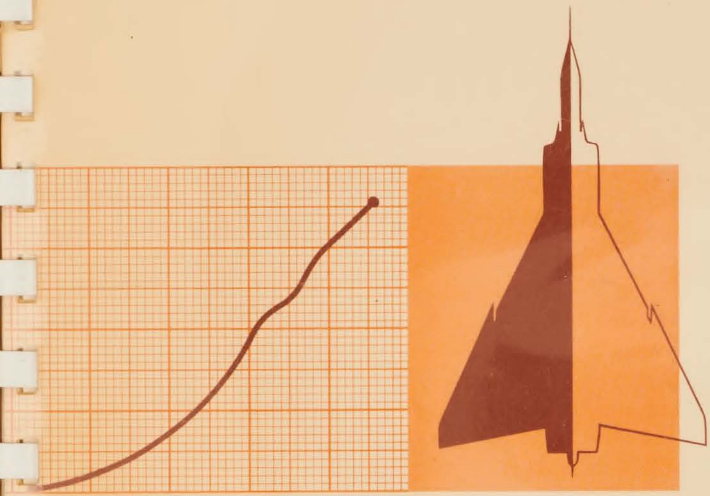
SECRET



AVRO ARROW
quarterly technical report

FOR THE PERIOD ENDING

Dec. 31 1957

AVRO AIRCRAFT
AVRO AIRCRAFT LIMITED

SECURITY WARNING

This document is classified SECRET and must be handled in accordance with the existing regulations pertaining to information classified SECRET published by the Canadian Government.

THE UNAUTHORIZED RETENTION OR DESTRUCTION OF THIS DOCUMENT IS PROHIBITED.

This brochure has been specially prepared for the Canadian Government and its attendant services. The information and data contained herein is the property of Avro Aircraft Limited and the recipient shall not transfer, copy, reprint, divulge or use any portion of it without the prior consent of Avro Aircraft Limited.

This is copy number.........15...........

Issued to......................

Date......................

SECRET

ARROW

QUARTERLY TECHNICAL REPORT

70/ENG PUB/5

FOR PERIOD ENDING 31 DECEMBER 1957

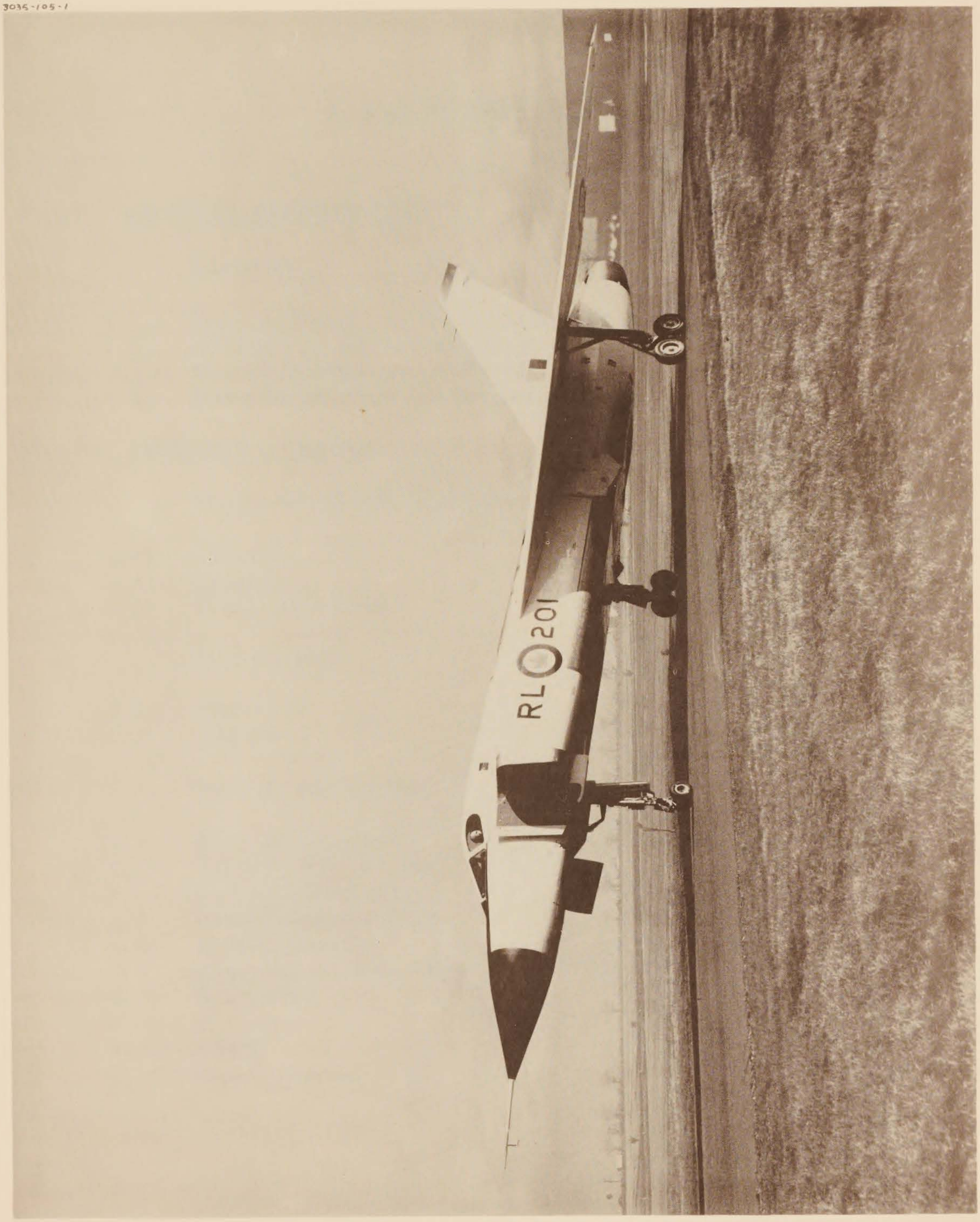
Classification cancelled/changed to UNCLASSIFIED
by authority of .... (date) ....
Signature .... Rank ....

Prepared By: PROJECT MANAGEMENT SERVICES
ENGINEERING DIVISION

AVRO AIRCRAFT LIMITED

MALTON - ONTARIO

3035-105-1



AVRO ARROW 1

TABLE OF CONTENTS

TABLE OF CONTENTS (Cont'd)

TABLE OF CONTENTS (Cont'd)

TABLE OF CONTENTS (Cont'd)

TABLE OF CONTENTS (Cont'd)

TABLE OF CONTENTS (Cont'd)

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (Cont'd)

PART 1

GENERAL INFORMATION

1.0 INTRODUCTION

1.1 SCOPE OF QUARTERLY TECHNICAL REPORT

This is the second Quarterly Technical Report on the AVRO ARROW aircraft project. The report is compiled with the primary object of informing the Canadian Government of technical development of the ARROW project during the three months ending 31 December 1957.

The report presents a description of work carried out and the 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 design, testing and development.

1.2 SUMMARY OF ARROW PROGRAM

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. Issue 4 of this specification has been issued by the RCAF and is in the course of negotiation between the RCAF and AVRO for acceptance.

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 Iroquois turbojets. The ARROW 1 is normally considered as an unarmed aircraft although one aircraft (25203) will carry a weapon pack with simulated air vehicle (SAV) missiles. All ARROW 1 aircraft will fulfill the role of development vehicles, 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.

Production of the first ARROW 1 began early in 1955, and its first flight is expected early in 1958. Manufacture of ARROW 2 details commenced in January 1957 and is proceeding with increasing volume. A considerable amount of design and test work relevant to the ARROW 2 has, of course, been accomplished in connection with the ARROW 1.

1.3 BRIEF DESCRIPTION OF ARROW

1.3.1 ARROW 1

The ARROW 1 carries a crew of two, pilot and flight observer, in a pressurized and air conditioned cockpit which is equipped with two split clam-shell type canopies and automatic upward ejection seats.

The airframe is an all-metal stressed-skin structure and consists of the following major sections: the radar nose, front fuselage, centre fuselage, duct bay, engine bay, rear fuselage, inner and outer wings, elevators, ailerons, fin, rudder and speed 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 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.

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.

1.3.2 ARROW 2

The ARROW 2 is an all weather, day or night interceptor and is the production version of the ARROW.

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 1 electronic system and replacement of the J75 engines with the Orenda Iroquois engines.

The mechanical proportioner type fuel centre of gravity control system of the ARROW 1 has been replaced by an electrically controlled sequencing system, and provision is made for a jettisonable external fuel tank.

