopt an AC system, which would then be more representative of the ARROW control system.

Implementation of this decision raised several problems, namely:

1. Conversion of DC signals into AC (as signals to and from the analog computer are DC).

2. Phase compensation.

3. Quadrature problems.

During the reporting period, development of the AC system has progressed in order to bring it to a state suitable for ground tests using the analog computer. Much of the work was concerned with development of modulators, demodulators and a phase lead network. However, the results were not satisfactory due to phase shift problems, and it was decided to employ Reeves electro-mechanical servos taken from the ARROW damper simulator. The phase advance network was finally abandoned.

Some tests were carried out using the analog computer and revealed a non-linearity in the parallel servo which was eventually eliminated by raising the gain of the parallel servo amplifier. This introduced instability caused by feedback through the control stick-elevator linkage, when using the stick force transducer in the system and not the computer. Efforts were then made to reduce instability by the use of filters at the stick force transducer output.

It should be noted that the instability is not a problem of mechanization of the specified system, but is a problem inherent in the system itself. From this point forward efforts were directed toward correction of this system deficiency.

Lengthy ground cables between the aircraft and the analog computer were successfully employed for computer tests. The technique established thereby will be applied for tests with the complete flying controls system test rig and the first ARROW aircraft.

8.18 ARROW FLIGHT SIMULATOR

Rapid progress is being made with the construction of the ARROW flight simulator and considerable effort is being expended to achieve its early completion. The simulator will eventually consist of a representative pilot's and obs/AI's cockpit rig complete with controls and instruments, an electronic damping system simulator and the analog computer. It will be used to evaluate and investigate stability and control characteristics related to landing techniques, aerodynamic behaviour, landing gear up and down, effects of the ram air turbine, pilot handling and other similar aspects.

The following work has been carried out during the reporting period:

(a) The construction of special function servos for the analog computer.

(b) The construction of scheduling servos for use with the damper simulator, the cockpit rig and the flying controls system rig.

(c) The completion of instrument servos to drive the pilot's display along with the construction of the associated amplifiers.

(d) The completion of all position feedback circuits (i. e. servo positions, throttle positions, stick positions, etc.).

(e) Construction of the cockpit rig, which is still proceeding.

(f) The completion of a pilot's instrument panel display and two scope displays for use with the flying controls test rig and with the first aircraft.

(g) Modifications to the damper simulator to incorporate automatic scheduling, landing gear switching, automatic trim, etc.

(h) Associated wiring, construction of panels and other equipment.

The simulator has been used to investigate problems concerned with instability in the stick force transducer/parallel servo loop and various filter configurations are being tried to solve them. The instability was discovered using a Bell Aircraft stick force transducer, and an F-100 stick force transducer subsequently acquired showed little improvement. A Humphrey unit tested on the simulator and on the flying controls rig showed some improvement, although instability still existed.

A combination of methods will be used to achieve a solution to this problem after further theoretical analysis is completed. These may be summarized as follows:

(a) Mass balancing of the stick. This is a partial solution in cutting down oscillations of the stick.

(b) The use of an electronic filter to limit the frequency of the DC signal into the Moog valve of the parallel servo. This is being tried on the flying controls rig.

(c) Velocity limiting of the parallel servo to achieve a mechanical equivalent of item (b).

9.0 ENGINE INSTALLATION

9.1 GENERAL

The engine installation for the ARROW aircraft is described in the following reports:

Engine Installation Data Manual for CF-105 Mk. 1 Aircraft (Pratt & Whitney J75P3-P5 Engines) April 1957 and ARROW 2 Engine Installation Report No. 72/AIREQ 25/1, June 1957. Engine installation will be discussed under the following sub-headings:

(i) Engines

(ii) Engine accessories

(iii) Engine mounting and installation

(iv) Power extraction systems

(v) Lubrication systems

(vi) Engine power control system

(vii) Intakes and ventilating

9.2 ENGINES

The ARROW aircraft are powered by two twin-spool, high thrust turbo-jet engines with integral afterburning units. The ARROW 1 aircraft is powered by the Pratt & Whitney J75 units and the ARROW 2 will be equipped with the Orenda Iroquois power units.

9.2.1 J75 POWER UNITS - ARROW 1

9.2.1.1 First Aircraft

The first ARROW 1 aircraft is being equipped with J75P3 engines. These are calibrated units which will be used to determine engine power losses due to installation characteristics and accessories. Four of these units have been delivered by Pratt & Whitney and are presently in the build-up stage at AVRO.

The engine build-up involves the mounting of AVRO specified components and parts on the bare engine as delivered by Pratt & Whitney. A list of these components and parts follows:

Engine Suspension and Attachment Fittings

Engine shrouds

Constant Speed Drive and Alternator

Nose Bullet Fairing

Adaptor Ring

Heat Exchange Duct

Power Take-off Gear Box

Output gear box and starter unit

Associated systems piping and wiring

Miscellaneous piping and wiring adaptors, connectors, etc.

Upon completion of the engine build-up, the power units will be installed in the aircraft, in preparation for the scheduled ground running tests.

9.2.1.2 Subsequent Aircraft

The remaining four aircraft of the series will be equipped with J75P5 units which will not be calibrated by the supplier. These are production units and differ from the P3 units only by virtue of the absence of calibration data as supplied by Pratt & Whitney. The performance of these units will be checked against the installed performance data of the P3 units.

9.2.2 ORENDA IROQUOIS POWER UNITS - ARROW 2

The Iroquois power unit is in an advanced state of development and testing. Static testing of the unit has been in progress for a considerable time and the unit has passed the required type tests. A unit is presently being prepared for installation in a Boeing B-47 flying test bed.

9.2.3 ADDITIONAL DATA

For further information regarding the J75 engine, reference should be made to the Pratt & Whitney "Gas Turbine Installation Handbook", and additional information on the Iroquois engine may be obtained from the Iroquois Installation Data Book.

9.3 ENGINE ACCESSORIES

9.3.1 GENERAL

The term "engine accessories" as used here encompasses all components and parts which are assembled to the engine during the engine build-up stage. This definition is intended, primarily, to cover all components and parts which are assembled by AVRO to the engine as it is delivered by the engine manufacturer. For convenience, however, the definition is extended to include the engine manufacturer's accessories which are an AVRO requirement, or the design of which is influenced by AVRO.

9.3.2 ORENDA ACCESSORIES TO THE IROQUOIS

9.3.2.1 Air-to-Air Heat Exchanger

The original engine configuration submitted to AVRO by Orenda Engines Limited included an externally mounted heat exchanger. Subsequently AVRO designed the engine tunnel to accommodate the given engine configuration and provide the required engine cooling air flow. Recently, Orenda informed AVRO of its decision to remove the heat exchanger since it was no longer required. An investigation was undertaken to determine the influence of the heat exchanger on the engine cooling air flow and the tunnel internal air pressure.

The heat exchanger offers a resisting surface in the cooling air flow path and consequently possesses the characteristics of a restrictor. Its removal would permit an increased flow into the cooling region with a subsequent build-up of internal pressure. The excess internal tunnel pressures will subject the tunnel wall structure to loadings in excess of the design loadings with the probability of subsequent fatigue failure of the structure.

To prevent such failures, AVRO has requested Orenda to replace the heat exchanger with a structure which duplicates the heat exchanger shape and location. The alternative to this solution was a redesign of the ejector, or a redesign of the engine tunnel which in terms of time and additional cost was considered uneconomical.

9.3.2.2 Nose Bullet Fairing

The nose bullet fairing, which encloses the constant speed drive unit and the alternator is being designed and developed by Orenda. AVRO had done some early investigations with regard to the design of the component and the related anti-icing problems. However, since a hot-air anti-icing system bleeding air directly from a low-pressure compressor bleed port was being considered, it was established that Orenda could handle the problem more conveniently. Subsequently, agreement was reached whereby Orenda accepted the responsibility of supplying the engine with a nose bullet fairing and anti-icing provisions.

9.3.3 CONSTANT SPEED DRIVES

A constant speed drive unit is a hydraulic transmission which accepts mechanical power at a variable speed input and delivers mechanical power at constant speed output. The constant speed drive (CSD) units for the ARROW aircraft are being supplied by General Electrical for the ARROW 1 and by Sundstrand for the ARROW 2.

9.3.3.1 The General Electric CSD Units

Based on the Avro Specification requirements, General Electric has developed and produced a CSD unit. During vendor qualification testing, however, the unit was observed to cease operating under simulated vibration conditions. Further testing has indicated that the specified power output of the unit at low engine speeds is not being obtained. Consequently, the functional integrity of the unit is presently in doubt. GE is continuing development work and testing on the unit in an effort to meet specification requirements.

The CSD units will not be fully qualified in time for the engine ground running tests but, limited approval for ground test will be granted with eventual approval for flight test.

9.3.3.2 The Sundstrand CSD Units

A 40 KVA output, constant speed drive is required for the ARROW 2. The Sundstrand CSD was selected since the low speed power output of the General Electric unit was inadequate.

These units are presently in the design and development stage at the vendor's facilities. The further progress on these units will be reported in subsequent Quarterly Reports, as information becomes available.

9.3.4 NOSE BULLET FAIRING

As noted in paragraph 9.3.2.2, Orenda Engines Limited is supplying the Iroquois engine with a nose bullet fairing. In the case of the J75 for the ARROW 1 aircraft, AVRO is responsible for providing the nose bullet fairing, and bullet anti-icing.

The portion of the fairing enclosing the alternator and CSD unit presents no problem, since sufficient heat is generated by the two units to prevent icing of the fairing. The forward portion of the nose bullet, however, is susceptible to icing and anti-icing provisions are required.

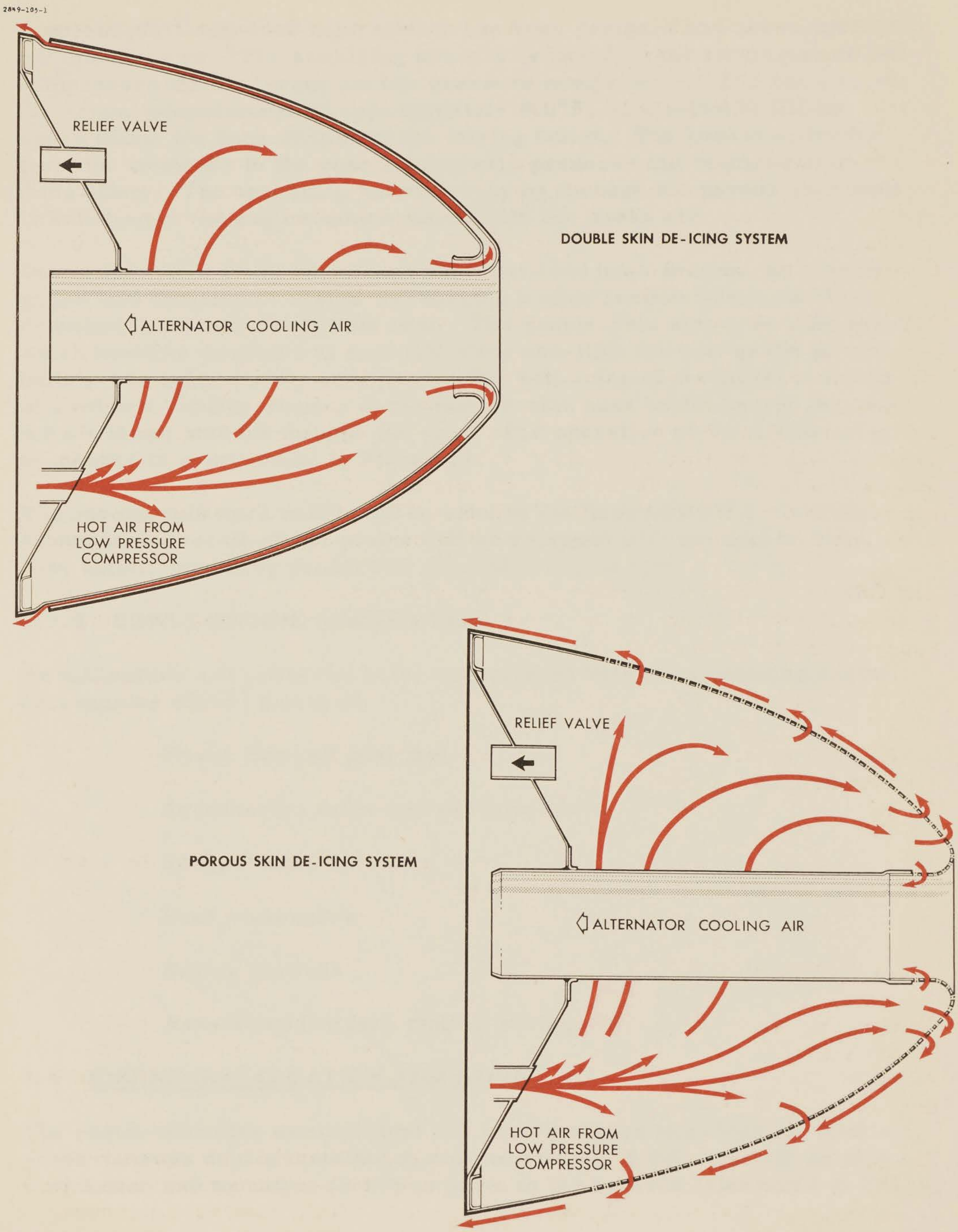


FIG. 17 NOSE BULLET - COMPARISON OF POROUS AND DOUBLE SKIN DE-ICING

A porous-skin anti-iced nose section has been designed and sub-contracted for manufacture. The anti-icing process is based on the admission of hot compressed air bled from the low pressure compressor. The hot air, at maximum temperatures of approximately 500°F, is allowed to fill the space within the nose section of the fairing bullet. The heat transferred from the bleed air to the nose section skin produces the required anti-icing effect. The anti-icing hot air escaping through the porous nose skin is returned to the engine compressors with the intake air.

Due to the high cost of the porous skin anti-iced nose section, an investigation was initiated to design and develop a nose section which could be produced at a more favourable cost. The double-skin anti-iced nose bullet, which could be produced at approximately one-fifth the cost of the porous bullet, was subsequently conceived. The anti-icing of the bullet is accomplished in a similar manner to the porous-skin nose bullet except that the hot air is not bled off through the skin. The operation of these anti-icing processes is represented in Figure 17.

The porous skin nose bullet will be used in the first ARROW 1 aircraft. Subsequent aircraft of the series will be equipped with the double-skin nose bullet, providing production schedules can be met.

9.3.5 OTHER ENGINE ACCESSORIES

No difficulties are presently being experienced with the remaining engine accessories which consist of:

- Power take-off gear box
- Accessories drive out-put gear box
- Starter unit
- Heat exchangers
- Engine shrouds
- Associated fittings, piping, wiring, etc.

9.4 ENGINE INSTALLATION AND MOUNTING

The engine mounting arrangement has been designed to reduce all trial and error methods during installation to a minimum. A full description of the installation and mounting details is given in the reports referenced in Part 8.

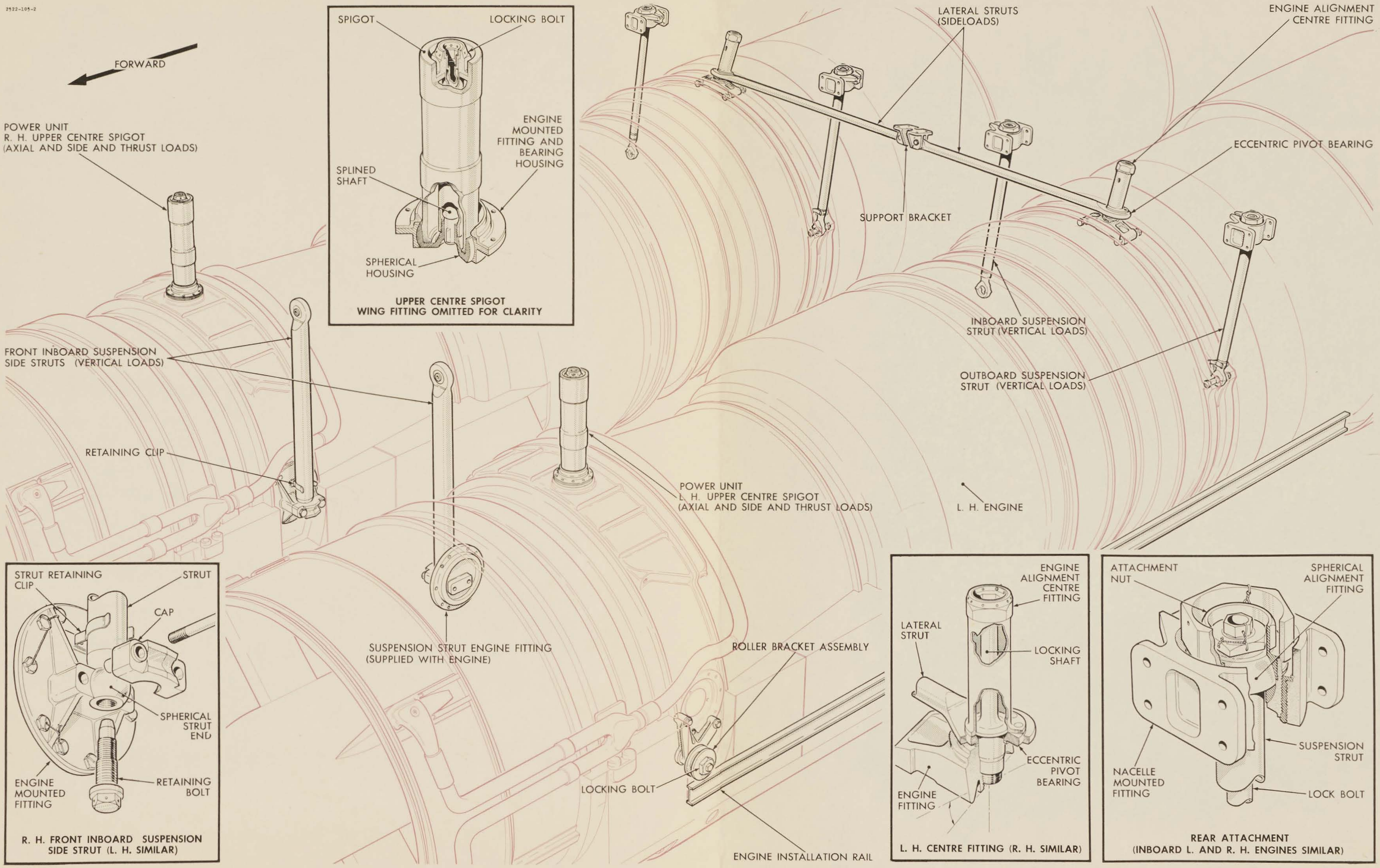


FIG. 18 IROQUOIS ENGINE ATTACHMENT AND SUSPENSION

9.4.1 STATUS

9.4.1.1 ARROW 1

The design of the mounting is complete and in production. Trial installation and mounting of the engines has been carried out on the first aircraft. Apart from minor interference problems of engine bracketry within the engine tunnel, the engine mounting arrangement and installation procedure are satisfactory. The interference problem has been eliminated by an adjustment of the interfering engine-mounted brackets.

9.4.1.2 ARROW 2

The design of the mounting has been completed and is now in production for fabrication. The mounting components will be fabricated employing the flash welding technique. The process is based on the fusion of metal, in which a high-strength weld is produced. This method of fabrication was adopted since no vendors would guarantee forged components to the specified strength and tolerance requirements.

9.5 POWER EXTRACTION SYSTEMS

Engine power is extracted to provide mechanical, electrical, hydraulic and pneumatic power to operate the various aircraft systems and services. The power extraction is based on either a mechanical or a pneumatic power take-off system. Each engine is provided with two mechanical power extraction points and several pneumatic power extraction points. These are utilized to provide three systems, two of which are mechanical and the third a pneumatic system.

9.5.1 PNEUMATIC SYSTEM

The pneumatic power extraction system consists simply of a shut-off valve and an air duct connected through a manifold to the engine bleed ports. Bleed ports are provided on the engine at selected stages of the high pressure compressor. The high energy air thus obtained is utilized to provide aircraft air conditioning and pressurization.

9.5.2 MECHANICAL SYSTEMS

The mechanical power take-off provisions are utilized to generate the aircraft electrical energy, produce hydraulic power and drive the fuel system booster pumps.

9.5.2.1 Low-Pressure Compressor Power Take-off

A constant speed drive unit is mounted directly to the engine's front power

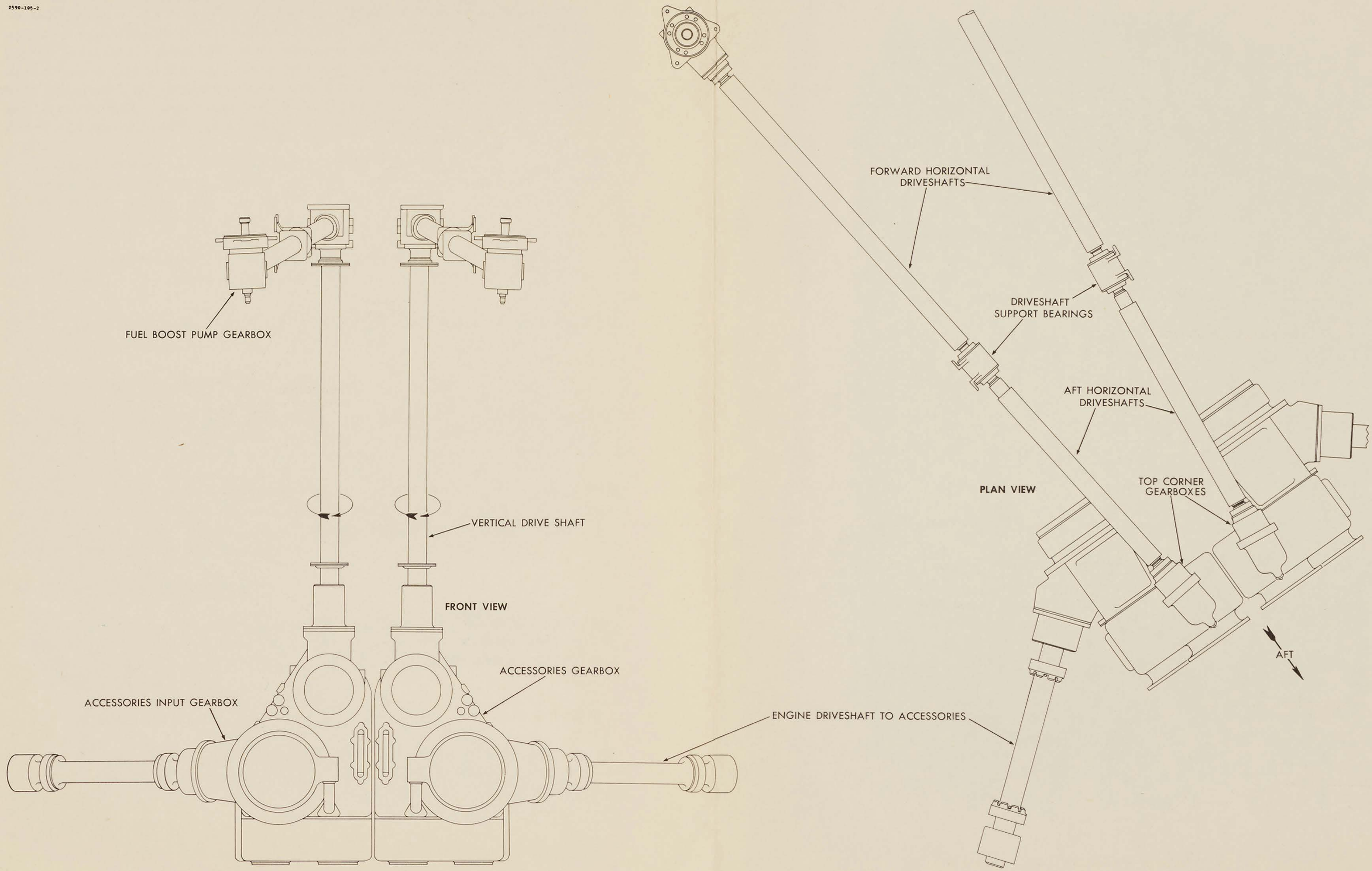
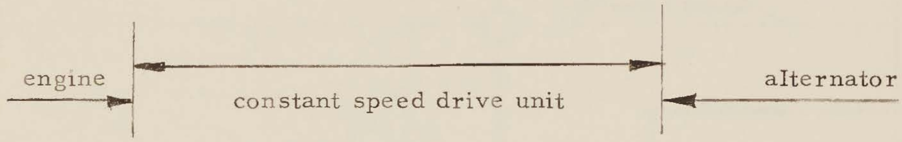


FIG. 19 MECHANICAL POWER TRANSMISSION FROM IROQUOIS ENGINES

take-off pad, located at the front frame inlet casing, the pad input shaft being coincident with and driven by the low pressure compressor rotor shaft. The constant speed drive unit accepts a variable input speed from the engine power pad and converts this mechanical power into a constant output speed at the output end of the unit, by a hydraulic coupling. Electrical power is then generated by an alternator coupled to the output shaft of the constant speed drive unit. The sequence of events in the production of electrical energy is conveniently summarized by the following:

{mechanical power at variable speed} → {hydraulic power} → {mechanical power at constant speed} → {electrical power}



The constant speed drive unit and the alternator are engine-mounted and consequently are considered as engine accessories.

9.5.2.2 High-Pressure Compressor Power Take-off

Engine power from the high-pressure compressor rotor is available at the engine power take-off pad on the underside of the engine. Power is extracted by an engine-mounted gear box from which the power is mechanically transmitted to the airframe-mounted gear box. These main airframe-mounted gear boxes are coupled directly to hydraulic pumps which produce the required hydraulic power. In addition, power is transmitted mechanically from the main gear boxes to provide power for driving the fuel booster pumps. See Figure 19.

9.6 LUBRICATION SYSTEMS

Independent oil systems are provided for right and left hand engine installations and related systems. Thus, the loss of either engines does not in any way influence the operation of the remaining engine and its associated accessories and systems.

9.6.1 ARROW 1 - OIL SYSTEMS

A single oil system is installed to supply lubricating oil to the gear boxes of the mechanical power extraction system and to supply oil to the constant speed drive unit. This system is independent of the engine oil system and is adequately described in the ARROW 1 report referenced in Part 8.

The engine oil system requires supplementary oil cooling, consequently

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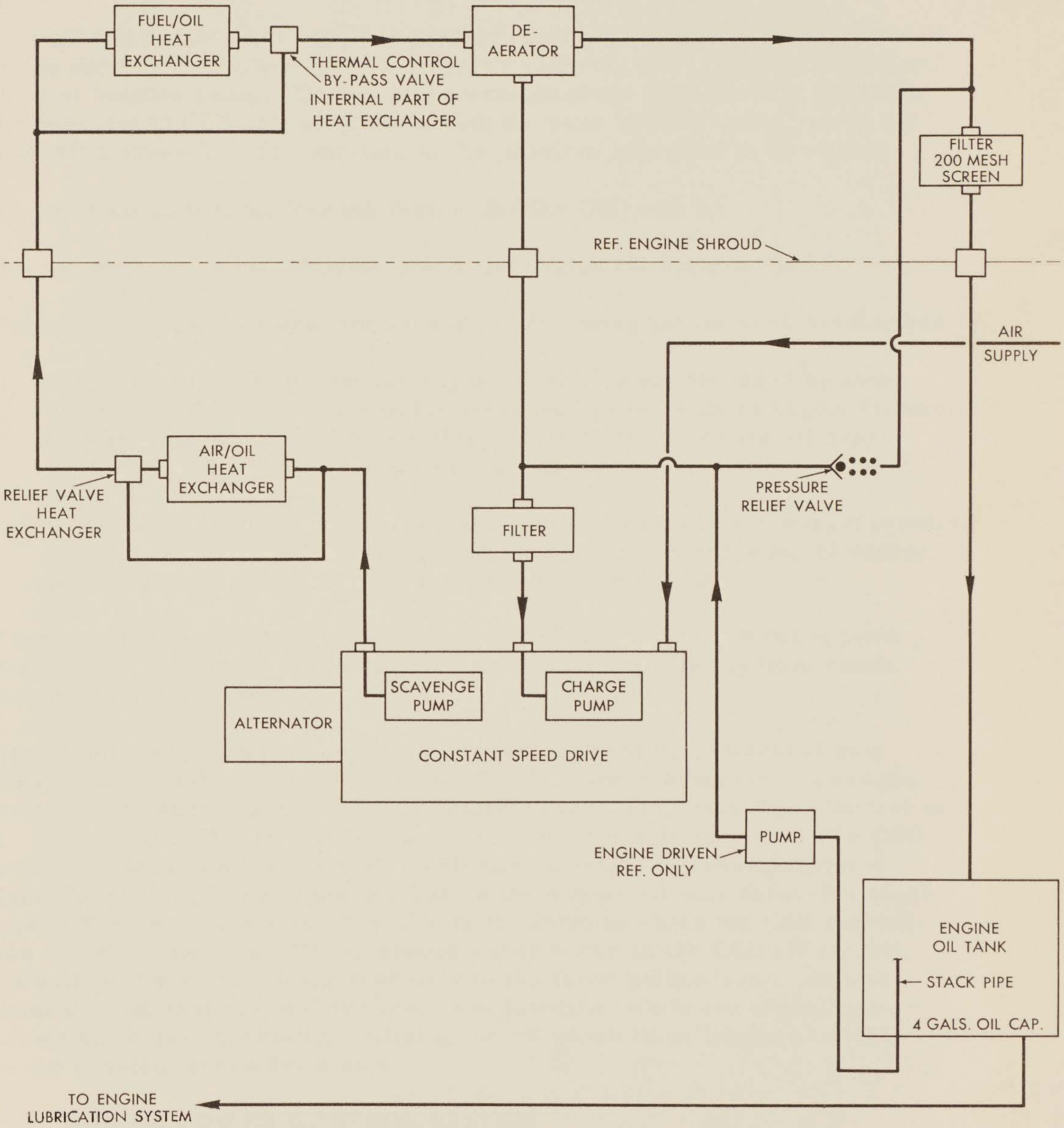

FIG. 20 ARROW 2 CONSTANT SPEED DRIVE INTEGRATED OIL SYSTEM

piping is installed to direct oil flow through an airframe-mounted oil-to-fuel heat exchanger.

9.6.2 ARROW 2 - OIL SYSTEMS

A single oil system has been designed to meet the lubrication requirements of the mechanical power extraction system driving the hydraulic pumps and the fuel booster pump. Design requirements made it difficult to integrate the Sundstrand CSD unit oil system with the gear box oil system as in the ARROW 1 aircraft. The solution to the problem appeared to be either:

(a) provide an independent oil system for the CSD unit or

(b) integrate the CSD oil system with the engine oil system.

The two alternatives were studied and the following points were established:

(i) A system integrated with the engine system is not favoured by most engine manufacturers due to the increased probability of engine failure. The argument here is that a failure of the CSD unit or its oil supply system would result in the loss of the engine.

(ii) Utilizing an independent CSD oil system would result in a weight penalty of 50 lb. over that of the integrated oil system, in addition, of course, to the space requirements of a separate oil tank.

These studies tended to favour the integrated oil system and subsequent discussions between AVRO and Orenda resulted in the oil system shown schematically in Figure 20.

The integrated oil system, as shown in the schematic, is integral only insofar as the oil tank is concerned. The CSD lubricating circuit and the engine lubricating circuit are completely independent, except for the use of a common oil tank. In order to preserve the engine in the event of a CSD failure or loss of oil in the CSD lubricating circuit, the constant speed drive lubricating system takes oil from the engine oil tank through a stack pipe. The use of the stack pipe limits the level to which the CSD system can drain the oil tank. Thus, should a leak occur in the CSD oil circuit, oil will be drained from the tank only to the three gallon level. At this point the constant speed drive ceases to function, while the engine continues to operate normally, utilizing the oil which is no longer available to the constant speed drive unit

9.7 ENGINE POWER CONTROL SYSTEM

9.7.1 GENERAL

The ARROW power control system is manually operated through a conventional twin power lever control box assembly which is mechanically linked by a system of cables, pulleys, bell-cranks and push rods to the engine fuel flow control units. The system is described in the reports referenced in Part 8.

The system for the ARROW 1 and ARROW 2 are functionally identical. Minor installation differences will result due to the different engine installations.

9.7.2 THROTTLE CONTROL BOX

Due to the proper control characteristics peculiar to a particular engine type, the throttle control boxes for the ARROW 1 and ARROW 2 aircraft are not identical. Although externally similar, the power lever geometry for the units is different. ARROW 2 power lever geometry is shown in Figure 21.

9.7.3 STATUS AND PROGRESS

9.7.3.1 ARROW 1 System

The design of the system has been complete for some time, and the fabrication and procurement of component parts is well in hand. Recent design modifications have been incorporated to accommodate system rigging and alignment procedures. These modifications provide for lock-pin holes in the control lever quadrants to permit the locking of the levers in the IDLE position, and the provision of corresponding alignment marks on the fuel flow control unit pulley quadrants. A preliminary rigging procedure for the system has been prepared and appended to the Engine Installation Data Manual.

The engine control system is presently being installed in the first aircraft of the series. Functional test results and a check of the rigging procedure will be available from the engine ground running tests.

9.7.3.2 ARROW 2 system

The ARROW 2 system design is complete and specifications for component parts have been prepared and submitted for tender to the vendors. Further development work on the system will depend on the ability of vendors to meet specification requirements and on installation problems as they arise.

9.8 ENGINE INTAKES AND VENTILATION

The engine air intake and ventilating system is described in the referenced reports. The intake system comprises an intake ramp at the air inlet located on the side of the fuselage at the pilot's station, a duct throughout

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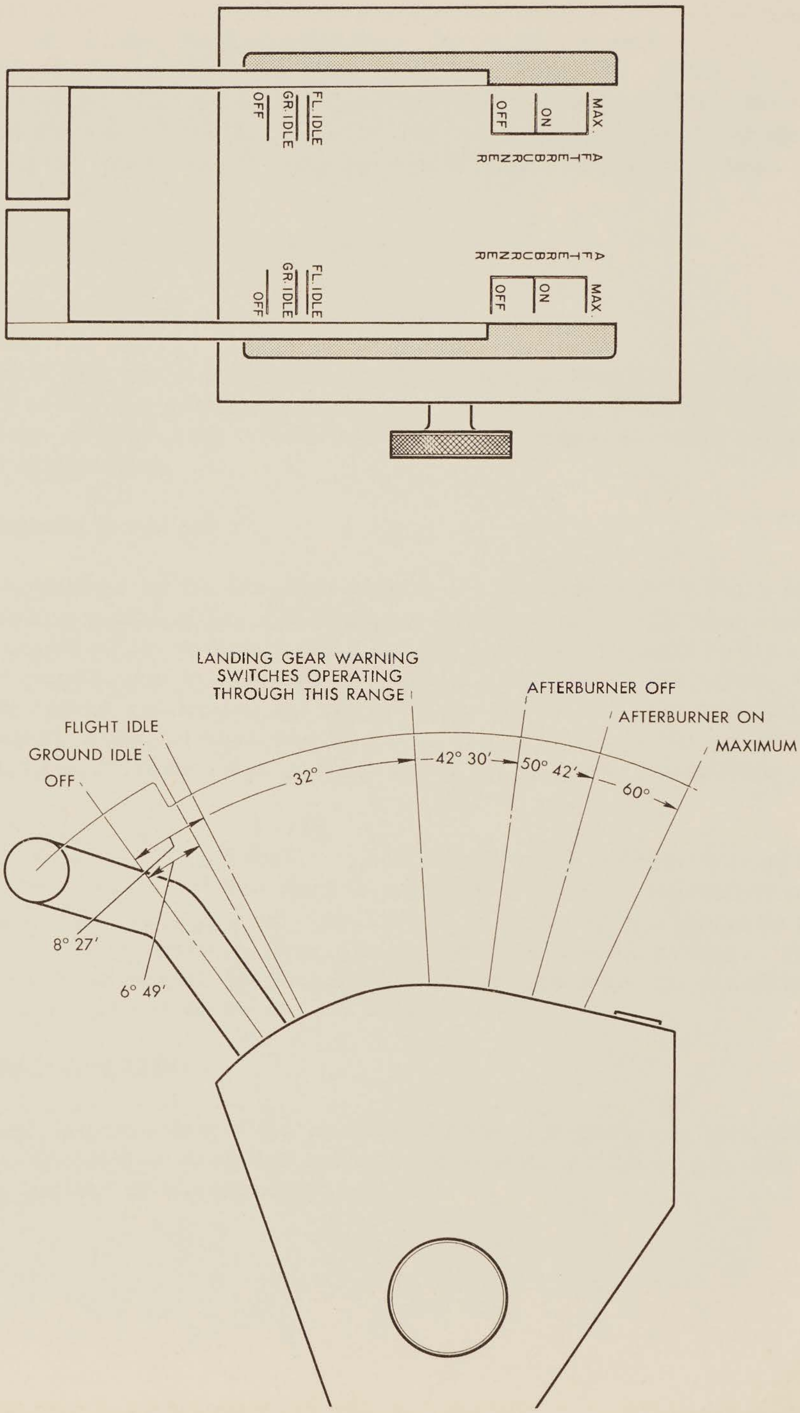

FIG. 21 ARROW 2-POWER LEVER GEOMETRY

the full length of the centre portion of the fuselage and an engine-to-duct connection. The intake ramp and the duct are primarily structural and consequently are discussed under Part 4, Product Design. The related aerodynamics of the air flow in the duct and at the intake are similarly considered in Part 2, Technical Design under the appropriate headings. The portion of the intake which links the air duct to the engine intake, however, is discussed here.