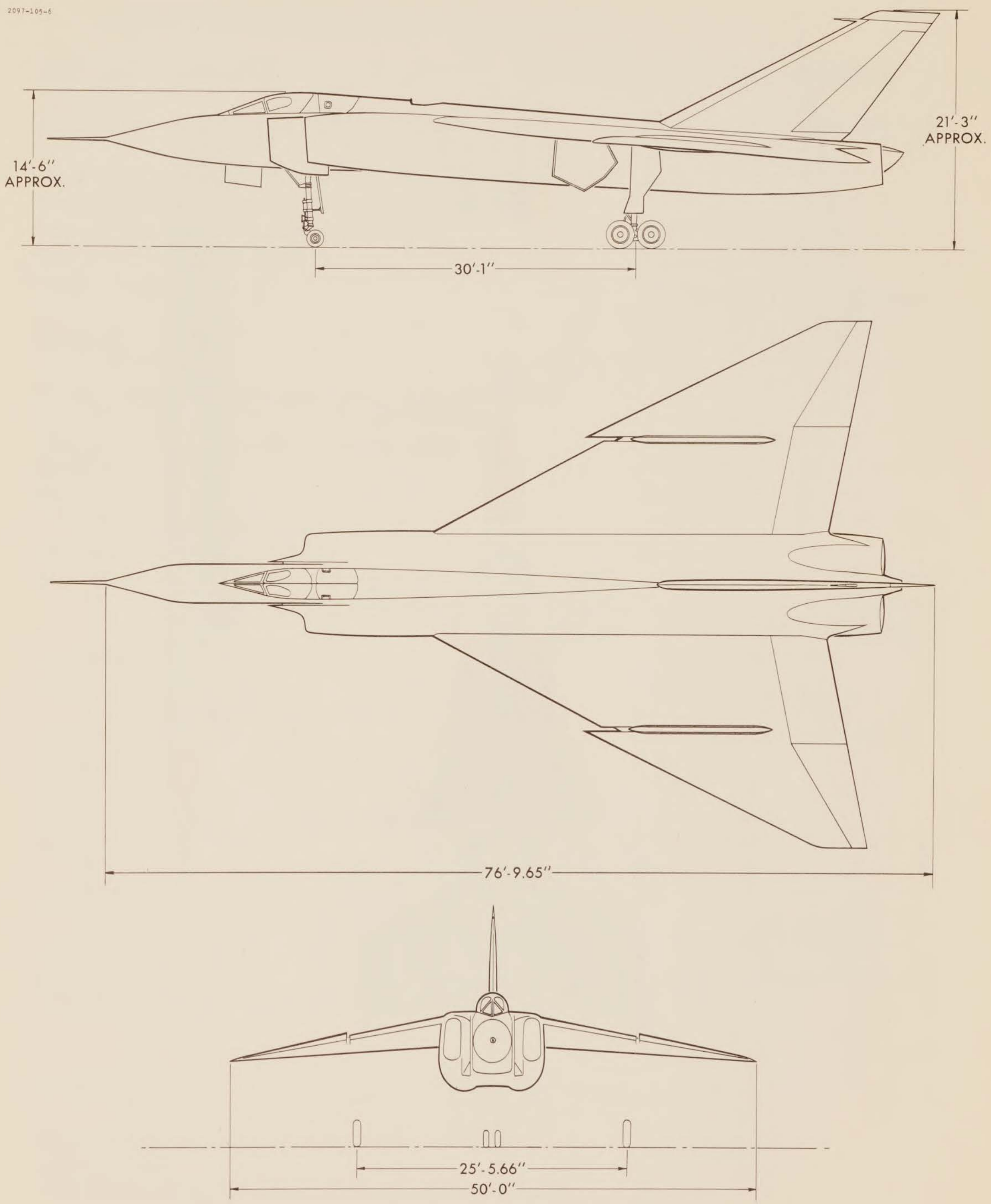


FIG. 1 3 VIEW GENERAL ARRANGEMENT

UNCLASSIFIED

2456-105-3

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

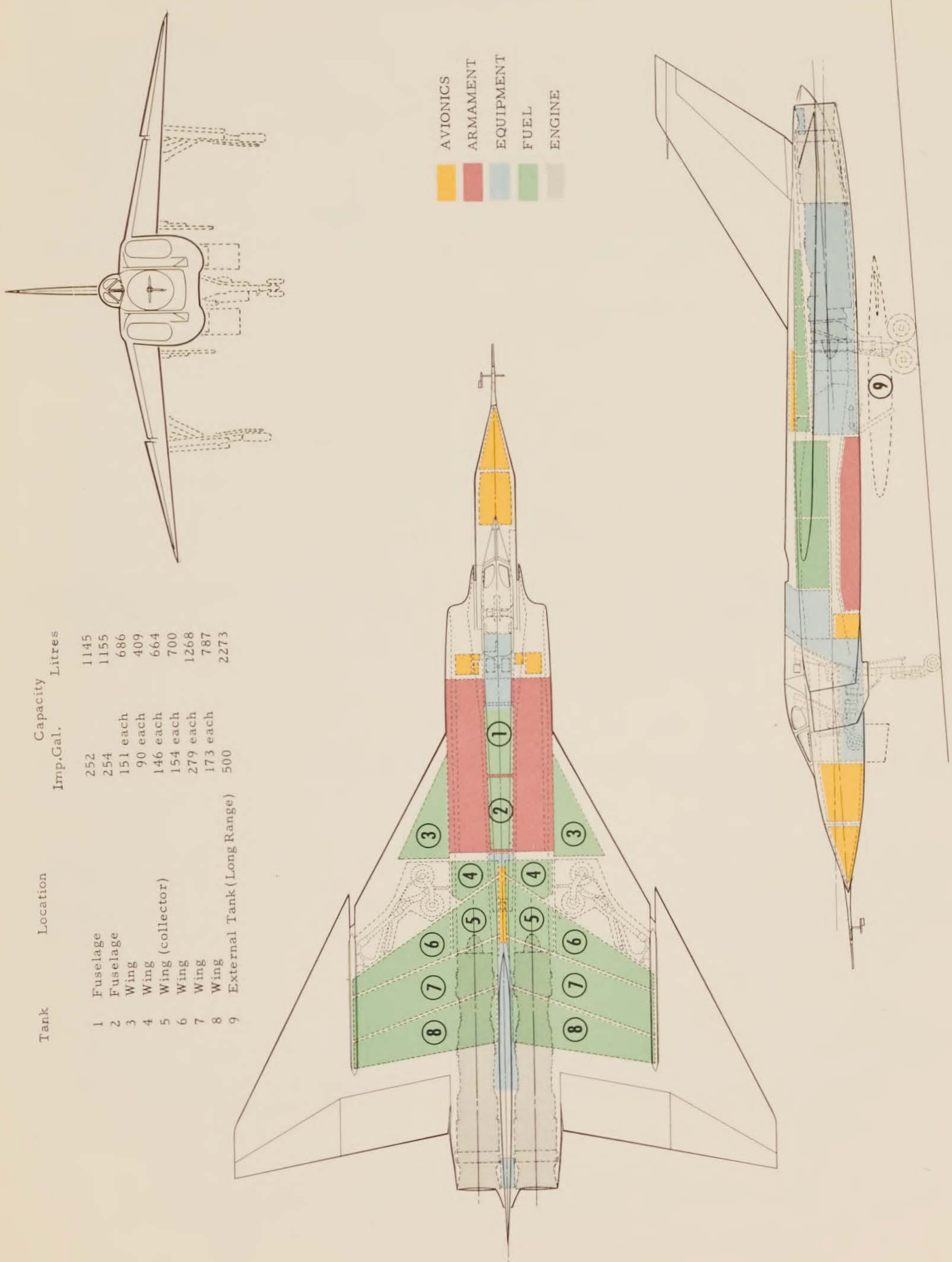


FIG. 2 EQUIPMENT ZONES

1.4 FIXED DIMENSIONS AND GENERAL DATA

CHARACTERISTICS:	ARROW 1 and ARROW 2
Length of Aircraft (excluding probe) -	77 ft 9.65 in (See note 1) 76 ft 9.65 in (" " 2)
Height of aircraft over highest portion of fin	21 ft 3.0 in
Ground angle (Angle between aircraft reference line and ground static line)	4.55 degrees
Tread of main wheels	25 ft 5.66 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	106.90 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 0.0 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 degrees
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

CONTROL SURFACES AND CORRESPONDING CONTROL MOVEMENTS

CHARACTERISTICS:	ARROW 1 and ARROW 2 Surface Movement	Control Movement
Ailerons: Up and Down	19°	4.98 in
Elevators: Up	30°	Aft. 6.63 in
Down	20°	Fwd. 4.37 in
Rudder: Left	30°	Fwd. 3.28 in
Right	30°	Aft. 3.03 in
Speed Brakes	60°	-

Note 1. Aircraft 25201, 25202, 25203

Note 2. Aircraft 25204 and subsequent aircraft.

2.0 WEIGHT AND CENTRE OF GRAVITY

2.1 GENERAL

2.1.1 WEIGHT REPORTING

Monthly weight reports are issued for the first ARROW 1 aircraft (25201) and the ARROW 2 production aircraft. Upon submission of the final weight and balance report for aircraft 25201, monthly weight reports will be prepared and issued for the subsequent ARROW 1 aircraft.

2.1.2 ACTUAL WEIGHING

The first aircraft has been weighed three times, using a Cox and Stevens electronic weighing kit. The aircraft was first weighed to supply weight and centre of gravity data for the first ground vibration tests. The two subsequent weighings provided centre of gravity data and checked the fuel load for the ground running tests and the low speed taxi trials. The weight and centre of gravity data determined from these weighings has checked closely with the data given in the monthly weight report.

The structural test aircraft has been weighed to provide the structural test department with weight and centre of gravity data. This information was required to permit the calculation of ballast requirements for pre-test loading of the structure.

2.2 WEIGHTS

A tabular statement of weight and a weight history chart are given for the ARROW 1 (Fig. 3) and ARROW 2 (Fig. 4). Both charts and tables are based on the monthly weight reports. Weight accounting, as used in the footnotes to the tables, refers to recorded weight changes due to revised weight estimates, based on production drawings, availability of actual weights or new weight quotations from vendors, and minor design changes.

2.3 CENTRE OF GRAVITY

2.3.1 ARROW 1 - AIRCRAFT 25201

The extreme aft centre of gravity (C. G.) position is limited to 31% of the mean aerodynamic chord (MAC) by the use of ballast. The extreme forward C. G. occurs at approximately 29.73% MAC under these conditions.

For first flights, the aircraft will be ballasted to limit the aft C. G. to 30% MAC.

2.3.2 ARROW 2

The in-flight C.G. of the ARROW 2 will be controlled by an automatic fuel management system. The fuel management system is dependent on the fuel tank draining sequence, and since a sequencing order has not yet been approved, a C.G. envelope for the ARROW 2 is not presently available.

The extreme points of C.G. travel are as follows:

Extreme forward C.G.	27.70% MAC
Extreme aft C.G.	30.35% MAC

Since the C.G. of the maximum internal fuel load is at approximately 30% MAC, it is reasonable to assume that a fuel sequencing order which limits the aircraft C.G. to a forward position of 28% MAC, and an aft position of 31% MAC, is possible without resorting to the use of ballast.