9.8.1 ENGINE TO DUCT CONNECTION

9.8.1.1 J75 Installation

The front end of the J75 is fitted with an adaptor ring which fits into the aft extremity of the air intake duct. No rigid connection is made between the duct and the adaptor ring. A metal-to-metal contact is established by virtue of the installation.

9.8.1.2 Iroquois Installation

The engine mountings on the Iroquois engine are considerably further aft than the mounting points of the J75 engine. Consequently, any wing deflections in the region of the forward and rear engine mounts would be transmitted to the engine with subsequent large displacements of the forward portion of the engine relative to the adjacent structure. To accommodate the displacement of the forward end of the engine relative to the structure, a floating duct was introduced to provide the necessary engine-to-duct connection.

The aft end of this articulated duct is fixed directly to the front face of the engine. The forward end of this duct is coupled to the rear extremity of the fixed intake duct through a universal joint. The universal coupling at the fixed duct permits vertical, transverse and longitudinal motion of the engine. This type of engine-to-duct joint will accommodate all possible engine displacements due to structural deflections.

9.8.2 ENGINE COOLING

The ventilating systems described in the referenced reports are provided to prevent excessive heat transfer from the engine to the structure and are adequately described in the reports.

10.0 ELECTRICAL SYSTEM

To explain the major differences between the ARROW 1 and the ARROW 2 electrical power systems a brief description of the relevant aspects is given below:

10.1 ARROW 1 ELECTRICAL SYSTEM

The ARROW 1 power system comprises two engine mounted, air cooled, 30 KVA alternators, each driven by its respective engine through a mechanical-hydraulic constant speed drive unit. DC power is provided by two 3 KW transformer-rectifier units, cooled by the air conditioning system. Each transformer-rectifier unit is supplied from its applicable alternator and feeds essential DC services together with the DC shedding bus. The latter is de-energized when one alternator becomes inoperative.

The left hand alternator normally supplies the left-hand AC shedding bus and left-hand intake duct de-icing. If the left-hand alternator is rendered inoperative the AC shedding bus and left-hand intake duct de-icing are shut down.

The right-hand alternator normally supplies the right-hand intake duct de-icing. The primary AC buses are normally supplied from the right-hand alternator and feed the electronic and electrical services, and the emergency AC bus. The primary AC buses are switched to the left-hand alternator if the right-hand alternator fails or the engine is shut down. The right-hand intake duct de-icing supply is then not available.

External AC power from a ground servicing unit supplies the AC buses and the transformer-rectifier units until "breakaway", even with both engines running. The left and right-hand AC warning lights are extinguished when the ground supply is energizing the AC buses.

During flight the emergency hydraulically driven alternator becomes operable on failure of shutdown of both alternators. Emergency DC power is provided by the battery.

10.2 ARROW 2 ELECTRICAL SYSTEM

The ARROW 2 power system comprises two engine-mounted, oil cooled, 40 KVA alternators, each driven by its respective engine through a mechanical-hydraulic constant-speed drive unit. DC power is provided by two 4.5 KW transformer-rectifier units, cooled by the air conditioning system.

The left alternator normally supplies the primary AC buses feeding the electronics services, the emergency AC bus and the left transformer-rectifier unit. On failure or shut-down of the left alternator, the primary

AC buses, emergency AC bus and the left transformer-rectifier unit are then supplied by the right alternator.

The right alternator supplies the primary AC buses feeding the electrical services, both left and right intake duct de-icing, and the right transformer-rectifier unit. On failure or shut-down of the right alternator, the primary AC buses and the right transformer-rectifier unit are then supplied by the left alternator. The DC shedding loads are removed.

External AC power from a ground servicing unit supplies both left and right AC buses and transformer-rectifier units. As each engine reaches a sufficient rpm for operation of its alternator, the applicable AC bus system and transformer-rectifier are transferred from ground supply to the aircraft alternator. The left and right AC warning lights remain on until each aircraft alternator is supplying its applicable system, therefore the pilot knows before rolling that the aircraft alternators are functioning.

During flight a ram air turbine is automatically extended on failure or shut-down of both alternators, and supplies emergency AC loads.

Should there be a bus fault in either the left or right AC system the AC loads of that particular system will not be transferred to the operating alternator, so that the fault is not transferred and the remaining alternator will not shut down. The pilot has a power distribution selector switch for selection of "MISSILES" or "DE-ICING". When "MISSILES" is selected, a shut-down of one alternator will disconnect AC and DC de-icing loads (with the exception of windshield and canopy de-icing) and the DC shedding bus. Under these conditions a combat mission may be completed without restriction since radar, computer, etc., are all functioning. When "DE-ICING" is selected, a shut-down of one side of the system will disconnect the armament AC and DC loads and the DC shedding bus and reconnect de-icing loads. In this case, the aircraft is essentially unarmed since the power to the fire control sub-system is cut off. Essential telecommunication and navigation services are retained, however. In either case the services are automatically reinstated when the alternator recommences operation.

10.3 ELECTRICAL SYSTEM DESIGN PROGRESS

Design of the ARROW 1 electrical system is complete and breadboard testing to check the functioning of circuits has been satisfactorily concluded. The ARROW 2 system, using 40 KVA oil-cooled, brushless alternators and 4.5 KW transformer-rectifier units, is described in reference 8. The reporting period has been mainly taken up with the production of ARROW 2 theoretical circuits and preparations for breadboard testing of the system.

Breadboard tests for ARROW 2 will be similar to those performed for the earlier aircraft, and will take the form of individual circuit functioning to prove the system on the basis of aircraft requirements. Loading tests, using actual equipment where available, will be carried out for various conditions of flight.

Theoretical electrical circuits for control of the armament hydraulic system have been designed, and a breadboard has been constructed to test the system. Launcher operation tests have been carried out using the breadboard.

10.3.1 INSTALLATION DESIGN

The design of theoretical circuits, equipment installation and wiring for the ARROW 1 is completed, although alterations and additions resulting from testing and changes of requirements are being incorporated as they arise.

Electrical installation design for the ARROW 2 is progressing on schedule and basic theoretical circuits have been established. These are, however, subject to frequent changes as they are in a stage of design refinement. Schedules are generally being met, although ARROW 1 changes have interfered with the progress of ARROW 2 wiring design to some extent. RCAF evaluation of the ARROW 2 mockup resulted in a number of change requests which are presently under design consideration.

A major problem affecting the ARROW 2 electrical system design is the decision as to the manufacturer of the power system. (Both Lucas-Rotax and Canadian Westinghouse are under consideration). As a result, design of the main power panel for ARROW 2 is held up until the source of equipment is decided upon.

However, engineering of the ARROW 2 system has been carried out, based on a Lucas-Rotax proposal. The Lucas-Rotax system was originally selected since it was lower in cost and comparable in weight and performance to the other system proposal.

10.3.2 LOAD ANALYSIS

The electrical load analysis for both the ARROW 1 and the ARROW 2 have been completed. In the case of the latter aircraft, the load analysis determined that two 40 KVA generators and two 4.5 KW transformer-rectifier units would be required.

Load analysis for the first three ARROW 1 aircraft has shown that no problems need be anticipated in this respect. For the fourth and fifth aircraft however, it will not be possible to use both de-icing and instrumentation simultaneously with ASTRA 1. Provision will be included in these aircraft to allow selection of either de-icing or instrumentation.

10.4 ELECTRICAL SYSTEM DEVELOPMENT

Electrical system alterations due to functioning tests and changes of requirements are continually being made, and are briefly described as follows:

10.4.1 REVERSE CURRENT RELAY

As a result of breadboard testing it was found necessary to introduce a reverse current relay, in place of the original relay, to isolate the battery and the emergency DC buses in the event of a line fault. It was discovered that the original relay allowed a lock-on circuit under these conditions, whereby the battery supplied the main bus, in addition to its designed function of supply to the emergency and battery buses during emergency conditions.

10.4.2 RAM AIR TURBINE

Provision has been made in the ARROW 1 electrical system for extension of the ram air turbine supplying emergency hydraulic power. Actuation is initiated by the operation of a cockpit switch, energizing a solenoid valve in the extension mechanism hydraulic circuit.

No action has yet been taken regarding the design of a ram air turbine electrical circuit for ARROW 2 as the location of the turbine in the aircraft is still being considered. It is anticipated however, that extension of the unit will be initiated automatically on failure of the normal electrical supply, the turbine being required to provide both hydraulic and electrical emergency power for ARROW 2.

10.4.3 NICKEL-CADMIUM BATTERY

A nickel-cadmium type battery is used in the ARROW electrical system. A thermal protection relay is highly desirable with this type of battery and will therefore be incorporated in the ARROW 2. Tests of battery operation at temperatures below -15°F indicate that a battery warm-up period would be necessary if it were subjected to "cold soaking" below this temperature. It is therefore proposed to replace the battery should the aircraft be required for immediate flight after cold soaking.

A report on battery procedures is being prepared.

10.4.4 FUEL SYSTEM

The 70% level warning signal has now been deleted from the fuel level warning system in ARROW 1, as changes in fuel system pressures have caused the 70% level to become a normal condition during flight.

A change has been made to the minimum fuel distribution shift sequence as a result of ARROW 1 flight requirements. The partial refuelling sequence has also been changed.

Circuit arrangements ensure that the ARROW 2 external fuel tank, which is wired into the existing C. G. control system, is drained first in the sequence. The tank jettison circuit is energized automatically when missile firing is initiated to ensure that the tank is jettisoned before the missile doors open.

An investigation is presently being conducted into total fuel indication in the rear cockpit for ARROW 2

10.4.5 AIR CONDITIONING

Minor improvements to ARROW 1 cockpit temperature control wiring have been incorporated.

An ARROW 2 turbine unit overspeed light has been added, and a press-to-check switch introduced for the evaporator venting control.

A light introduced into the cockpit provides warning of air conditioned equipment overheating, and replaces the signal into the ground check annunciator box for both ARROW 1 and ARROW 2.

10.4.6 LANDING GEAR CONTROL

The ARROW 2 "landing gear up" warning circuit has been changed to include a cut-out which operates at 10,000 feet, to prevent operation of the warning above that altitude.

Provision has been made for the supply of an electrical signal to the damping system on commencement of landing gear lowering. This is required to switch the damping system to low speed operation configuration with landing gear down.

10.4.7 DE-ICING

Radome de-icing is no longer incorporated, although provision has been made for its inclusion at a later date, if required. A single de-icing control box has been introduced for windshield and canopies on ARROW 2, and de-icing of the obs/AI's panel has been added.

10.4.8 FIRE PROTECTION

The inertia crash switch has been deleted as it was considered that inadvertent operation of the unit due to vibration was a possible disadvantage outweighing the advantages of using the switch.

10.4.9 FLIGHT SERVICES

A position potentiometer, driven by the right aileron, has been added, and is connected to the cockpit indicator to provide separate indication of both left and right ailerons. This allows correct indication of aileron position for ARROW 2 with the addition of 4° up actuation at 45,000 feet.

Wiring of the Obs/AI's bail-out warning has been changed. The circuit is now supplied directly from the battery. This is a result of an RCAF change request for ARROW 2.

10.4.10 LANDING AND TAXI LIGHT SWITCH

The landing and taxi light switch has been altered to read LANDING-TAXI-OFF. It originally read LANDING-OFF-TAXI, but was changed at RCAF request.

10.4.11 MISSILE LAUNCHER RETRACTION

The ARROW 2 armament system circuits have been altered to allow the forward missile doors to remain closed until the rear missiles are fired, when in the "ALL" mode. This will minimize the entry of fumes into the armament bay.

11.0 AIR CONDITIONING SYSTEM

11.1 GENERAL

The ARROW is equipped with a simple air cycle air conditioning system with a water evaporator included in the system to increase the cooling capacity. The evaporator in located between the heat exchanger and the expansion turbine is a series circuit arrangement.

The results of recent testing have required the introduction of a number of modifications to the ARROW 1 system. An elaboration of the necessary modifications is given in the following discussions.

11.2 ARROW 1 AIR CONDITIONING SYSTEM

The system for the first three aircraft is shown schematically in Figure 22. The system will be modified for the fourth and fifth aircraft, to provide for the increased cooling load resulting from the introduction of the ASTRA 1 test installation.

11.2.1 FIRST AIRCRAFT

All system components, with the exception of the turbine-fan unit, heat exchanger and boiler are being subjected to pre-installation tests on the air conditioning system test rig. By this procedure, the necessary system modifications and component adjustments can be made, and installation procedures and techniques established, before final assembly in the aircraft. System components are being transferred directly to the aircraft, once test requirements have been met.

Some modifications to the system have been necessary as a result of these tests. The areas affected are indicated in Figure 22.

11.2.1.1 Equipment cooling circuit

Tests on the system test rig have revealed a temperature stratification in the equipment cooling circuit ducts. The apparent cause of the unsatisfactory temperature distribution was inadequate mixing of the hot and cold airflows. The situation was corrected by inserting flow deflectors in the hot air by-pass duct and the augmentor air duct (Figure 23) where they enter the main cooling-air duct. This modification is being incorporated in the ARROW 1 system design.

11.2.1.2 Cockpit temperature control circuit

An unstable cockpit temperature was observed when the temperature setting of the pilot's control thermostat was suddenly changed. A sudden demand for increased cockpit temperature resulted in the following sequence of events:

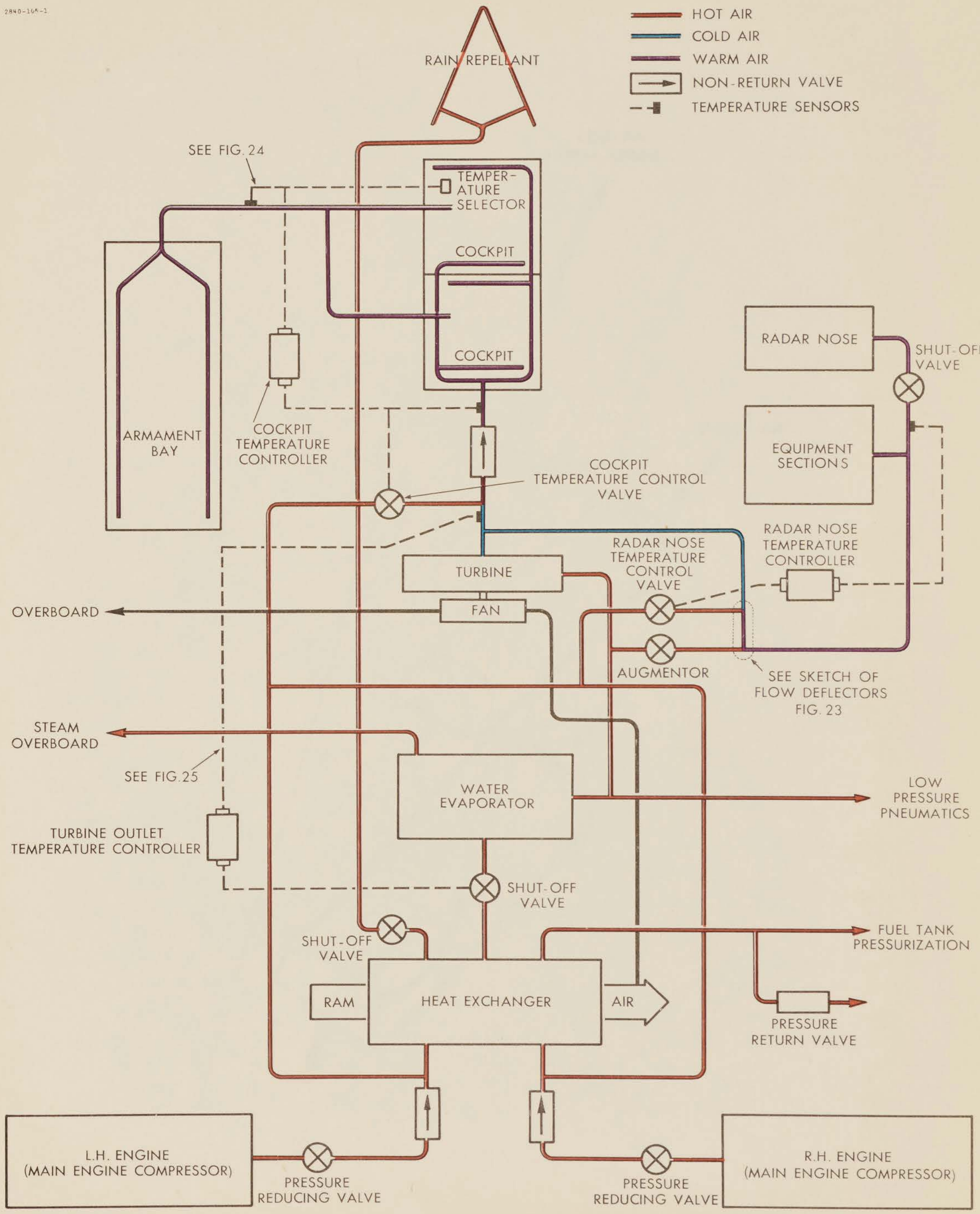

FIG. 22 ARROW 1 AIR CONDITIONING SYSTEM

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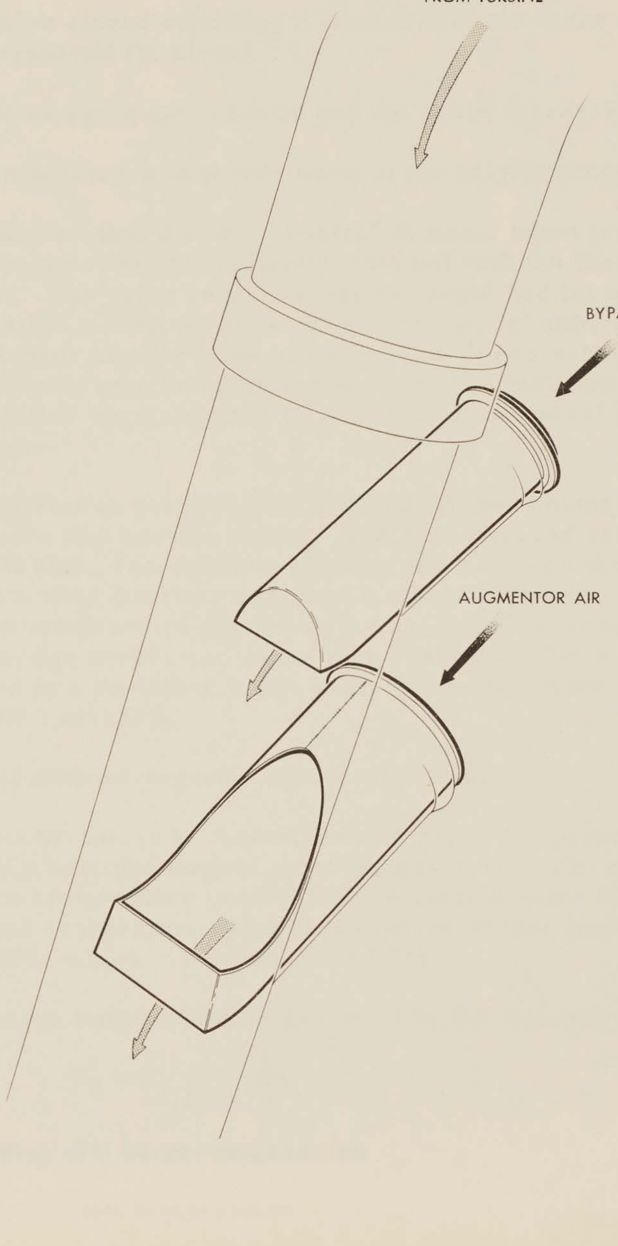

FIG. 23 FLOW DEFLECTORS

(i) The hot-air bypass valve opened fully to accommodate the increased temperature demands of the cockpit.

(ii) The flow of air into the cockpit at a temperature well above the selected temperature operated the overheat thermostat (to open).

(iii) The bypass valve closed admitting a flow of cold air to the cockpit which cooled the thermostat (to close).

(iv) The bypass valve again opened fully and the cycle repeated.

Maximum inlet temperatures observed were in the neighborhood of 250°F.

An investigation showed that the basic control system, apart from the control thermostat feature, was not satisfactorily matched with the thermal lag in the cockpit circuit. The valve response was too rapid and its sensitivity too great. The derivative circuit of the temperature control unit, interpretating the temperature sensor signals from upstream and downstream of the cockpit, was not sufficiently matched with actual thermal conditions. Improved control characteristics were obtained by modifying the control circuit as shown in Figure 24.

The 4000 ohm, resistor in parallel with the control thermostat reduces the response of the valve to a control signal. The 2 microfarad capacitor in series with the 2000 ohm, resistor are wired in parallel with the control valve to introduce a time constant and reduce valve sensitivity. The derivative circuit of the temperature control unit was improved by connecting a 4 microfarad capacitor across connector pins F and H. These components are being packaged in a container which will be installed in the first and subsequent ARROW 1 aircraft.

11.2.1.3 Turbine outlet-temperature control circuit

The purpose of this circuit is to control turbine outlet temperatures within specified limits of a selected control point temperature. The components in the circuit are a temperature control unit, a control valve located upstream of the water evaporator, and a temperature sensor located downstream of the turbine outlet. (Figures 22 and 25).

The turbine discharge temperature is governed by the following expression:

$$T_2 = T_1 \Big/ \left(\frac{P_1}{P_2}\right)^{\frac{k-1}{k}}$$

where: T_2 = turbine discharge temperature

T_1 = turbine inlet temperature

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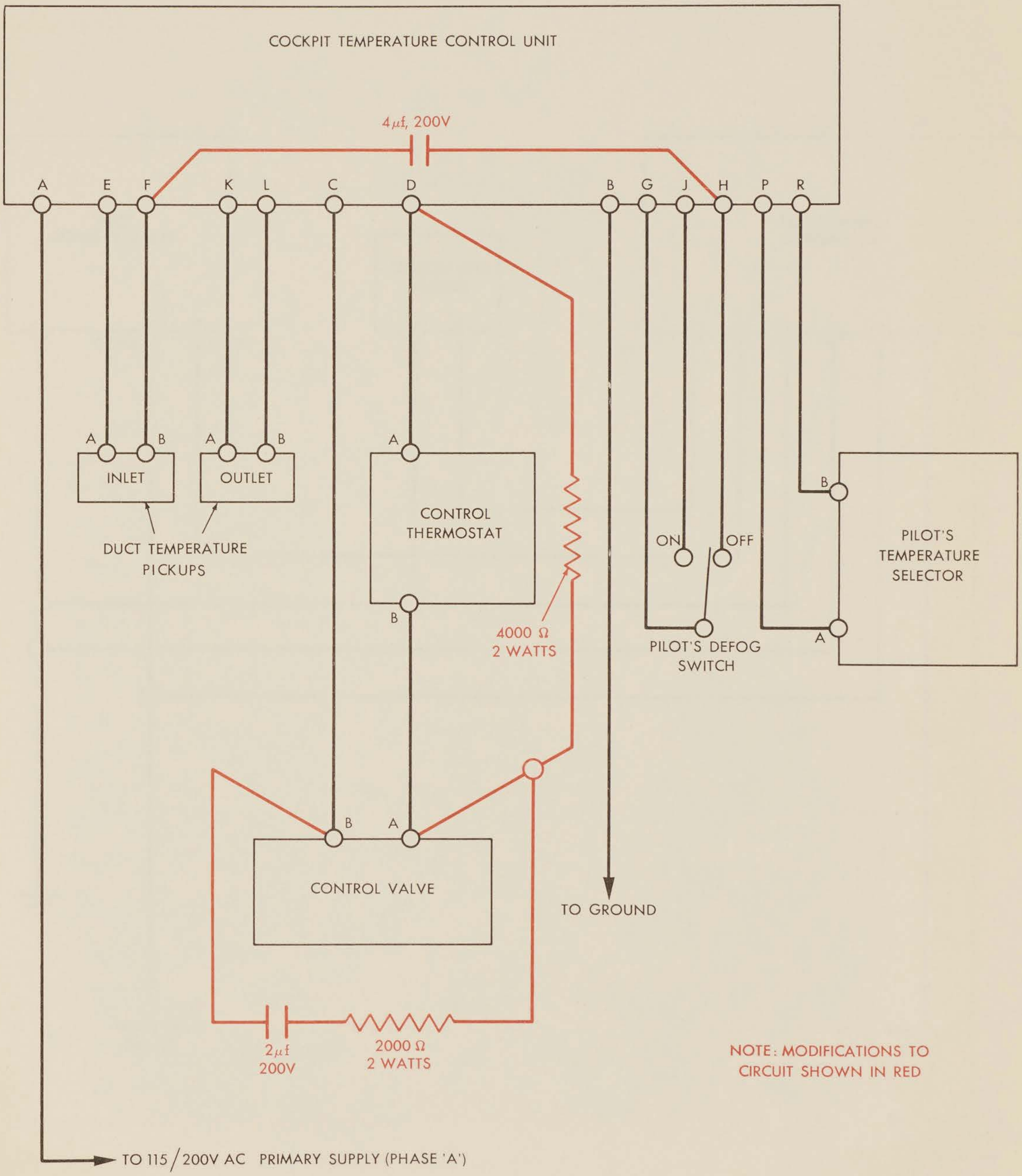

FIG. 24 COCKPIT TEMPERATURE CONTROL CIRCUIT

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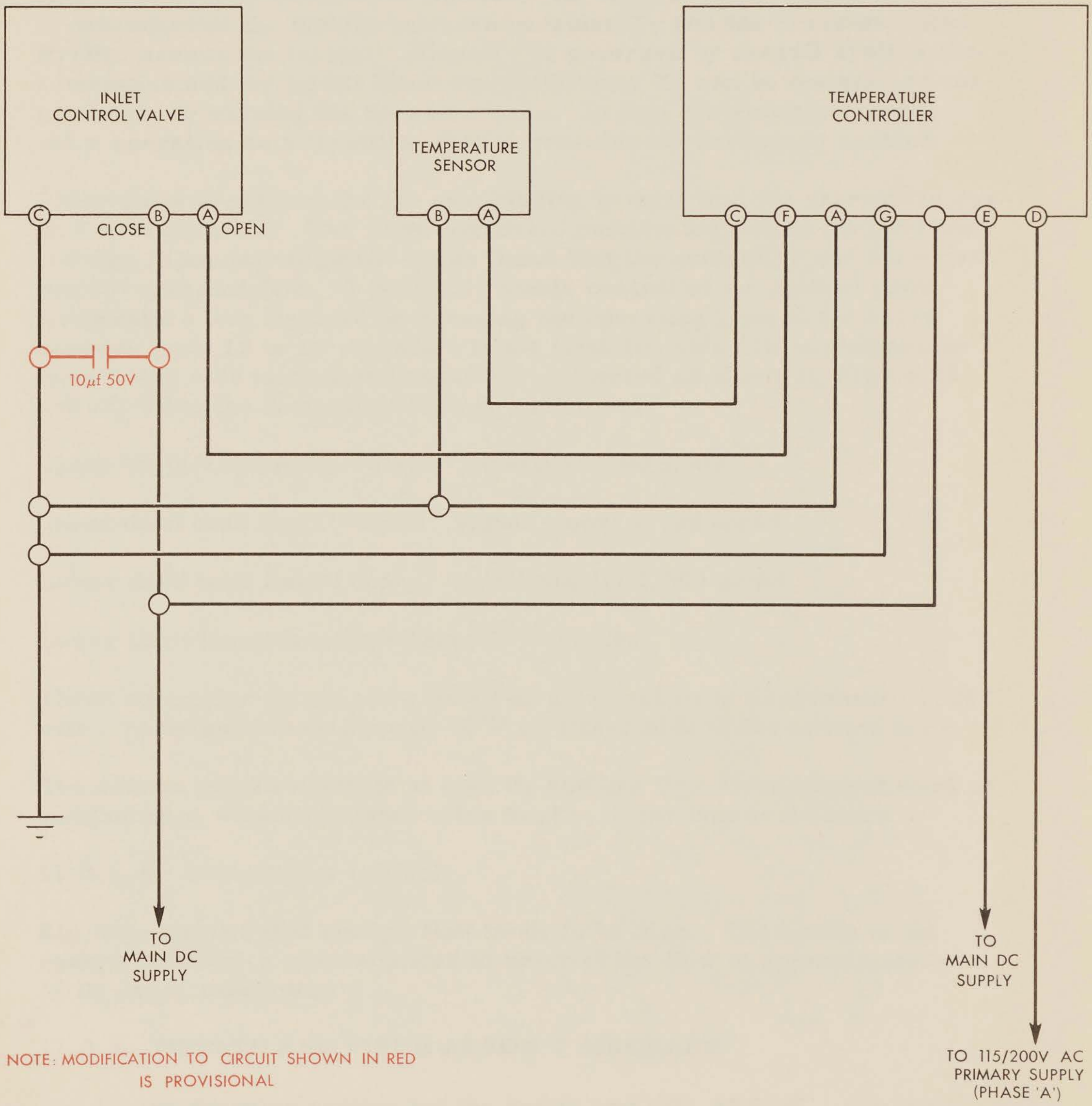

FIG. 25 TURBINE OUTLET TEMPERATURE CONTROL CIRCUIT

P_2 = turbine discharge pressure

P_1 = turbine inlet pressure

k = c_p c_v = ratio of specific heats

Since k is usually considered constant, the control point temperature, T_2, is dependent on the turbine inlet temperature T_1 and the pressure ratio, P_1/P_2, across the turbine. Since T_1 is governed by overall system characteristics and the initial bleed air conditions, T_2 can be maintained constant only by varying the pressure ratio. In this system, the control valve operating as a throttling device provides the necessary control.

The results of tests in the air conditioning system test rig showed the system to be unstable. The valve oscillated continuously about its optimum setting. Upon investigation it was found that the controller and the valve were poorly matched. A generally stable control of the control point temperature was obtained by changing the operating time of the valve actuator from 10 to 20 seconds. In the first aircraft this is obtained by introducing a 10 microfarad capacitor connected as shown in Figure 25 and adjusting the controller settings as follows:

Upper limit (continuous "close" signal) 70,000 ohms.

Upper dead band limit ("close" signal stops) 8,800 ohms.

Lower dead band limit ("open" signal starts) 4,800 ohms.

Lower limit (continuous "open" signal) 50 ohms.

These controller settings are based on a control point temperature of 20°F with a permissible variation of 10°F on either side of the control point.

The system is still unstable at high P_1 and low T_1. Development work is continuing on a cam-operated valve to give linear characteristics.

11.2.1.4 Cockpit flow control

Rig tests showed that cockpit flow tends to be high. The limits to the cockpit inlet valve were adjusted to restrict the flow to approximately 30 lb./min. maximum.

11.2.2 FOURTH AND FIFTH ARROW 1 AIRCRAFT

The air conditioning system for the fourth and fifth ARROW 1 aircraft is essentially the same as the system shown in Figure 22. The main difference is the deletion of the rain repellent circuit. The rain repellent air supply line is being re-routed to supply air to a small water evaporator and expansion turbine installed as an integral part of the armament bay instrument pack.

The design of this system is progressing satisfactorily. Obviously, development flight testing of the system in the first three aircraft will influence the design of this variant of the system.

11.2.3 MAINTENANCE

11.2.3.1 Water boiler filling technique

The development of a filling technique which eliminates the use of a dipstick is presently being handled by Wayne Pump Co. The method proposed uses an off-the-shelf filling nozzle modified to AVRO requirements. The development of this item is well in hand; further details should be available in the near future.

11.3 ARROW 2 AIR CONDITIONING SYSTEM

The ARROW 2 system is similar to the ARROW 1 system in principle only. The installation of the Iroquois engine has resulted in an entirely new set of system inlet-air conditions. In addition, a modified flight envelope, combined with the installation of additional electronic equipment and missile armament, has led to an increased cooling load. Consequently, for the ARROW 2, the system has had to be completely re-engineered.

The system as presently conceived is adequately described in Report No. 72/SYSTEMS/22/48, ARROW 2 Air Conditioning System June 1957.

The design of the system is presently in an advanced stage; however, the results of ARROW 1 testing and performance analysis could very quickly change the status of the design program. Performance analysis, utilizing automatic computing machinery, is presently being prepared. Vendors for the development and supply of the required system components have been contacted and the progress in this regard is satisfactory at present.

11.4 GENERAL DEVELOPMENT PROGRAM - AIR CONDITIONING

A general test program for system development, utilizing the static test rigs, the metal airplane mock-up, and the aircraft test vehicles is being considered.

11.4.1 TEST RIGS

The ARROW 1 static test rig has been used most extensively up to the present time. The test program utilizing this rig is discussed in greater detail in Part 6, para 28, under Testing.

11.4.2 METAL AIRPLANE MOCK-UP

A series of tests have been scheduled to investigate the problem of cockpit environment. In conjunction with the RCAF Institute of Aviation Medicine, consideration is being given towards extending these tests to explore pilot reaction and behaviour under prolonged exposure to the cockpit environment.

11.4.3 FLIGHT TESTING

The flight testing of the system will be discussed in subsequent quarterly reports.

12.0 LOW PRESSURE PNEUMATIC SYSTEM

12.1 GENERAL

The low pressure pneumatic provisions in the ARROW aircraft consists of two independent systems, namely:

1. A pressurizing system supplied by compressed air tapped from the air conditioning system downstream of the water evaporator.

2. A pitot-static system using air pressure reference from a nose boom and a fin probe.

The systems for the ARROW 1 and ARROW 2 aircraft are essentially the same. The ARROW 2 system is adequately described in Report No. 72/ SYSTEMS/18/29, June 1957 entitled ARROW 2 Low Pressure Pneumatic System.

12.2 PRESSURIZATION SYSTEM

The system provides air pressure for canopy sealing, crew anti-G suits, armament pack sealing, ASTRA 1 hydraulic system pressurization, and ASTRA 1 waveguide and radar pressurization in the ARROW 2 aircraft.

In the ARROW 1 aircraft, no provision is made for the pressurization of the ASTRA 1 since it is not being installed in the aircraft. Provision similar to armament pack sealing is being made for the flight test equipment pack which replaces the armament pack.

12.3 PITOT-STATIC SYSTEM

This system is identical for both ARROW 1 and ARROW 2 aircraft except where piping runs are re-routed to allow for structural or equipment installation differences in the two airplane variants.

12.4 PROGRESS AND STATUS OF LOW PRESSURE PNEUMATICS

ARROW 2 installation design is in progress.

Functional testing for the ARROW 1 system has been scheduled and will be reported in subsequent issues of this publication.

13.0 FIRE PROTECTION SYSTEM

13.1 GENERAL

The fire protection system for the ARROW aircraft consists of the following systems:

1. Fire detection sub-system, employing continuous wire detectors coupled to a pilot warning light system.

2. High rate discharge fire extinguishing system, employing $CF_2 Br_2$ (Freon) as the extinguishing agent.

The systems for the ARROW 1 and ARROW 2 aircraft are fully described in the following reports:

Engine Installation Data Manual for CF-105 Mk. 1 Aircraft (Pratt & Whitney J75P3-P5 Engines) Part 2, Section 4, April 1957.

ARROW 2 Fire Protection System, Report No. 72/SYSTEMS/23/31, June 1957.

The system is basically the same for the ARROW 1 and ARROW 2 aircraft; System differences are apparent in Figures 26 and 27. The primary difference between the two systems is the installation of a tertiary fire zone in each of the engine fire areas of the ARROW 2 aircraft. This tertiary zone is on the underside of the Iroqouis engine, where the engine accessories are enclosed within a shrouded compartment.

13.2 FIRE PROTECTION SYSTEM TEST PROGRAM

A test program has been scheduled based on the ARROW 1 installation. The results of these tests are likely to influence system design for the ARROW 2 aircraft.

13.2.1 FIRE DETECTION SUB-SYSTEM

A full-scale test rig is being prepared to check the operation of the complete sub-system. The detector control units and the detector loops will be subjected to pre-installation tests. A complete report of these tests will be available within the next quarter.

Preliminary testing of the fire detector control unit has indicated that the trim resistors will have to be replaced by resistors of lower value. This will be required to prevent premature warning of temperature rises in the potential fire areas.

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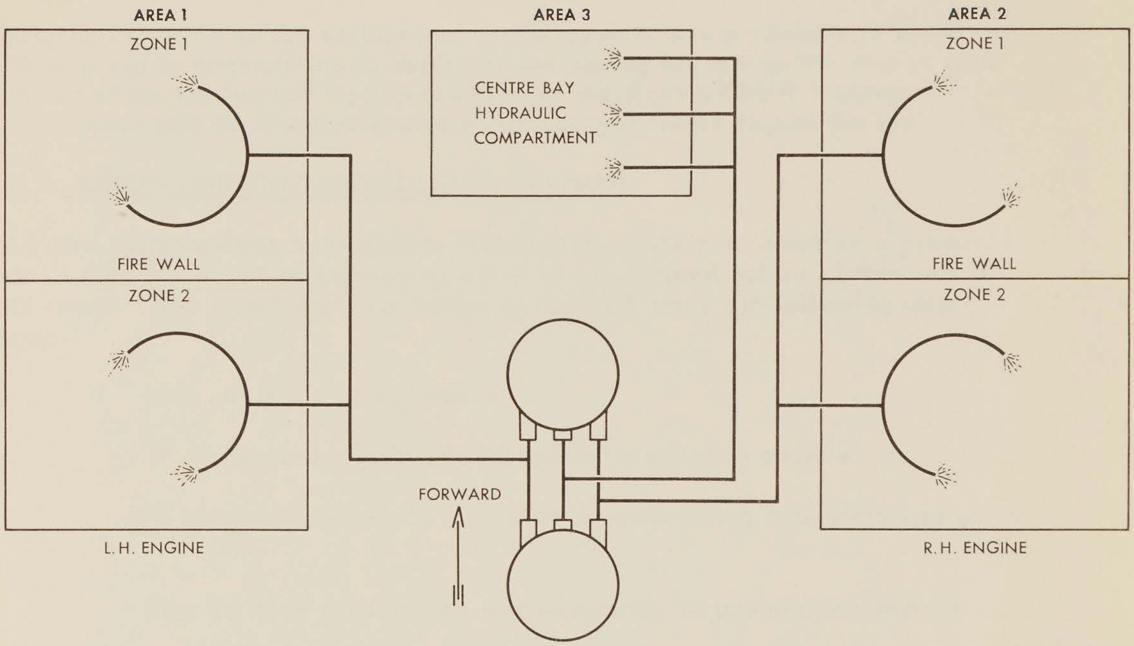


FIG. 26 FIRE EXTINGUISHING SYSTEM ARROW 1

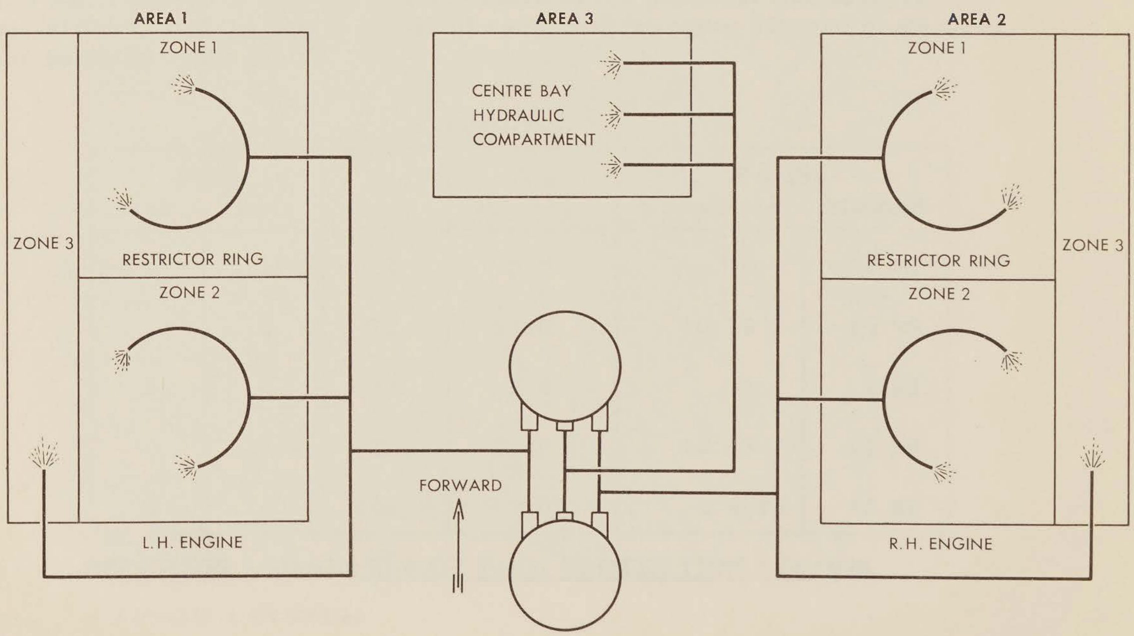


FIG. 27 FIRE EXTINGUISHING SYSTEM ARROW 2

13.2.2 FIRE EXTINGUISHING SUB-SYSTEM

Distribution tests on the extinguishing sub-system are presently in progress. The test rig is constructed to duplicate the piping for one of the engine fire areas and the centre bay equipment area of the first ARROW 1 aircraft. These tests will be completed within the next quarterly report period.

13.3 EXTINGUISHING AGENT DISTRIBUTION

The fire extinguishing sub-system distribution tests are based on a predetermined distribution of extinguishing agent to the defined potential fire areas and zones. The required distribution is derived using the following expression:

$W = .02\,V + .25\,W_a$ where

W = required weight of extinguishing agent in pounds

V = the volume of the fire area or zone being considered, in cubic feet

W_a = air flow through the compartment, in pounds per second

The required distribution for the ARROW 1, based on the above formula, is given in Table 'A'. The calculated values of the potential fire area volume and airflow, and the actual weight of extinguishing agent allotted to the area, are included in the table.

TABLE A

Fire		V	W_a	W - lbs.	
Area	Zone	cu. ft.	lb./sec.	Calculated	Allotted
1	1	23	2.0	.96	1.05
	2	70	35.0	10.15	10.95
2	1	23	2.0	.96	1.05
	2	70	35.0	10.15	10.95
3	-	220	.03	4.4075	12.00

13.4 PROGRESS AND STATUS OF FIRE PROTECTION SYSTEM

13.4.1 ARROW 1 SYSTEM

The design of the system is complete and with the exception of the items to be subjected to pre-installation testing, the system is installed in the first aircraft. The necessary adjustments to the discharge nozzles of the extinguishing sub-system are dependent on the outcome of the distribution tests.

Additional fire protection has been provided in the first aircraft for first flights to meet the specification requirements of CAP 479 and specification MIL-E-5352A (USAF) para. 3.3.4. The system consists of three bottles, each containing a charge of 22 pounds of Freon, mounted in the armament bay and connected to the basic extinguishing system, permitting discharge through the same nozzles.

13.4.2 ARROW 2 SYSTEM

This system is identical to the ARROW 1 system, except for the provision of discharge nozzles for the tertiary zone of the engine fire areas. Re-routing of piping runs and detector wiring will be necessary due to the difference in engine installation.

Provision for overheat warning, in addition to fire warning is now a requirement for the ARROW 2. A proposal for cockpit display of overheat warning and fire warning signals has been submitted to the RCAF for approval.

14.0 FUEL SYSTEM

14.1 GENERAL

A high pressure fuel system employing tank pressurization by air for fuel transfer and a remotely located engine driven booster pump for engine fuel supply has been designed and developed for the ARROW aircraft. The system for the two aircraft variants is adequately described in the following reports:

CF-105, Brochure F-1, Fuel System, February 1956

ARROW 2 Fuel System, Report No. 72/SYSTEMS/16/21, June 1957

Development testing of the ARROW fuel system has been in progress for several months. Full-scale test rigs have been the principal tools in this testing program.

All system components for the first aircraft have been subjected to pre-installation tests and the system is now completely installed in the first aircraft.

The development program for the ARROW fuel system has encountered the usual procurement difficulties and qualification problems. Examples of items which present procurement difficulties are the fuel booster pumps for the engine supply sub-system and the proportioners for the fuel transfer sub-system. The qualification problems usually result in the relaxation of AVRO specification requirements to permit restricted flight approval of components.

14.2 ENGINE SUPPLY SUB-SYSTEM

Vendor qualification tests for the fuel booster pump indicate that the AVRO specified fuel delivery rates for the pump are not being achieved. A delivery rate of 65,000 lb. per hour is reported instead of the specified 100,000 lb. per hour. The vendor agrees that the specified delivery rates can be achieved. However, development time and cost are involved.

The fuel booster pump is being accepted for ARROW 1 installation with limited flight approval since previously specified delivery requirements are not required for the J75 engines. A re-appraisal of the maximum fuel flow rates required by the Iroquois engine in its developed state has been initiated with a view to saving part of the cost of developing pumps with the specified delivery capacity.

14.3 FUEL TRANSFER SUB-SYSTEM

The fuel transfer system provides for the conveyance of fuel from the tributary wing tanks and fuselage tanks to the wing collector tanks. The fuel is forced through the transfer system by the internal tank pressures supplied by the air pressurization sub-system.

Since the aerodynamic characteristics of the aircraft demand a fairly close control of its centre of gravity, some provision for the control of centre of gravity of the fuel load is necessary. Two proposals to meet these requirements were submitted and are presently in advanced stages of development.

The proportioning method of fuel centre of gravity control provides for the emptying of the tributary tanks at rates proportional to the respective tank capacities. Thus, starting with all tanks full, all tanks are emptied in the same elapsed time.

The sequencing method for the control of fuel centre of gravity provides for the emptying of individual tanks in a pre-determined order. The order in which the tanks are emptied is such that the aircraft centre of gravity remains within the specified limits of 28% to 31% MAC.

14.3.1 FUEL PROPORTIONER (FIGURE 28)

The fuel proportioner showed evidence of metering inaccuracies under low flow conditions together with the apparent seizing of the bypass valve, when subjected to tests in the system test rig. An examination of the unit revealed a swelling of the metering vanes and corrosion of the seals between chambers of the bypass valve.

Redesign of the metering vanes has corrected the swelling. A carbon-base material is now used in place of the original phenolic-base material. Further testing with the modified proportioner has shown that the metering inaccuracies are now tolerable for low flow conditions over extended periods.

The difficulties with the bypass valve are expected to be overcome by a change of the material used for the inter-chamber sealing.

Fuel proportioners are now installed in the first aircraft of the ARROW 1 series. Ten of these units were originally ordered for installation in the five aircraft of this series. However, with the units on test in the ground test rig, and making some provision for spares, it will be possible to provide only the first three aircraft with proportioner units. Hence, depending on the availability of new units, the fourth and fifth aircraft may be equipped with either the fuel flow proportioning units or the fuel sequencing sub-systems.

14.3.2 FUEL SEQUENCING SYSTEM

A set of fuel tank drainage sequencing units together with the associated

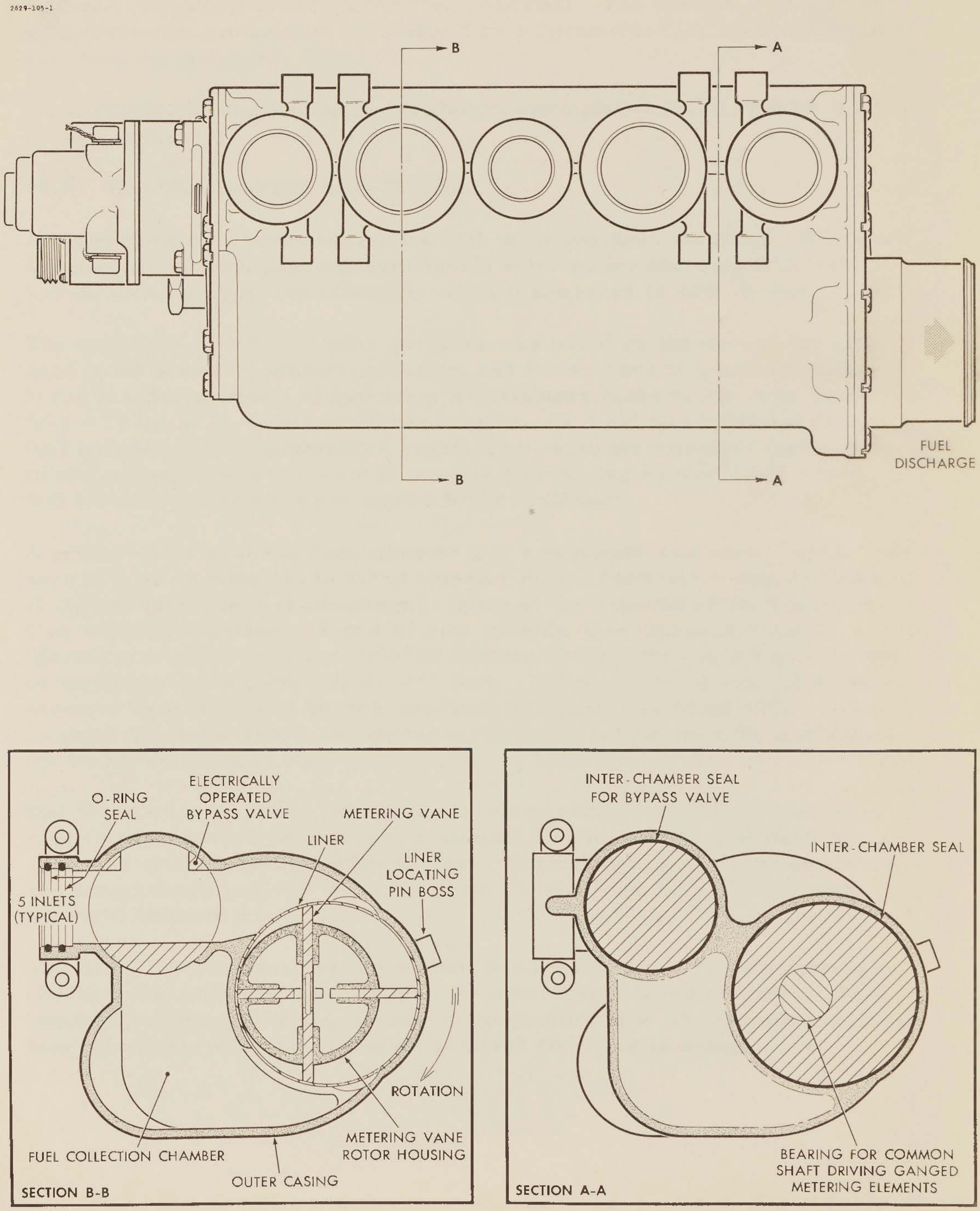


FIG. 28 FUEL PROPORTIONER

system components have been received by AVRO. The system is being installed in the ground test rig and will be subjected to functional testing, commencing in January 1958.

Fuel transfer by sequencing is presently contemplated in the ARROW 2 fuel system design.

14.4 PRESSURIZATION SUB-SYSTEM

A reduced internal pressure for the fuel tanks has been specified. Whereas the nominal internal pressure previously selected for the wing fuel tanks was 25 psia, the reduced allowable nominal pressure is now 19 psia.

The originally selected 25 psia pressure was based on the sum of the standard ICAO sea-level ambient pressure and the necessary pressure differential to effect transfer of fuel from the tributary tanks to the collector tank. Thus, at 14.7 psia ambient pressure, 10.3 psi was available for fuel transfer. By maintaining a constant 25 psi in the tributary tanks, sufficient pressure was available to prevent fuel boiling as well as to effect fuel transfer under the most severe flight conditions.

A review of the problem has indicated that a minimum differential pressure of 8 psi is adequate to effect transfer of fuel from the tributary tanks to the collector tanks if advantage is taken of the capacity of the collector tank as an accumulator. For a 19 psia system, this means that some special provisions for the overboard release of air from the collector tanks is necessary for altitudes up to 8000 feet. The minimum internal pressure required to prevent fuel boiling was fixed at 8 psia (see Figure 29). Thus, as indicated in the chart, the maximum pressure which could be available for fuel transfer is 11 psi.

The installation of an air extractor capable of reducing collector tank pressure from a nominal 14.7 psia to a nominal 11 psia established the pressure available for fuel transfer at 8 psi. Consequently, air-release valves and air admission valves were redesigned to maintain the collector tank pressure between 8 and 11 psi.

The ARROW 1 fuel tank venting system was modified to a 19 psia system by resetting the pressure regulators in the compressor bleed air lines. This involved replacing a spring, which, in conjunction with an evacuated bellows, determines the reference pressure to which the valve is designed.

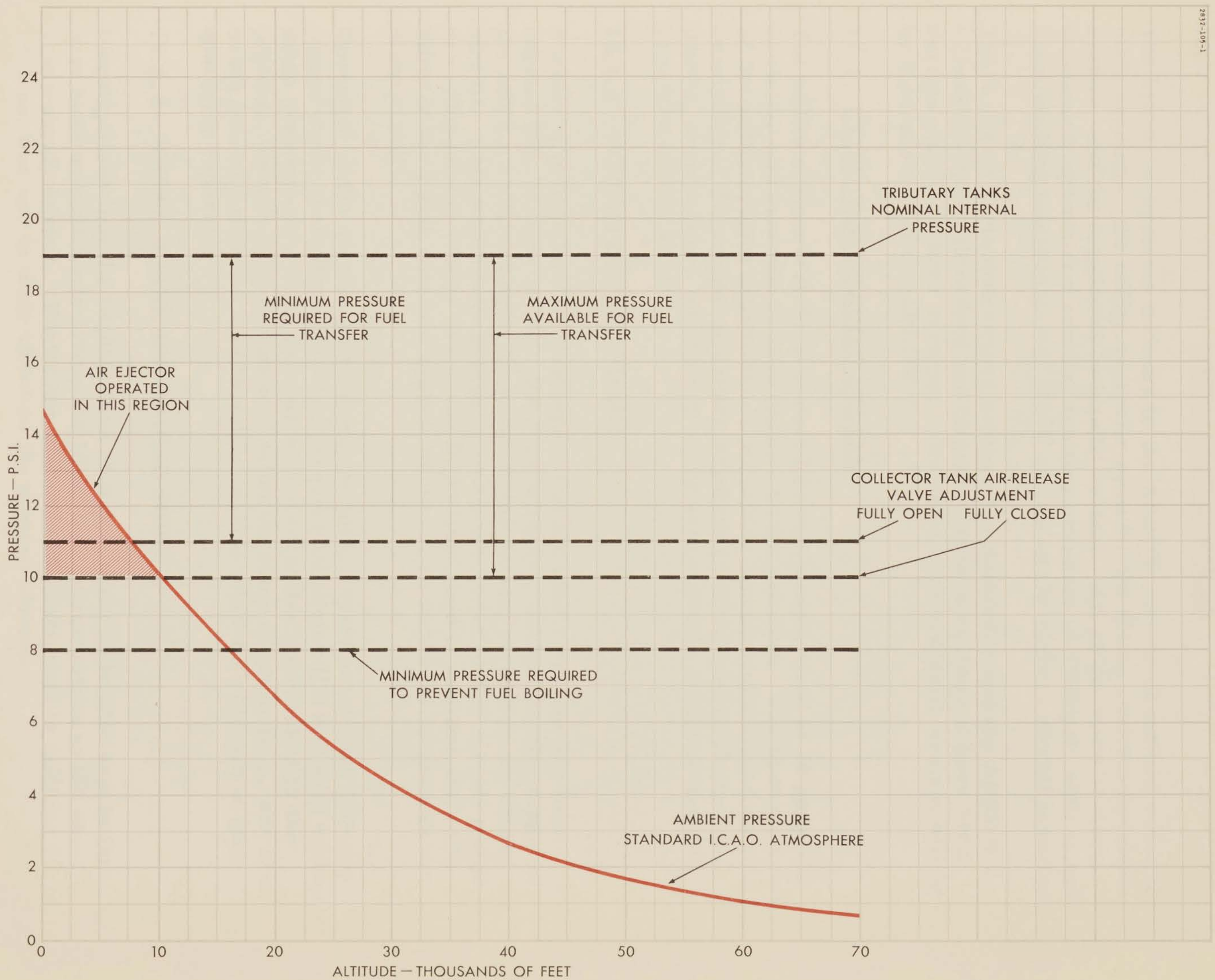


FIG. 29 ARROW 1 - FUEL PRESSURIZATION

15.0 HYDRAULIC SYSTEMS

The hydraulic systems included in the ARROW may be described under three main headings: the utility hydraulic system, the flying controls hydraulic system, and the armament hydraulic system. All of the three systems mentioned operate at a nominal working pressure of 4000 psi, with an optimum working temperature of 250°F and using the airless circuit principle.

There are no basic differences between the ARROW 1 and the ARROW 2, although certain alterations and improvements to equipment have been carried out in the later version. The armament hydraulic system will be be incorporated in the ARROW 1.

15.1 UTILITY HYDRAULIC SYSTEM

The utility system (ref. 9) is powered by two pumps, each of 20 gpm (US) capacity, one mounted on each engine-driven gearbox. The power circuit of the utility system constitutes an unloading circuit and a loaded circuit, consisting of the pumps, pressure regulating valve, heat exchangers, filters, accumulator, emergency brake accumulator, compensator and pressure control valve.

15.1.1 SYSTEM DEVELOPMENT

Further investigation into development of the utility system has been carried out, bearing in mind the possibility of transferring the supply of the ASTRA 1 antenna drive from the flying control circuit to the utility circuit. This would have obvious advantages as the utility circuit pumps are unloaded for 75% to 90% of the flight time, and in addition the reliability of the flying controls system would not be impaired by the antenna drive circuit.

Operation of the antenna drive using the utility system constant delivery pumps would present problems of pressure regulator life and heat exchange; these problems are being considered. Alternative approaches to the problem, namely the evaluation of different pump combinations, are also under investigation and it is hoped to provide some conclusions in later quarterly reports.

15.1.2 WHEEL BRAKE HYDRAULIC SUPPLY

Investigations into the implications of wheel brake hydraulic supply failure have been undertaken to formulate ideas for improvements to the emergency supply system. Particular attention has been paid to failures in the 1500 psi line and to their effect on the normal system.

Fusing the 1500 psi line at the pressure control valve was evaluated as a method of retaining the normal 4000 psi pressure in the event of failure of the 1500 psi line. This was not pursued however, as loss of pressure in the 1500 psi system would de-pressurize the utility compensator and allow the pumps to cavitate, resulting in loss of normal pressure.

The alternative considered was to replace the 200 cu. in. emergency accumulator with one of 100 cu. in. capacity and introduce a second accumulator of 100 cu. in., charged from the normal 4000 psi supply, for emergency braking only. An 80 cu. in. capacity fuse would be incorporated in the off-shoot line to protect the 4000 psi normal braking supply in the event of a failure in the off-shoot.

15.1.3 PRESSURE REGULATOR VALVE

The utility circuit pressure regulator valve diverts pump flow into the unloading circuit when the main accumulator pressure builds up to a maximum of 4350 psi, and conversely it diverts the pump flow back into the main circuit when the accumulator pressure falls to 3850 psi. This loading and unloading feature was found on test to produce pressure surges above the tolerable maximum in both the pressure and the return lines, in addition to rapid cycling of the valve. To obviate these surges, which could become detrimental to the equipment, the 80 cu. in. accumulator in the pressure line has been replaced by one of 200 cu. in. capacity and a small spherical type accumulator has been added to the return line. These alterations, together with modification of the valve, have reduced the surge values to an acceptable figure.

15.1.4 ANTI-SKID SYSTEM

An appraisal of various types of anti-skid equipment has been made with a view to installation and evaluation of an anti-skid system in the ARROW. The Goodyear and Hydro-Aire companies have been asked to tender proposals for test installations of electrically operated anti-skid systems to be fitted to an aircraft for braking tests.

AVRO is making an analytical attempt to determine the parameters which have a bearing on the natural frequency of the landing gear. Initial taxi trials will supply dynamic data to permit prediction of the braking characteristics. This will determine what anti-skid characteristics are required.

15.1.5 BRAKE CONTROL VALVES

The wheel brake hydraulic control valves were found during testing to require an excessive pedal load for their correct operation and to require an undue amount of pedal deflection, i.e.: excessive travel of the valve operating lever.

To remedy this undesirable feature it was necessary to reduce the travel of the valve operating lever from 3 inches to 2 5/8 inches and to reduce the maximum valve operating load from 90 to 80 lb., thereby lessening pedal loads.

15.1.6 PRESSURE CONTROL VALVE

Choice of constant delivery pumps for the utility system made it necessary to modify the combined pressure control and pressure reducing valve to handle the low pressure pump bypass. The modification took the form of the addition of a second relief valve in parallel with that existing for the ARROW 1. Failure of the pressure regulator would have meant that one relief valve had to handle a flow of 40 gpm from the two pumps under the previous arrangement, whereas this would now be divided between two relief valves. A single, larger capacity relief valve was introduced for ARROW 2.

Testing of the pressure control valve induced a failure in the valve body under hydraulic pressure. Revision of the valve body design to improve its structural strength rectified the fault.

15.1.7 SPEED BRAKE JACK

A restrictor has been incorporated in the speed brake jack to limit its extension pressure during retraction of the speed brake. This prevents an excessive pressure build-up in the jack caused by the air loads assisting retraction, and by the restriction of return pressure.

15.2 RADAR ANTENNA DRIVE HYDRAULIC SUB-SYSTEM

The ASTRA 1 radar antenna drive hydraulic system is powered by a motor-pump combination, the motor portion of which is driven by a pressure of 4000 psi from the flying controls hydraulics 'A' system, and is controlled to 3 1/2 gpm by a built-in flow control valve. The pump is of the constant delivery type, supplying the antenna drive system with 13 gpm at a pressure of 1000 psi. This motor-pump combination is designed to stall with a load of 1000 psi in the antenna drive system. An accumulator in the motor drive lines maintains the pressure should the flying controls demand sufficient fluid to starve the motor system. System overloading is prevented by a relief valve set at 1250 psi. A diagram of the system is shown in Figure 30. The motor-pump and major items of equipment in the 1000 psi portion of the system are government furnished as part of ASTRA 1.

15.2.1 EFFECTS OF FAILURES

The effects of failures in the antenna drive system have been analyzed as follows:

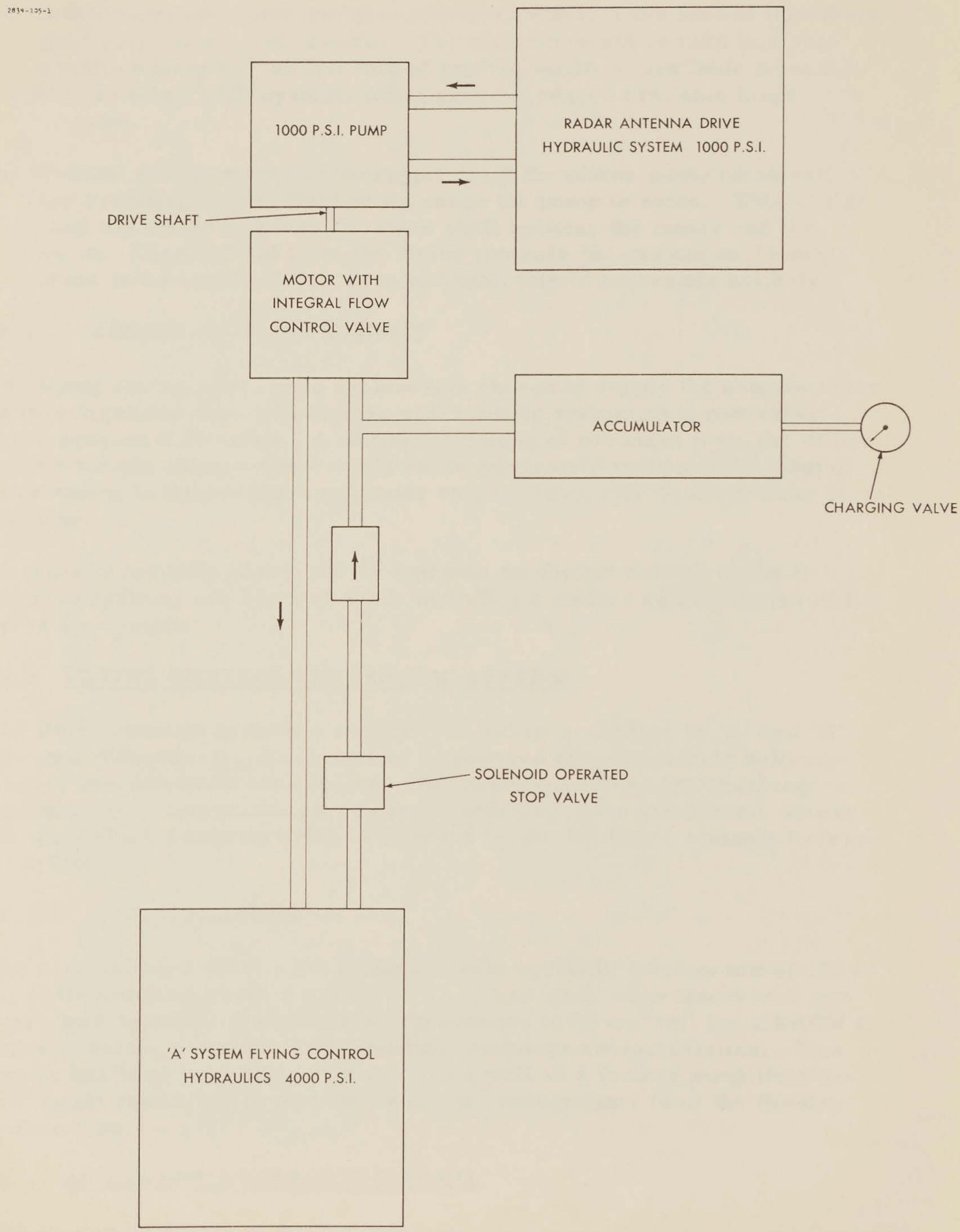


FIG. 30 RADAR ANTENNA DRIVE SYSTEM

(a) A failure of the motor-pump supply lines would cause loss of fluid from the flying controls 'A' system. The aircraft would remain in a safe condition however, as full rate of control would be available from the flying controls 'B' system, although at a reduced available hinge moment.

(b) Failure in the antenna drive supply from the motor-pump combination and resulting loss of fluid would cause the pump to seize. This could stall the motor or shear the drive shaft between the motor and the pump. Loss of fluid from the flying controls 'A' system could only occur if the motor casing were to burst; this is extremely unlikely.

15.2.2 CHOICE OF DRIVE SOURCE

The flying controls hydraulic system was chosen to supply the antenna drive as investigations show that the use of the utility system as it now exists would present difficulties. A constant bleeding of pressure from the utility system for the antenna drive would cause continuous cycling from loading to unloading in the pressure regulator valve, giving rise to rapid wear of this unit.

Alternative methods of driving the antenna, to obviate the use of the flying controls system, are being studied, including a method employing the utility hydraulic system.

15.3 FLYING CONTROLS HYDRAULIC SYSTEM

The flying controls hydraulic system (ref. 10) is duplicated as 'A' and 'B' systems respectively. Each system is powered by two variable delivery pumps, one driven by each engine. The system includes heat exchangers, accumulators, compensators, pressure reducing valve and filters. Power for the ASTRA 1 antenna drive is supplied by the 'A' flying controls hydraulic system.

15.3.1 ACCUMULATORS

The accumulators used in the flying controls hydraulic system are of 100 cu. in. self-displacing type for ARROW 1. It has since been discovered however, that the capacity could be safely reduced to 25 cu. in.. for ARROW 2 and still retain adequate valve frequency response characteristics. This change has been incorporated in design as well as a further simplification and weight reduction, in that the 25 cu. in. accumulator is of the floating system type.

15.4 ARMAMENT HYDRAULIC SYSTEM

The schematic design of the armament hydraulic system (ref. 11) was finalized in late 1956 and no major alterations have since been necessary. Detailed engineering work has subsequently been concentrated on analytically proving the performance and studying the expected characteristics of the system. Progress has mainly consisted of definition of equipment and specification requirements.

The testing about to commence for the armament hydraulic system will take the form of:

1. Functioning tests.

2. Temperature tests.

3. Ground firing tests.

4. Leakage tests.

Two problems connected with the system have recently been solved. These are:

1. Deflections of the forward missile jack drag links during extension have caused the change from series to parallel in the hydraulic circuit at the incorrect moment. This is attributable to premature operation of the micro-switch, which is actuated by the drag link. This trouble was overcome by attaching the micro-switch to the jack shroud and causing it to be actuated directly by extension of the jack, thus providing positive operation.

2. Dampers were found necessary in all four rear jacks to relieve the impact of extension by reducing the velocity of the last portion of the extension stroke.

15.5 EMERGENCY POWER

An investigation of the problem of emergency provision of electrical and hydraulic power after double engine flame-out has been completed. Two aspects of the problem were considered:

(a) Providing sufficient electrical AC power to bring the aircraft into the engine relight zone after double engine flame-out.

(b) The use of a ram air driven turbine to provide sufficient electrical AC power and hydraulic power to land the aircraft after a double engine flame-out.

15.5.1 RELIGHTING - BOTH ENGINES WINDMILLING OR ONE ENGINE SEIZED

Reference to the curve of flying control pump delivery vs. Mach number and altitude (Figure 31) will show that the output limitations of the engine-driven alternators with J75 engines windmilling are Mach 1.0 at sea level to Mach 1.5 at 55,000 feet. It can be seen, therefore, that to bring the aircraft into the relight zone an emergency source of AC electrical power is necessary, DC being available from the battery.

To meet the essential electrical power requirements for ARROW 1 it was decided to utilize a hydraulic motor-driven alternator, powered from the utility hydraulic system. The windmilling engines driving the utility system pumps also provide sufficient hydraulic power for limited flying control actuation. In the "one engine seized" case, emergency electrical power would still be provided due to the duplication of the utility hydraulic system.

15.5.2 LANDING - BOTH ENGINES WINDMILLING OR ONE ENGINE SEIZED

It will be observed from Figure 31 that the windmilling engines are capable of supplying sufficient hydraulic power to control the aircraft down to Mach 0.4 at sea level or Mach 0.85 at 55,000 feet. Therefore, for landing the aircraft, it is necessary to provide emergency hydraulic power in addition to electrical power.

A ram air driven turbine will be installed on the first three ARROW aircraft of each version, thus providing an insurance for aircraft using relatively undeveloped engines. Beyond these six aircraft the reliability of the engines and associated systems will be more adequately proved and it should then only be necessary to supply emergency electrical power. In the ARROW 1 case the ram air turbine will be required to supply emergency hydraulic power only, as DC is available for telecommunications equipment from the battery and structural considerations limit deployment of the turbine to below 350 knots. It is intended that the ARROW 2 installation supply both hydraulic and AC electrical power for landing.