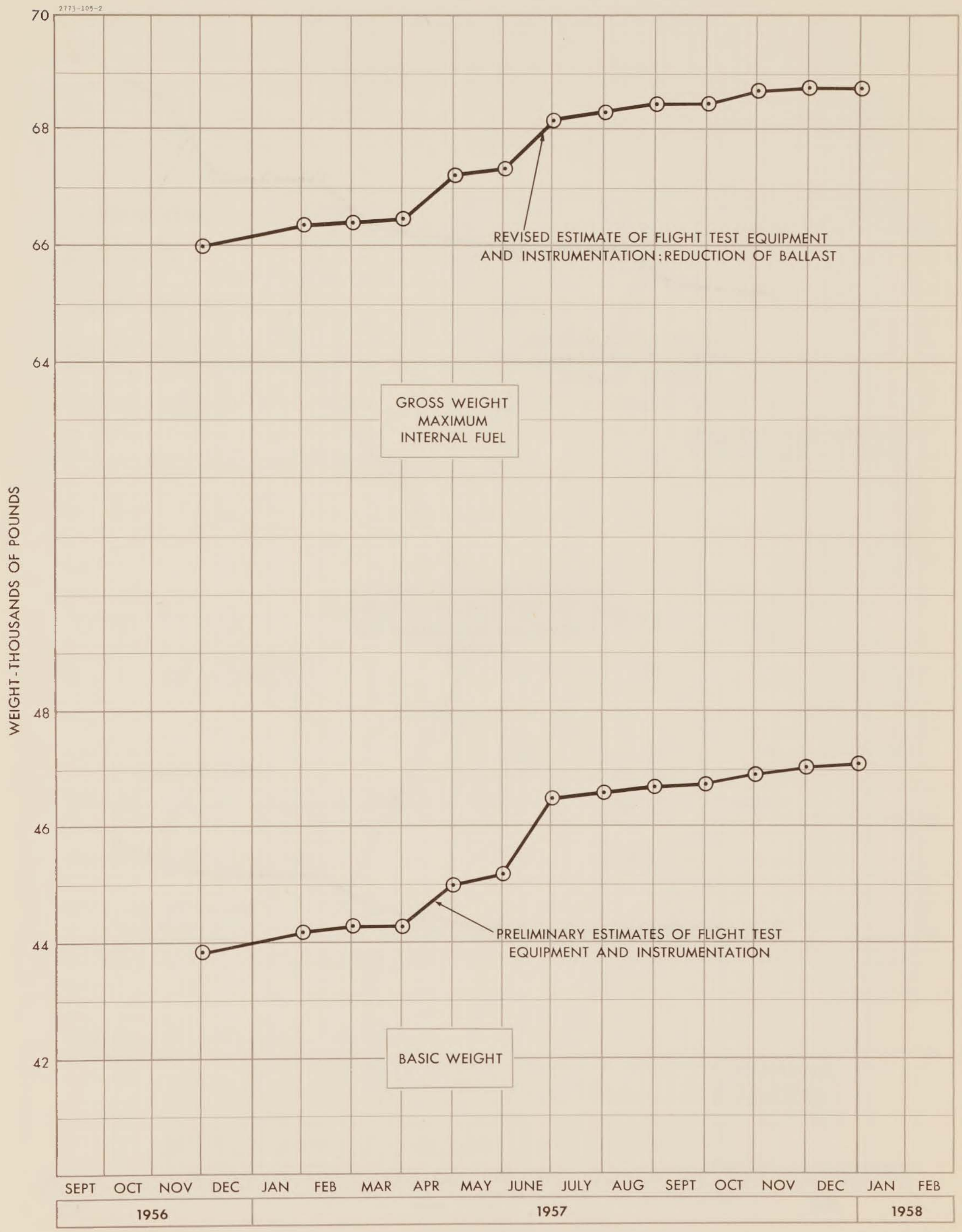


FIG. 3 WEIGHT HISTORY-ARROW 1, AIRCRAFT 25201

UNCLASSIFIED

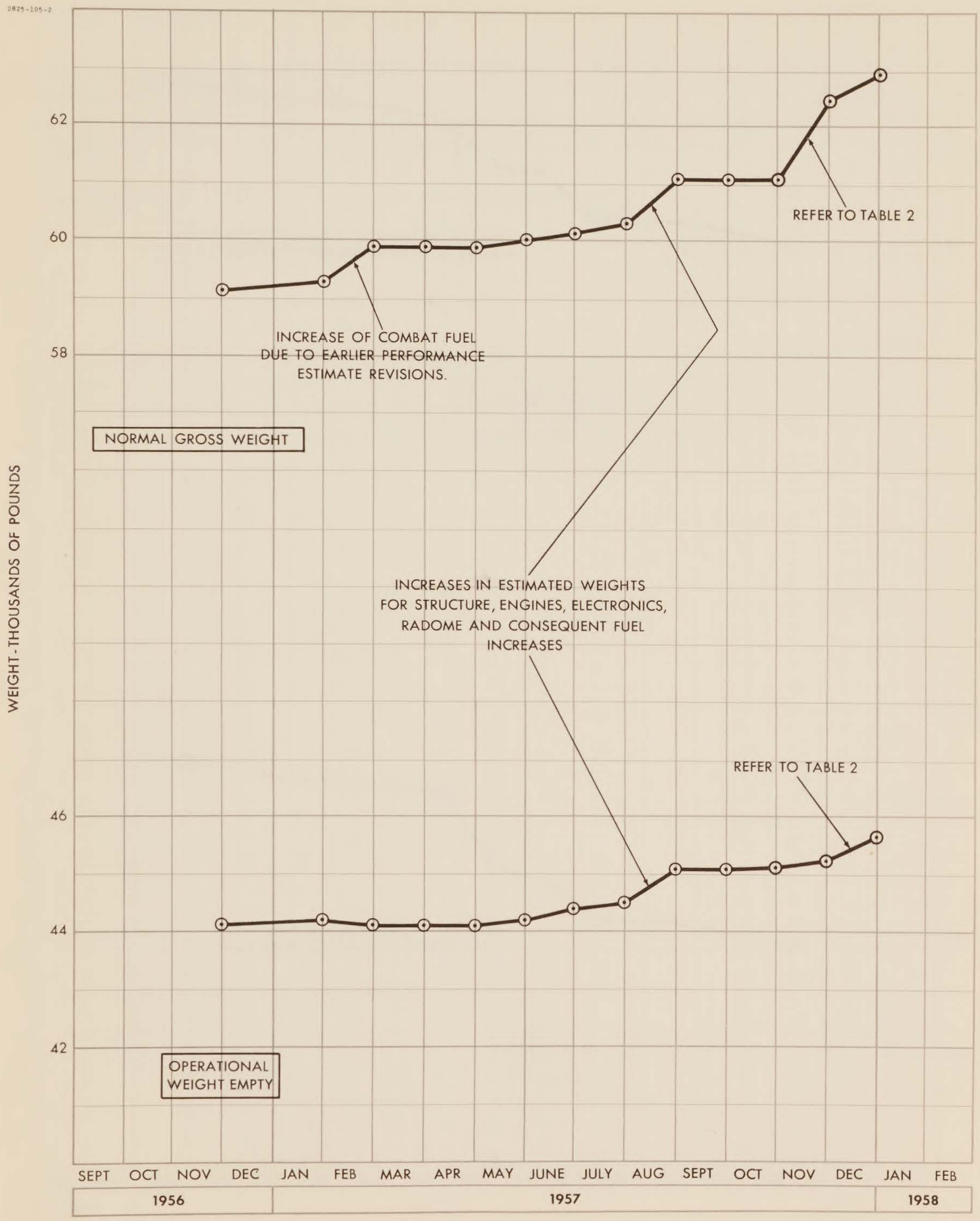


FIG. 4 WEIGHT HISTORY-ARROW 2 (OPERATIONAL AIRCRAFT)

TABLE 1 - STATEMENT OF WEIGHT

ARROW 1 - AIRCRAFT 25201

	Weight - Pounds			
	Present	Previous	Change	Notes
Structure	18606	18531	+ 75	(a)
Landing gear	2601	2610	- 9	(a)
Power plant and services	14392	14366	+ 26	(a)
Flying controls	1853	1857	- 4	(a)
Fixed and removable equipment	9627	9412	+215	(a) +175 (b) + 16 (c) - 9 (d) -100 (e) +133
BASIC WEIGHT	47079	46776	+303	
Useful load (less fuel)	921	983	- 62	(f) - 40 (c) - 22
Ballast	905	959	- 54	(g) - 54
OPERATIONAL WEIGHT EMPTY	48905	48718	+187	
Maximum internal fuel	19843	19843	-	
ALL UP WEIGHT - MAXIMUM INTERNAL FUEL	68748	68561	+187	

Notes:
(a) Weight accounting
(b) Introduction of emergency ram air turbine system.
(c) Deletion of radome de-icing system.
(d) Weight accounting - Doppler equipment weight deleted.
(e) Additional flight test equipment introduced.
(f) Weight accounting - crew parachutes now included with seats.
(g) Ballast adjusted to limit aft C.G. to 31% MAC.

TABLE 2 - STATEMENT OF WEIGHTS

ARROW 2 - OPERATIONAL AIRCRAFT

	Weight - Pounds			
	Present	Previous	Change	Notes
Structure	19044	18896	+ 148	(a)
Landing gear	2542	2552	- 10	(a)
Power plant and services	10800	10800	-	
Flying controls group	1793	1791	+ 2	(a)
Fixed and removable equipment	8641	8271	+ 370	(a) + 360 (b) + 19 (c) - 9
BASIC WEIGHT	42820	42310	+ 510	
Useful Load (less fuel)	2789	2851	- 62	(d) - 40 (c) - 22
OPERATIONAL WEIGHT EMPTY	45609	45161	+ 448	
Normal combat mission fuel	17370	15940	+1430	(e) +1430
NORMAL COMBAT WEIGHT	62979	61101	+1878	

Notes:
- (a) Weight accounting
- (b) Electrical power system changed from Lucas-Rotax to Westinghouse system.
- (c) Deletion of radome de-icing.
- (d) Weight accounting - crew parachutes now included with ejection seats.
- (e) Normal combat mission fuel load changed due to revised performance estimates based on new engine performance data.

3.0 PERFORMANCE

3.1 ARROW 1

A revision of the ARROW 1 performance estimates was necessitated by the change in the ejector size from 39 in. to 45 in. and the need for a revised form of performance presentation for the ARROW 1, in its role as test vehicle. This performance estimate is based on drag data from Periodic Performance Report No. 9, and revised J75 installed engine data. Complete graphical details are contained in Periodic Performance Report No. 11, October 1957.

The performance shown is for an aircraft having J75 engines with 45 in. divergent ejectors, a 4° up aileron deflection above 45,000 ft., C.G. at 29.5% MAC and ICAO standard atmospheric conditions. It should be noted that the Model Specification performance figures differ from these shown here since they are based on 0° aileron deflection at all altitudes. The performance estimates with aileron deflection are given since it is planned to fix the ailerons in this position for the high altitude portion of the test program.

Aircraft weights used in the performance estimates are from Weights Report No. 7-0400-44, Issue 10, October 1957, suitably modified for the larger ejector. These weights include test equipment and ballast to limit the aft C.G. travel to 31% MAC.

The first flights of the ARROW 1 will be made with the 39 in. ejector as the 45 in. ejector will not be available at the time.

3.1.1 DRAG DATA

No drag data revision has been made since Periodic Performance Report No. 9 was issued. Changes in aircraft weight have almost no effect on drag calculations, since in this case, they are based on coefficients of lift.

The increase in ejector diameter contributes to the reduction of "boat tail" drag, as indicated in area rule studies. Boat tail drag results from the projected area between the tangent to the slope of the engine nacelles, and the periphery of the ejector. This is the area over which the boat tail drag coefficient acts. The area rule theory is applicable only at supersonic speeds. However, the effect of the jet stream in this area renders the drag calculations dubious.

3.1.2 PROPULSION DATA

As a result of the change from a 39 in. cylindrical ejector to a 45 in. divergent ejector, the J75 installed engine data has been revised. The larger ejector contributes to the reduction of boat tail drag and results in some increase in thrust at high speeds. This will allow a larger test regime of altitude and speed for the ARROW 1. In addition, the increased ejector diameter reduces the size of the ARROW 1's stinger. The revised propulsion data, with the exception of afterburner-off thrust characteristics, is based on the ejector pumping and thrust characteristics experimentally determined by the Orenda Engines Limited test facility at Nobel. The afterburner-off characteristics were obtained from NACA experimental data as the Nobel results showed too much scatter to correlate. The reason for scatter in these results from Nobel could only be conclusively determined through an extensive model testing program. It is thought, however, that the constriction in the bypass, followed by excessive expansion, causes erroneous pitot readings.

The nozzle geometry used in the tests referred to above, differs slightly from the actual aircraft geometry (i.e. the ratio of primary nozzle diameter to ejector throat diameter) particularly in the afterburner-off cases. This leads to some doubt as to whether the discharge flow is attached or detached, particularly at M = .92 at 40,000 ft. At low primary nozzle pressure ratios, the exhaust gases detach from the wall of the ejector, and as a result there is an increase in thrust. The exact pressure ratio for detachment is sensitive to geometry change. From the available model test data it is not possible to determine with certainty whether the flow is or is not attached for one case of subsonic cruise. In the present propulsion estimates the more conservative value obtained with attached flow has been used.

3.2 ARROW 2

A revision of the ARROW 2 performance estimates was made necessary as a result of the change in the operational weight empty of the aircraft and completely revised installed engine thrust calculations. This performance estimate is based on drag data from Periodic Performance Report No. 9, and revised installed engine data. (See paragraphs 3.2.1 and 3.2.2 below). Complete graphical details are contained in Periodic Performance Report No. 12, November 1957.