A single ram air turbine is being used to power an alternator and a hydraulic pump on a common drive shaft. The turbine is installed inside the aircraft and extended into the slipstream when emergency power is required. Hydraulic diagrams for extension of the unit and connection into the flying controls hydraulic system are shown in Figures 32 and 33.

To meet damping system requirements at high Mach numbers the emergency electrical power will be available within 2 to 3 seconds after main supply failure, actuation of the turbine extension mechanism being initiated by main AC supply failure for ARROW 2 and by a cockpit switch for ARROW 1. At

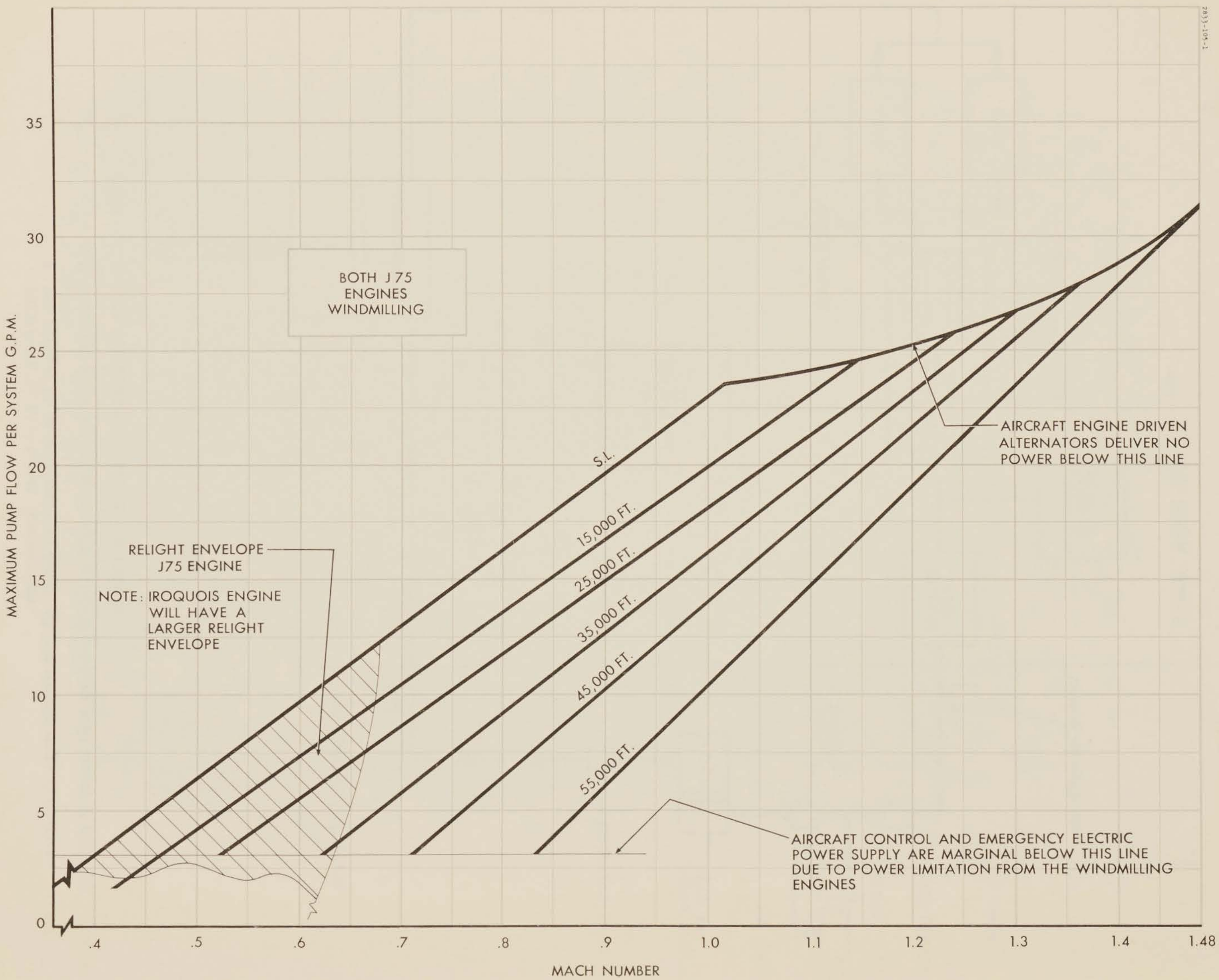
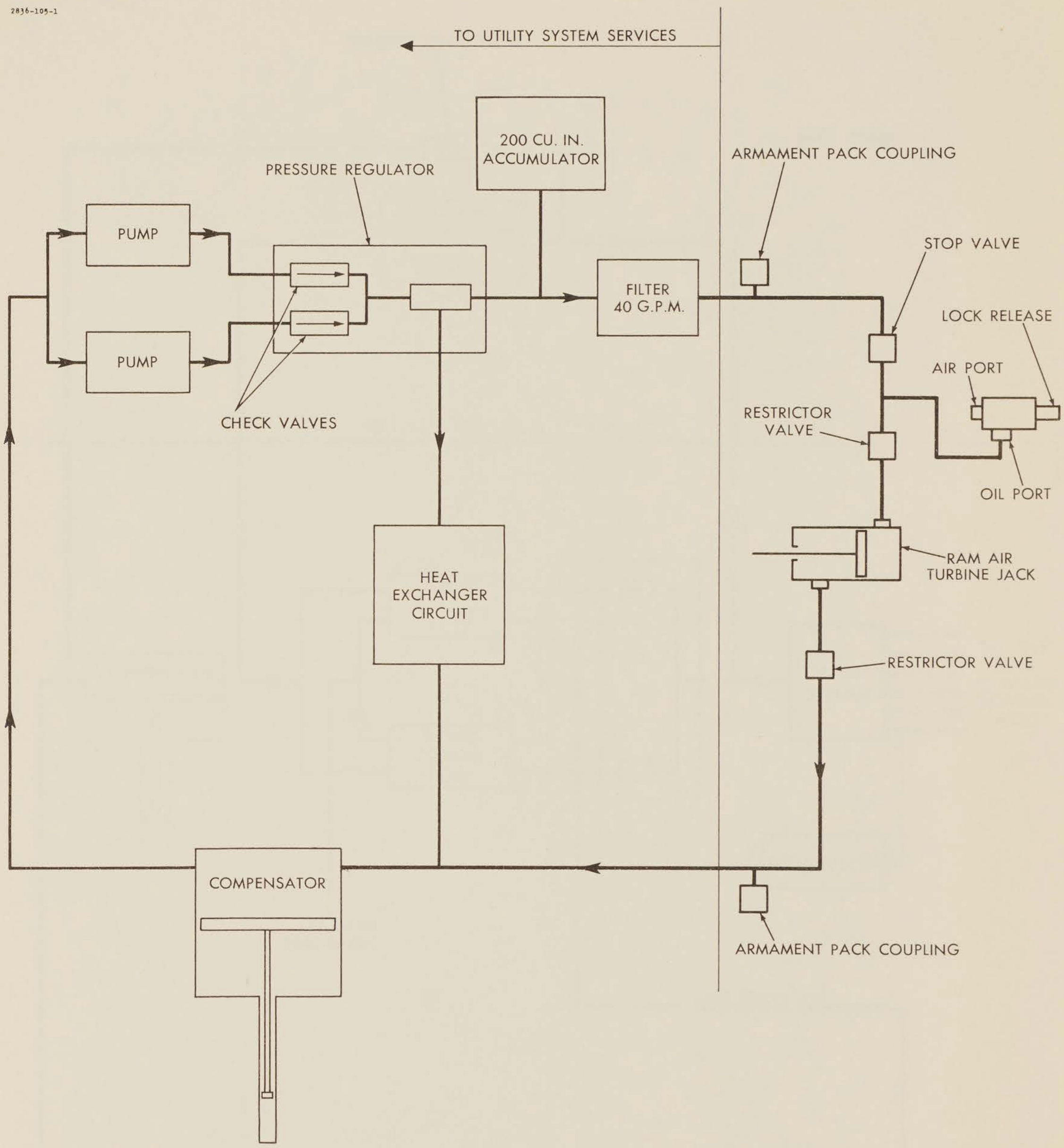


FIG. 31 FLYING CONTROL PUMP DELIVERY VS. MACH NUMBER AND ALTITUDE



NOTE:
1. JACK ROD EXTENDS TO EXTEND THE TURBINE
2. JACK WILL HAVE AN INTERNAL LOCK TO LOCK THE JACK ROD WHEN IN EXTENDED POSITION.
3. THE DOOR IS TO BE MECHANICALLY LINKED TO THE TURBINE AND TO BE LOCKED CLOSED MECHANICALLY BY A SPRING-LOADED LOCK

FIG. 32 RAM AIR TURBINE EXTENSION - HYDRAULIC CIRCUIT

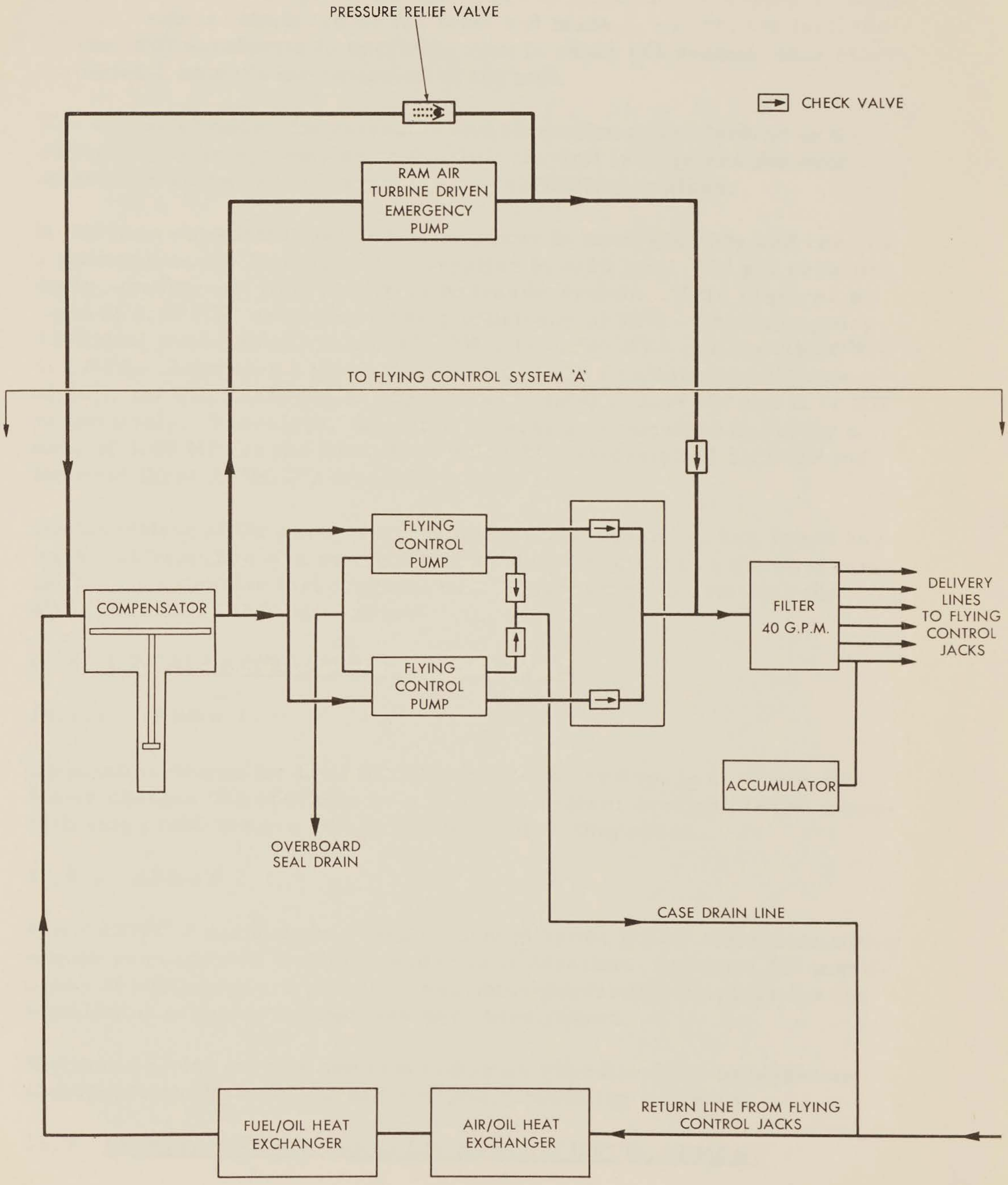


FIG. 33 RAM AIR TURBINE CIRCUIT - FLYING CONTROL HYDRAULICS

the limiting aircraft speeds for operation of the main AC supply by windmilling engines (Mach 1.0 at sea level and Mach 1.5 at 55,000 feet) the turbine will accelerate to operating rpm in about 1/2 second, thus allowing about 2 seconds for extension of the unit.

The source of power for extension and retraction of the turbine is the utility hydraulic system, as at the time the unit is required the hydraulic system pressure is maintained by the windmilling engines.

It has been calculated that hydraulic power to meet stability and control requirements for landing can be supplied by a 10 gpm, 500 psi pressure pump, feeding one flying controls hydraulic system. This requires an input of 3.25 HP, assuming a pump efficiency of 90%. The emergency electrical power required for ARROW 1 is 0.425 KVA and for ARROW 2 is 1.4 KVA. Assuming a power factor of 0.9 and an alternator efficiency of 80%, the electrical power requirements totaled 0.64 HP and 2.11 HP respectively. Therefore, the ram air turbine is required to supply a total of 3.89 HP for the first three ARROW 1 aircraft and 5.36 HP for the first three ARROW 2 aircraft.

Consideration of the power requirements and aircraft landing speed has led to the selection of a suitable turbine unit for testing in the first aircraft. An extensive test program will be necessary to establish the airworthiness of the installation.

15.6 INSTALLATION DESIGN - HYDRAULICS

15.6.1 ARROW 1

Installation design for the ARROW 1 hydraulic system is now complete. Minor changes and additions as a result of system development or manufacturing problems are being incorporated as they arise.

15.6.2 ARROW 2

The ARROW 2 mock-up has been evaluated by the RCAF and the resulting change requests are presently under consideration. Schemes for installation of equipment are complete and some production drawings for the installation of major components have been issued.

Hydraulic piping layouts are proceeding on schedule from information obtained from the mock-up and equipment mounting schemes.

15.7 PROGRAM FOR HYDRAULIC SYSTEM DEVELOPMENT

The program of work to be carried out on the hydraulic system includes:

1. An investigation into the effects of windmilling Iroquois engines.

2. An analysis of braking characteristics.

3. Further study of the source of power for the ASTRA I radar antenna drive, associated with revision of the utility hydraulic system.

4. Installation details, analysis and testing of the emergency ram air turbine.

16.0 FLYING CONTROLS

16.1 GENERAL

The ARROW flying control system (ref. 12) comprises fully powered, hydraulically actuated control surfaces. Control in normal or auto-pilot operation is provided by a servo system using electrical input signals and hydraulic output servos to provide control surface actuator movement. An emergency system of control is provided which employs a mechanical linkage from the cockpit controls to the control surface actuator valves (ref. Figures 35, 36 and 37)

The functions of the complete flying control system are to control the aircraft in arbitrarily commanded manoeuvres and to stabilize the aircraft in these manoeuvres. Stabilization is accomplished by the damping system (ref. para. 4).

The flying control system, while being fairly well established, has undergone several changes to improve its operation. The following comments are relevant to these changes.

16.2 ELEVATOR CONTROL

To supplement the feel springs, a bob weight is installed in the torque tube which provides some natural feel proportional to g in the pitching plane. This formerly produced a stick force of 3 lb/g but has now been increased to 4.25 lb/g. A bob weight balancing spring is provided to eliminate stick forces when in level flight.

The parallel servo is an electro-hydraulic actuator which performs the function of initiating control movement upon command from the stick force system on the automatic flight control system. The servo was formerly limited to an output force of 165 lbs which has been increased to 175 lbs.

The linkage sensitivity has been changed from 2 1/2% to 5%, where sensitivity is defined as the ratio of the stick movement required for total valve movement (fully open to fully close), to the total available stick movement. This was accomplished by altering the actuator linkage. (Figure 38). The servo elevator and aileron stroke was then changed from ± .6 inches to ± .375 inches.

The elevator feel trim unit is connected to the rear fuselage elevator quadrant. In the event that the feel unit becomes jammed or seized or runs away, an emergency release mechanism is provided which is controlled by a manual switch in the pilot's cockpit. The release of the mechanism will then allow the system to operate freely but with the absence

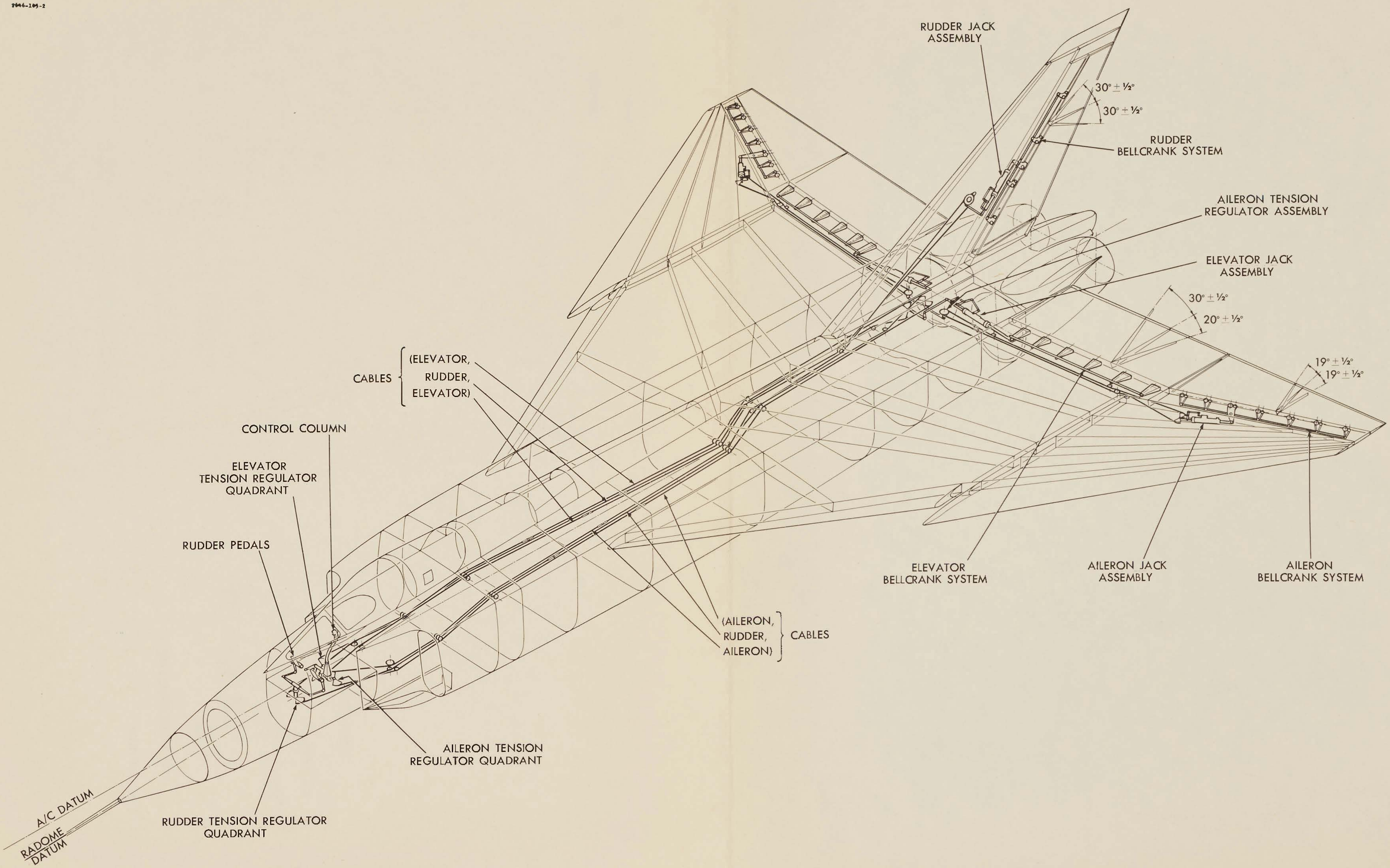


FIG. 34 SCHEMATIC - FLYING CONTROLS

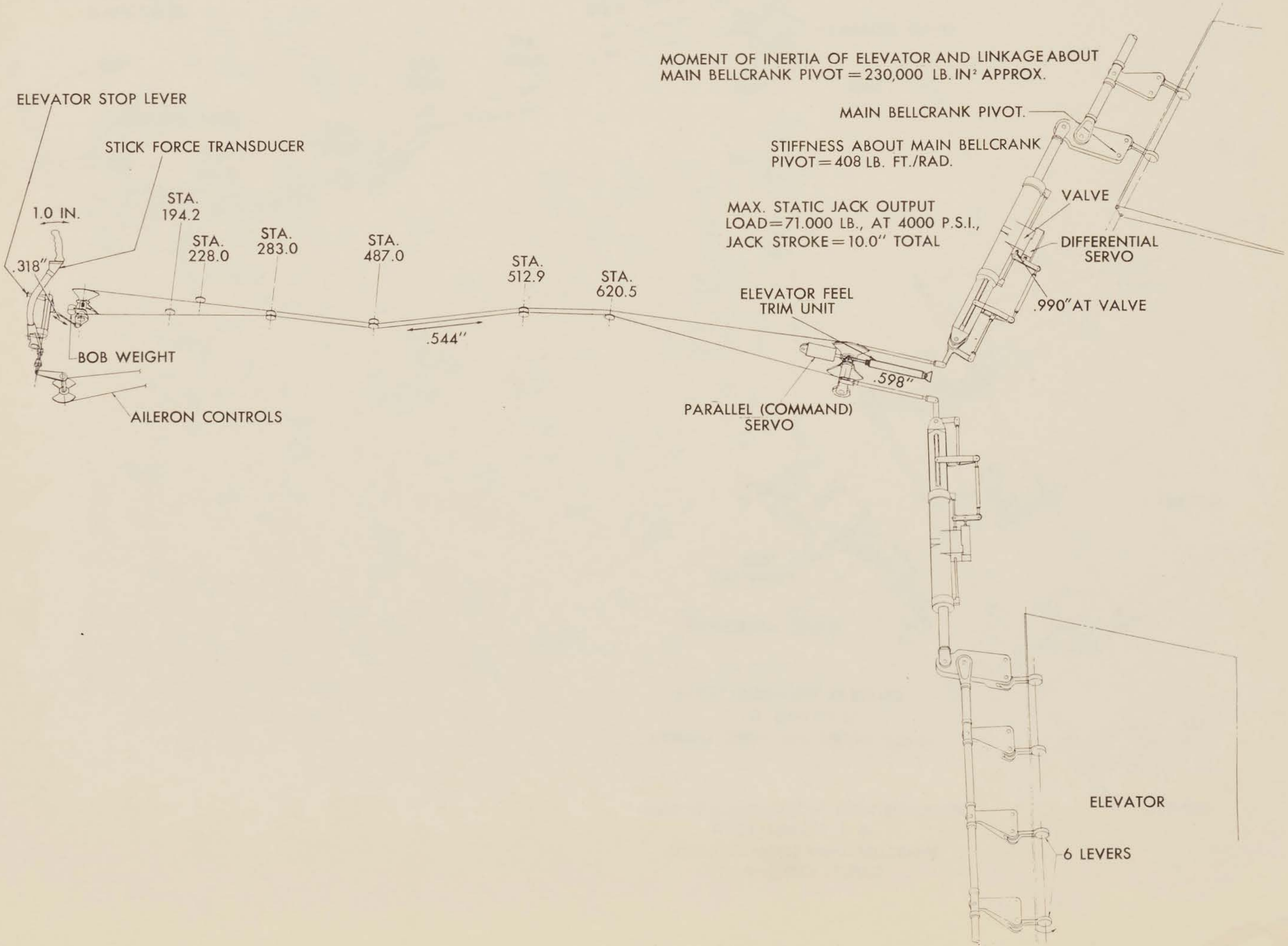


FIG.35 ELEVATOR SYSTEM DATA

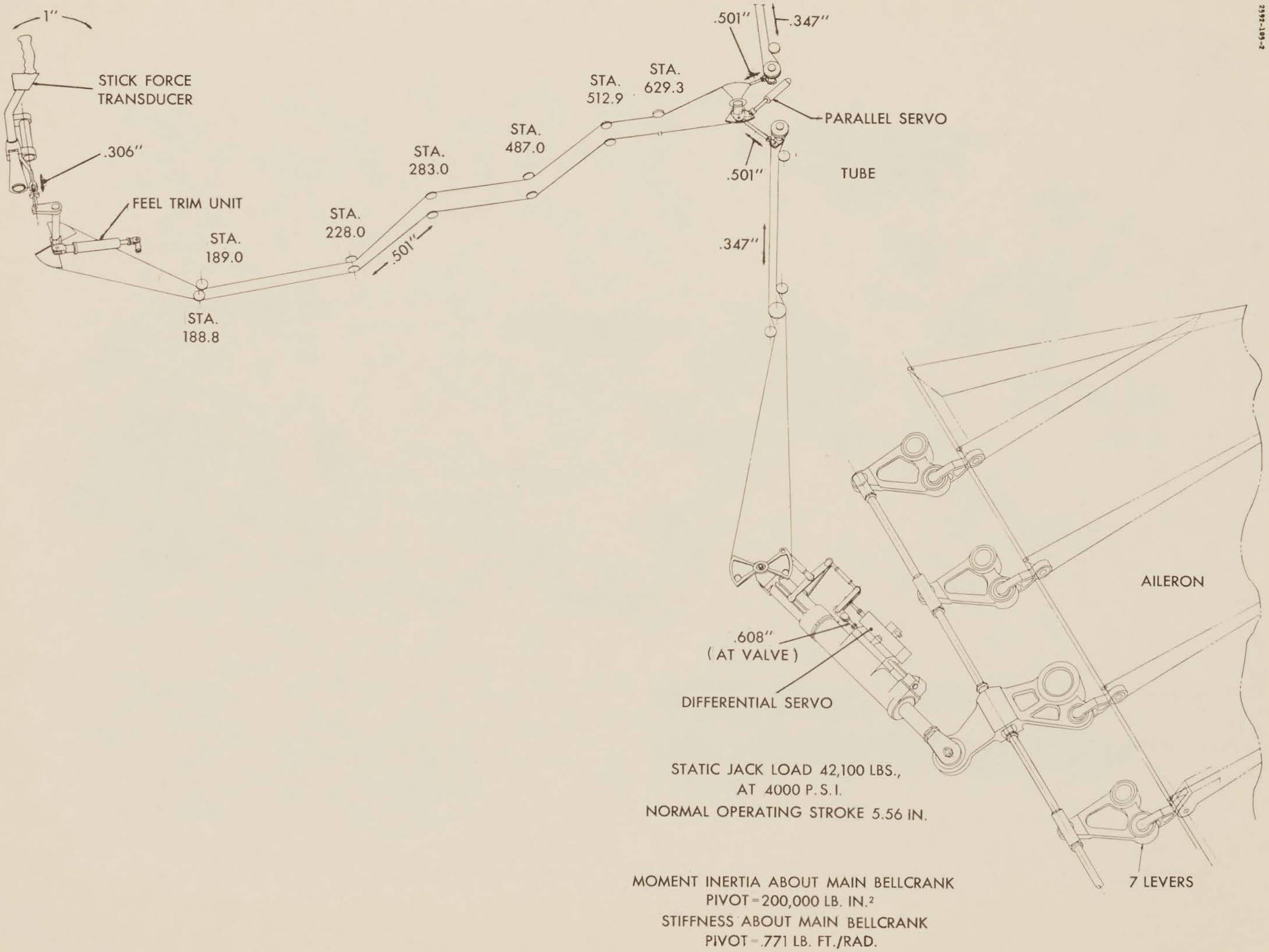
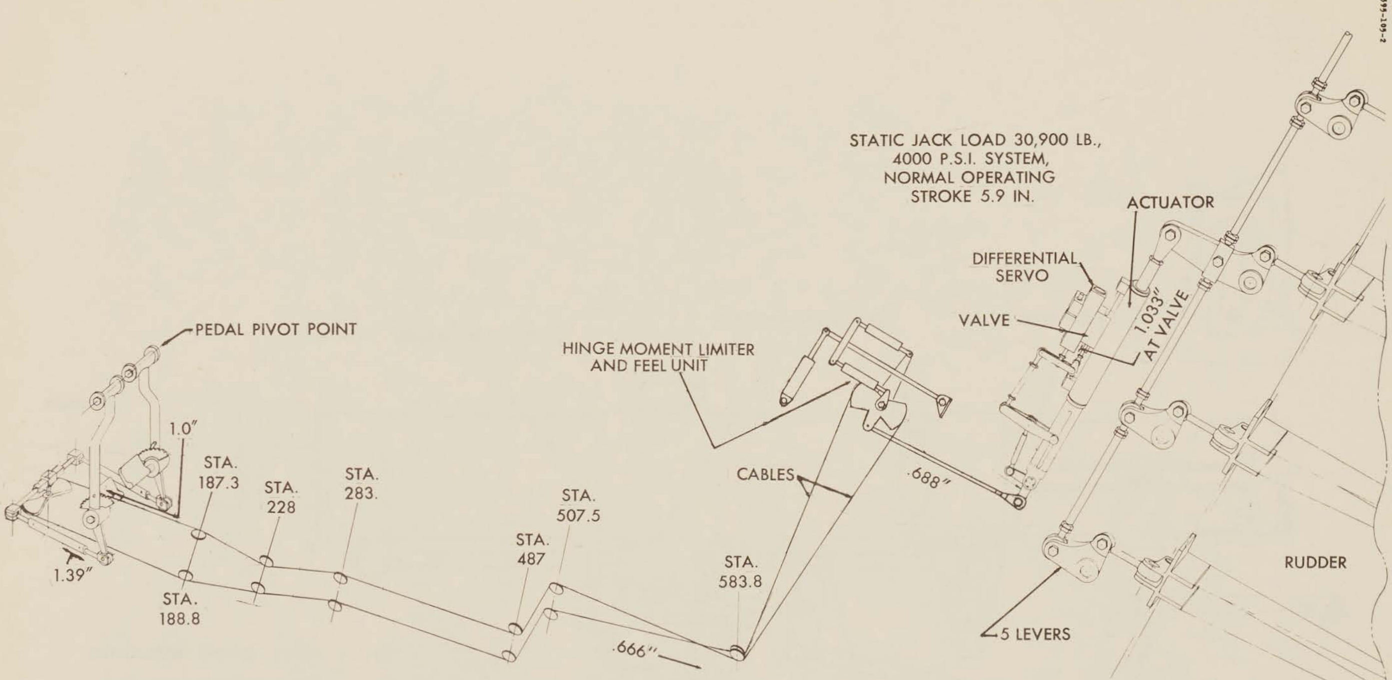


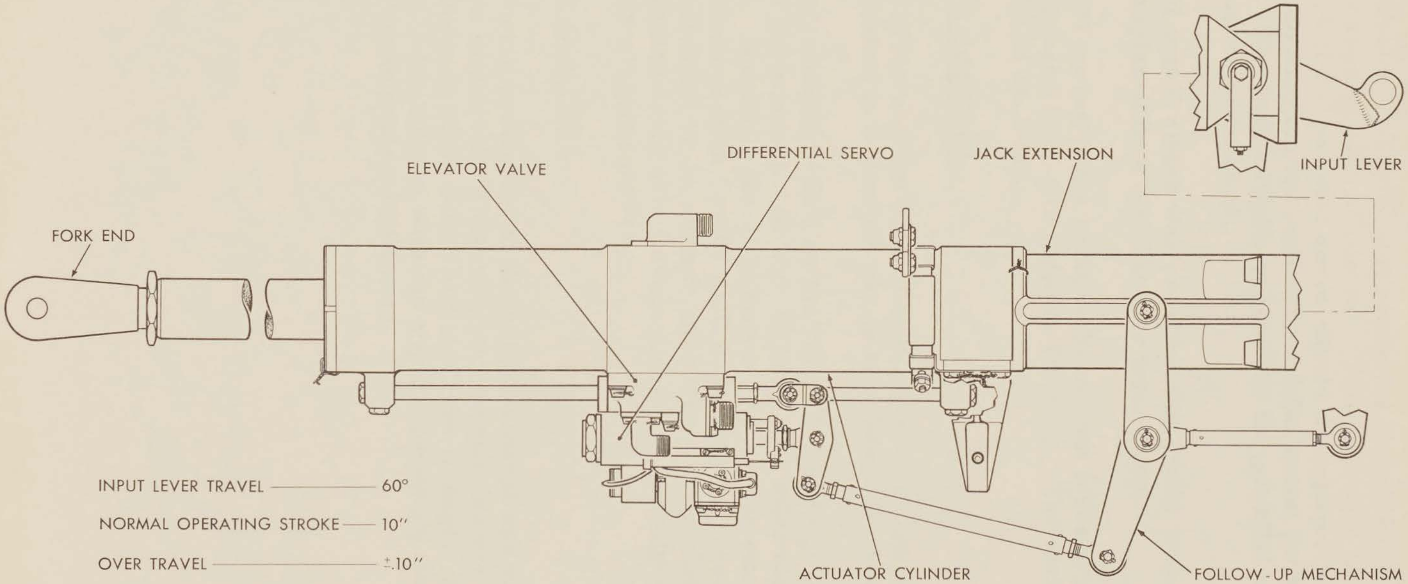
FIG.36 AILERON SYSTEM DATA



MOMENT OF INERTIA OF RUDDER AND LINKAGE 34,100 LB. IN.[2] (APPROX.) ABOUT C/L RUDDER HINGE

STIFFNESS OF RUDDER AND LINKAGE .16X10^{-6} RAD. /LB. IN. (APPROX.) ABOUT C/L RUDDER HINGE FROM C. G. OF RUDDER TO JACK ROD END.

FIG.37 RUDDER SYSTEM DATA



INPUT LEVER TRAVEL	60°
NORMAL OPERATING STROKE	10"
OVER TRAVEL	±.10"
TOTAL STROKE	10.2"
MAXIMUM STATIC OUTPUT	71,000 LBS. AT 4000 P.S.I. SYSTEM PRESSURE
VALVE MOVEMENT	±.125" MAX.
DIFFERENTIAL SERVO MOVEMENT	±.375"
LIMIT LOAD AT INPUT LEVER	615 LBS.

FIG. 38 ASSEMBLY OF ELEVATOR POWER CONTROL ACTUATOR, VALVE AND DIFFERENTIAL SERVO

of feel. The elevator feel trim unit performs an auto trim function during automatic and manual modes of control. This prevents the sudden "bump" force at the stick if the parallel servo should disengage, and is accomplished by having the auto function controlled by a differential pressure switch in the parallel servo. The load initially reacted by the parallel servo is then transferred to the auto trim function.

Formerly a ball detent clutch was provided integrally with the feel trim unit. This has been replaced by a friction disc type clutch.

The elevator feel spring is the positive break-out force type. The spring rate has been changed from 113 lb/in to 75 lb/in. The feel spring is adjustable between 0 and 24.5 lbs.

16.3 AILERON CONTROL

The sensitivity of the aileron control system is also changed from 2 1/2% to 5% and accomplished in a similar way to the change of elevator sensitivity.

A feature has been incorporated on the ARROW 2 to improve the high altitude performance of the aircraft. At high altitude both ailerons are automatically deflected up 4°. Normally the elevators would impose excessive trim drag in providing sufficient trim for the shift of centre of gravity. The aileron deflection is controlled by an altitude sensitive switch which controls an electrically actuated adjustable linkage and automatically varies the angle between the rear quadrant levers.

16.4 RUDDER CONTROL

It was found, during evaluation of the rudder control system, that some valve oscillation occurred. This very undesirable feature is apparently due to the lack of damping of the valve and feedback linkage. A damper is being arranged on the valve spindle with the object of proving the effectiveness of damping in preventing valve oscillation.

17.0 OXYGEN SYSTEM

The ARROW is equipped with normal and emergency oxygen systems consisting primarily of a liquid oxygen converter and pressure regulator, with emergency gaseous oxygen cylinders for each crew member (ref. 13).

17.1 OXYGEN SYSTEM DEVELOPMENT

Since its inception the oxygen system has undergone various changes, mainly of equipment, which incorporated a reduction of system working pressure from 300 psi to 70 psi. The major changes are as follows.

17.1.1 OXYGEN REGULATOR

The oxygen regulator is of the pressure demand type with an optimum working pressure of 70 psi. This unit was recently changed, at the request of the RCAF, to facilitate the use of small bore tubing, and a Firewel type F2400 regulator.

17.1.2 LIQUID OXYGEN CONVERTER

When the ARROW oxygen system design was in its early stages the only available liquid oxygen converters required charging from outside the aircraft, which was not compatible with ARROW turnaround time requirements. Consequently action was taken to secure a specially designed converter with a quickly exchangeable liquid oxygen container. Several problems arose with design and qualification of this device. Meanwhile, "off-the-shelf" equipment meeting ARROW requirements became available and it was decided to supersede the special design with the standard item which is now available.

17.1.3 QUANTITY INDICATION

A quantity indicating repeater gauge has been added to the obs/AI's cockpit, which was previously equipped with a low level warning light only. In the ARROW 1 aircraft, the pilot's indicator will be calibrated in litres and the repeater unit in percentage. The ARROW 2 system will have both instruments calibrated in percent. A "power-off" warning flag in each gauge indicates failure of its electrical supply.

17.1.4 EMERGENCY OXYGEN CYLINDERS

The introduction of Mk. C-5 Martin-Baker ejection seats necessitated repositioning the emergency oxygen cylinders, originally situated behind the seat, to the location beneath the seat pan. Restricted space in the new position required the employment of special "L" shaped oxygen cylinders of small diameter.

18.0 ARMAMENT SYSTEM

18.1 GENERAL

The armament of the ARROW 2 consists of four Sparrow 2D missiles housed in an interchangeable pack. In general there has been very little alteration from the original armament configuration.

18.2 ARMAMENT SYSTEM CHANGES

The following items are currently under review for design improvement or because of changes in requirements.

(1) Missile protection

(2) Rigidity effects on Antenna line of sight

(3) Dynamic study of the missile launcher

(4) Missile electrical circuitry

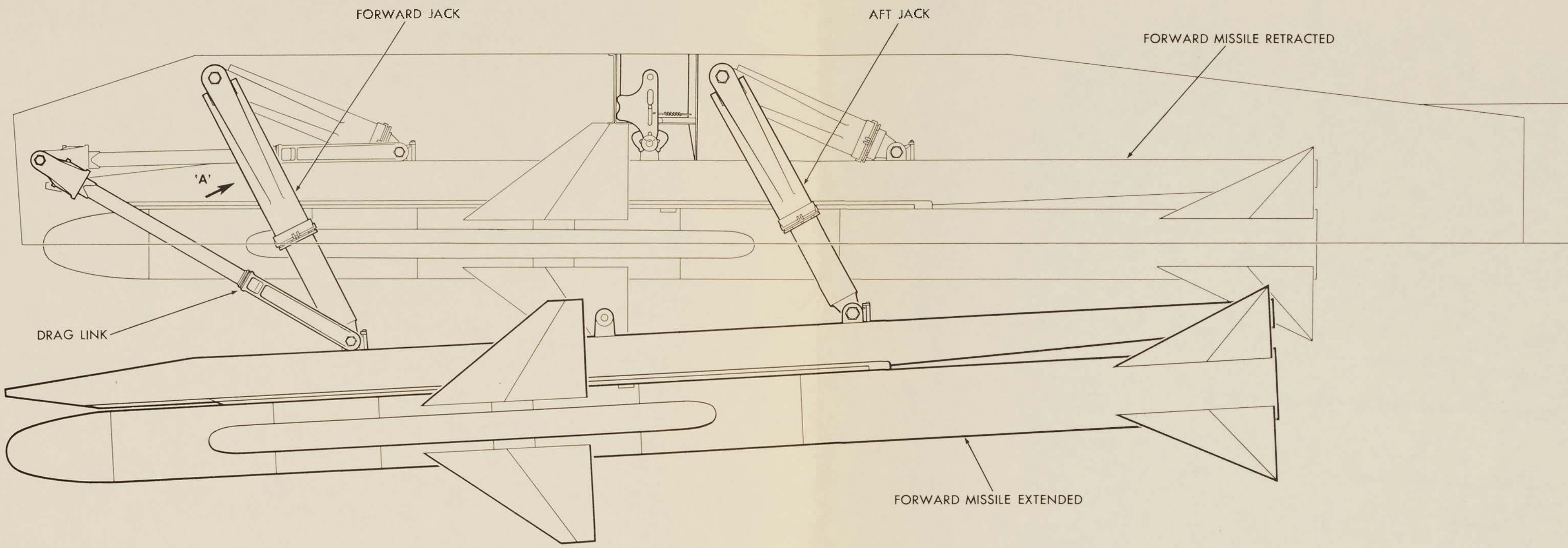
18.2.1 MISSILE PROTECTION

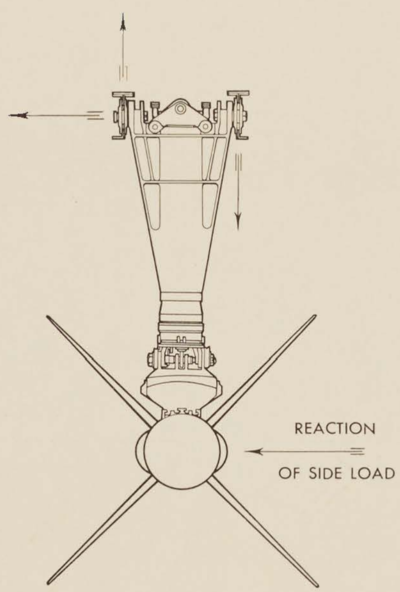
Due to the location of the missiles in their semi-submerged position in the armament pack, they are subject to a temperature rise from skin friction at high Mach numbers.

The missiles have been designed to withstand temperatures that may be encountered during prolonged periods of flight at Mach 1.5, whereas the aircraft has capabilities of Mach 2 for periods between 10 and 15 minutes, with a subsequent skin temperature rise to about 275°F. This temperature could be detrimental to the performance of the missile.

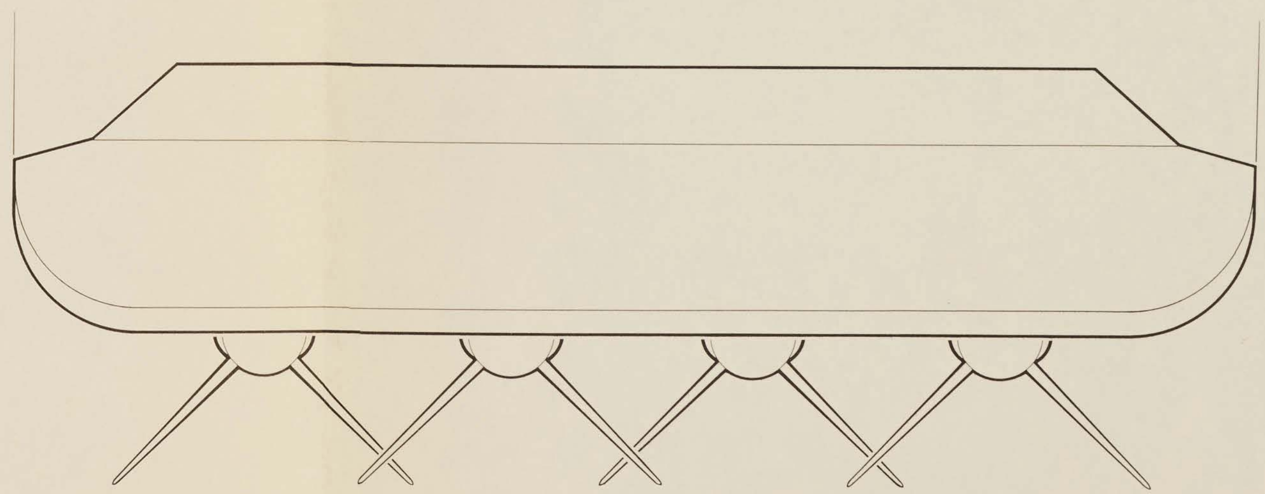
Canadair Aircraft Limited has agreed to conduct tests to obtain the temperature tolerance of the missile. These tests have been scheduled for January or February of 1958. In the meantime AVRO is investigating the possibility of protecting the missiles from the ambient environment, with either a low density or frangible cocoon around each missile. A further cooling aid for the missiles is available, by ducting the exhaust air from the cockpit air conditioning supply through an airspace between the lower surface of the missiles and their cocooning material. This would be in addition to circulating cooling air above the missiles in the weapon pack.

In addition to the temperature problems of the missile, which may be solved by the actual missile temperatures falling within the temperature tolerance, there will still remain the necessity of protection to the missile radome from stones, slush, etc., that can be thrown up by the aircraft





VIEW ON ARROW 'A'



FRONT VIEW OF PACK SHOWING SEMI-SUBMERGED STOWED MISSILES

FIG.39 MISSILE EXTENSION MECHANISM

nose wheel and also the effect of rain erosion during flight. If a cocoon is adopted for temperature control, it would also provide the necessary protection from physcial contact with the radome.

AVRO's studies on missile protection are still of a preliminary nature.

18.2.2 RIGIDITY EFFECTS ON ANTENNA LINE OF SIGHT

Although lines of sight of the aircraft radar antenna and the missile antenna can be parallel within acceptable limits on a static check, it is possible that they may be unacceptable during flight, due to lack of structural rigidity. This fact has been apparent for some time, but due to pressure of other work, a study of rigidity effects on antenna line of sight has not been started.

18.2.3 DYNAMIC STUDY OF THE MISSILE LAUNCHER

A study has been undertaken to analyze the effect that the increased length of launcher rail (60 inches total length) will have on the trajectory of the missile. The results of this study are not yet available.

18.2.4 ELECTRICAL CIRCUITRY

The electronic circuits can be divided into two parts.

(a) Actuating circuits

(b) Firing circuits

The actuating circuits consist of the missile lowering and door operation, while the firing circuits consist of missile sequence, firing, jettison and launcher retraction.

18.2.4.1

During the early stages of the design investigation for a Sparrow 2D missile installation on ARROW 2 aircraft, certain assumptions had to be made by AVRO, due to the lack of adequate data on the Sparrow missile system. One of these assumptions was that automatic missile jettison of any unfired missile would be accomplished just prior to aircraft breakaway. This was due to a 4g restriction on aircraft manoeuvre while carrying missiles.

From data now available at AVRO, it was learned that it is not practicable to apply a jettison signal just prior to a breakaway, hence a redesign of the missile firing circuitry has been undertaken. A stress investigation is being conducted to check the existing structure for a 7.33g manoeuvre with missiles in any position. This investigation will not be completed until December 1957.

18.2.4.2

It was originally thought that if the aft pair of missiles was fired before the forward pair, there was a possibility of the missiles fouling each other. Recent tunnel tests provide a sufficient weight of evidence to conclude that a restriction on the firing order is not necessary.

The firing order will be arranged such that whichever missiles are locked on to the target will be fired when the firing pulses from the intervalometer are fed through at half-second intervals. If two or more missiles are locked on at the instant of a firing pulse, they will be fired in ascending numerical order of missile arrangement.

18.3 MISSILE CHANGES

The following changes to the Sparrow 2D missile have been requested.

(a) Umbilical plug

(b) Missile arming time

18.3.1 THE UMBILICAL PLUG

Because of the structural requirements for a semi-submerged missile installation, it is not possible to insert the umbilical plug by hand, as is the current practice on missile installations involving the use of a pylon-suspended missile.

AVRO is proposing to use an adaptation of a proposal presented by Douglas Aircraft Company involving a self-engaging umbilical plug. The use of this scheme would involve a change to the umbilical plug receptacle in the missile body. The RCAF was notified of this on 25 February 1957 (AVRO letter Ref. 5746/03/J).

18.3.2 MISSILE ARMING TIME

Due to the unguided distance from the missile launcher to the aircraft nose, it is considered necessary from an aircraft safety aspect to increase the period of time required for functioning of the missile arming device. The RCAF was advised of this on 24 July 1957 (AVRO letter Ref. 9174/03/J).

Mk. 4 ejection seat, and the decision to adopt the Mk. C-5 seat led to installation alterations. Structural improvements to the seat caused a reduction in the space between the rear of the seat and the cockpit bulkhead. This necessitated repositioning the emergency oxygen cylinder to a position beneath the seat pan, which in turn required the introduction of a special "L" shaped cylinder of small diameter to utilize the available space. Minor structural alterations to bulkhead members were also carried out to allow the seat to be fitted. These alterations are now complete.

19.1.3 OXYGEN REGULATOR

A change of oxygen regulator instituted by the RCAF, and the relocations of the emergency oxygen bottle, made it necessary to slightly reposition this unit. This has now been done and the required Firewel type F2400 regulator is fitted beneath the seat pan.

19.1.4 QUICK DISCONNECT

To provide for the use of a small bore oxygen lead for the crewman's supply it was necessary to make a minor modification to the complete leads quick disconnect on the seat.

19.1.5 MOCK-UP CONFERENCE

The escape system was inspected by the RCAF at the ARROW 2 mock-up conference and as a result a change request for an additional canopy firing cartridge was raised. The requirement is being complied with and will provide additional internal explosive opening of the canopy, in case the normal cartridge fails to detonate.

19.1.6 HUMAN FACTORS ENGINEERING

A human engineering study to determine the nature of delays in escape due to communication between the pilot and observer is being carried out at AVRO using RCAF and Company aircrew as subjects. The results of the study will be discussed in the next issue of this report. A minor outcome of the study was the transfer of the pilot's bail-out warning switch from the fire and fuel panel to the aft side of the throttle box on the ARROW 2 where the switch (now of a recessed push button type) is isolated.

19.1.7 FUNCTIONING TESTS

Tests have been carried out using a representative canopy operation system with various applied hinge moment cases, covering the critical portions of the flight envelope. The operation of the canopy mechanism has been investigated and opening times established at various temperatures. Cartridge consistency tests will be carried out in the near future using the latest air loads data available.

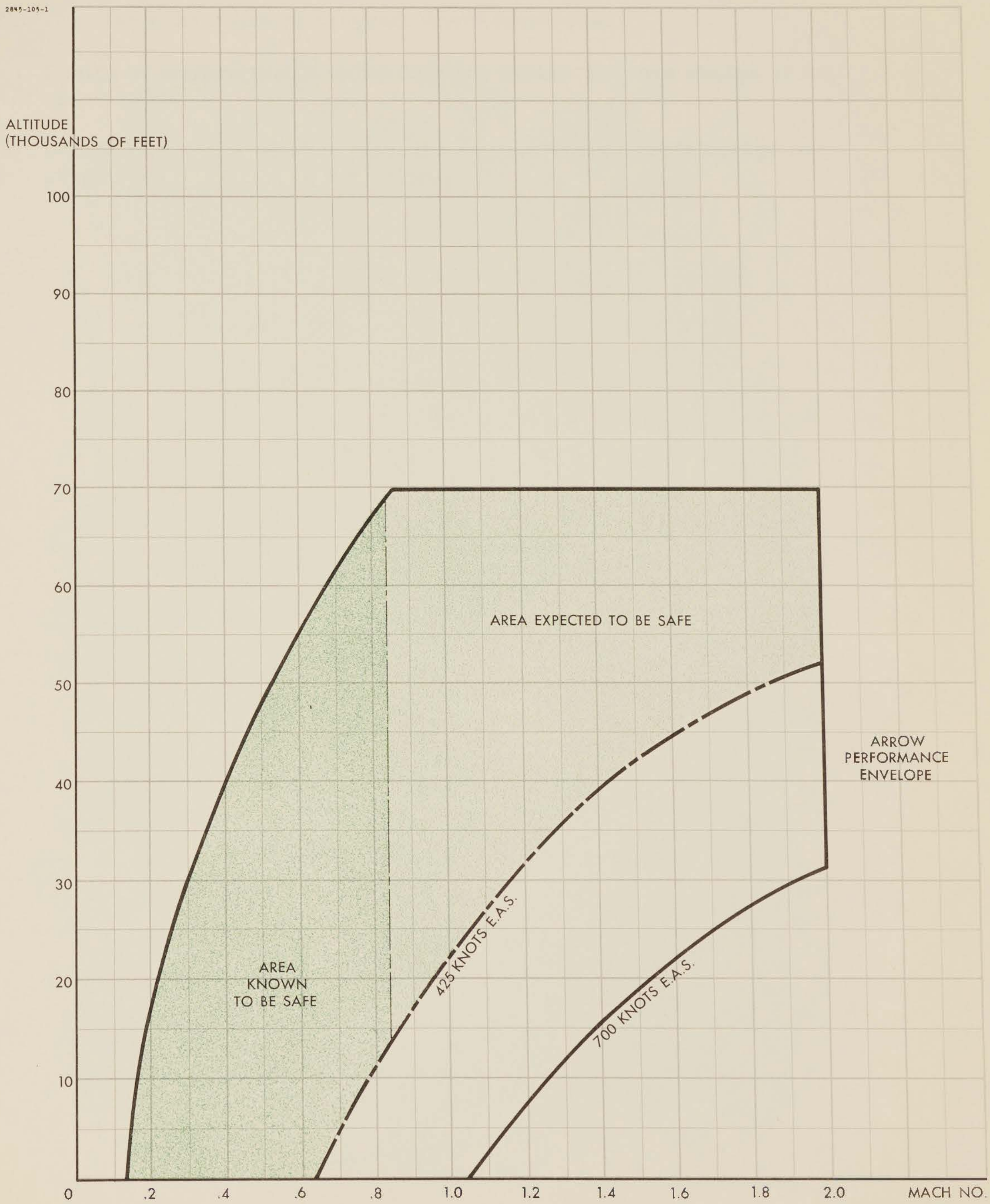


FIG.40 PERFORMANCE LIMITATIONS OF MARTIN-BAKER C5 SEAT

The scope of a development program would encompass:

(a) Design of improvements to the existing system to make escape at high speed safer.

(b) Design of a new system to meet the requirements of safe escape over all points within the flight envelope.

20.0 STRESSING

20.1 STRUCTURAL ANALYSIS

It ean be easily recognized that the structure is highly redundant, i.e. the loads carried by each element are dependent not only on the externally applied loads, but also on the deformation within the structure. This led to the neeessity of using the matrix method of solution paralleled with an independent standard engineering analysis using broad assumptions. It has been found that the two methods complement each other and provide a good check.

The stressing of a supersonic delta-wing aireraft requires an analysis which treats the wing as a flat plate and is therefore a two-dimensional problem. This method proceeds entirely with redundant stress distribution and accounts correctly for taper and sweep effects, Poisson's ratio, torsional warping and for the shear lag effect. The so-called lumping method, while omitting several valuable factors in the strain energy formula, becomes a very useful mathematical tool. When the large number of simultaneous equations are found, the problem is set up in matrix form and thus enables the greater portion of caleulations to be performed by the high speed digital computer.

The many applied concentrated loads are established by rational prediction of air loads and although structural flexibility makes it impossible to predict the precise effects on the aircraft in manoeuvers, a large number of stressing cases have been investigated.

The final proof will come through structural and flight testing.

20.2 DETERMINATION OF STRUCTURAL CRITERIA FOR STRUCTURAL TESTING OF COMPLETE AIRCRAFT

The preparation for the structural testing of the aircraft has consumed a great deal of time in establishing loading conditions, strain gauge locations and the arrangements for processing the resulting information.

The structural testing of the ARROW will be carried out for three reasons:

(a) To substantiate the structural integrity of the aircraft. - Only limit loads will be used so as not to distort the structure permanently. A large number of strain gauges will be required to prove that the yield stresses of the structural material are not exceeded at the limit load condition.

(b) To substantiate the theoretical analysis - The ARROW airframe, being a complex redundant structure, must be examined thoroughly to ensure that it is a fail-safe structure. The strain gauging throughout the structure will verify the theoretical analysis and strength of the structure.

(c) To comply with the requirements of MIL-S-5710 - The static test aircraft will be structurally complete; however, the following items may be omitted: radar nose, air brakes, armament pack, floating duct, engines, dorsal fairing, instruments, accessories, equipment and control circuits. It will be essential, however, that the flying controls linkage systems be included and that rigid struts be used in place of jacks.

To ensure that all possible sources of failure will be checked, it is essential that several loading cases be examined.

(a) Rolling pull-out

This case will be given top priority. Asymmetric loads will be applied to the wing and the rolling moment reacted by loads applied to the fin. The side loads on the fin will be reacted by loads distributed along the fuselage.

The aircraft, in this case, is under a normal acceleration factor of 4.89 g (limit) with no pitch condition. There will be an aerodynamic load of 36,500 lb (limit) on the fin which will act from the right side to the left side. An aerodynamic load will also act on the fuselage and in the same direction as the loading on the fin.

In addition, to simulate the in flight condition, the pilot's fuselage fuel tanks will be pressurized to 10.0 p.s.i. (limit) and the wing fuel tanks to 21.0 p.s.i. (limit).

(b) Symmetric case with pitch

All loads will be symmetric and the pitch effects will be simulated by variation in the loads applied. This case will give the critical case on the aft portion of the wing and fuselage.

The C.G. is at 31% M.A.C. and the normal acceleration factor is 7.204g. The pitching acceleration is 4.692 rad./sec^2 nose down. Rolling will not be taken into account. The fuselage fuel tanks and wing tanks will be pressurized to the limit values.

(c) Symmetric case - no pitch

This case gives the highest bending moment in the nose fuselage and the highest loads on the forward part of the wing. The C.G. location will be 28% M.A.C. and the normal acceleration factor produced is 7.33 g (limit). No pitch or roll forces will be introduced. The fuselage tanks and wing fuel tanks will be pressurized to limit values.

(d) Symmetric case - no pitch

This case is similar to case (c) except that the over-ride aerodynamic load gives the highest bending moment on the rear fuselage. The aircraft will be in the same balanced state as for case 3. The over-ride aerodynamic down load will be distributed on the rear fuselage which is balanced by a reduction in elevator angle.

The cockpits, the fuselage fuel tanks and wing tanks will again be pressurized to limit values.

20.3 ACCELERATED FATIGUE

Fatigue may be generally thought of as the result of stress cycles produced by gusts, cabin pressurization, manoeuvres, take-offs and landings. A large part of fatigue may, however, be attributed to noise, both jet engine and aerodynamic. Surveys have shown that the maximum noise intensity can occur between frequencies of 100 c.p.s. and 1,000 c.p.s. The susceptible aircraft panels will then vibrate at their natural frequencies with the fluctuating load being equivalent to the local noise level.

AVRO began its investigation of accelerated fatigue in October 1956. At that time test equipment was assembled to aid in the development of the tail-plane for the CF-100 Mk. 6 aircraft with afterburner. In May, 1957 on completion of this CF-100 test program, the test equipment was made available for the testing of ARROW structural panels.

The experimental method of investigating accelerated fatigue consists of breaking down the structure into several representative panels. Each panel is a built-up structure with construction representative of the location on the aircraft. Each panel is placed in the test chamber and subjected to the desired noise intensity. When the natural frequency of the panel is established, the test is continued until failure develops. With these results as reference, the structure can be modified where necessary to enable it to withstand the fluctuating load for a satisfactory time duration.

Before actual testing began, it was necessary to establish the sound levels which could be expected at various positions on the aircraft. These estimated sound levels were considered sufficiently accurate for testing and are as follows:

Engine nacelle area	164 decibels
Stinger area	164 decibels
Rudder area	140 decibels
Fuselage side area	140 and 149 decibels

Since May, 1957, when testing began, the following panel areas have been tested and improved.

Engine nacelle panels - The original material was to be titanium but was changed to stainless steel. Various modifications were made to subsequent panels, after the first of each type had been tested.

In general, the nacelle panels showed a weakness in the riveting of the inner skin, although the first panel tested produced failure of the skin along spot welds. One panel employed five different types of riveting combinations. These were blind or explosive rivets substituted for the original explosive rivets. A fatigue life of 12 hours at 164 db noise level was considered satisfactory.

Magnesium rudder skin panels - These panels were of ZE41-H26 magnesium alloy of .040 skin thickness. Failures developed in the skin along the rivet lines. Design was improved by employing double ribs with skin backing plates, and satisfactory fatigue life was achieved.

Fuselage side panels - The material was ZE41-H26 magnesium alloy with a skin thickness of .040 and later .051. Four panels were tested at 149 db and two at 140 db. Several cleats, attaching the longeron to the formers, failed and their thickness was subsequently increased from .032 to .040.

Stinger panels - These were of stainless steel construction. The main source of trouble appeared to be the single skin with its large flat side having the noise impinge on it.

At present, the investigation is being continued on stainless steel stinger panels and fuselage side skin panels.

The causes of failure with accelerated fatigue are many; however, with the panels tested, the main sources of failure are skin cracks along spot welds, stress concentrations and structure discontinuity.

In order that the actual fatigue life of a structure can be predicted to some degree of accuracy, theoretical work is being given consideration. The testing of representative panels will then serve as a check for the basis of calculations.

20.4 HEAT

The problem of aerodynamic heating during supersonic flight is becoming increasingly great as higher performance aircraft are developed. This gives rise to the necessity of more accurate evaluation of the effects on the aircraft structure. It has been estimated that the skin temperature of the ARROW due to boundary layer heating will be about 250°F while flying at M - 2.0. Under transient heating conditions, sudden increases in temperature in some parts of the structure will cause differences in temperature between adjoining components which leads to warping, development of thermal stresses and reduction of torsional stiffness. The governing factors of this temperature distribution are many and their determination by theory or experiment is not a simple problem.

Further development of the ARROW will lead to increased supersonic speed and consequently higher skin temperatures. To combat this higher temperture, it will be necessary that the aircraft have some type of insulating coating. The insulation will then lower the heat transferred to the structure to the extent of simplifying the problem considerably.

At present, tests are being carried out to determine the effects of combined loading and transient heating on portions of the main wing torque box with its integral fuel cell areas. The specimens are subjected to loads in bending and shear while the skin is heated to a temperature of 250°F. The integral fuel cells are filled to various fuel levels, and temperatures, pressures and strains, at various points throughout the test specimen, are recorded as testing continues.

In the attempt to reach agreement with theoretical work, tests are also being carried out on fuselage sections to determine the temperature distribution and stresses developed. Other tests include the determination of thermal resistances in typical aircraft joints. Fairly accurate correlation has been established regarding temperature distribution by theoretical and experimental means.

Tests are progressing satisfactorily on the determination of the heat transfer coefficient between a typical wing skin and an integral fuel cell containing fuel. The tests involve heating the skin at a given rate while checking the heat transfer through the skin to the contained fuel.

For reasons of safety, it is desirable to use some liquid which is not as flammable as the standard type of fuel. This liquid must possess properties very close to those of the fuel such as viscosity, specific heat and specific gravity. In addition, a high boiling point is essential. Several liquids have been investigated; however, an adequate substitute has not yet been found.

The test equipment consists of an aluminum tank with straight sides. The bottom of the tank represents an aircraft skin. Thermocouples have been carefully placed on the inner and outer surfaces of the skin to determine the temperature gradient through the skin. Temperatures are also taken in the fuel near the skin and at various levels through the fuel.

An insulated cylinder is placed in the centre of the box. This cylinder then supports a column of fuel and isolates it from the uneven fuel temperature existing in the corners of the box.

The completion of this series of tests will aid greatly in the thermal analysis of the inner wing structure which at present would impose a difficult task due to the absence of an accurate heat transfer coefficient.

21.0 ARROW 2 MOCK-UP

21.1 GENERAL

As the result of ARROW 2 the mock-up conference held at Avro Aircraft Limited on 18 - 24 September 1957 by the RCAF, 266 change requests were presented. Of these 5 were withdrawn by the RCAF. A further 9 were not demonstrated at the mock-up and therefore were not evaluated.

21.2 CHANGE REQUEST CATEGORIES

CATEGORY I

Inspection Change To be accomplished prior to delivery of any aircraft. This category will include all items on which the company have been given direction through authorized channels.

CATEGORY II

Mandatory Change To be accomplished prior to delivery of any aircraft. This category will include all changes considered essential for the operation and safety of the aircraft and crew.

CATEGORY III

Changes of a nature requiring a study by the contractor or the RCAF.

CATEGORY IV

Changes not acceptable.

21.3 TABULATION OF CHANGE REQUESTS

SUBJECT	NUMBER OF CHANGE REQUESTS PER CATEGORY					
	Category # 1	Category # 2	Category # 3	Category # 4	Not Eval.	With-drawn
Cockpit	33	-	23	6	1	1
Structure	26	-	20	5	-	-
Engine Installation	16	-	1	1	3	-
Electrical	16	-	5	1	-	-
Air Conditioning	4	-	2	-	1	-
Low Pressure Pneumatics	1	-	-	-	-	-
Fire Extinguisher System	3	-	-	-	-	-
De-Icing	2	-	-	-	-	-
Fuel System	9	-	1	1	-	-
Hydraulics	8	-	6	1	-	-
Oxygen System	4	-	1	-	2	-
Instruments	3	-	4	-	-	-
ASTRA I	20	-	12	2	1	3
Armament	5	-	7	3	1	1
SUB-TOTALS	150	0	82	20	9	5

200-CA-2

FIG. 41 STRUCTURE - ARROW 1

200-CA-2

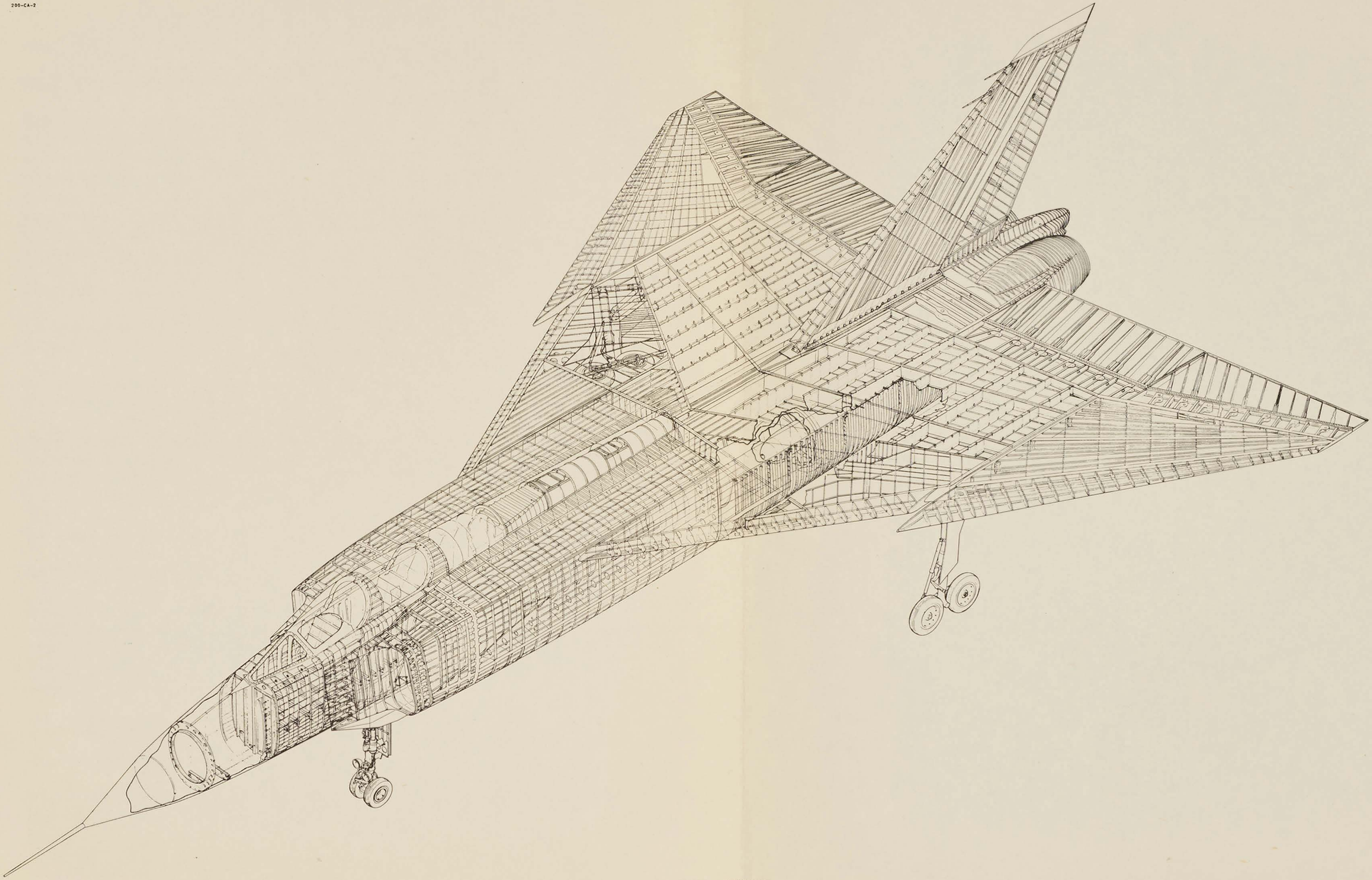


FIG. 41 STRUCTURE - ARROW 1

NOTE: THIS DRAWING IS INCOMPLETE

22.0 COMPONENT DESIGN

22.1 WING

22.1.1 WING DESIGN

The ARROW 2 inner wing is essentially the same as that of the ARROW 1. The J75 engine mounts are superseded by Iroquois engine mounts which necessitate considerable changes to the inner wing structure. Changes to the main torque box and tanks 3 and 4 were caused by the modification of the fuel system involving new bracketry, piping and structural modifications. The new engine mounts also involved changes to the centre trailing edge and the wing centre box. The fuselage vertical struts were modified and the joint at station 742 (rear fuselage joint) required changes. There is also a re-design of the aileron control linkage in the control box because of fouling condition on the ARROW 1. The above work is approximately 90% completed.

On account of the enlarging of the fuselage to accommodate the Iroquois engines it was necessary to modify the elevator to give correct clearance.

A new landing gear pivot door is necessitated by a revision to the landing gear shortening device.

Some structural improvements will be incorporated on the ARROW 2. These improvements, which are presently in work, will include changes to the joint of rib 24 to the front spar and the joint of the landing gear jack pick-up fitting to rib 10 and the main spar.

No major changes are expected on the outer wing and aileron.

22.1.2 WING STRESS

The stress approving of ARROW 2 new drawings is nearly complete. Present work involves the completion of ARROW 1 stress reports and a detailed coverage of the effects of the modified fuel system on ARROW 1 and 2. The stress group is also engaged in preparing for the static testing of the complete aircraft.

22.2 FUSELAGE

22.2.1 FRONT FUSELAGE DESIGN

The basic change from the ARROW 1 concerns the intake geometry due to the installation of the Iroquois engines. Although the Intake lines are modified, the basic structure concept remain as for ARROW 1. The revised air conditioning system and the installation of ASTRA I have also led to design changes. To date, 75% of the production drawings have been issued for ARROW 2. The remaining drawings have been completed.