The pertinent performance changes since Periodic Performance Report No. 10 are:

Combat g at M = 1.5 and 50,000 feet.......from 1.63 to 1.56

Combat ceiling.......from 63,000 ft. at Mach 2 with afterburner
to 60,000 ft. at Mach 1.8 with afterburner

Combat mission fuel, (200 MM radius).......from 15,672 to
17,270 lb.

The performance shown is for an aircraft having Iroquois Series 2 engines with 49 in. divergent ejectors, a 4^{o} up-aileron deflection over 45,000 ft., CG at 29.5% MAC, and ICAO standard atmospheric conditions.

Aircraft weights used in the performance estimate are from Weight Report No. 7-0400-34 issue 2-12. An increase of 1,497 lb. in the operational weight empty has occurred since Periodic Performance Report No. 10, December 1956.

3.2.1 DRAG DATA

The total drag used in the ARROW 2 performance estimate should prove to be conservative, since no allowance has been made for the possible saving in boat-tail drag. The more conservative drag estimate has been used in this report because of the difficulty in accurately determining the decrease in boat-tail drag.

3.2.2 PROPULSION

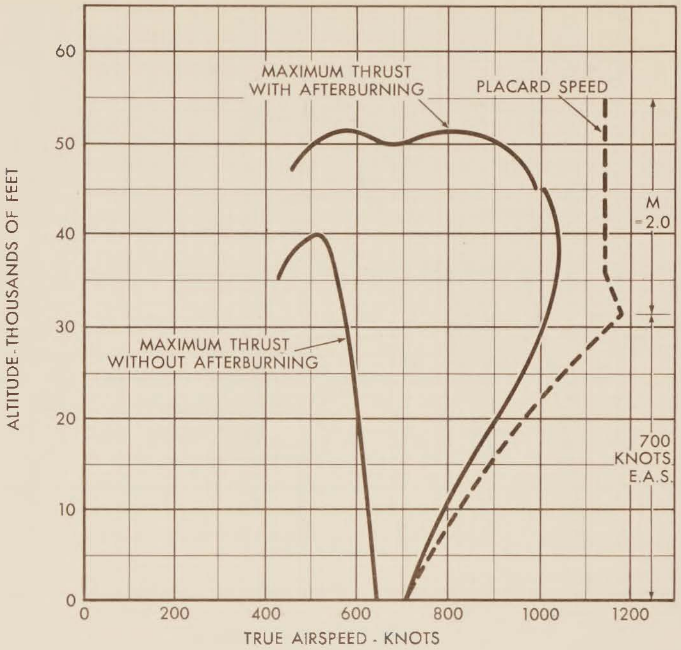
The engine data has been completely revised since Periodic Performance Report No. 10. The engine data used is for a divergent ejector with a final nozzle diameter of 49 in. and a throat diameter of 40 in. The ejector size has been increased to reduce boat-tail drag, eliminate the large stinger (as described for the ARROW 1) and to improve the ejector for higher speeds.

The ejector pumping and thrust characteristics for determining the installed thrust have been taken from NACA experimental data. (See NACA RM E55G21a). The thrust was calculated for various diameter ratios and was plotted to obtain the values for the actual ratio of throat and exit nozzle diameter to jet pipe nozzle diameter.

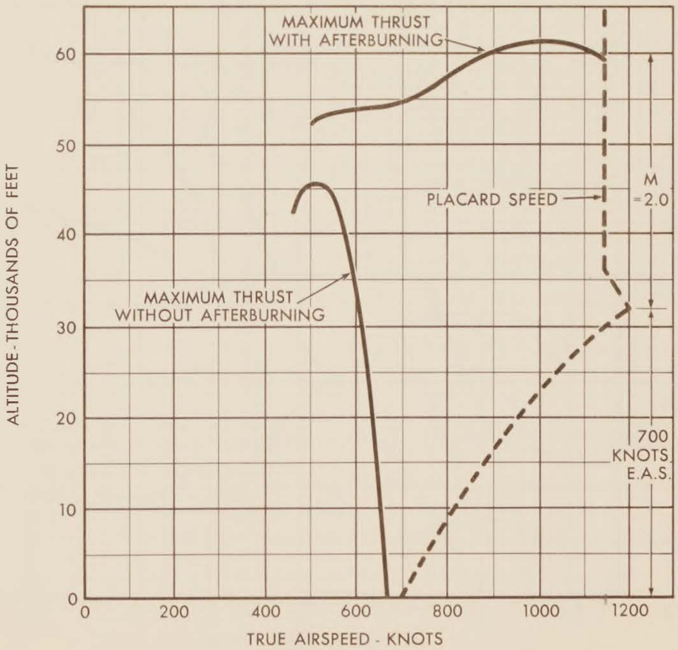
The calculation of installed thrust and fuel flow has been made using the non-dimensional engine curves on which the engine model specification (EMS 8 Issue 2) is based.

The Lucas-Rotax engine control system with a trimming device has been used in the present estimate. This enables both high and low pressure rotors to be operated at maximum RPM throughout the major part of the flight envelope, at both maximum thrust and maximum thrust with after-burner.