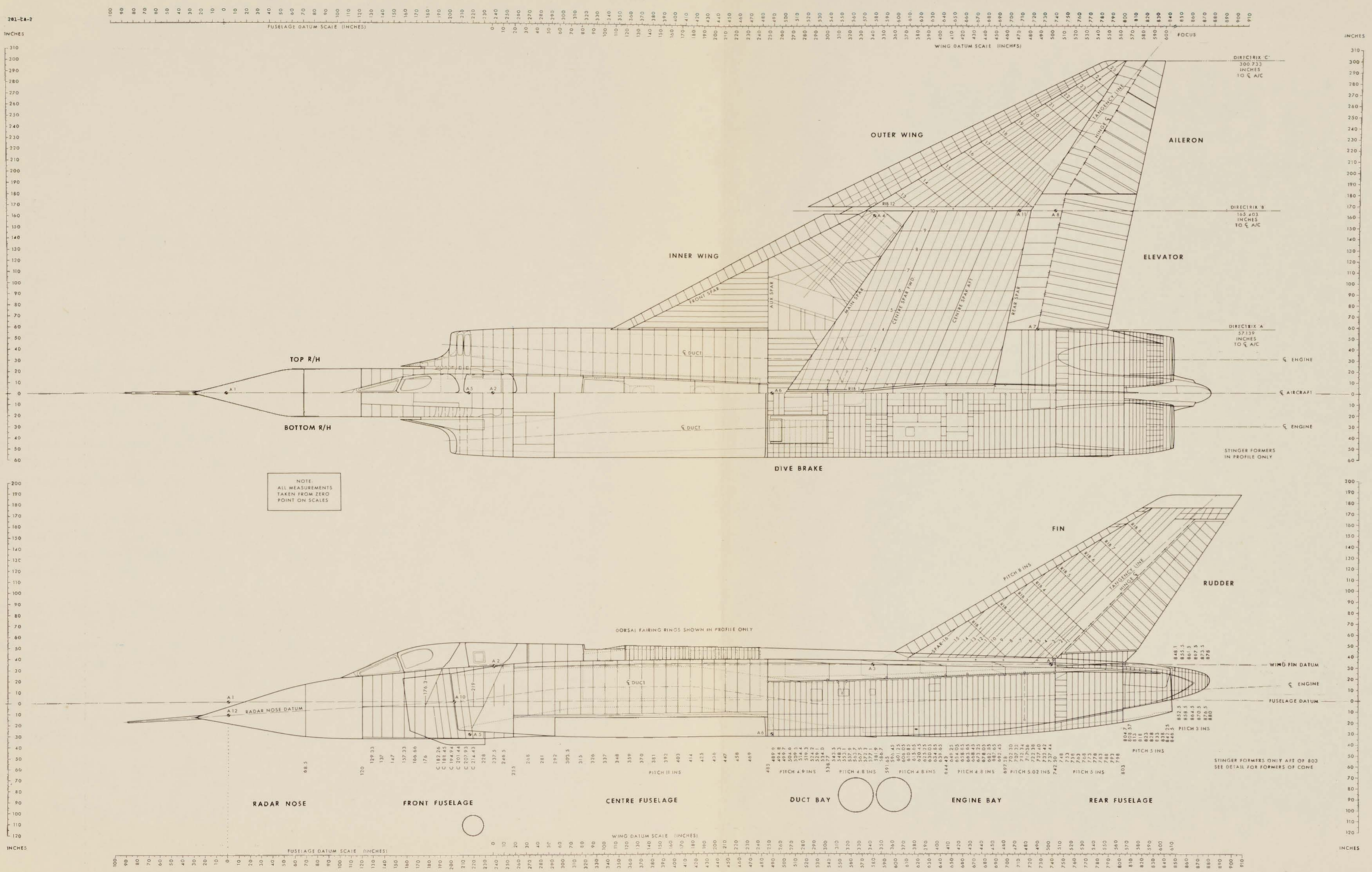


FIG. 42 ARROW STATION AND DATUM LINES

Some minor changes were requested by Production.

22.2.2 FRONT FUSELAGE STRESS

Stressing has been completed on the ARROW 2 obs/AI's canopy with the enlarged window. The strain energy analysis of the ramp and fuselage structure is under way for the assymetric ramp loading cases.

22.2.3 CENTRE FUSELAGE DESIGN

The basic changes from ARROW 1 include provisions for access to the air conditioning ducts, attachment for the external fuel tank, and structural changes to suit the new air conditioning system and the installation of ASTRA I. The increased fuel tank pressure has also caused structural redesign.

The design of the basic structure is complete and issued. At present, scheming is in progress on ASTRA I electronic equipment.

The structural design of the armament pack has been completed and production drawings issued although several proposed changes have resulted from the mock-up evaluation conference. Present work involves the installation of air conditioning piping in the pack. A minor change has been effected on the drag link doors. Formerly a single door was used but it is now replaced by two hinge doors to allow for the extension of the forward lowering jack as well as the movement of the drag link as before.

The launchers are progressing satisfactorily. Approximately 25% of the production drawings have been issued and stressing has almost been completed. The first three sets of launchers will be manufactured by Experimental rather than on a production basis.

22.2.4 CENTRE FUSELAGE STRESS - ARROW 1

The leak rate tests on the fuel tanks No. 1 and 2 of ARROW 1 have been successfully completed at 6.0 p.s.i.g.

Tests carried out on the Marman joint of the air conditioning ducts have been successfully completed for both static and cycling load conditions. The loading cases were:

(a) 90 p.s.i. at 920°F.

(b) 350 p.s.i. at 645°F.

22.2.5 CENTRE FUSELAGE STRESS - ARROW 2

All ARROW 2 drawings on the centre fuselage, except for the heat exchanger outlet ducts and the electronic bay structure, have been stress approved. The armament structure and the missile lowering mechanism have been stress approved except for the pack seals.

Matrix analyses will be carried out for the pack structure and for the lowering mechanism complete. From these analyses more accurate stress distributions and deflections can be obtained.

At present an investigation is being carried out on the structural possibility of having the missiles in the fully extended position under full flight envelope conditions.

22.2.6 DUCT BAY AND ENGINE BAY DESIGN

The basic changes from the ARROW 1 are caused by the installation of the Iroquois engine and the provision for an external fuel tank. The fuel system and revised air conditioning system have also meant several changes.

The heat exchanger has presented an assembly problem and is being investigated. On the Iroquois installation, because of the rate of deflection due to the change in position of the engine mounts, an articulating duct had to be provided between the front end of the engine and the floating duct. At present these schemes are being stress approved and the production drawings are expected to be completed and issued by the end of January, 1958. The duct bay structure, excluding the heat exchanger, articulating duct and aft portion of the floating duct have been completed and issued.

The problem of access to the engine-to-intake assembly was provided by increasing the length of the existing access door immediately below the assembly.

22.2.7 DUCT BAY AND ENGINE BAY STRESS

A strength study has been carried out to access the changes required to increase the speed restrictions on the dive brakes. A static test was also performed to assist in the strength check. Approximately 40 drawings will be affected; however, the changes are minor and no difficulty is anticipated.

A program has been started to study the effect of the engine doors on the load distribution in the bottom skin. The study will be more comprehensive than that done in the past to enable the refinement of stressing of the bottom skin and inner longeron.

22.2.8 REAR FUSELAGE DESIGN - ARROW 1

A redesigned tailcone was proposed to provide a divergent shape of ejector to increase the engine performance. It was later found that afterburner details did not agree with information previously supplied on which the original design was based. This necessitated the redesign of the tailcone. An assignment was then raised to overcome this problem and the divergent shape of ejector included in the redesign.

Revised tailcones and stingers are expected to be available for installation in March 1958. Flight testing to that point can be conducted with existing nozzles.

22.2.9 REAR FUSELAGE DESIGN - ARROW 2

The rear fuselage is divided into two main sections: a fixed portion and a removable section. The work on the fixed portion is complete and issued and the removable portion is expected to be completed by March 1958.

22.2.10 REAR FUSELAGE STRESS

ARROW 2 production drawings are progressing satisfactorily. Nearly 2,000 drawings have been processed and approved since March, 1957.

The fatigue of the structure in a noise field is still under investigation. Tests so far have shown that the tailcones on the ARROW 1 are satisfactory but further work will have to be carried out on the stinger. A more comprehensive program is being planned for the testing of the tailcone and stinger of the ARROW 2. These will be constructed of N155 material, an iron base alloy. The ARROW 2 tailcone is quite similar to that of the ARROW 1 redesign and although temperatures are higher, the survey of materials showed that the N155 material will be satisfactory.

22.3 <u>FIN AND RUDDER</u>

The fin and rudder for ARROW 2 will be the same as for ARROW 1 except for minor changes. Structural revisions to the fin will accommodate the redesigned air data system.

The geometry of the rudder hinge moment limitation system has been altered on ARROW 1 and 2 in order to improve the undersirable characteristics of the original system. It involves modifications to skins and the hydraulic actuator access door and the relocation of the modified link assembly. This work was completed in May, 1957, except for the access door for which schemes have been completed and stress approved.

22.4 LANDING GEAR

The sub-contractor developed a shortening and twisting device which saves weight and reduces the complications of the mechanism. The sub-contractor is presently engaged in eliminating faults of the design, and these changes must be incorporated before the first taxi tests are carried out.

22.5 RADAR NOSE

22.5.1 RADAR NOSE DESIGN - ARROW 2

The ARROW 1 radar nose is being completely redesigned for the ARROW 2 to accommodate the ASTRA I system.

The radome for ARROW 2 is similar in design to the ARROW 1 radome but now has to meet electrical requirements. Detailed electrical and structural design of the radome was sub-contracted to a company specializing in that type of work.

Approximately 80% of the production drawings for the radar nose have been issued and the remaining drawings have been completed. It is expected that several minor changes will be required when the ASTRA I system is installed.

22.5.2 RADAR NOSE STRESS - ARROW 2

All ARROW 2 drawings have been stress approved, except for G.A. drawings, of which 75% have been stress approved.

22.6 LONG RANGE TANK

The ARROW 2 will have provisions for carrying a jettisonable, external fuel tank which will be used for long range ferry missions.

With the objective of using a free drop release system, a design has been established on the basis of wind tunnel test information.

The design of the external fuel tank has been temporarily suspended due to higher priority work.

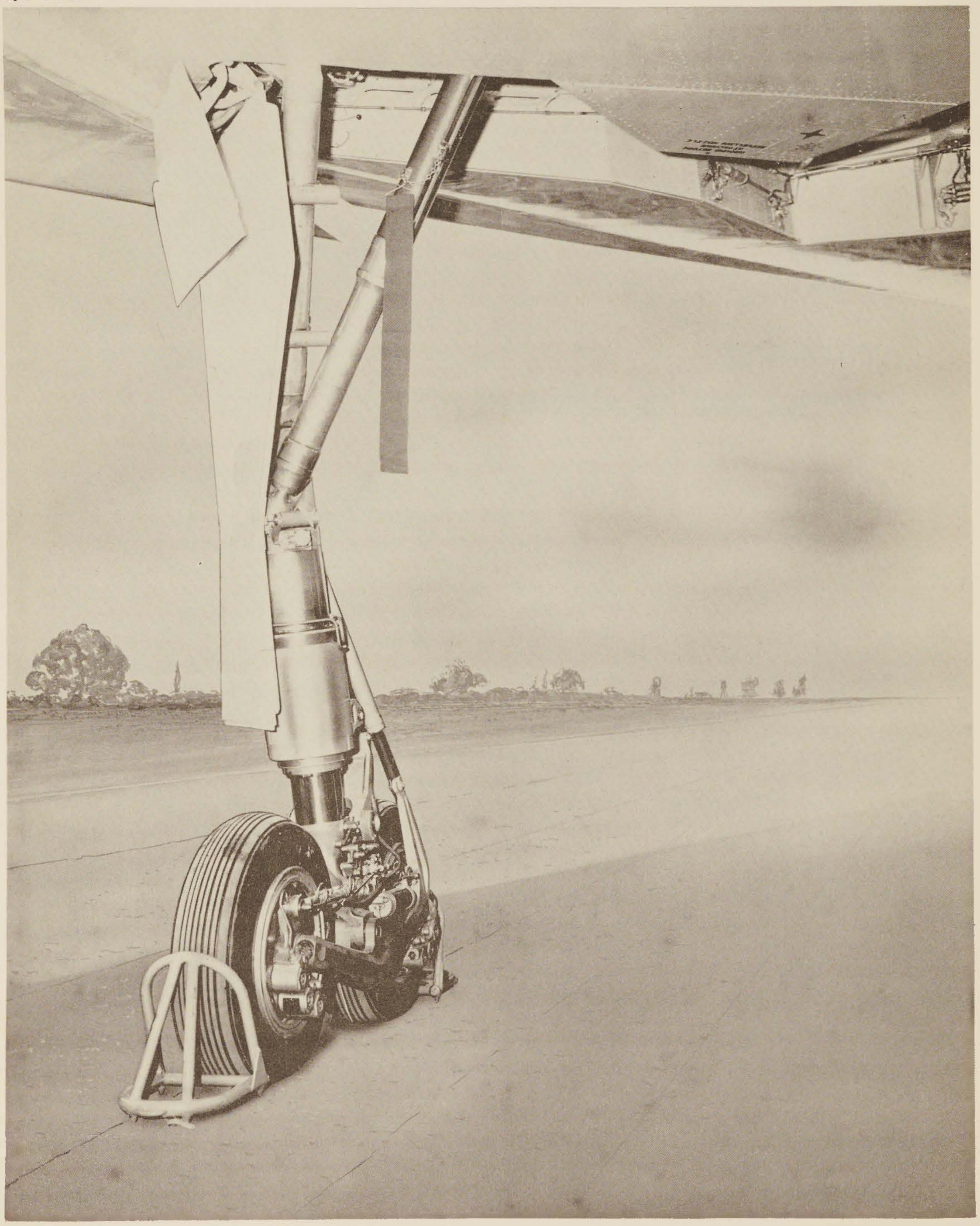


FIG. 43 LANDING GEAR

23.0 MAINTENANCE AND RELIABILITY

AVRO'S maintenance and reliability engineering section is divided into three groups: Qualification Engineering, Maintenance Engineering and Reliability Engineering.

23.1 MAINTENANCE ENGINEERING

The function of the maintenance engineering group is to analyze all engineering drawings and advise the design engineers on the various maintenance characteristics of their designs. At this stage vital information is recorded in the form of Maintenance Data Records. These records are fundamental in that they form the basis for Engineering Orders compiled by the Sales and Service Division for the RCAF, Preliminary Inspection Schedules, and Maintenance Instructions. Eventually the Maintenance Data Records will cover every maintenance aspect of every component and system of the aircraft.

The maintenance engineering group is also responsible for the analysis and satisfactory maintenance capability of the Ground Support Equipment. Each item of equipment will be analyzed as and when it becomes available.

Since its formation this group has issued approximately 1000 Maintenance Data Records, of which 250 covering the ARROW 2 have been issued since April 1957. Between April and September twenty-two Maintenance Instructions were issued. In June a Preliminary Inspection Schedule covering the turn-around, primary and 25-hour inspection requirements was issued.

The group has initiated 265 design maintenance changes, 121 of which have been incorporated into the basic design. 92 are under consideration for incorporation.

23.1.1 FLIGHT DEVELOPMENT PROGRAM

The development or flight testing phase constitutes a logical follow-through of the design phase for the maintenance engineering group. It is in this phase that the theories on the maintenance of the new weapons system, developed in the design stage, can be tried and proved in actual practice for the first time. Organization and procedure arrangements have been made to enable maintenance analysts to take their place alongside flight test engineers and flight servicing crews to ensure that:

(a) Maintenance characteristics are adequate.

(b) Maintenance records data are accurate or modified as required.

(c) Immediate corrective action is taken within the Engineering division.

In this field very close liaison with the RCAF maintenance appraisal team is maintained.

23.1.2 ACCESSIBILITY

One of the most important aspects of the maintenance engineering group is to ensure maximum accessibility to all equipment requiring servicing, such as engines, armament and electronics etc.

The aircraft was examined from all aspects of accessibility to determine the removal and installation of equipment for the various systems, culminating in the RCAF Evaluation at the ARROW 2 Mock-up Conference in September 1957. Report 72/MAINT 00/1 covering the Maintenance Accessibility Survey was issued in September 1957.

23.1.3 MAINTENANCE AND OVERHAUL OF THE ASTRA 1 SYSTEM

AVRO has prepared a report on the Maintenance and Overhaul of the ASTRA 1 System. This report covered the broad aspects of the subject and was presented at the Mock-up Evaluation Conference in September 1957 (Ref 6). Avro Aircraft Limited and RCA are currently preparing a joint report entitled "Preliminary Report on Maintenance and Overhaul of the ASTRA 1 Electronics System". Although considerable thought has been given to the maintenance procedures and maintenance test equipment required to support the ASTRA 1 system, much of the detail design has not been finalized.

The ASTRA I system is presently in the initial stages of development and numerous and significant changes will occur which will affect maintenance procedures before the equipment is delivered to squadron service. It is intended to amend the above report as frequently as possible in order to maintain it as an effective document.

23.2 PERSONNEL REQUIREMENTS DATA

The group is currently engaged in preparing a proposal for a method of providing a Personnel Requirements Data for submission to the RCAF which will include the following:

(a) An early systematic identification of skills and knowledge required to maintain the ARROW weapon system.

(b) A prediction of the number and type of personnel required to maintain the ARROW weapon system.

(c) A recommendation for the type and amount of maintenance required for the ARROW weapon system.

(d) A recommendation for the organization of maintenance personnel.

(e) A recommendation for a training program for maintenance personnel.

The preparation of the Personnel Requirements Data report will be a continuous process commencing at the design stage. As the development program progresses and more knowledge on the systems become available, more, and more accurate Personnel Requirements Data will be accumulated and periodically issued to the RCAF.

The Company will act in an advisory capacity only in passing on to the RCAF the "know-how" acquired during the design and development stage. The recommendations will assist the RCAF in efficiently planning the maintenance and operation of the complete ARROW weapons system. To date the task descriptions of three systems of the aircraft have been completed, one of which has been forwarded to the RCAF for its comments. This will form the basis for discussion for final approval of the Personnel Requirements Data procedure to be held in Ottawa in mid-October.

23.3 RELIABILITY AND QUALIFICATION TESTING

It is generally recognized that the problem of equipment reliability in modern aerial weapon systems requires a co-ordinated reliability program which must be integrated into the design and development of the aircraft. With this in view, AVRO organized such a program at the commencement of the ARROW project.

23.3.1 EXACTING EQUIPMENT SPECIFICATION

Little existing equipment could be used to meet the servicing requirements of the ARROW. Numerous items of mechanical, hydraulic and electronic equipment are required to operate in environments which are completely foreign to items of equipment that could be bought "off the shelf". The first step in the reliability program was the provision of specifications to outline the requirements which the equipment must meet.

Approximately 370 specifications have been issued, and approximately 110 vendors have benefitted from the amount of work involved in supplying the required airborne equipment.

23.3.2 FREQUENT TECHNICAL LIAISON

Experience has proved that it is desirable to maintain a close liaison with the vendor's development engineers, from the time a vendor has been given a contract for the development of a piece of equipment. This is necessary to keep abreast of the progress of the design and development of the item until it is satisfactorily completed.

23.3.3 RIGID CONTROL OF QUALIFICATION TESTING

AVRO'S equipment specifications call for a variety of stringent tests to which the equipment must be subjected before it is approved for production. These are called qualification tests. To date, of the 916 items of equipment to be qualified 353 have been approved and 206 have received a limited flight approval, leaving a balance of 357 items outstanding.

23.3.4 FUNCTIONAL TESTING OF A COMPLETE SYSTEM

After due consideration it was decided to build functional test rigs of every major sub-system in the aircraft. This was done to check the performance of the equipment systems under simulated flight conditions. Under these conditions it becomes possible to locate and overcome any major troubles that may occur much earlier than leaving to be found in the flight testing stage.

24.0 GROUND SUPPORT EQUIPMENT

The engineering of the ground support equipment required for the first flight of the ARROW 1 is virtually complete. AVRO-manufactured equipment and procurement of bought-out items are proceeding satisfactorily. More than half of the essential equipment required for the first flight has been delivered.

The outstanding items of equipment from an engineering standpoint are the liquid oxygen converter trailer, fuselage maintenance stand and the probe cover. Work on these items is proceeding with the exception of the probe cover, which is being held in abeyance until the design of the probe has been finalized. The engine exhausts cover is being revised.

This equipment, which is required during the development program will serve two purposes. It will make it possible for AVRO to operate the aircraft and enable AVRO and the RCAF to assess the suitability of the pre-production equipment prior to squadron use.

A large proportion of the ground support equipment for the ARROW 1 aircraft will also be used with the ARROW 2. The following items of equipment specifically required for the ARROW 2 have still to be designed.

- Engine starting truck
- Power and air conditioning truck
- Multiple missile trailer
- Armament pack test stand
- Armament pack test control console
- Iroquois engine change stand
- Iroquois maintenance trailer
- Engine intake cover
- Engine exhausts cover
- Air conditioning inlet cover
- Air conditioning outlet cover
- Probe cover
- Radome cover

Materiel Procurement to consider ways and means of overcoming the problems presented by the DDP cancellation of contractor assistance items. A meeting was held at Air Materiel Command, Ottawa, on 29 August 1957, to discuss ground handling and test equipment. The problem was referred to Air Force Head Quarters.

24.1.2 AIR CONDITIONING AND POWER UNIT

RCA has recommended that the temperature of the air entering the electronic equipment should not exceed 70°F. Since the temperature of this air is raised by approximately 15°F in the ASTRA oil-to-air heat exchanger, it is necessary to supply the air to the aircraft at 55°F. Also the ground air conditioning system must be capable of supplying this air with no free moisture. The possibility of attaining this cooling capacity (150 lb of air per minute minimum) with the current air cycle refrigeration equipment is marginal providing the present ambient requirements (of -20°F to 110°F for the ARROW 1 and -65°F to 120°F for the ARROW 2) are met. Since RCA maintains that the lower inlet temperature is necessary to ensure the desired reliability of the electronic equipment, the decision was made that RCA's requirement be met regardless of the air conditioner design implications. This was subsequently confirmed by the 35th ARROW Development Co-ordinating Committee meeting on 25 July 1957.

AVRO has investigated the whole question of environmental requirements which affect the starter unit as well as the air conditioner and power unit. A visit was made by AVRO representatives to Wright Air Development Centre, Dayton, Ohio, for the purpose of obtaining USAF opinions regarding equipment for the ARROW 2 air conditioner power unit.

A report is in preparation which will make recommendations for meeting these requirements. On completion, this report will be submitted to the RCAF for comments and approval.

Auxiliary external fuel tank trailer

Fuel tank test intercooler unit

Aircraft component slings for:

Station 255

Rudder control box

Rudder

Elevator control box

Elevator

Aileron control box

Aileron

Air conditioning pack

Main landing gear installation stand

Nose landing gear installation stand

Universal stand for removal of aileron and elevator control boxes

Canopy locking actuator

Rigging boards

Radar maintenance stands

24.1 GROUND SUPPORT EQUIPMENT PROBLEMS

24.1.1 PROVISION OF GOVERNMENT FURNISHED TEST EQUIPMENT

In June 1957 AVRO'S request for a quantity of government-furnished test equipment, urgently required as "contractor's assistance items" for ground servicing the ARROW aircraft, was cancelled by the Department of Defense Production (Letter reference 266-35B Ottawa June 1957). This was taken up for urgent consideration at the 35th ARROW Development Co-ordinating Committee Meeting, 25 July 1957. It was recommended that the RCAF convene a meeting with AVRO and the Director of

25.0 AIR BASE FACILITIES

Under the authority of an RCAF letter ref. AMC 1038 CN-100 (ACT-2-1) dated September 27, 1955, AVRO is currently studying the requirements for air base facilities with the view to making recommendations to the RCAF.

A report on the ARROW 2 Readiness Facility Log 105/9 was prepared and issued by AVRO in May 1957.

The purpose of the report was twofold:

(a) To outline the requirements for facilities and equipment which will be needed in order to maintain the ARROW 2 aircraft at various states of readiness.

(b) To make recommendations concerning the facilities and to specify equipment.

The following conclusions were drawn and recommendations made:

(a) That the standard RCAF readiness hangars will be suitable for use with the ARROW 2 aircraft, but some alterations will be required to accommodate ground support equipment and services to the aircraft.

(b) It is noted with the present siting arrangements of the standard readiness hangars at least 20 seconds will be required to taxi from the readiness hangars to the runway with the ARROW 2 aircraft.

(c) A taxi strip is recommended from the maintenance area to the rear of the readiness hangars.

(d) The present layout for readiness hangars and crew rooms is considered satisfactory for use with the ARROW 2.

(e) To be compatible with the Arrow 2 engine starting time, it is recommended that the hangar doors be capable of being fully opened within 20 seconds.

(f) For meeting the "scramble" and "standby" requirements the ARROW 2 aircraft will have provision for accepting the following services from the ground support equipment.

(i) Hot, medium pressure air for starting the two jet engines. Simultaneous starting of both engines is required in order to meet the scramble requirements.

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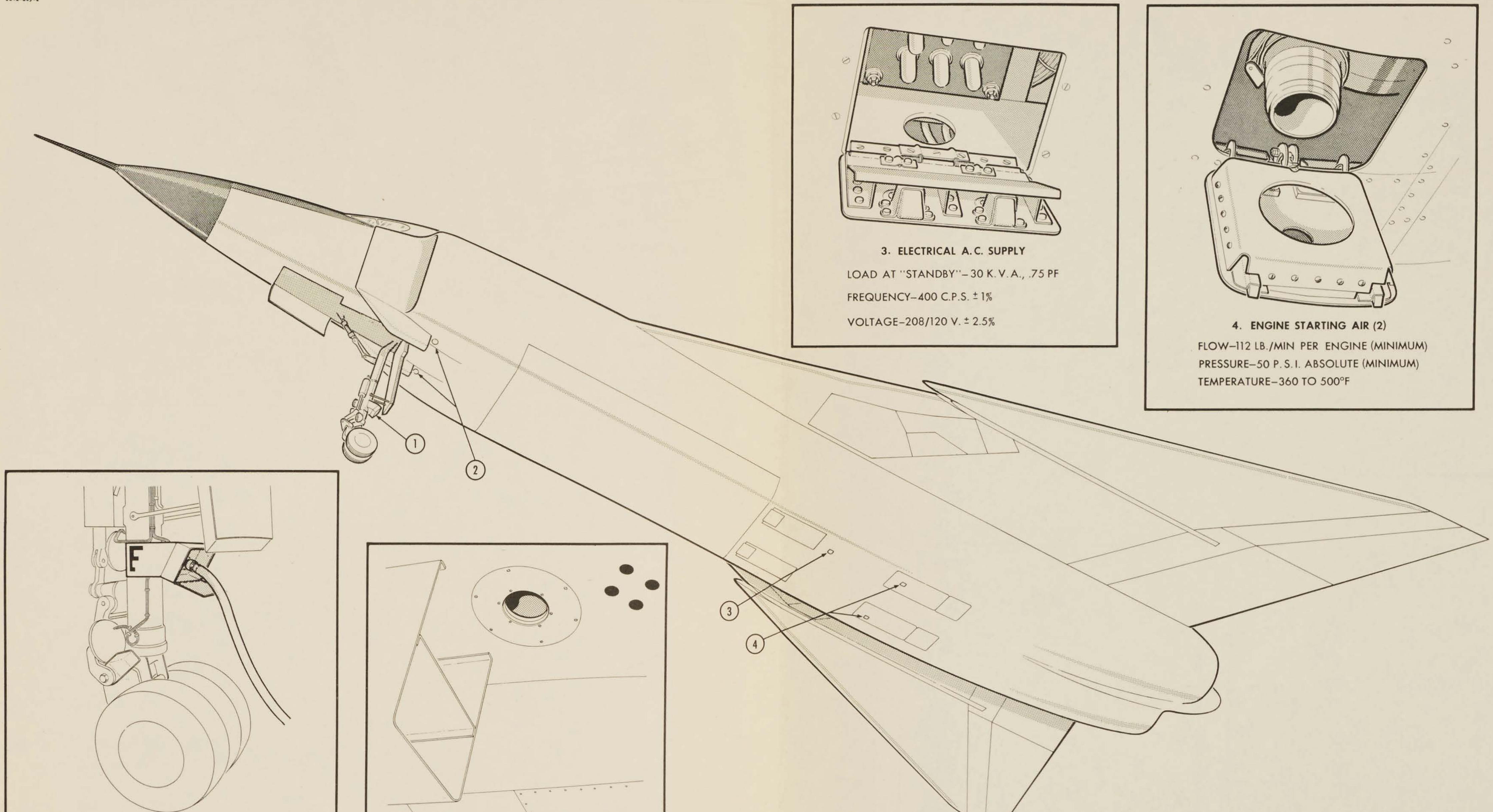

1. NOSE LANDING GEAR CONNECTOR

SERVICES
- 27.5 V. D.C. POWER
- ENGINE STARTING CONTROLS
- ENGINE FUEL FLOW CONTROL HEATERS
- TELESCRAMBLE
- INTERCOMMUNICATION

2. COOLING AIR SUPPLY (2)

FLOW–150 LB./MIN. (TOTAL)

TEMPERATURE–55 TO 80° F.

PRESSURE–4.5 P.S.I. GAUGE

3. ELECTRICAL A.C. SUPPLY

LOAD AT "STANDBY"–30 K.V.A., .75 PF

FREQUENCY–400 C.P.S. ± 1%

VOLTAGE–208/120 V. ± 2.5%

4. ENGINE STARTING AIR (2)

FLOW–112 LB./MIN PER ENGINE (MINIMUM)

PRESSURE–50 P.S.I. ABSOLUTE (MINIMUM)

TEMPERATURE–360 TO 500°F

FIG. 44 ARROW 2 CONNECTIONS FOR GROUND SUPPORT SERVICES

(ii) Up to 40 K.V.A. of 208/120V, 400 c.p.s. 3 phase AC power

(iii) 5 amperes at 27.5 VDC with the main AC cable

(iv) Cooling air

(v) Ground wire

(vi) Nose landing gear cable

It is recommended that provision be made in the RCAF standard readiness hangars for running these services under the floor to suitable points below the aircraft.

(g) Although the readiness characteristics of the ASTRA I electronic system are not known at present, it is considered desirable to be able to warm and operate the fire control system prior to take-off for a combat mission. On this basis the following services will be required for each ARROW 2 aircraft at standby:

(i) Hot, medium pressure air.

(ii) Electrical power (208/120V, 400 c.p.s., 3 phase AC) up to 30 K.V.A. at 0.75 power factor will be required.

(iii) Cooling air

(iv) Services through the nose landing gear receptacle:

(a). 27.5V DC signal wires

(b) Intercommunication with ground crew

(c) Telescramble

(v) Ground Wire

(h) For engine starting only, without supplying power and cooling air for the ASTRA I system, the following services will be required.

(i) Hot, medium pressure air.

(ii) Services through the nose landing gear receptacle:

50 amperes at 27.5V DC

27.5V DC control wire

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Task	PREPARE FOR STANDBY ; / SCRAMBLE — TIME - MINUTES TO TAKE OFF (5 4 3 2 1 0)
AIR CREWS	
1. LEAVE CREW ROOM, GO TO #1 A/C	20 SECS
2. LEAVE CREW ROOM, GO TO #2 A/C	35 SECS
3. ENTER & CHECK COCKPITS #1 A/C	30 SECS
4. ENTER & CHECK COCKPITS #2 A/C	30 SECS
5. CONNECT HARNESS, SERVICES #1 A/C	1 MIN 30 SECS
6. CONNECT HARNESS, SERVICES #2 A/C	1 MIN 30 SECS
7. WAIT FOR SCRAMBLE ORDER #1 A/C	40 SECS
8. WAIT FOR SCRAMBLE ORDER #2 A/C	50 SECS
9. START ENGINES #1 A/C	20 SECS
10. START ENGINES #2 A/C	20 SECS
11. ACCELERATE ENGINES & TAXI #1 A/C	25 SECS
12. ACCELERATE ENGINES & TAXI #2 A/C	25 SECS
13. TAKE-OFF #1 A/C	30 SECS
14. TAKE-OFF #2 A/C	30 SECS
GROUND CREW #1	
1. PROCEED TO CONTROL ROOM	30 SECS
2. START COOLING AIR SUPPLY, ENGINE STARTERS	45 SECS
3. MONITOR GROUND SUPPORT EQUIPMENT	2 MIN 5 SECS
4. CHECK RELEASE OF COUPLINGS #1 A/C	10 SECS
5. CHECK RELEASE OF COUPLINGS #2 A/C	10 SECS
6. TURN OFF EQUIPMENT	10 SECS
GROUND CREW #2 & 3	
1. REMOVE GROUND LOCKS ASSIST AIRCREW #1 A/C	2 MIN 20 SECS
2. REMOVE ACCESS STAIRS	20 SECS
3. OPEN HANGAR DOORS	20 SECS
4. PROCEED TO CREW ROOMS	15 SECS
GROUND CREW #4 & 5	
1. REMOVE GROUND LOCKS ASSIST AIRCREW #2 A/C	2 MIN 35 SECS
2. REMOVE ACCESS STAIRS	20 SECS
3. OPEN HANGAR DOORS	20 SECS
4. PROCEED TO CONTROL ROOM	15 SECS

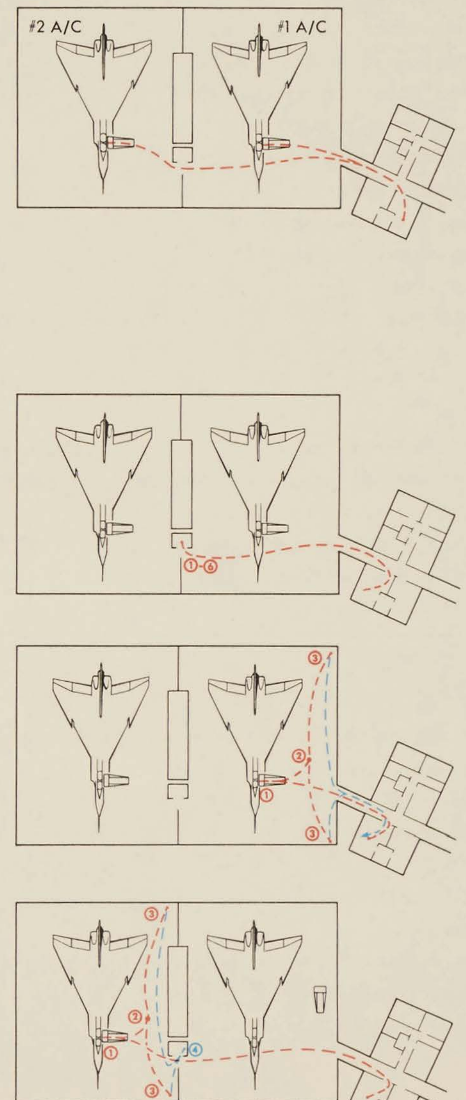

FIG. 45 TIMES AND MOTIONS FOR STANDBY AND SCRAMBLE

27.5V DC signal wire

500 VA, 400 c.p.s., 1 phase AC power

The aircraft might be operated with only these services for training and ferry flights or when AC electrical power is not available.

(i) Gas turbine compressors are recommended for supplying the hot, medium pressure air.

(j) It is recommended that the 3-phase, 400 c.p.s., AC power for standby be obtained from an ARROW 2 generator driven by a 50 H.P. synchronous motor through gearing.

(k) For providing cooling air to the aircraft, an electrically driven compressor and a Freon refrigeration system with an air cooled condenser is recommended.

(l) A 500 VA inverter is recommended for supplying 115 volt single phase AC power to each aircraft under emergency conditions.

(m) Existing types of RCAF equipment will be suitable for supplying 28 VDC power for the ARROW 2 aircraft.

(n) The AN/AIC-17 ground intercommunications system is recommended for use in the readiness hangars.

(o) The mobile retractable stairs designed by Avro Aircraft Limited are recommended for access to the cockpits of the ARROW 2 in the readiness hangars.

(p) It is recommended that all of the controls for the ground support equipment for two aircraft be brought to one control panel and that this be in a room near the front of the readiness hangar.

(q) The main electrical supply to the readiness hangars should provide at least 838 K.V.A. in order to maintain four ARROW 2 aircraft at "standby", assuming that the compressors for engine starting are not powered by electricity.

(r) If the electrical supply to the air base should fail, emergency electrical power and cooling air for aircraft at standby should be obtained from mobile power/air conditioning units.