3022-103-1



ARROW 1 - AT HALF FUEL WEIGHT

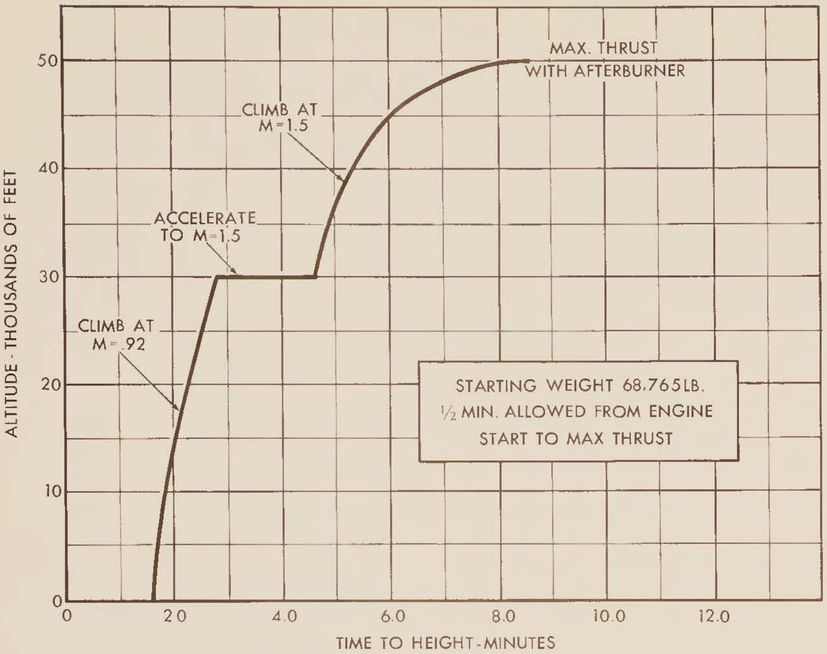


ARROW 2 - AT COMBAT WEIGHT

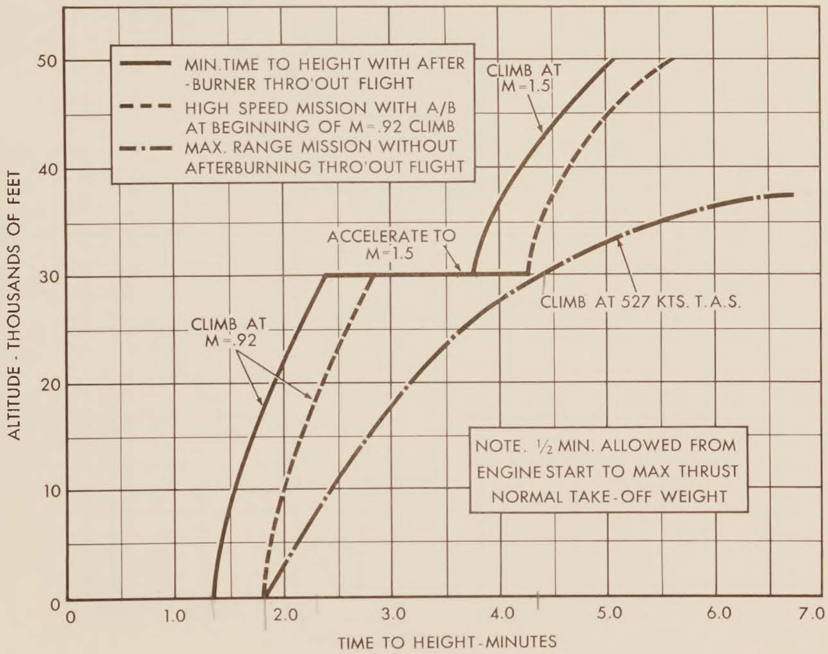
M 1.5
N = 1.5

FIG. 5 MAXIMUM LEVEL SPEED

3025-103-1



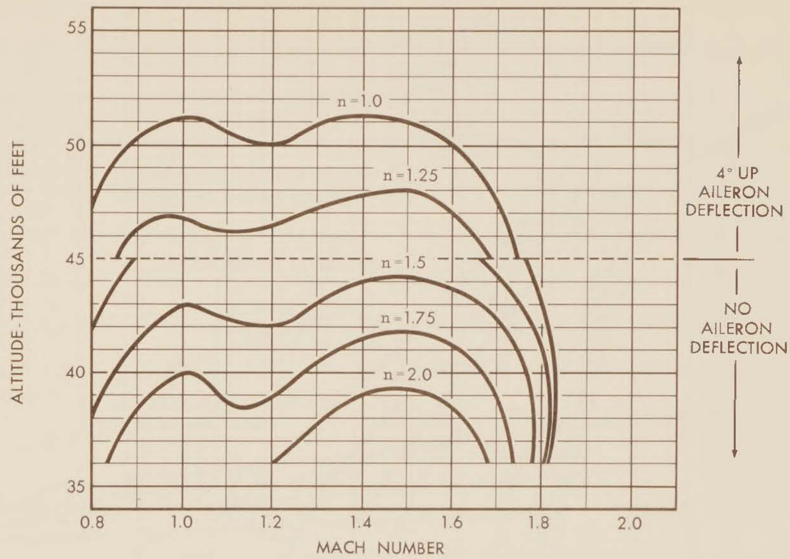
ARROW 1



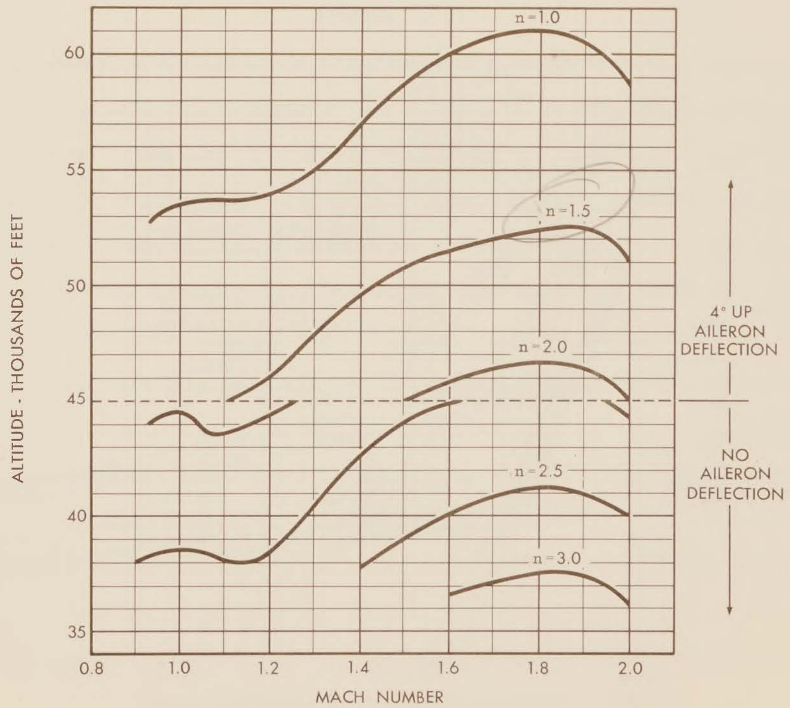
ARROW 2

FIG. 6 TIME TO HEIGHT

3024-105-1

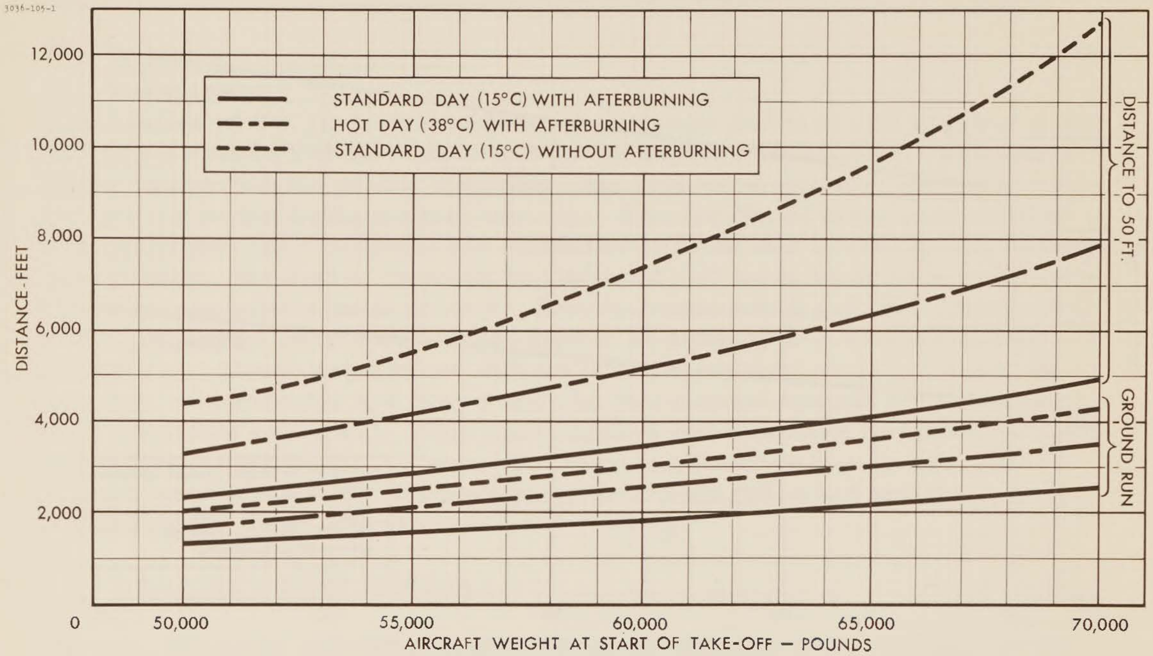


ARROW 1 - HALF FUEL WEIGHT, MAXIMUM THRUST

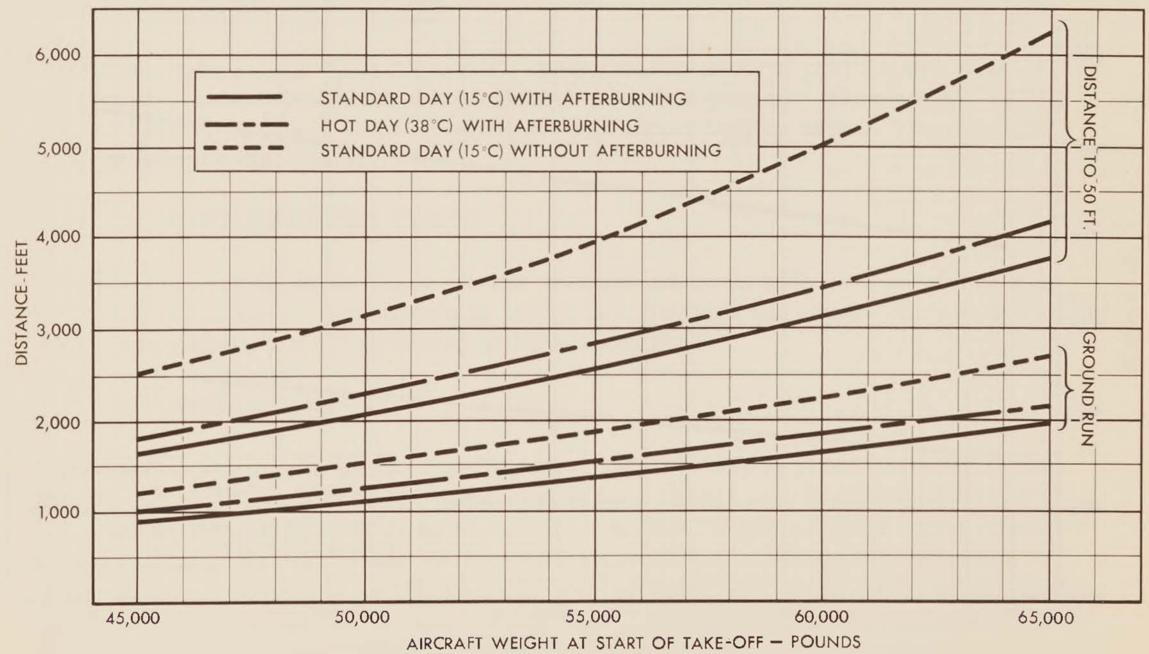


ARROW 2 - COMBAT WEIGHT, MAXIMUM THRUST

FIG. 7 MANOEUVRABILITY



ARROW 1



ARROW 2

FIG. 9 TAKE-OFF DISTANCE AT SEA LEVEL

4.0 STABILITY AND CONTROL

4.1 AIRCRAFT SIMULATOR

Modification of the aircraft simulator is complete and it is now capable of simulating all speeds and all altitudes in seven degrees of freedom. The simulator comprises the analog computer, the damper simulator, and either the cockpit rig or the flying control test rig. The results of tests completed on this equipment have indicated the existence of problems in the control systems. In particular, the feel of controls in pitch and roll were found to be unacceptable from the pilot's point of view. Several combinations of feel springs and break-out forces have been investigated in an attempt to overcome these difficulties. It was determined that the existing system could not meet specification requirements and it was decided that a boost system for the control stick should be evaluated. This boost system has now been installed, in part, on the flying control rig. The preliminary results have been good. Final assessment will be made when all parts have been delivered and the system is completed. (Ref: paras 15.2.2 and 15.3.4).

4.2 RELATIVE WIND SENSOR

Dynamic response and calibration tests on the relative wind sensor have not been performed as yet, due to delays in delivery of the sensor.

4.3 SEAT JETTISON TESTS

Jettison tests on the observer/AI's seat (not the pilot's seat as previously reported), have been completed at the NAE wind tunnel. These tests have shown fair agreement with predictions and other known data. The results are still being analyzed.

4.4 CANOPY OPENING TESTS

Canopy opening dynamic cases were calculated to establish the loads on the aircraft structure. It is hoped to clear the aircraft structure for canopy operation when the aircraft is flying at 720 knots EAS.

4.5 RUDDER MONITOR

The two systems, presently mechanized, did not seem to be entirely satisfactory, since each of them appeared to be subject to a number of flight conditions where disengagement occurs, or the ability of the rudder monitor to protect the aircraft from damaging manoeuvres is marginal. These difficulties may be due to the use of some revised values of derivatives that are now known to be incorrect. Study with corrected derivatives is now continuing, and it is hoped that the problem areas will be eliminated.

4.6 STICK-FORCE MODE "GEAR DOWN" INSTABILITY

The instability becomes apparent when the landing gear is down, as a high frequency, low amplitude oscillation of the control column when a control correction force is applied by the pilot. The damping system circuitry is altered for "gear down" operation and this is thought to be responsible for the instability. Relocation of the parallel servo to a forward position on the simulator, combined with a modified gain for the servo, provided a cure to this instability problem. However, further work is required to verify this, under all conditions of damper operation, and to ensure compatibility with damper performance.

4.7 g LIMITER

A number of problems have developed in the g limit function. These problems are being resolved by Honeywell.

4.8 IR SEEKER

Data on aeroelastic effects of the fin and air turbulence were submitted to RCA to permit a dynamic analysis of the IR system.

4.9 AFCS

Block diagrams and gain schedules for the various modes of the AFCS have been obtained from Honeywell and detailed analysis is being undertaken.

4.10 MODEL TESTING

Preparation for post stall gyration testing of the aircraft are in progress. Testing is being performed in two stages. In the first stage, crude, expendable 1/24 scale wooden models will be used to determine techniques and requirements. The second stage of testing will employ the present, more refined, 1/24 scale ARROW spin tunnel model for final analysis.

The free-flight catapult launch method of testing will be employed. A dynamically similar scale model will be launched horizontally into still air from a suitable height so that it will attain a stalled flight condition. The subsequent motion will be observed and recorded photographically. This method has been used successfully by NACA.

It is essential, that sufficient still air space be provided for the successful completion of these tests, to ensure undisturbed flight and the full development of the post stall gyrations. It is estimated that these tests will require an enclosed space of approximately 100 feet square x 75 feet high, with suitable netting. A request was made to the RCAF to supply such facilities, but no space of sufficient height was available. As yet, no other suitable facility has been offered for consideration.

To date, the crude model catapult rail has been completed and the model launching platform is being manufactured. Preliminary crude model launchings are expected to be completed in early 1958.

The basic subsonic spin and recovery characteristics testing program is continuing at the NAE spin tunnel. The program consists of a minimum number of configurations and conditions (i.e. control setting combinations, recovery combinations and various attitudes) that should be investigated for a range of model inertia characteristics. This range covers the empty weight, half-fuel weight and full-fuel weight conditions at 10,000 ft. simulated altitude. Testing at the higher simulated altitudes of 30,000 and 40,000 ft. is considered to be desirable. These tests, however, cannot be performed on the existing 1/24 scale spin tunnel model due to high model wing loadings, which even at the lowest testing altitudes, results in handling difficulties. In addition, at the higher simulated altitudes, the high model wing loadings would result in a sink velocity which would exceed tunnel velocity. The handling problems could be reduced by testing a smaller scale model (of the order of 1/36) in the N.A.E. spin tunnel or preferably in a larger and more powerful tunnel.

A preliminary spin recovery parachute investigation has been completed, but model handling difficulties described above have precluded the optimizing of the anti-spin parachute configuration.

This investigation will continue following the completion of the basic spin test program.

5.0 THERMODYNAMICS

5.1 ARROW 1 THERMODYNAMICS ACTIVITY

During the last quarter, thermodynamics work on the ARROW 1 has been confined to monitoring the results of engine ground running tests. Structural and system temperatures for the sea level static case cannot, in many cases, be calculated with accuracy due to uncertainty of heat transfer coefficients, cooling air flow at low engine RPM, and transient effects. For these reasons, the ground running tests were of importance as a prerequisite to taxi tests.

5.1.1 STRUCTURAL HEATING

A preliminary review of the engine ground running results indicates that the structural temperatures are far below their critical values (taken as 250°F), and that the engines may be successfully idled for reasonably long periods, even during hot weather. More complete testing will be necessary to confirm the initial ground test results.

5.1.2 OIL SYSTEMS HEATING

It was also found during engine ground running tests that the aircraft oil systems took considerably longer to warm up than expected. In particular, the gear box and constant speed drive oils were much cooler than anticipated.

5.2 ARROW 2 THERMODYNAMICS ACTIVITY

Work on the ARROW 2 has, for the most part, been devoted to the study of heating effects (at speeds above Mach 1.5) on the following:

5.2.1 MISSILE HEATING

An analysis of the kinetic heating effects on the Sparrow 2D missiles is underway and means of protection are being investigated. An insulated cocoon which meets the aerodynamic requirements, is under consideration, but installation problems have not yet been resolved. The progress of the program is being hampered by lack of data on the missile components thermal paths and temperature tolerance.

5.2.2 FUEL TEMPERATURES

Fuel temperatures are now being estimated for a few typical combat missions. This work is being done in more detail than was done for the ARROW 1, as ground test results on fuel heating have become available since the first estimates were made. Preliminary results show that the fuel temperatures will not vary appreciably from the ARROW 1 estimates. These results, however, are not yet considered to be sufficiently accurate, to provide the data necessary to determine the fuel over-temperature time in flight.

5.2.3 REAR FUSELAGE TEMPERATURES

Testing of the ARROW 1 has indicated that no overheating is occurring on the rear fuselage structure, but more complete testing will be necessary to confirm the initial ground test results. These results are also applicable to the ARROW 2 rear fuselage structure.

5.2.4 STRUCTURAL TEMPERATURE DISTRIBUTIONS

The transient temperature distributions over the exposed surfaces of the aircraft are being investigated. Methods of analysis and the accuracy of various approximations are being examined in order to reduce the time required for calculations. The relative importance of various input data parameters, such as angle of attack, camber and thickness-chord ratio are being evaluated to determine what assumptions can be made. The missions which will involve the most severe conditions are also being determined.

5.3 ENGINE PERFORMANCE INDICATOR

The engine performance indicator was briefly described in the last Quarterly Report. The following is a more complete description. Full details and related graphs are contained in Report 72/POWER/3, "An Engine Performance Indicating System for the Iroquois with Afterburner", November 1957.

This instrument is to installed in the ARROW 2 to provide the pilot with a means of:

(a) Checking the "integrity" or "health" of the engines.

(b) Selecting and setting power output from the engines on the ground or in flight.

At the present time these functions are performed by JPT (jet pipe temperature) and RPM indicators for single spool engines, with the addition of an engine pressure ratio indicator for twin spool engines. These instruments, however, are not temperature compensated, and strictly speaking, their readings can only be correlated if the ambient temperature and flight conditions are know. In addition, the pressure ratio instrument presents a display which requires interpretation, since high readings are not necessarily related to high power outputs, or low readings to low outputs. With the advent of two-spool engines and higher aircraft speeds, a requirement exists for a new type of instrument which is temperature compensated, related to throttle position and which has a simplified display. The instrument display should indicate the percentage of power being produced in relation to the total amount of power available under the ambient conditions.

Since thrust is not exactly proportional to throttle position, the dial reading is only an approximate indication of percent thrust.

5.3.1 NOTATION

This notation applies to the discussion of theory which follows:

N_H = High pressure rotor speed

N_L = Low pressure rotor speed

T_1 = Compressor inlet total temperature

T_7 = Jet pipe total temperature

P_1 = Compressor inlet total pressure

P_6 = Turbine outlet total pressure

P_7 = Nozzle total pressure

γ = Ratio of specific heats

M^* = Choking mass flow

$\frac{P_6}{P_1}$ = Total engine pressure ratio

$\frac{N_L}{\sqrt{T_1}}$ = "Non dimensional" low pressure rotor speed

$\frac{N_H}{\sqrt{T_1}}$ = "Non dimensional" high pressure rotor speed

A = Nozzle area

The last five items are referred to in the text as engine parameters.

5.3.2 THEORY OF OPERATION

The engine performance indicator has been designed to operate on the basis of a fully choked engine condition (i.e. where exhaust gases at the nozzle are at M = 1). This covers most operating conditions but leads to error during low speed operation, as on landing approach. At low speeds, the instrument will not be as accurate, but will provide the pilot with an indication of response to throttle movement.

With a single spool, constant nozzle area jet engine operating in a choked condition, the engine shaft operates at a speed proportional to the square root of the combustion chamber temperature. This relationship is the basis of the JPT and RPM indicator method of checking engine "health". With the Iroquois engine, allowance must be made for twin spools and, if possible, a variable nozzle area. Fuel to the Iroquois engine is scheduled to give a constant low pressure rotor speed for a particular throttle setting, independent of altitude, Mach number or ambient temperature In the development of this instrument it was desirable to measure the low pressure rotor speed through other engine parameters, and relate it to throttle angle. By calculating the low pressure rotor speed in this way, a check is provided on the functional integrity of the engine. With a variable nozzle area, only two engine parameters are needed to determine all the remaining parameters. If nozzle area is considered to be constant, only one parameter is necessary. These relationships can be shown as:

$$\frac{N_L}{\sqrt{T_1}} = f_1\left(\frac{N_H}{\sqrt{T_1}}, A\right) \quad (1)$$

and

$$\frac{N_L}{\sqrt{T_1}} = f_2\left(\frac{P_6}{P_1}, A\right) \quad (2)$$

(For dimensional identity in these equations and following equations,

$\sqrt{T_1}$ must be multiplied by $\sqrt{\gamma R g}$

where R is the gas constant for air (53.3)

The first equation is the more accurate for calculating N_L, since there is one less variable, and N_H is easier to measure accurately than the pressure ratio in equation (2). In addition, the parameter of $N_H/\sqrt{T_1}$ eliminates the need for probes in the intake ducts required in the measurement of P. These advantages indicate the desirability of developing an instrument using equation (1) as the measured parameter. However, the pressure ratio type of instrument is in an advanced stage of development, and, with temperature compensation, can be adapted to give satisfactory results.

If the area (A) in equation (2) is considered to be a constant, the loss in accuracy is small in the case of the Iroquois. For most engine conditions, the area is very close to constant, at 668 sq. in. but it increases under certain conditions by approximately 5%. By assuming the area to be at maximum value at all times, the instrument can be made to read correctly at maximum throttle setting for all flight conditions. This assumption causes the instrument to read too high for some throttle settings. The error is acceptable however, since it is small or zero during equilibrium flight and climbs, or accelerations at maximum power.

Equation (2) can be re-written to state approximately:

$$\frac{N_L}{\sqrt{T_1}} = f_2\left(\frac{P_6}{P_1}, \text{A max.}\right)$$

but A max. is a function of T_1, therefore

$$\frac{N_L}{\sqrt{T_1}} = f_3\left(\frac{P_6}{P_1}, T_1\right)$$

$$= f_4\left(\frac{P_6}{P_1}\right) \times f_5\left(T_1\right)$$

so

$$\frac{N_L}{f_5\ (T_1)\sqrt{T_1}} = f_4\left(\frac{P_6}{P_1}\right)$$

or

$$\frac{N_L}{f_6\ (T_1)} = f_4\left(\frac{P_6}{P_1}\right)$$

These are the engine parameters through which the low pressure rotor speed is computed and converted to throttle angle.

5 3.3 ESTIMATE OF ERROR

Under conditions of high inlet temperatures and low power settings, large errors occur between actual and calculated throttle angle. These are transient errors, however, since under these conditions the aircraft must be decelerating. An important error of 5 degrees between calculated and actual throttle angle does occur at high power setting with high inlet temperatures, as a result of ignoring the parameter of engine area. In this case pressure ratio reaches a maximum below full throttle and there is no further increase in pressure ratio as the throttle is moved to the full position, but thrust is increased due to the same pressure acting over a larger nozzle area. This error can be eliminated only through the measurement of another engine parameter, such as area. An adjustment screw will be provided for ground calibration. This will reduce the error at sea level static, maximum power, to zero.

5.3.4 ADVANTAGES OF THE ENGINE PERFORMANCE INDICATOR

1. A single instrument provides a simple percentage reading which should always be proportional to the throttle angle position. Since the theory is based on a properly operating engine, any malfunctioning will result in a lack of correlation between throttle angle and instrument reading.

2. For the same reasons, any deterioration of engine performance will show during ground running as a decrease from 100%.

3. The pilot may select and set by the instrument any desired thrust setting which will not change with changes in flight condition (except as noted above).

4. High throttle settings correspond to high power outputs and high instrument readings, and vice versa for low settings. This is in contrast to the uncompensated pressure ratio instrument, in which the reverse is generally the case.

5. Nozzle area malfunction is detectable, although not measured directly. A decrease in normal operating area would result in a higher pressure ratio causing the instrument to read over 100%. A reading lower than 100% is apparent when nozzle area is too large, under normal operating conditions.

5.3.5 DISADVANTAGES OF THE ENGINE PERFORMANCE INDICATOR

1. The accuracy of the instrument may deteriorate when the nozzle is operating outside the choked condition.

2. Under certain conditions the instrument tends to over-read. The error is small until high speeds are reached where it may indicate full throttle (90 degrees) when the throttles are actually at 85 degrees.

5.3.6 OPERATION OF AFTERBURNER

Instrumentation to inform the pilot of the afterburner performance is also proposed. As before, it is desirable to have a direct correlation between afterburner throttle position and the power delivered by the afterburner, that is, the degree of afterburning.

5.3.7 THEORY OF INSTRUMENT OPERATION WITH AFTERBURNER OPERATING

During afterburning, extra thrust is obtained by burning additional fuel in the excess air of the engine exhaust. There is no significant increase in total exit pressure or in mass flow. The increase in thrust is the result of the same total pressure acting over an increased exit area sufficient to accommodate the expanded gases.

The relationships may be stated as:

$$M^*\sqrt{T_7} = f(\gamma,\ P_7,\ A)$$

This shows that for M* constant and γ, P_7 constant, A should vary as $\sqrt{T_7}$

If the area is measured and compared to the area for maximum afterburning conditions, a figure is obtained which represents the power being delivered as a percentage of the maximum power available. For the Iroquois engine, the nozzle area required for various degrees of afterburning is proportional to the afterburning temperature. The temperature depends on fuel flow which is controlled by throttle position. The variation of fuel flow with throttle is roughly linear with throttle angle. Under these relationships, a given throttle position should always correspond approximately with a particular nozzle area, for a properly functioning unit.