(s) Mobile ground support equipment will be developed by Avro Aircraft Limited for the RCAF in the form of two vehicles, one to provide all of the services related to starting the aircraft, the other to provide electrical power and cooling air for standby and for maintenance work.

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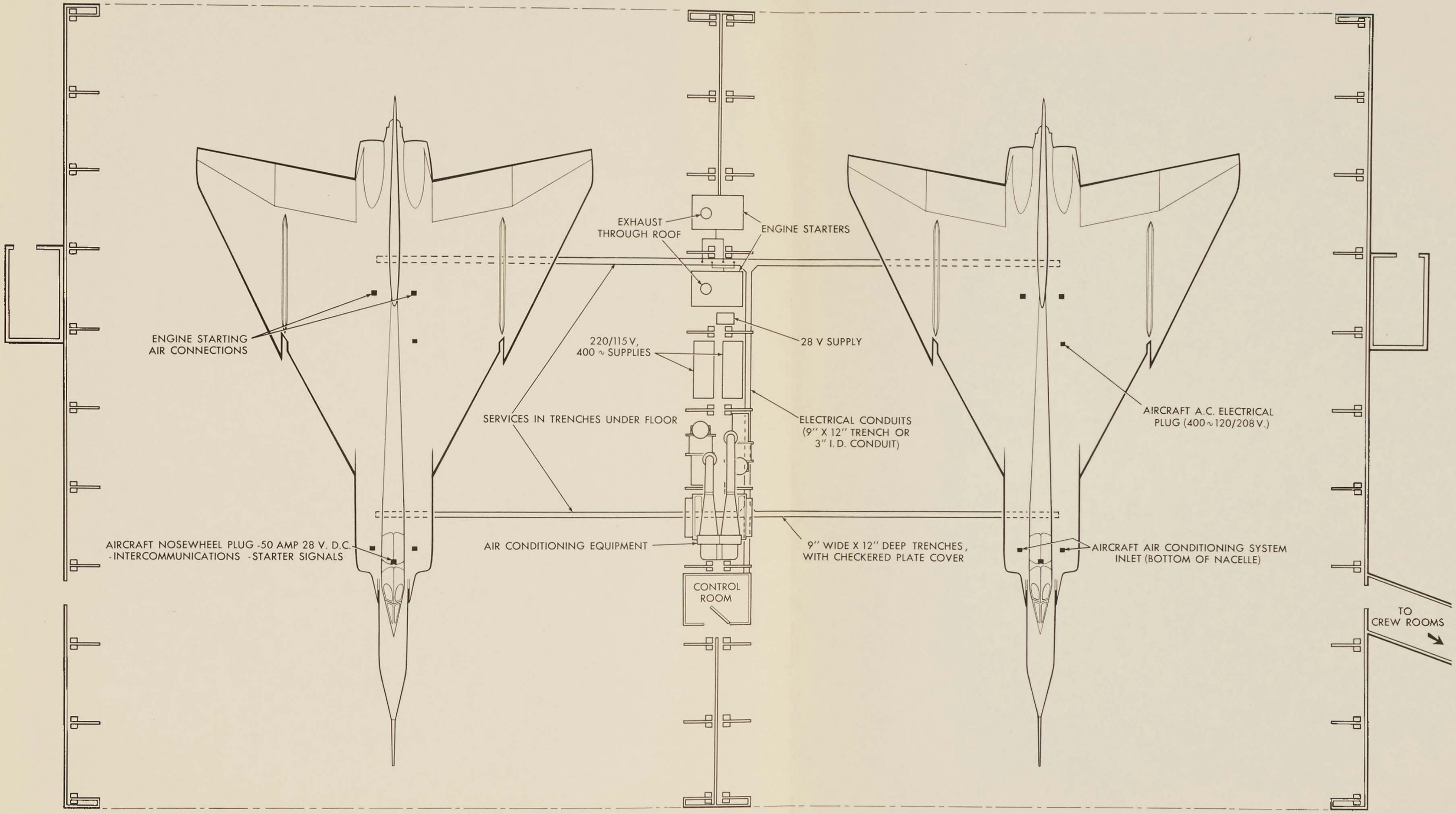

FIG. 46 LAYOUT OF GROUND SUPPORT EQUIPMENT AND SERVICES IN R.C.A.F. STANDARD READINESS HANGARS

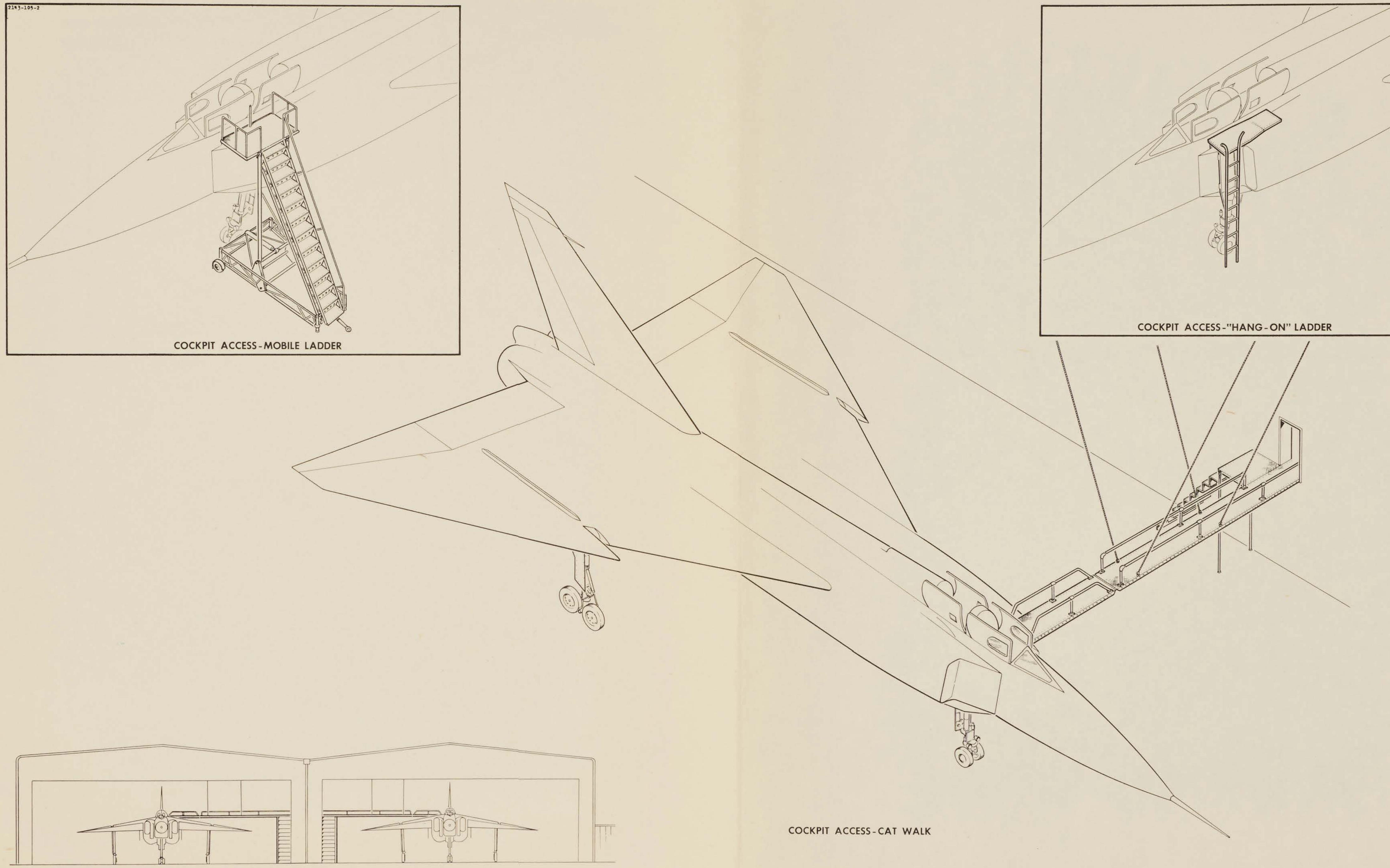

FIG. 47 ARROW 2 - COCKPIT ACCESS EQUIPMENT

(t) At each ARROW 2 base there should be towing vehicles available that are capable of providing a drawbar pull of 7000 lb. under all surface conditions that are likely to be encountered.

26.0 AIRCRAFT SYSTEMS TRAINER

26.1 GENERAL

For the purpose of training Air Force technicians in the most efficient maintenance of the various systems and services in the aircraft, it is customary to provide an Aircraft Systems Trainer (AST). An AST consists of accurate representations of the various equipment systems and services, in which the actual equipment or models are installed, having their controls mounted in such a manner to facilitate the visual instruction of those systems and services.

26.2 STATUS OF AIR SYSTEMS TRAINER

Under the authority of the AD-44 Statement of Work, AVRO is currently engaged in a design study which will embody recommendations for an AST, due for submission to the RCAF in December of this year.

At an ad hoc meeting of the ARROW Development Co-ordinating Committee Meeting on 4 September 1957 it was agreed that the AST study should cover the complete weapon system and should be co-ordinated with Orenda Engines Limited, RCA and Canadair Limited, this being subject to the necessary contractual action being initiated by the RCAF with these contractors to ensure the necessary inputs. Such action has not yet been taken, and hence no inputs have been received. It is therefore clear that the present AST study will not include the full information contemplated by the above meeting.

The AST for the ARROW will consist of a number of panels, each designed to portray the function of a particular system or portion of a system. The AST will demonstrate the principles of operation, location and function of components or components systems, safety precautions, servicing and maintenance procedures and the use of special test equipment where applicable. Projection transparencies to supplement the trainer as a training aid will be supplied as necessary.

At present no statement of the number of panels which will comprise an AST can be made. This will be decided during the design study.

As a result of the recommendations of the 4th Meeting of the Training Aids Committee held in Ottawa 4 July 1957, the 36th ARROW Development Co-ordinating Committee of 17 September 1957 issued the following requirements which will be included in AVRO Aircraft Systems Trainer design study.

(a) The design study is to determine the systems that must be grouped together and those that could exist as independent units. This requirement was predicted by the integration factors involved in the ARROW aircraft systems. It was envisioned that to demonstrate a particular system, the trainer configuration would have to reflect a series of sub-systems tied together without regard to the trade structure.

(b) The trainer design will be of a simple construction, sufficient to permit movement of a training unit from one room to another within the Field Technical Training Unit.

(c) Since inter-station mobility is not a requirement, heavy bases, castors and skids are redundant.

(d) The trainer is to be limited generally to a size that will allow movement of the AST through double doors (RCAF to define). The AST bearing load should not exceed normal floor loading (RCAF to define).

(e) Suitable snap-on dust covers of plastic materials are to be provided and the requirements for AST shipping and/or storage containers will be deleted.

(f) The use of large components for training is to be discouraged; however, this will be dependent on the manual skills required to be taught, and the decision in each instance will be based on the merits of the requirement.

(g) The AST configuration is to take into account the concept that the animated type trainer could be effectively used to teach operation, but it would be ineffective in training maintenance procedures and fault finding techniques. AVRO is to consider the possibility of utilizing the two types of configurations per system, to teach maintenance to a higher level than at present.

(h) The factors involved in the missile marry-up task are to be considerered so that the requirement for an AST and test equipment can be assessed.

(i) The test equipment is to be considered in the design study so that the AST is designed as a training system.

(j) The use of 110 V 60 cycle and/or 50 cycle AC would be satisfactory for illuminated, animated schematic type trainer configurations. Where the use of 28 V DC and 400 cps power is required, in the case of operable type trainers, power supplies are normally to be part of the building and not supplied as part of the AST. AVRO's study should specify where special power arrangements are necessary and state the type and quality of power required.

(k) All telecommunications installations in the ARROW are part of the ASTRA I system.
Refer to item (g) above.

(l) Engine parts are to be provided as part of the Engine Air Systems Trainer if a complete engine is not practical.

(m) AVRO is to consider a propulsion system AST to demonstrate after-burner, fuel and ignition system, etc. as one unit. Use of coloured film and/or transparency technique to teach the propulsion system, is not to be discounted.

(n) AVRO is to attempt to estimate the AST space and power requirements. The report will assist the RCAF in formulating a more detailed specification.

It is not considered necessary that an AST should be produced for the ARROW 1. The first trainer will portray the first aircraft to be provided for RCAF use.

The current trend of thinking appears to indicate that an alteration in the RCAF trade structure will be necessary to provide personnel qualified to service and maintain ARROW aircraft. Comptability with the present RCAF trade structure will therefore not be considered in designing the trainer.

27.0 STRUCTURAL GROUND TEST PROGRAM

27.1 STATIC TESTING OF COMPLETE AIRCRAFT

One of the most important objectives of the static testing of the complete aircraft is the confirmation of the calculated internal load distribution. The test series will cover the strength of the main landing gear bay structure under dynamic loading conditions as well as the strength of primary structure under loads representative of manoeuvres. A test program has been established and is as follows:

1. Design and construction of the test rig (fig. 48), building of the test aircraft and the installation of internal strain gauges.

2. Cockpit limit and proof pressure tests.

3. Seat ejection tests.

4. The initial setting-up of the aircraft in the test rig and external strain gauging.

5. Landing gear spring-back case to limit load.

6. Rolling pull-out case to limit load.

7. Main landing gear uplocks and doors integrity test.

8. The symmetric case with no pitch to limit load; test on front fuselage (ref. para. 20.2).

9. The symmetric case with no pitch to limit load; test on rear fuselage. (ref. para. 20.2).

27.1.1 TEST PROGRESS

Construction of the test rig, the design of which began in 1955, is nearly complete. It is designed so that the rig structure can be altered conveniently to permit the various tests to be carried out. The test loads are applied through a system of whiffle trees to obtain the desired load distribution. Rubber patches are cemented to the skin surface to transmit the loads to the aircraft structure. Normal aircraft load pick-up points are used wherever possible. The first ground tests will be performed using calculated loads. The tests will employ approximately 7,500 strain gauges strategically placed throughout the aircraft strucutre. Automatic strain recording equipment will then transfer the strain values to typed sheets which will be suitable for direct inclusion in subsequence reports. Punch cards will be used in conjunction with automatic plotting apparatus to provide rapid inspection of the data.

The design work on the test rig is complete except for the lower priority cases. The main structure of the test rig is approximately 90% complete. The design of the whiffle trees and tension patch assemblies is complete; however, their erection cannot be started until the aircraft is installed in the test rig.

Approximately 50% of the required number of strain gauges have been installed with complete wiring. The majority of those installed are internal strain gauges.

The recording equipment has been received and is being set up. The equipment will soon be put in operation in the testing of the engine shroud. This test employs a large number of strain gauges and will serve as a useful test run for the recording equipment and punch card system which will subsequently be used in the static testing of the complete aircraft.

28.0 SYSTEMS GROUND TEST PROGRAM

28.1 FUEL SYSTEM

28.1.1 INTRODUCTION

The fuel system test rig has been designed to simulate the left wing and fuselage fuel tank installation in the ARROW. The tank capacities, geometric relationship, shape, and the position of ancillary equipment, including booster pump and pressurization system, have been duplicated. The assembly is capable of limited rotation on gimbals about the pitch and roll axes, although not at representative rates of roll. External facilities provide for altitude, climb and dive simulation, fuel heating and simulated engine consumption, as well as pressure refuelling and defuelling.

The test program includes investigations of the pressurization system, normal in-flight fuel transfer, refuelling and defuelling, simulated flight sequences and emergency operations.

Initial tests were started in 1954 on a preliminary rig. The completion of a suitable building to house the main rig provides separate test benches for equipment items requiring special investigations, as well as auxiliary apparatus for plumbing, filtering, heating, measuring and storing fuel. There are also isolated rooms for control and recording purposes and for electrical generators and hydraulic pumps which provide the rig motive power.

28.1.2 PROGRAM

The ARROW fuel system tests have been divided into six main sections, to be carried out in approximately the following order:

(i) Calibrations and start-up

(ii) Initial testing under static conditions -
- Pressure refuelling and defuelling
- Pressurization system check on pressure build-up and relief valve
- In-flight fuel transfer
- Damaged proportioner cases

(iii) Complete investigations of all operating aspects for full ranges of parameters under static conditions. This includes all tests listed in (ii) plus:
- Collector tank pressurization
- In-flight fuel transfer tests (inclined attitudes)
- Fuel pressure regulator tests

(iv) Simulated mission tests

(v) Changeover to low pressure system test

(vi) Changeover to fuel management system

(vii) Fuel management system tests

The above program was started in 1956 and is scheduled for completion in November 1958. The general progress has been satisfactory. The next series of tests will be simulated mission tests.

28.1.3 BRIEF HISTORY OF TESTS

Pressurization tests included the fuel pressure regulators, pressure refuelling and individual tank refuelling. Aero Supply fuel-no-air valves proved a source of trouble, though continuous development has made them satisfactory for rig operation. In-flight fuel transfer tests at inclined attitudes were completed, but trouble in refuelling tank #5 was experienced. Further development work to clear this problem is in progress.

Checks were done to determine the effectiveness of the Chan-o-Seal principle of sealing integral tanks on a quarter scale tank. Leaks occurred under high temperature, but were re-injected successfully. The tests were continued and finally stopped when a crack appeared on the lower skin of the tank. The Stress Office, has confirmed that this failure is of no significance with regard to wing structure as the specimen was non-representative of the actual structure and was only used for leak testing. These tests have been completed.

Fuel flow proportioner tests were carried out by Eclipse-Pioneer Division of Bendix Aviation Corp. The tests have been completed and the reports received by AVRO. For further details refer to Part 3 para. 14.3.1.

Severe leaks developed during testing of the stringer seal but after modification the testing was satisfactorily completed without further breakdown.

The collector tank air ejector has been tested, modified and retested several times with improving results. In its present state, it is just about acceptable for use in the first aircraft, but further development is needed.

28.2 FLYING CONTROLS SYSTEM

28.2.1 INTRODUCTION

The main testing carried out will consist of an evaluation of the complete flying control system, from the cockpit controls to the control surfaces (Fig. 50).

Testing is further broken down into individual tests on hydraulic and mechanical components. The program for flying controls system testing is as follows:

1. Elevator frequency response tests without hinge moment (completed in 1956)

2. Elevator frequency response tests with hinge moment (begun in 1956)

3. Duty cyling of elevator system at room temperature (begun in 1956)

4. Development of elevator input circuit for manual and stick force modes

5. Hydraulic system investigations etc.

6. Aileron and rudder frequency response tests without hinge moments.

7. Aileron and rudder frequency response tests with hinge moments

8. Combined system tests with hinge moments at room temperature

9. Simulated flight tests with full system and analog computors at normal, high and low temperatures.

28.2.2 GENERAL SYSTEMS TESTS

Tests were carried out in 25 cu. in. and 100 cu. in. accumulators in regard to pressure surges in the hydraulic system. The 25 cu. in. accumulator proved satisfactory.

Hydraulic connections for 4,000 p.s.i. system

Tests were carried out at NAE to determine the torque vs performance of the boss fittings for hydraulic connections.

The results of flexural fatigue tests on hydraulic connections at elevated temperatures were very unsatisfactory. Fretting occurred between the stainless steel tubing and the cadmium plated sleeves. This was thought to be due to electrolytic action and tests were repeated using nickel plated sleeves and mating surfaces coated with Epon resin. This showed no improvement; however, NAE suggested an alternative method of swaging the connections, in which the tubing is expanded out into the sleeves, rather than the existing method of swaging the sleeves into the tubing. This method proved very satisfactory, although production of the joint is more difficult. The remainder of the initial airworthiness tests are now being completed at NAE.

Suitability of Wig-O-Flex pipe and boss couplings for low pressure hydraulics

The original test requirement for these specimens was quite severe. The requirement was revised, to the deflection imparted to the specimen during the cycling operations at - 65°F and 250°F.

Leakage was the main fault and it was decided that the O-ring must be made of an improved material. MS28784 O-rings were used which reduced the leakage to acceptable limits.

Tests of steel tubing for hydraulic systems

The early failure of steel tubing in testing was attributed to internal scoring during bending operations and in other cases due to faulty welding of the seam.

It has been decided to discontinue the use of steel tubing to Avrocan specification M-7-6 and use tubing of specification MIL-T-6845.

Control system bearing selection

Torrington plain bearings which have electro molybdenum disulphide surface finish have been tested. One bearing suffered early failure which was attributed to the interference fit between the inner and outer races. Testing was discontinued after one million cycles on the remaining bearings as the decision was taken not to install plain bearings in the aircraft.

28.2.3 ELEVATOR

On testing the emergency mode of the flying control system the stick forces were too high. A further investigation was carried out with a modified control valve on one jack, and the feel unit break-out force adjusted to obtain good stick centring and trim run on characteristics compatible with reasonable stick break-out forces. This was found to be satisfactory.

28.2.3.1 Evaluation of Elevator Control Valves

Two elevator control valves were first tested in the emergency mode which exhibited high friction forces. Tests were then carried out on two modified valves at room temperature. The valves had an excessive leak rate from pressure to return ports.

Tests were also carried out to determine the pressure drop across the valve for various flow rates with the valve fully opened. The valve displacement was then reduced to .07 inch to simulate an aileron valve operation. Again poor force characteristics were exhibited.

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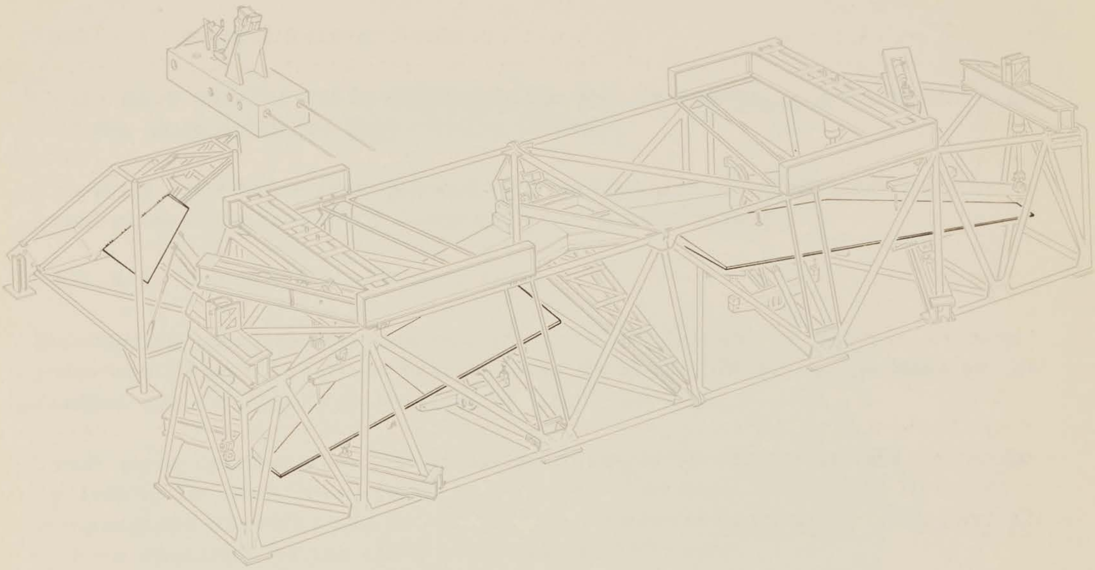


FIG. 50 FLYING CONTROL SYSTEM TEST

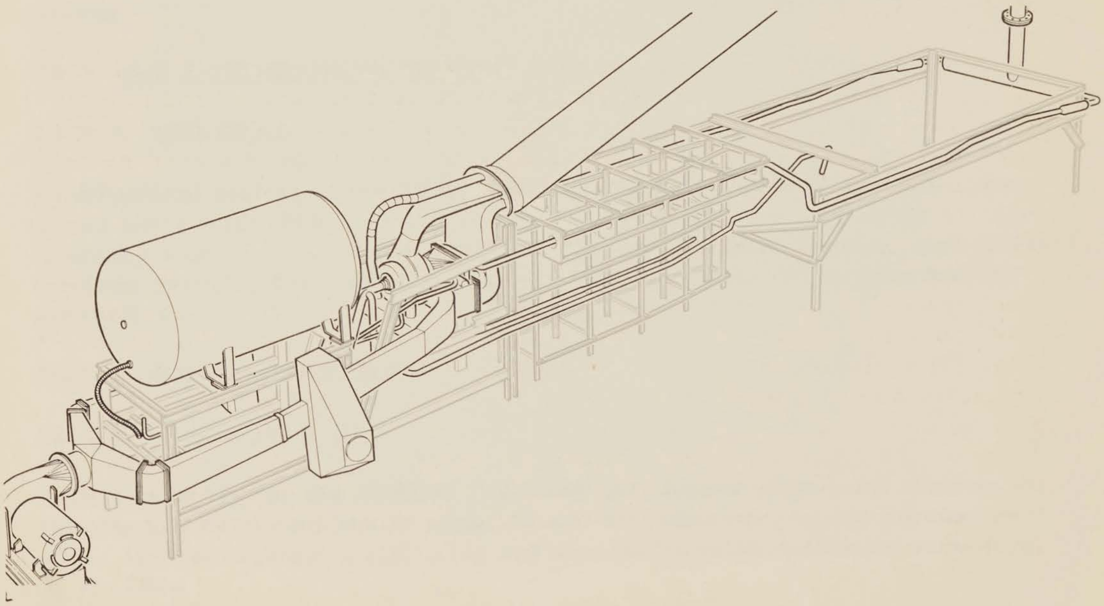


FIG. 51 AIR CONDITIONING SYSTEM TEST

28.2.4 RUDDER

28.2.4.1 Rudder System Tests

Tests were carried out to determine the feel unit spring rates. It was found that the units exhibited high friction forces.

The tests for rudder strength and stiffness have been completed and frequency response tests will commence in the near future.

28.2.4.2 Rudder jack and valve tests

During frequency response tests, valve oscillation was observed. Further tests are being carried out to determine the effectiveness of damping on the control valve and back-lash in the follow-up linkage.

Tests were conducted to determine the changeover time from one hydraulic system to the other when failure develops. The tests were done at room temperature as well as at low and high ambient temperatures. The period of time required for the changeover was satisfactory.

28.2.4.3 Evaluation of the rudder control valve

Tests were carried out to determine the pressure drop across the rudder control valve when fully opened for various flow rates. This was done at room temperature and at 250°F. Further testing will be done at low temperatures.

28.3 <u>AIR CONDITIONING SYSTEM</u>

28.3.1 GENERAL

Development testing of the ARROW air conditioning system has been in progress since mid-1956. The basic tool in this program has been the air conditioning test rig. Recently, the testing program on the rig has been directed towards ensuring the installation of a workable system in the first ARROW 1 aircraft.

28.3.2 SYSTEM TEST RIGS

28.3.2.1 ARROW 1 Rig

A full-scale rig for the ARROW 1 aircraft was constructed some time ago. The rig has been continually modified and developed to provide simulation of the aircraft system installation and system flight performance as nearly as possible.

The air supply for the rig is provided by a remotely located compressor plant. The plant consists of three centrifugal compressors driven by electric motors. A set of heat exchangers are employed to control the simulated bleed air temperatures. The hot compressed air is ducted to the rig-mounted system through heavily insulated ducts in which the charge air temperature losses are held to a minimum.

The system ducting and equipment are mounted on a stationary rig which has recently been enclosed by a sound-proofing structure to localize the noise generated by the system. The aircraft system ducting and equipment are duplicated on the rig installation as closely as possible. Substitute equipment, however, is used when specified aircraft equipment is not available.

The ram cooling air for the system is supplied by a blower driven by a tandem electric motor installation. The ram cooling air is drawn by the blower from the general test area outside of the rig enclosure and exhausted to the atmosphere outside the main test building. No attempts have been made to simulate the actual flight temperature and pressure conditions of the ram cooling air.

28.3.2.2 ARROW 2 Rig

A full-scale rig for the ARROW 2 aircraft, similar to the ARROW 1 rig, will be constructed. The design of the test rig is nearing completion and a substantial portion of the component parts are presently in manufacture. An ARROW 2 rig is necessary since the systems for the two aircraft variants are quite different and simultaneous testing of ARROW 1 and ARROW 2 system is scheduled.

28.3.3 TEST PROGRAM AND PROGRESS

The tests making up the air conditioning system test program, and their status to date, are as follows:

1. ARROW 1 air conditioning system tests. This is a series of general tests to determine overall system performance and to check system functional integrity. Tests have been completed on:

   (a) Flow distribution

   (b) Duct characteristics

   (c) Equipment and rain repellent circuits

   (d) Simulated taxi case

2. ARROW 2 air conditioning system tests. These are performance and

functional integrity tests similar to the ARROW 1 tests. The test rig is in process of design and construction.

3. Cockpit environment tests. This is a general test program utilizing the aircraft metal mock-up. The metal mock-up is presently being prepared for the tests.

4. Performance check of ARROW 1 first aircraft equipment. This comprises pre-installation tests of system equipment on the test rig. Scheduled tests have been completed.

5. Pre-flight testing of ARROW 1 first aircraft system. This comprises testing of the aircraft installed system prior to first flight.

6. Performance and pre-flight testing of ARROW 1 fourth and fifth aircraft system and components. These constitute tests similar to the tests on the first aircraft outlined in items 4 and 5 above, to provide pre-installation and pre-flight checks of the specially modified air conditioning system of the ASTRA I test vehicles.

7. Flow control test. Preparation for these tests are in progress.

8. Miscellaneous tests:

   (a) Duct insulation tests have been completed recently, and have established insulation requirements for the bleed air ducts in the ARROW 2.

   (b) Clamp joint leak tests to establish efficiency of duct joints are presently well advanced.

   (c) Leak detection tests for bleed air lines are scheduled.

   (d) Pressure drop tests in the vaned elbows are scheduled.

Additional tests will be requested and scheduled as the design and development program progresses.

28.3.4 TEST RESULTS

The performance tests of the ARROW I first aircraft equipment have shown that the turbine, augmentor, heat exchanger, water evaporator and pressure reducing valves operate satisfactorily. Some difficulties, however, were experienced with the temperature control circuits.

The cockpit temperature control circuit and the turbine outlet-temperature control circuit were found to be unstable on test. The nature of the instability of the circuits and the modifications made to obtain satisfactory circuit operation are discussed in greater detail in Part 3, Section 11 of this report. The equipment temperature control circuit exhibited satisfactory operating characteristics.

The duct characteristics tests have indicated that duct losses are from 30% to 60% lower then the estimated values.

28.4 ELECTRICAL SYSTEM

28.4.1 GENERAL

The test program scheduled for the electrical system has been completed. The tests, however, produced the requirement for several modifications to the system breadboard. The modified circuits then required further testing. A brief summary of electrical systems testing is given below:

28.4.2 ELECTRICAL SYSTEM FUNCTIONAL AND OPERATIONAL TEST

The operation of the complete electrical system has been completed. The input terminal conditions were typical of the aircraft power supply, and service loads were represented electrically at the output terminals. It was desired to determine the effects of typical load situations upon the voltage and current at test points in the circuit, while considering continuous service loads and probable arrangements of the longer transient loads.

The tests simulated the operation of the various power system circuits as installed in the elctrical system breadboard. The functioning of each system could then be observed under typical flight and ground operations including conditions of typical failure. Tests were conducted on the following systems:

(a) Start and ignition system

(b) Engine service system

(c) Landing system circuit

(d) Fuel system circuit

(e) Fire protection circuit

(f) Canopy actuator circuit

(g) Air conditioning circuit

(h) Flight services circuit

(i) Internal and external light circuits

(j) Duct de-icing circuits, and nose, windshield and canopy circuits. Several faults were observed during these tests, these have been corrected in the ARROW 1 electrical system. A brief description of the major recommended changes will be found in section 10. of this report.

28.4.3 30 KVA ALTERNATOR PRESSURE DROP

During the testing of the 30 KVA alternator, a high pressure drop was noted. Investigation later revealed that the position of the cooling duct in the alternator was critical. A more detailed test was then carried out under more rigid regulations with an exactly representative inlet duct placed in various positions. Tests were also conducted using ducts of different configurations. These tests were followed by pressure drop tests made at higher air densities. The completion of these tests produced satisfactory verification of aerodynamic calculations. The design of the duct was changed to give the optimum cooling.

28.5 LANDING GEAR SYSTEM

28.5.1 GENERAL

The objective of the ground test program for the landing gear system is to perform realistic function tests which will include variables such as aircraft speed, accelerations, altitude, temperature and operation of emergency system. The program will also provide studies of the interactions of the three landing gears, and their doors, and the times required for raising and lowering.

Test rigs have been constructed with all appropriate pneumatic and hydraulic piping, control apparatus and actuating gear. These rigs contain the mechanisms to reproduce the moments on the legs and doors for all positions of the gear and for any flight case.

The rig structure itself has been designed to simulate the deflections of that part of the airframe it replaces. A method of varying the distances between various elements of the system, such as hinges, pivot points, latch positions etc., has been devised to represent structural distortion corresponding to the flight case being considered. Extreme temperature tests will also be made with allowances for the difference in thermal expansion and contraction between the rig and the actual airframe.

The landing gear test program will require no additional special facilities since the environmental chambers, pumping, recording and refrigeration equipment required for the flying control tests are to be used.

All testing bearing on the safety of the aircraft will be completed before first flight, with the remainder being completed as time and facilities permit.

28.5.2 STATUS OF LANDING GEAR SYSTEM TESTING

Testing of the nose landing gear retraction jack for hydraulic damping has shown the action to be satisfactory.

Tests have been completed for retraction and extension under two flight case loads (1 and 7) with satisfactory results.

The flow rate to retract the nose gear in 2 seconds was found to be approximately 10 gallons per minute.

Tests have been completed for retraction and extension under two flight case loads (1 and 4) during which it was found that the door uplock rollers struck the latches unevenly causing a small amount of pin bending. In addition, the door jack was applying load to the door before the uplock was released. Further testing has been carried out to investigate these faults.

The rig is now having nose-wheel steering incorporated into it, for testing in conjunction with the flying control rig.

Preliminary extension and retraetion tests of the main landing gear are planned for the immediate future.

28.6 CANOPY AND ESCAPE SYSTEM

28.6.1 INTRODUCTION

Initial tests on the canopy and escape system may be divided into two main stages: mechanical operation of the system and operation under simulated flight loads. The first of these has already been completed and the second is about to commence. In addition planning of rocket sled tests is under way. No contractual authority has yet been granted for this test work.

28.6.2 PROGRAM

The program on the canopy and escape meehanisms has been divided into four main sections and the tests will be done in approximately the following order:

(i) Rig tests with dummy canopy to develop and prove emergency unlatching and opening.

(ii) Rig tests with actual canopy and representative cockpit volume, pressurized, but without external load, to demonstrate canopy integrity.

(iii) Demonstrations of canopy emergency actuation and seat ejection sequences at zero speed, without simulated airloads.

(iv) Rocket sled tests with dummy ejections

To date section (i) has been completed and section (ii) tests are about to commence.

28.6.3 A BRIEF HISTORY OF TESTS

Twenty four initial canopy functioning tests, for cartridge evaluation, were done with no loading on the canopy. Opening times varied from .37 to .76 seconds, at room temperatures, but results were more consistent with high and low cartridge temperatures. Further testing will be conducted with the aircraft canopy subjected to initial pressure on another rig configuration which is now under construction. This testing must await availability of canopies, which are expected in October.

Ejection system tests will be performed using the cockpit of the static test aircraft. Efforts continue to obtain a suitable net to catch the seat and dummy.

Considerable preparatory work has been done towards rocket sled testing of the ARROW escape system. No material orders will be placed until contractual authority is granted. Test specifications have been sent to prospective contractors and their proposals are being evaluated.

28.7 SPARROW MISSILE PACKAGE

28.7.1 HYDRAULIC MISSILE LOWERING AND RETRACTION SYSTEM

As part of the overall armament test program, a series of tests has been conducted to prove the principle of the hydraulic system for the missile lowering and retraction mechanism.

The test rig had facilities for one only simulated Sparrow 2D missile and missile launcher. Actual missile lowering and retraction hardware was not used, as these items were not available. Although the complete test was not very satisfactory from a mechanical aspect, results were obtained that indicated the hydraulic pressures and operation time were within the design requirements. Further testing will be continued when the actual aircraft hardware is available.

28.7.2 MECHANICAL WEAR TESTS

In an attempt to reduce the wear of the magnesium sideload reacting link of the missile lowering mechanism, a series of tests is being carried out to obtain a suitable liner that will reduce the coefficient of friction. The side-load reacting link acts in a similar manner to a cylinder and piston with two sets of piston rings and a side force acting on the piston rod.

As a first attempt to reduce the wear, the magnesium bore was lined with .005" thick Teflon tape, cemented to the walls with Garlock 201. The piston was actuated at the designed speed and applied loads. After 102 cycles the Teflon was worn through to the metal surface at the bearing areas of the piston rings. The piston ring bearing areas were then increased by 50%. Although similar wear results were obtained, it was thought that the wear may be due to heat generated by continuous cycling. The Garlock 201 cement should be limited to 150°F. Friction temperatures were not taken during the tests.

Testing will be continued using Teflon at single cycle operation and temperature limited to 130-140°F. Later tests will use nylon sheet liners in place of Teflon tape.

28.7.3 WEAPON PACK-FRONT SEAL

Various tests have been conducted to evolve an effective seal between the forward edge of the weapon pack and adjacent airframe structure. This seal has to accommodate a structural movement of over one inch.

Following the findings of these tests, a suitable seal has been designed. This seal consists of a strip of stainless steel fingers, covered with Neoprene rubber and then covered with Teflon sheet. This sandwich construction bridges the gap with one end attached to the airframe, while the other rests on an aluminum block. The sandwich construction is backed by an inflatable air bag. The stainless steel strip is in a finger formation to allow it to follow the aircraft contours.

The concluding tests indicated that this will be a serviceable seal, retaining the maximum allowable leak rate, with a force of 35 lbs/in of seal length applied to the adjacent structures.

28.7.4 DOOR OPERATING LINKAGE

Recently a series of tests has been planned to investigate the operation of the roller-type doors closing about the missile body. The object of these tests will be to check the actual mechanical function of the doors and link mechanism at the design speeds of door operation. Testing has not yet started.

29.0 FLIGHT TESTING

29.1 GENERAL

To date the Flight Testing Department has been engaged in preparation of the flight test program for the ARROW, which has consisted of the following major projects:

1. Evaluation and final selection of both airborne and ground instrumentation.

2. The design of aircraft instrumentation and installation.

3. The breadboard testing of the instrumentation.

4. The flight testing, on CF-100 aircraft, of both the recording and telemetering facilities.

5. The testing of a simulated ARROW flying control system on CF-100 aircraft 18107.

29.2 FLIGHT TEST INSTRUMENTATION

Three ARROW aircraft are scheduled for full instrumentation. Each aircraft will have an instrument pack which is a complete sub-assembly in itself. This pack is designed to carry both the airborne magnetic tape and the telemetering system. It is interchangeable with the weapon pack. The data processing system is fully automatic in collection, storage, recovery, corrections and presentation. Test data will be available in both tabular and graphical form.

The flight testing of the instrumentation is currently under way in aircraft 18185 and 18186. 18185 is being used for Datatape and also for radio compass installation development, including measurement of sense antenna capacitance and effective height. 18186 is the vehicle used for telemetry development.

29.3 PRE-FLIGHT TESTS

As applied to the ARROW flight test program, the term "pre-flight" refers to tests performed during build and prior to first flight. Extensive pre-flight tests of both aircraft systems and instrumentation are scheduled; these will include initial engine runs, control settings and functional tests on all the systems.

On completion of quality control tests, the pre-flight tests on instrumentation check-out and calibration will be done.

29.4 FLIGHT TEST PROGRAM FOR FIRST AIRCRAFT

It has become obvious that nearly all preliminary flying will be based on the stability and control program. Testing on other subjects will, in the main, be byproducts of this program.

The object of this program is to examine the airworthiness, safety of flight and systems functioning of the aircraft and to establish preliminary flight envelope boundaries for safe operation of the aircraft. It is anticipated that this will encompass the greater part of the flight envelope.

In order to proceed with the flight test program it will be necessary to ensure that it is safe to do so from a structural point of view. Initial tests will be made at low normal load factors and with limited manoeuvring. With the installed instrumentation it will be possible to assess the stress levels achieved at critical points and obtain approval to continue to higher factors. This procedure will be followed throughout the program until the limits of the damper system have been covered. It will be necessary to carry out specific tests for this purpose but these will be limited in number and will be integrated with other flights.

A similar probing program may be necessary for flutter testing in the region of critical design points. Owing to the stringent recording requirements no other testing will take place during these tests.

29.4.1 TESTING - PHASE 1 - AIRCRAFT ALLOCATION

It can be seen from Fig. 54 that in all, three aircraft have been allocated for Phase 1 trials. The second of these will then be made available to the RCAF as a safe functional vehicle in order that a Phase 2 program may be executed.

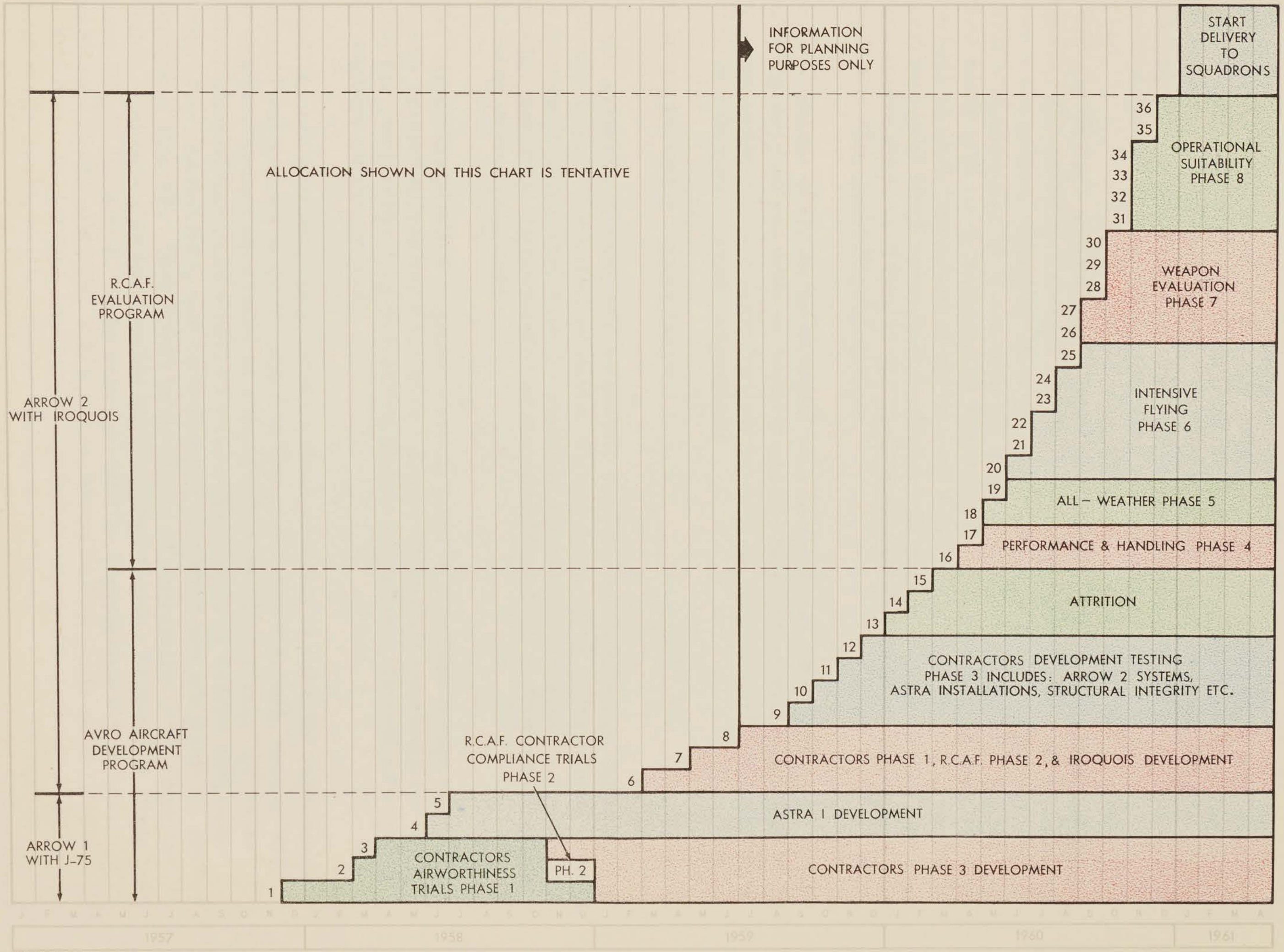
Upon its return, it is scheduled for use (together with the first and second aircraft) in contractor's Phase 3 development work. Phase 1 testing is divided into two main sections, namely Stability and Control, and Systems. As has already been mentioned, the priority on the former is such as to make it the focal point of the initial flights.

29.4.2 PHASE 1 - STABILITY AND CONTROL

The principle of testing will be one of probing flights. A segment of the flight envelope will be examined at each step and each mode of control will be developed within this segment.

It is hoped that damper development on the flight simulator, together with pilot familiarization on this simulator, will permit safe take-off and landing with the damper system inoperative. It is estimated that a large part of the flight envelope may be flown with no artificial damping and in order to avoid

ALLOCATION OF AIRCRAFT



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the possibility of damper failure, the first flight will be made with this mode of control. It is hoped to test the emergency mode during the first flight, but at a medium altitude. The first flight will be limited to Mach numbers in the medium subsonic range. Within this envelope all three modes of control will be tested during subsequent flights and developed up to a limited amount of normal acceleration and rate of roll.

Once the damper systems have been developed at the lower altitudes and speeds, the stability envelope will be gradually extended into those regions where the natural damping alone is inadequate. In these regions, the aircraft will normally be flown with the damper system operative, but the thresholds will be determined by reverting to natural damping only for short periods at each increment of speed over a range of altitudes. Acceptable increments of Mach number and altitude will depend on day-to-day results. Successive steps in filling in the flight envelope will be made with due regard to structural considerations.

One advantage of flying the aircraft with no artificial damping is that stability derivatives may be derived and any sources of disagreement between estimated and flight test values may be distinguished between aircraft and damper system parameters.

29.4.3 PHASE 1 - SYSTEMS

On the systems, testing will be mainly confined to monitoring, although specific flights, on a limited basis, will be done. The flying control system is so closely integrated with the stability and control program as to warrant special attention. Examination of the operation of the control system will be made during this time.

The engine installation requires certain tests to simulate design conditions mostly related to cooling problems. These will entail operating the aircraft for reasonably long periods at steady conditions, as well as running the engines on the ground.

Since the air conditioning system on these aircraft is not fully representative of the system to be installed in the production aircraft, testing will be confined to that necessary to ensure adequate functioning. Tests on telecommunications and antennas will be necessary to ensure that communication, navigation, and identification facilities are adequate for the flight test aircraft. Tests will be arranged in conjunction with other test flights whenever this is possible but a limited number of specific flight tests will be required. This program will be complementary to tests made on CF-100 aircraft, full scale mockups and on scale models prior to ARROW 1 first flight. It is not intended to carry out generalized performance tests during the Phase 1 program. However, in the course of other tests it will be possible to obtain some data on maximum speeds and altitude performance.

Ref: Proposed Flight Test Program - ARROW 1 - Phase 1 (Issue 2)
22 May 1957.

TECHNICAL DATA

GENERAL

Very many forms of technical data of varied presentation have been produced during the course of the investigation and design of the ARROW aircraft. Some of these documents have become obsolete before completion, while others have remained valid but are not in a reproducible form due to various reasons, although they could be either made available for inspection, or made into an issuable document, if specifically required.

PRESENTATION OF TECHNICAL DATA

Technical data relative to the AVRO ARROW aircraft is presented in the forms listed below. Where applicable they are sub-divided into ARROW 1 and ARROW 2.

Model Specifications

AVROCAN Specifications

Technical Reports

Maintenance and Ground Support

30.0 SPECIFICATIONS ISSUED

30.1 MODEL SPECIFICATIONS

	Number	Date Issued
ARROW 1 - Model Specification	AAMS-105/1	March 1957
ARROW 1 - Equipment List (Issue 3)	Appendix to AAMS-105/1	May 1957
ARROW 2 - Model Specification (Draft)	AAMS-105-2	August 1957

30.2 AVROCAN SPECIFICATIONS

The AVROCAN specifications are currently sub-divided into the following sections:

Equipment (E)

Process (P)

Material (M)

Manufacturing (Ma)

Tests (T)

Test Equipment (TA)

Of these sections, only the equipment (E) is further sub-divided into CF-100 and ARROW aircraft. To date approximately 350 AVROCAN (E) specifications have been issued specifically for the ARROW aircraft. An index of these specification numbers and titles dated July 31, 1957 was issued to the RCAF.

In addition to above, the following indexes of standards and specifications have been issued to the RCAF at its request.

(1) GEN/STDS/3 — Design and Data Standards

(2) GEN/STDS/8 — Process Standards

(3) GEN/STDS/9 — Process (P) Specifications

Additions to this list of standards and specifications will be issued at a later date.

31.0 REPORTS ISSUED

31.1 TECHNICAL REPORTS

The technical reports have been sub-divided into the following categories.

(a) Preliminary Design Proposal

(b) Wind Tunnel Data

(c) Weight and Balance

(d) Performance

(e) Stress Analysis

(f) Structural Strength Tests

(g) Electrical Load Analysis

(h) Weapon System Analysis

(j) Weapon Ground Support and Personnel

(k) Aircraft Ground and Flight Tests

(l) Functional Type Tests

(m) Vendor's Reports

(n) ASTRA I System

(o) Systems

31.1.1 PRELIMINARY DESIGN PROPOSAL

Report #	Description	Issued
P/C105/1	Preliminary Design Proposal	May 1953
P/C105/2	Preliminary Design Proposal	June 1953

31.1.2 WIND TUNNEL DATA

TUNNEL AND REPORT NUMBER	DESCRIPTION	DATE OF TEST AND ISSUE
C.A.L. (Buffalo)	SERIES I (.03 scale)	SEPT. 1953 (TEST DATE)
P/WT/6	Preliminary Plots	(Sept. 53)
P/WT/7	Final Plots	(Sept. 53)
P/WT/8	Derivatives and Zero Values	(Sept. 53)
C.A.L. (Buffalo)	SERIES II (.03 scale)	APR. 1954 (TEST DATE)
P/WT/19	Corrected Plots	(May 54)
P/WT/19a	Rough Plots	(Apr. 54)
P/WT/20	Derivatives and Zero Values	(June 54)
C.A.L. (Buffalo)	SERIES III (.03 scale)	JUNE 1954 (TEST DATE)
P/WT/27	Rough Plots	(June 54)
P/WT/29	Corrected Plots	(July 54)
P/WT/30	Derivatives and Zero Values	(Oct. 54)
C.A.L. (Buffalo)	SERIES IV (.03 scale)	JULY 1954 (TEST DATE)
P/WT/39	Corrected Plots	(Aug. 54)
P/WT/40	Derivatives and Zero Values	(Aug. 54)
P/WT/41	Rough Plots	(Aug. 54)

TUNNEL AND REPORT NUMBER	DESCRIPTION	DATE OF TEST AND ISSUE
C. A. L. (Buffalo)	SERIES V (.03 scale)	OCT. 1954 (TEST DATE)
P/WT/47	Rough Plots	(Oct. 54)
P/WT/49	Corrected Plots	(Oct. 54)
P/WT/50	Derivatives and Zero Values	(Dec. 54)
P/WT/52	Configuration and Reynolds Number Investigation	(Dec. 54)
C. A. L. (Buffalo)	PERIOD I, II, and III (.04 scale)	MAR. 1955 (TEST DATE)
P/WT/58	Rough Plots (Phases I, II, & III	(Mar. 55)
P/WT/60	Final Plots (Phase I)	(Mar. 55)
P/WT/61	Final Plots (Phase II)	(Mar. 55)
P/WT/62	Final Plots (Phase III) and comparison with .03 scale plots	(Mar. 55)
P/WT/71	Aileron Pressure Plots (Phase III)	(Mar. 55)
C. A. L. (Buffalo)	PERIOD II (.04 scale)	APR. 1955 (TEST DATE)
P/WT/66	Rough Plots	(Apr. 55)
P/WT/68	Final Plots	(Apr. 55)
P/WT/70	Cross Plots	(May 55)
C. A. L. (Buffalo)	PERIOD III (.04 scale)	MAY 1955 (TEST DATE)
P/WT/76	Rough Plots	(June 55)
P/WT/79	Final Plots	(June 55)
P/WT/80	Derivatives and Zero Values	(June 55)
P/WT/81	Effect of Droop on C_L, C_D, & C_m	(Aug. 55)
P/WT/82	Final Plots (High Reynolds No. and High Angle of Attack at M = 0.5)	(June 55)
P/WT/84	Variation of Derivatives with Angle of Attack	(June 55)
P/WT/121	Fin Pitot Position Errors	(July 56)
N. A. E. (Ottawa)	(.0125 scale)	SEPT. 1955 (TEST DATE)
P/WT/85	Asymmetric Intake Flow	(Sept. 55)
N. A. E. (Ottawa)	PERIOD I (.07 scale)	DEC. 1955 (TEST DATE)
P/WT/90	Plots and Corrections	(Jan. 56)
P/WT/93	Plots	(Jan. 56)
P/WT/97	Plots and Corrections	(Mar. 56)
P/WT/98	Corrected Plots	(Apr. 56)

TUNNEL AND REPORT NUMBER	DESCRIPTION	DATE OF TEST AND ISSUE
N.A.E. (Ottawa)	Reflection Plane Model (.02 scale)	FEB. 1956 (TEST DATE)
P/WT/102	Plots	(Feb. 56)
N.A.C.A. (Langley)	(.03 scale) M = 1.41	APR. 1956 (TEST DATE)
P/WT/111	Plots	(May 56)
P/WT/112	Cross Plots	(May 56)
P/WT/114	Rough plots and Calculations	(May 56)
N.A.E. (Ottawa)	PERIODS II and III (.07 scale)	MAY - JULY 1956 (TEST DATE)
P/WT/119	Plots	(July 56)
P/WT/126	Photographs in Tunnel	(Sept. 56)
P/WT/129	Miscellaneous Effects	(Nov. 57)
N.A.E. (Ottawa)	(.0125 scale)	MAY - AUG. 1956 (TEST DATE)
P/WT/135	1/80th scale tests at N.A.E.	(Oct. 56)
N.A.C.A. (Langley)	(.03 scale) M = 1.6, 1.8, 2.0	JULY 1956 (TEST DATE)
P/WT/122	Plots in Body Axes	(Sept. 56)
P/WT/123	Plots in Stability Axes	(Sept. 56)
P/WT/125	Cross Plots and Derivatives in Stability Axes	(Sept. 56)
P/WT/127	Photographs in Tunnel	(Sept. 56)
C.A.L. (Buffalo)	(.04 scale)	FEB - MAR. 1957 (TEST DATE)
P/WT/147	Rough Plots	(Mar. 57)
P/WT/148	Final Plots (Armament)	(June 57)
P/WT/149	Final Plots (Canopy)	(Apr. 57)
P/WT/150	Final Plots (Aircraft)	(June 57)

31.1.3 WEIGHT AND BALANCE

The following reports are issued monthly, as required by CAP 479. Therefore, a further index of weight and balance reports will not be included in the Quarterly Technical Report.

31.1.4 PERFORMANCE

Report #	Description		Date
1	Monthly Performance Report	-	September 1955
2	Monthly Performance Report	-	November 1955
3	Monthly Performance Report	-	December 1955
4	Monthly Performance Report	-	January 1956
5	Monthly Performance Report	-	Not Issued
6	Monthly Performance Report	-	March 1956
7	Monthly Performance Report	-	April 1956
8	Monthly Performance Report	-	May 1956
9	Periodic Performance Report	-	November 1956
10	Periodic Performance Report	-	December 1956

31.1.5 STRESS ANALYSIS

The following index of stress reports is a revision of the index previously sent to the RCAF on July 12, 1956 (Ref. 1399/15A/J).

ARROW 1

Report #	Description
7/0500/24	Derivation of Loads for Asymmetric Flight Cases
7/0500/25	Wing Airload for Asymmetric Flight Cases 15, 16 and 17
7/0500/26	Comparison of Rigid Wing with Elastic Wing Loads
7/0500/27	Rigid Wing Landing Case Loads
7/0500/28	Sample Problem - Trapezoidal Structure Elements
7/0500/29	Thermal Analysis
7/0500/30	Comparison of Free Flight and Test Heating
7/0500/31	Comparison of the Accuracy of the Finite Difference and Analytical Method in the Problem of Linear Heat Transfer
7/0510/17	Loads System anti-Symmetric Case
7/0511/2	Structural Requirements for the UHF Annular Slot Antenna
7/0525/9	Unit Analysis of J75 Engine Reactions
7/0525/10	J75 Engine Reactions
7/0525/11	Engine Rail Analysis
7/0533/3	Drop Tank Release System
7/0551/1	Structural Requirements C-105 Radome
7/0552/49	Frame - Longeron - Inter Action
7/0558/84	Preliminary Loading Centre Section
7/0558/85	Heat Exchanger Cut-Outs
7/0562/65	Upper Inner Longeron and Dorsal Attachments
7/0562/66	Rib. No. 4 - Main Spar to Auxiliary Spar
7/0562/67	Fuselage Side Rib and Front Spar to Auxiliary Spar
7/0562/68	Tank Pressure Strength Summary

Report #	Description
7/0564/41	Outer Wing Technical Data Sheets
7/0564/42	Loads, Bending Moments, Shears and Torques
7/0564/43	Trailing Edge Loading Analysis
7/0564/44	Outer Wing Slinging Points
7/0574/3	Aileron Buzz Dampers
7/0574/4	Aileron Control System Tie Rod
7/0574/5	Aileron Control System Steel Bell Cranks
7/0574/6	Safe Life Limit for Aileron and Rudder
7/0582/2	Elevator Buzz Damper Mechanism
7/0583/13	Fin Trailing Edge Skins
7/0583/14	Fin Rib 92.25
7/0583/15	Fin Ribs (1) In Detachable Trailing Edge (2) In Main Structure (3) Holes etc. for Fair Leads
7/0583/16	Matrix Ribs
7/0583/17	Fin - Engineers Bending Distribution
7/0583/18	Fin and Rudder Loads for Design Cases

ARROW 2

Report #	Description
7/0511/200	Installation of Electrical Equipment
7/0514/200	Engine Controls P.S. - 13
7/0516/200	Fuel System
7/0522/200	Air Conditioning and Pressurization
7/0532/200	Hydraulic System - Flying Controls, Utility and Accessory Drive
7/0551/200	Radar Nose
7/0552/200	Forward Dorsal Fairing - Station 268 - 317
7/0554/200	Air Conditioning Tray Station 255 - 315
7/0554/201	Transition Duct Station 255 - 291
7/0554/202	By-Pass Door and Mechanism Cut-Outs in Intake Duct
7/0554/203	Compressor Outlet Duct Attached to Dorsal Station 291
7/0554/204	Air Conditioning Bay Side Walls Station 255 - 315
7/0554/205	Fuel Tank
7/0554/206	Deflector Shield
7/0554/207	Engine Bleed Pipe Support Structure
7/0554/208	Sparrow Pack Pick-up Structure
7/0554/209	Upper Access Panel Station 268 - 292
7/0554/210	Equipment Bay Formers
7/0554/211	Former Tubes and End Fittings
7/0554/212	Load and Stress Distribution from General Aircraft Analysis
7/0554/213	Heat Exchanger Exhaust Duct
7/0554/214	Transition Duct
7/0554/215	Tank Bay Formers Full Fuel Case
7/0554/200	Frame 214
7/0554/201	Frame 228
7/0555/202	Frame 237

Report #	Description
7/0555/203	Frame 246
7/0555/204	Frame 207
7/0556/200	External Drop Tank - Attached Loads
7/0556/201	External Drop Tank Attachment Structure Detail Stressing
7/0556/202	Floating Duct
7/0556/203	Oil/Air Heat Exchanger Installation
7/0556/204	Heat Exchanger Structure
7/0556/205	Gill Structure
7/0556/206	Former 562
7/0556/207	Pressure Regulator Access Door
7/0562/208	Light Formers 543 - 557
7/0558/233	Intercostal Beam and Transverse Diaphrams
7/0558/234	Starter Shroud
7/0558/235	Restrictor
7/0558/236	Wing to Fuselage Joint Station 697-742
7/0558/237	Engine Door Fairing
7/0558/238	No. 2 Engine Door Restrictor
7/0558/239	Service Doors No. 1, 2 and 3
7/0558/240	Starter Door
7/0558/241	Cross Beams Station 606 and 625
7/0559/200	Loadings
7/0559/201	Section Properties Longeron End Loads and Shear Flows
7/0559/202	Frame 753
7/0559/203	Frame 803
7/0559/204	Light Formers 758-798
7/0559/205	Skinning
7/0559/206	Longerons
7/0559/207	Centre Box Station 742.5 - 753
7/0559/208	Engine Door Station 742.5 - 803
7/0559/209	Rudder and Dorsal Fairing
7/0559/210	Latches
7/0559/211	Tail Cone Formers
7/0559/212	Tail Cone Skinning
7/0559/213	Tail Cone Longerons
7/0562/200	Front Centre Engine Mount
7/0562/201	Inboard Engine Mount
7/0562/202	Rib. No. 5
7/0562/203	Rib. No. 6
7/0562/204	Rib. No. 7
7/0562/205	Rib. No. 8
7/0562/206	Rib. No. 9
7/0562/207	Inner Wing Spar Shears
7/0562/208	Auxiliary Spar
7/0594/30	Preliminary Telescopic Link Sizing Sparrow 2 Installation
7/0594/31	Missile Operating Mechanism Carrier Preliminary Sizing Sparrow 2 Installation

Report #	Description
7/0594/32	Telescopic Link Stress Analysis
7/0594/33	Missile Operating Mechanism - Carrier Stress Analysis Sparrow 2 Installation
7/0594/34	Sparrow 2 Drag Link Stress Analysis
7/0594/35	Sparrow 2 Missile Pack Load Distribution and Deflection Analysis
7/0594/36	Sparrow 2 Missile Pack Trunnion Mount Deflection and Analysis
7/0594/37	Sparrow Missile Pack Parts Lists, Margins of Safety and Stress Report Reference

31.1.6 STRUCTURAL STRENGTH TESTS

No formal, or reproducible structural strength test reports are available, but some reporting has been compiled on "Advance Test Result Sheets" for use within the Engineering Departments of AVRO aircraft.

31.1.7 ELECTRICAL LOAD ANALYSIS

Report #	Description	Issued
E2	Estimated Power Loads for CF-105 Aircraft	Oct. 12/55
E3	Estimated power loads for CF-105 Aircraft using Sparrow Missiles	Oct. 12/55
P/System/8R	Load Analysis & Power System Avro ARROW 2	Mar. 27/57
P/System/42	Load Analysis ARROW 1 Aircraft 1, 2 & 3	June 5/57
P/System/49	Load Analysis ARROW 1 Aircraft 4 & 5	July-Aug/57

31.1.8 WEAPON SYSTEM ANALYSIS

NO REPORTS AVAILABLE

31.1.9 AIRCRAFT GROUND AND FLIGHT TEST

No ground or flight test reports have been prepared.

31.1.10 FUNCTIONAL TYPE TESTS

Each item of equipment procured to an AVROCAN equipment specification will undergo qualification testing. All functional type test data and qualification test reports for bought-out equipment will be indexed under AVRO drawing numbers and retained in AVRO'S Central Engineering Files.

On satisfactory completion of tests to the AVROCAN equipment specification, an approval statement is issued to the RCAF. Up to date approximately 535 equipment approval statements have been issued. Approval of the remaining 40% of the current equipment ordered, should be issued by December 1957, with approval, for at least the first aircraft flight.

31.1.11 VENDOR'S REPORTS

Vendor's reports on equipment supplied to AVRO, for use in the ARROW aircraft, will be retained on file at AVRO. Their use will be required in the preparation of the equipment approval statement issued for each item of equipment procured to an AVROCAN specification. An index of these reports will not be issued.

31.1.12 ASTRA 1 SYSTEM

NO AVRO REPORTS AVAILABLE

31.1.13 SYSTEMS

Report #		Description	Issued
CF-105 Brochure L. P. -1	ARROW 1	Low Pressure Pneumatic System	Feb. 1956
CF-105 Brochure H-1A	ARROW 1	Flying Control Hydraulic System	Feb. 1956
CF-105 Brochure H-2A	ARROW 1	Utility Hydraulic System	Feb. 1956
CF-105 Brochure E-4	ARROW 1	Electrical System	Feb. 1956
CF-105 Brochure O-1	ARROW 1	Oxygen System	Jan. 1956
CF-105 Brochure F-1	ARROW 1	Fuel System	Feb. 1956
CF-105 Brochure F. C. -1	ARROW 1	Flying Control System	Feb. 1956
CF-105 Brochure F. P. -1	ARROW 1	Fire Protection System	Dec. 1955
CF-105 Brochure(no number)	ARROW 1	Protection Against Ice	Feb. 1956
CF-105 Brochure(no number)	ARROW 1	Electronic System	Feb. 1956
CF-105 Brochure(no number)	ARROW 1	Armament Package Concept	Feb. 1956
P/EQUIP/62/1	ARROW 1	Air Conditioning System Complete with data sheets (J75 engine)	Feb. 1956
72/SYSTEM 13/7	ARROW 2	ASTRA 1 System	June 1957
72/AIREQ 25/1	ARROW 2	Engine Installation	June 1957
72/SYSTEM 11/27	ARROW 2	Electrical System	June 1957

Report #	Description	Issued
72/SYSTEM 22/48	ARROW 2 Air Conditioning System	June 1957
72/SYSTEM 18/29	ARROW 2 Low Pressure Pneumatic System	June 1957
72/SYSTEM 23/31	ARROW 2 Fire Protection System	June 1957
72/SYSTEM 20/51	ARROW 2 Protection Against Ice	June 1957
72/SYSTEM 16/21	ARROW 2 Fuel System	June 1957
72/SYSTEM 19/26	ARROW 2 Hydraulics - Utility	June 1957
72/SYSTEM 32/25	ARROW 2 Hydraulics - Flying Controls	June 1957
72/SYSTEM 19/40	ARROW 2 Hydraulics - Armament	June 1957
72/SYSTEM 15/28	ARROW 2 Flying Control System	June 1957
72/SYSTEM 26/8	ARROW 2 Armament System	June 1957
72/ENG PUB/2	ARROW 2 Escape System	June 1957
72/SYSTEM 21/30	ARROW 2 Oxygen System	June 1957

31.2 MAINTENANCE GROUND SUPPORT AND PERSONNEL REQUIREMENT REPORTS

Report #	Description
70/GEQ/1	Ground Support Equipment
70/GEQ/1	Proposal for Ground Support Equipment Demonstration and Evaluation Conferences
70/MAINT 00/1	Aircraft Jacking Requirements
71/MAINT 00/2	ARROW 1 Preliminary Maintenance Schedule
71/MAINT 11/2	Maintenance Instructions - Lighting Electrics
71/MAINT 11/9	Maintenance Instructions - Engine Services Electrics
71/MAINT 11/3	Maintenance Instructions - Electrics - Windscreen and Canopy De-Icing
71/MAINT 13/1	Maintenance Instructions - Electronics - J4 Compass
71/MAINT 13/9	Maintenance Instructions - Electronics - Radio Compass
71/MAINT 16/5	Maintenance Instructions - Integral Tank Sealing
71/MAINT 31/1	Maintenance Instructions - Arrester Gear
71/MAINT 92/1	Maintenance Instructions - Main Landing Gear
72/MAINT 00/1	ARROW 2 Accessibility Report
72/MAINT 00/2	Personnel Requirements Data
MAINT 105-15-6	Lubrication of Flying Controls - Link Rod Bearings
MAINT 105-01	Pilot's Orders - Ground Checks Required
LOG/105/1	Preliminary Requirements Analysis of Operational Ground Facilities
LOG/105/2	Preliminary Description of Major Ground Equipment
LOG/105/3	Towing from Nose Landing Gear
LOG/105/4	Automatic Disconnect Couplings
LOG/105/5	Estimate of Electrical Power required for the Ground Power Unit

Report #	Description
LOG/105/7	Ground Installations - 28v DC Power Supply and Interphone
LOG/105/8	Mobile Ground Power Units
LOG/105/9	Alert Shelters, General Requirements
LOG/105/10	CF-105 Access Door Investigation
LOG/105/13	CF-105 Environmental Requirements for Mobile Ground Power Equipment
LOG/105/14	Removal and Installation of Trailing Edge Control Boxes of Flying Control Mechanism
LOG/105/15	CF-105 Systems Maintenance
LOG/105/16	Evaluation of the Method and Time required to carry out an Engine Change on the CF-105
LOG/105/18	CF-105 Refuelling and Defueling
LOG/105/19	Maintenance Features of Rubber Fuel Cells
LOG/105/20	CF-105 Undercarriage
LOG/105/24	Evaluation Study of Proposed Mobile Ground Power Equipment
LOG/105/25	Report on Visits to W. A. D. C. & Pratt & Whitney - 6, 7 March and 13, 14 March 1956
LOG/105/26	Supplementary Information on the Solar T-300J-2, Gas Turbine Air Compressor
LOG/105/29	Description and Maintenance Instructions - General Electric Constant Speed Drive
LOG/105/30	Maintenance Operations during Engine Running
LOG/105/32	CF-105 Maintenance Testing - Hydraulic Systems and Pumps and Fuel Booster Pumps
LOG/105/33	CF-105 Hydraulic Systems Fluid Dispenser
LOG/105/34	CF-105 Engine and Gearbox Oil Dispenser
LOG/105/36	Preliminary Study of Proposed Armament Storage
LOG/105/38	CF-105 Noise (Report on Jet Noise Symposium June 23, 1956)
LOG/105/39	CF-105 Cockpit Pressurization Tests
LOG/105/40	Air Conditioning System Ground Test
LOG/105/43	CF-105 Engine Handling Equipment
LOG/105/44	Notes on Proposed Design of CF-105 Armament Pack Hoist/ Transport Trailer
LOG/105/45	Sparrow 2 Loading Transporter
LOG/105/46	Air Conditioning Ground Test Panel
LOG/105/47	Maintenance Accessibility Survey
LOG/105/48	CF-105 Complete Aircraft Sling
LOG/105/49	Various Methods of Defueling the CF-105 and the Requirement for Pressurized Air
LOG/105/50	CF-105 Runway Surface Requirements

REFERENCES

Ref. No.	Title	Report No.	Date
1	Monthly Weight and Balance Report	7-0400-34 Issue 12	
2	Monthly Weight and Balance Report	7-0400-44 Issue 10	
3	Damping System Development	P/Stability/137 page 18a	
4	Damping System Development	P/Stability/137 page 18c	
5	Temperature of Centre Rear Engine Mount	72/THERMO/3	Aug. 1957
6	ARROW 2, ASTRA I System	72/SYSTEMS 13/7	June 1957
7	Interim Supplement Brochure, ARROW 2, ASTRA I System	72/ENG PUB/3	Sept.1957
8	ARROW 2 Electrical System,	72/SYSTEMS 11/27	June 1957
9	ARROW 2 Hydraulic System	72/SYSTEMS 19/26	June 1957
10	ARROW 2 Flying Controls Hydraulic System	72/SYSTEMS 32/25	June 1957
11	ARROW 2 Armament Hydraulic System	72/SYSTEMS 19/48	June 1957
12	ARROW 2 Flying Control System	72/SYSTEMS 15/28	June 1957
13	ARROW 2 Oxygen System	72/SYSTEMS 21/30	June 1957
14	ARROW 2 Escape System	72/ENG PUB/2	June 1957
15	RD84A and B ARROW escape system specifications for rocket sled testing of the ARROW escape system		Sept. 1957
16	A proposed escape system for the ARROW	P/SYSTEMS/45	Aug. 1957