5.3.8 ESTIMATE OF ERROR

The accuracy of the afterburner portion of the engine performance indicator will be lower than the non-afterburner portion of the indicator due to the non-linear relationships that are used. The error expected will be around 6 or 7 percent.

5.3.9 INSTRUMENT DISPLAY (See Fig. 11)

The instrument face will have approximately 360° of graduated readings, in terms of percentage thrust, which will be divided into two scales. The first scale, over approximately 190° of the face, will have a pointer to indicate military thrust from 0 to 100% plus 10% for over-reading. The second scale, which continues from the 110% mark of the first scale and reads from 0 to 105% over approximately 160° of the instrument face, will have a separate pointer to indicate percentage of afterburner thrust. The second pointer will not appear until the first pointer is reading 100% military thrust and the afterburner is ignited. A smaller scale is located in the centre of the instrument face. This scale reads the jet pipe temperature from 500 to 950°C. It should be noted that this temperature indication is not an integral part of the Engine Performance Indicator, and is mounted in this position only as a convenience.

A contract for the manufacture of this instrument has recently been placed.

6.0 AEROELASTICITY

6.1 HIGH SPEED FLUTTER TESTING

Transonic flutter tests have been carried out on ARROW wing models at the Massachusettes Institute of Technology. The object of the tests was to determine the effects of stiffness on the flutter boundary of the models. This was accomplished by testing models with varying stiffness properties.

The wind tunnel used was a 24 inch octagonal slotted wall type, with a speed range from M = .7 to M = 1.2 and maximum dynamic pressure (q) of 8.5 psi. The models tested were 1/40th scale half span and full span delta wings with the sting mounted in the centre of the tunnel. An end plate attached to the mount of the half span model was used to approximate fuselage effects and to shield the model from distributed flow from the mount. (See figure 12). The end plate was slotted to eliminate interference with the model. The models were structural replicas of the aircraft, with a torsion box scaled to the primary structure and with cover skin from .0025 in. to .003 in. thick.

Each model was strain gauged to indicate bending and torsion. Traces from these gauges are used to determine the frequency of oscillations only, since amplitude response was not calibrated.

The models were mounted in the tunnel and vibrated in still air to determine the frequencies and node lines of the lower natural modes. The tunnel was then blown down, (i.e. air passed from a sphere under pressure to a sphere under partial vacuum) starting at M = 1.2. After each run, the model was vibration tested again to determine if any changes had occurred in model stiffness. If the model being tested does not flutter, the only way to determine the effects of stiffness, without changing velocity range or density, is to test a weaker model. Only three models of varying stiffness were available. Intermediate stiffness values were obtained by making spanwise saw cuts in the models that had not fluttered. A total of eleven flutter runs were made and the models were tested in the order of decreasing stiffness. Each of the models exhibited a limited amplitude tip instability at the lower Mach numbers, (M = 0.7 to M = 0.9) gradually becoming stable at the higher Mach numbers. Strain gauge traces showed the instabilities to be of an intermittent nature, but of increasing amplitude and duration, with decreasing model stiffness.

No destructive flutter was encountered during the tests, although the instability amplitude of the weakest model was sufficient to buckle the lower inboard skin. With this exception, all models were unaffected by the tests and had the same frequencies before and after the run. The full span models had very small fuselages and required external masses for engine simulation. However, it is considered that this had no effect on flutter results. The results of the tests are compared with predicted modes below. (See figure 14).

Figure 15 shows a typical detailed tunnel time history of Mach number (M), stagnation temperature (T_o), and dynamic pressure (q).

6.2 PROPOSED FLUTTER TESTING

Since a wing and a fin remain relatively undamaged after low speed flutter testing, it is proposed to modify them to agree with the results of the ground resonance tests, and repeat some of the flutter tests.

6.3 GROUND RESONANCE TESTS

The ground resonance tests on the ARROW 1 have been performed to check the calculations for the dynamic structural characteristics of the aircraft.

The problem of flexible support for an aircraft as large as the ARROW is a considerable one. It was decided, therefore, that in preference to a complex free suspension system (to simulate flight conditions) a support of known stiffness should be used. The predicted vibration modes of the aircraft were calculated using the three-point suspension of the landing gear as the known support. The ground resonance tests provided a check on these calculations since the landing gear (with partly deflated tires) was common to both the predicted modes and experimental results. With the method of calculation shown to be satisfactory, it follows that calculations to predict the vibration modes for a freely suspended aircraft, (as in flight) will be accurate.

The aircraft was tested, in a level attitude, at a weight of 48,600 lb. Vibration was introduced by attaching electro-magnetic shakers to the sting, the wing tips and the wing leading edges. The fin was excited at its leading edge by a smaller shaker. The amplitudes of the response, at a succession of frequencies, were measured at selected points by velocity pickups and recording apparatus. For each measured point, a plot of amplitude vs frequency was made on which the resonances were indicated. The resonance frequencies were then investigated in greater detail, and amplitudes over the whole aircraft were plotted to indicate the nodal lines.

The results of ground tests, comparing test results with predictions, are shown in Figure 16. Since re-evaluation of stiffness is not complete, the following comments must be regarded as preliminary:

Symmetric Modes - Fundamental mode shows conformity to prediction. Second mode indicates fuselage bending stiffness to be higher than expected. The third and fourth modes indicated that the outer wing stiffness is also higher than expected. The torsional stiffness, in particular, is considerably higher than was estimated.

Antisymmetric Modes - These are, in general, similar to the symmetric modes. The fundamental mode indicates agreement with predictions. The fuselage torsional stiffness and the outer wing torsional stiffness are much higher than estimated.

Fin Modes - No clear torsion mode emerged from the first tests, due to the shaker placement and excessive control surface coupling. Tests were repeated using a tip contour board to allow the shaker to apply greater torque, in an attempt to drive the second and third modes in a more satisfactory manner. This proved successful, and the modes obtained were in agreement with calculations. Strong rudder torsion coupling with the third mode was found, and this is now being investigated from the flutter point of view.

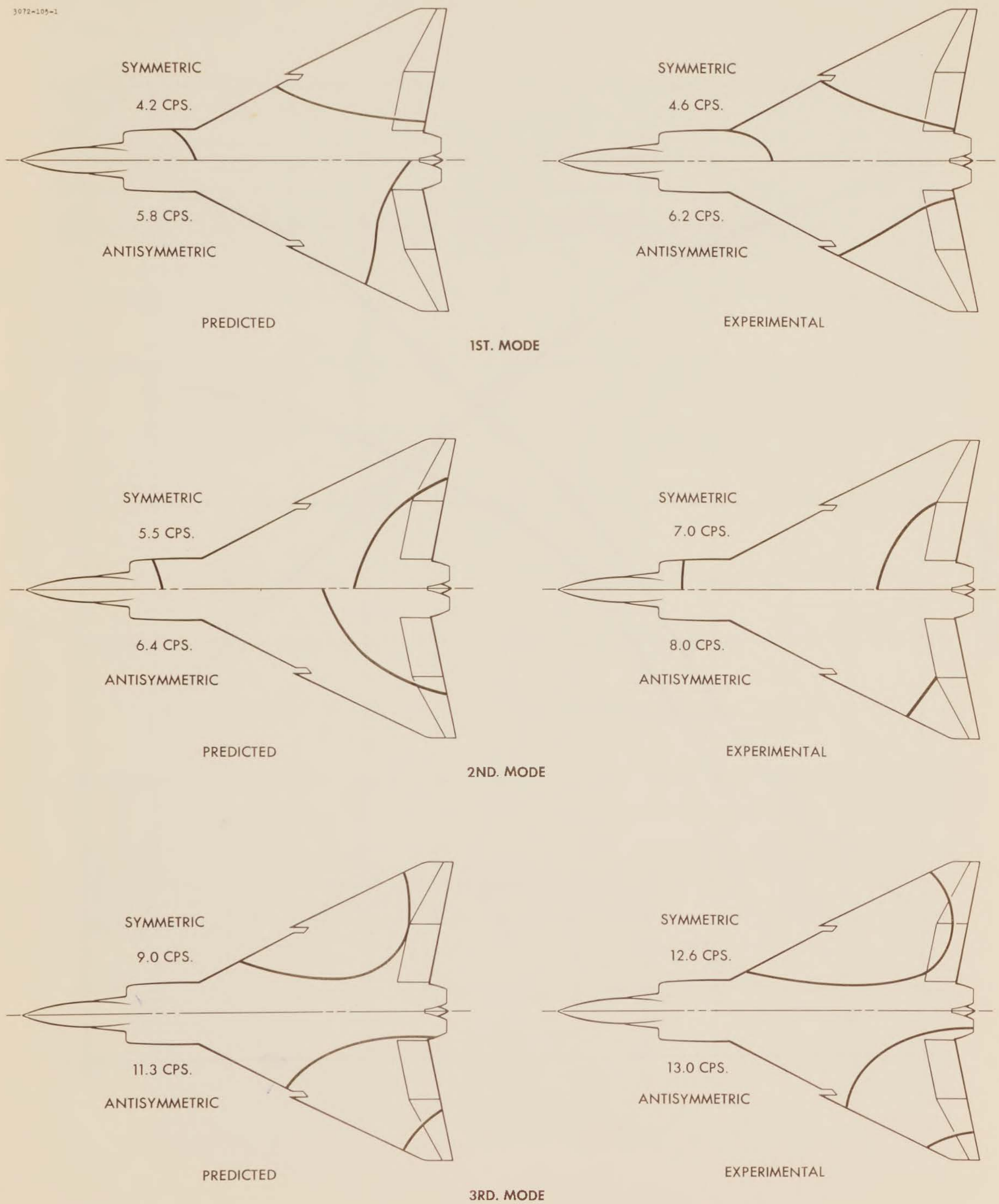


FIG. 16 GROUND RESONANCE MODES

PART 3

SYSTEMS AND EQUIPMENT DESIGN

7.0 ELECTRONIC SYSTEM

7.1 SCOPE OF ELECTRONIC SYSTEM REPORT

Liaison with RCA and with other companies connected with development of the ARROW's electronic system has remained the major effort throughout the report period. The areas of development which are AVRO's responsibility are discussed in this report.

7.2 ASTRA I INSTALLATION DESIGN

Wiring information required from RCA, to permit completion of aircraft and instrumentation wiring design for the ASTRA I development vehicle (aircraft 25204 and 25205), is not yet complete. Therefore, in order to meet schedules for aircraft completion, it has been decided to complete these aircraft without ASTRA I wiring and to temporarily incorporate the interim electronic system designed for aircraft 25201, 2 and 3. This involves the installation of temporary equipment mounting schemes and cockpit panels. After first flight (and RCAF acceptance of the aircraft) and prior to the aircraft being delivered to RCA, the interim system will be removed and replaced with ASTRA I wiring and panels.

On [illegible] only flying

The majority of the partial ASTRA system wiring requirements have now been received from RCA and the preparation of wiring schemes is in progress. Information still outstanding consists mainly of details of the compass system, air data computer and instrumentation. The overall wiring requirements for the full ASTRA I system are presently being studied, together with the missile firing system.

7.3 RADOME

7.3.1 ARROW RADOME DESIGN PROBLEMS

The accuracy of the fire control radar in the ARROW 2 will be influenced by the effect of the nose radome on the ASTRA I radar antenna pattern. The two major problems which are apparent in the design of an electrically acceptable radome are achievement of good transmission efficiency, and minimum distortion of the antenna pattern with maximum boresight accuracy.

Every precaution will be taken to maintain a high percentage of transmitted power, while reducing the boresight errors as much as possible. Intensive study and testing will be required to produce a satisfactory radome.

Boresight errors have two components: the in-plane error, or the component in a plane containing the radome axis and the radar line of sight, and the crosstalk error which is in a plane normal to the boresight error.

The primary cause of boresight error is refraction of rays by the radome wall, which produces a variation in phase delay across the antenna aperture. This causes the antenna to provide incorrect target angular information to the radar. Rate of change of boresight error with antenna look angle is of greater consequences than actual total error, as it determines the degree to which the tracking behaviour of the radar is affected.

Other sources of error, such as reflection, are less important and transmission losses due to these effects can be largely overcome by the appropriate choice of radome materials and wall dimensions.

The radome must not only have satisfactory electrical performance, but it must also have optimum aerodynamic shape and must be able to withstand the effects of aerodynamic heating, ice accretion and other environmental conditions.

7.3.1.1 Angle of Incidence

Boresight error is largely due to the insertion phase delay at the dielectric (the radome wall). The phase delay depends upon the path length of the ray in the dielectric, and varies with the angle of incidence of the transmitted ray, which in turn varies with antenna look angle. Thus the boresight error is complicated by variation of angle of incidence and can be minimized by design of a radome wall with a minimum variation in angle of incidence.

7.3.1.2 Polarization

The multiple polarization modes of the ASTRA I antenna complicate the problem of refraction at the radome wall. Optimum design for parallel, perpendicular and circular polarization poses some very difficult problems in boresight error rate reduction, as the radome has a depolarizing effect which influences the boresight error to a degree dependent upon the type of polarization employed. In general, extremes of boresight error are evident with linear polarizations. Circular or elliptical polarizations, which are combinations of parallel and perpendicular polarization, produce smaller errors.

7.3.2 RADOME DEVELOPMENT PROGRAM

As stated in the last ARROW Quarterly Technical Report,the radome is being designed and built by the Brunswick-Balke-Collender Company, to AVRO requirements. The B-B-C program may be summarized as follows:

<u>Design</u>

(a) Shape study and ray analysis.

(b) Electrical and structural analysis.

(c) Optimization and design of radome wall and electrical performance.

Convenience Tooling

(a) Specification and production of tooling.

(b) Construction and set-up of tooling.

Correction

(a) Testing and correction of developmental radome shells.

(b) Installation of hardward and final correction.

Testing

(a) Qualification testing of two radomes (environmental, mechanical, weathering and electrical tests).

(b) Electrical testing.

Production

(a) Manufacture and correction of two radomes.

(b) Delivery of two radomes to AVRO.

The shape study and ray analysis have been completed and the rest of the design phase is in an advanced stage.

7.3.3 RADOME REQUIREMENTS

The requirements for the ARROW 2 radome are as follows:

(a) Good transmission properties and high transmission efficiency.

(b) Sufficient structural strength and quality to withstand air loads and environmental effects.

(c) Efficient aerodynamic shape.

(d) Optimum electrical shape and thickness (within the bounds of aerodynamic requirements) to obtain a distortion-free antenna pattern.

The radome wall will be designed for transmission characteristics, and boresight error corrections will be made with dielectric rings or lenses. A major problem will be to correct for the varying polarization, and some relaxation of the requirements in this respect may be necessary. Corrections will be applied when simulation and analytical studies provide a satisfactory basis for design of the radome wall.

7.3.4 STATUS OF RADOME PROGRAM

Since the previous Quarterly Technical Report, certain important developments affecting the radome have occurred. Perhaps the most important is the shift away from the straight-sided cone originally proposed. It became apparent that the range of incidence angles was so large for this contour that it would result in unacceptably large boresight errors and error rates. The outer contour, which is now definitely defined, is ogival and is approximately 12 inches shorter than the original shape (See Figure 17). This shape presents a compromise between optimum electrical and aerodynamic efficiency, having a considerably reduced range of ray incidence angles.

Prior to the evolution of the improved shape, considerable effort had been expended in assessing the original conical shape and a second shape intermediate between the cone and the contour finally adopted. Ray studies, estimates of range of incidence angles, and predictions of transmission efficiency, based on an assumed wall thickness, were undertaken A digital computer program for computation of percent transmission and boresight error was evolved and checked out, but it was based on the conical shape radome All of this work had to be repeated for the improved contour and only recently have predictions become available for the new radome shape.

The digital computer program has not yet yielded any design information, since the modified program is still being checked out. Estimates of boresight error performance to date have been based solely on radome design experience. The computer program is not considered capable of accuracy in boresight error prediction of better than ± 50%. However, it should enable design trends to be investigated more readily than is possible by trial and error methods involving very costly measurement programs.

There are certain unresolved problems at present. One of the problems is how to optimize the radome when the variable antenna polarization requirements are considered. Optimum design for two linear polarizations and circular polarization will require an acceptable compromise for good all round performance. A more satisfactory result could be obtained if this requirement was relaxed, and definite plans have been evolved to study several optimum designs, so that a decision on this point can be reached within an acceptable time scale.

It has been decided that the radome will be of solid polyester resin glass cloth laminate, with half-wavelength wall thickness. The polyester resin to

be used in the construction of the radome is Bakelite Company BRSQ142. Test samples of the wall laminate material made with this resin have a dielectric constant of 4.07 and loss tangent of 0.012.

7.3.5 RADOME DESIGN STUDIES

A ray analysis for the final radome shape has permitted a preliminary study leading to an optimum radome wall thickness of 0.348 inches. Preliminary investigations have also been performed for a varying thickness wall.

Procedures for calculating transmission through the optimum thickness wall, for all look angles, have been formulated. These are based on the ray study and a computer program.

To determine the optimum thickness for the radome, graphs of transmission, phase delay and relative phase shift were plotted against thickness, for parallel and perpendicular polarization at several angles of incidence. Transmission for optimum thickness vs incidence angle was then plotted (Figure 18).

The effective power transmission coefficient for each ray was calculated for all look positions and the average power transmission coefficient determined by numerical integration.

7.3.5.1 Error Prediction

Predictions of boresight and crosstalk errors, together with probable corrections, are shown in Figure 19. Variables of polarization, frequency and look angle produce a band of errors and the shaded areas portray the expected spread of the band for linear polarizations. The upper edge of the band represents parallel polarization error and the lower edge represents perpendicular polarization error. Correction will result in lowering the whole band to reduce the maximum errors and also to reduce the error slope (i.e. the error rate).

As the radome must have optimum electrical performance for parallel, perpendicular and circular polarization it will not be possible to vary the wall thickness as a function of angular position about the radome axis (which could be done to optimize for a given linear polarization). A uniform wall thickness will be used at any given radome station, although fore and aft taper may be employed.

Curves of phase delay vs position of ray along the antenna dish, for various angles of incidence and look angle, are given in Figure 20. These are based on the ray analysis and the graph of phase delay vs angle of incidence shown in Figure 21.

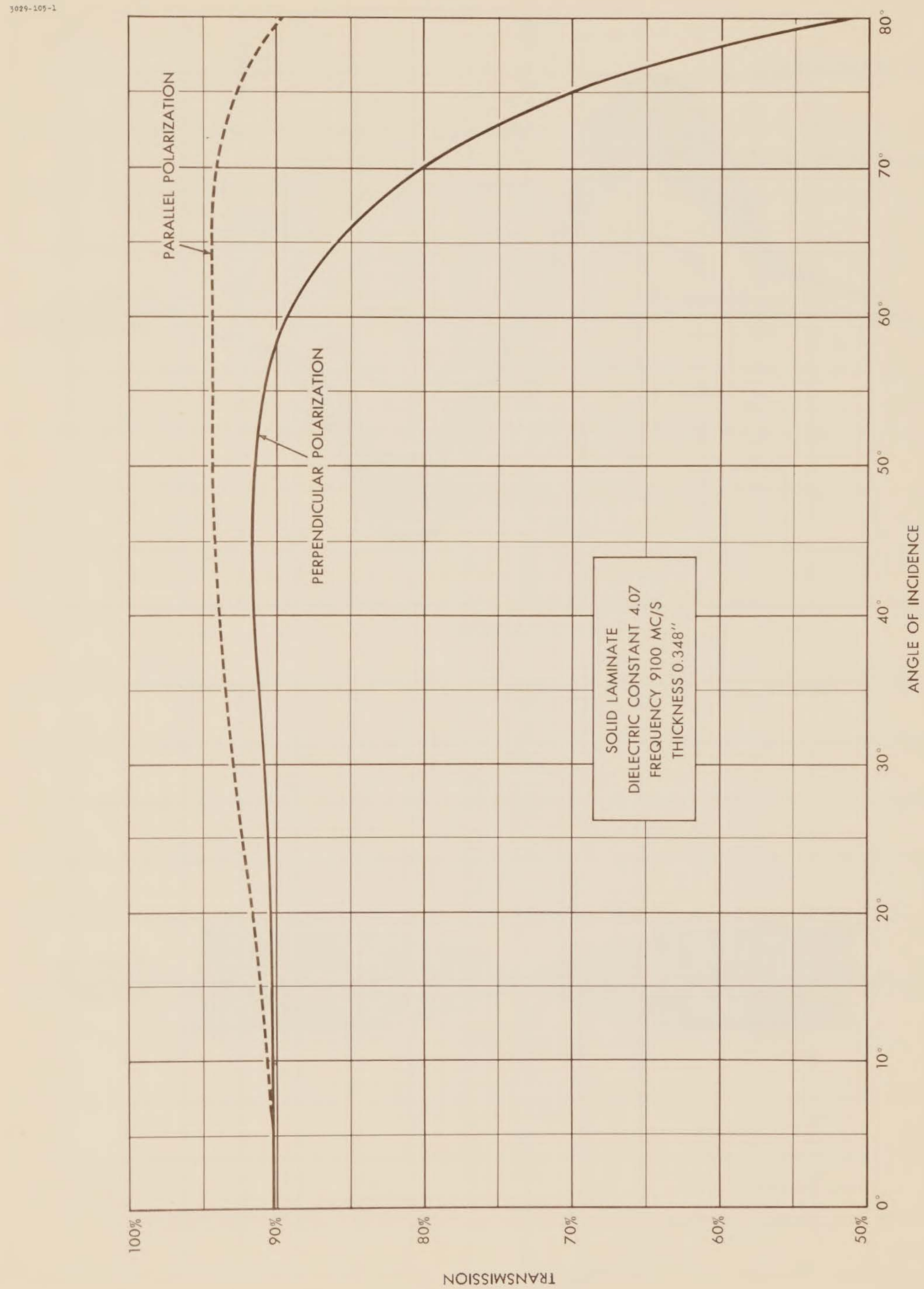


FIG.18 OPTIMUM THICKNESS RADOME WALL TRANSMISSION CHARACTERISTICS

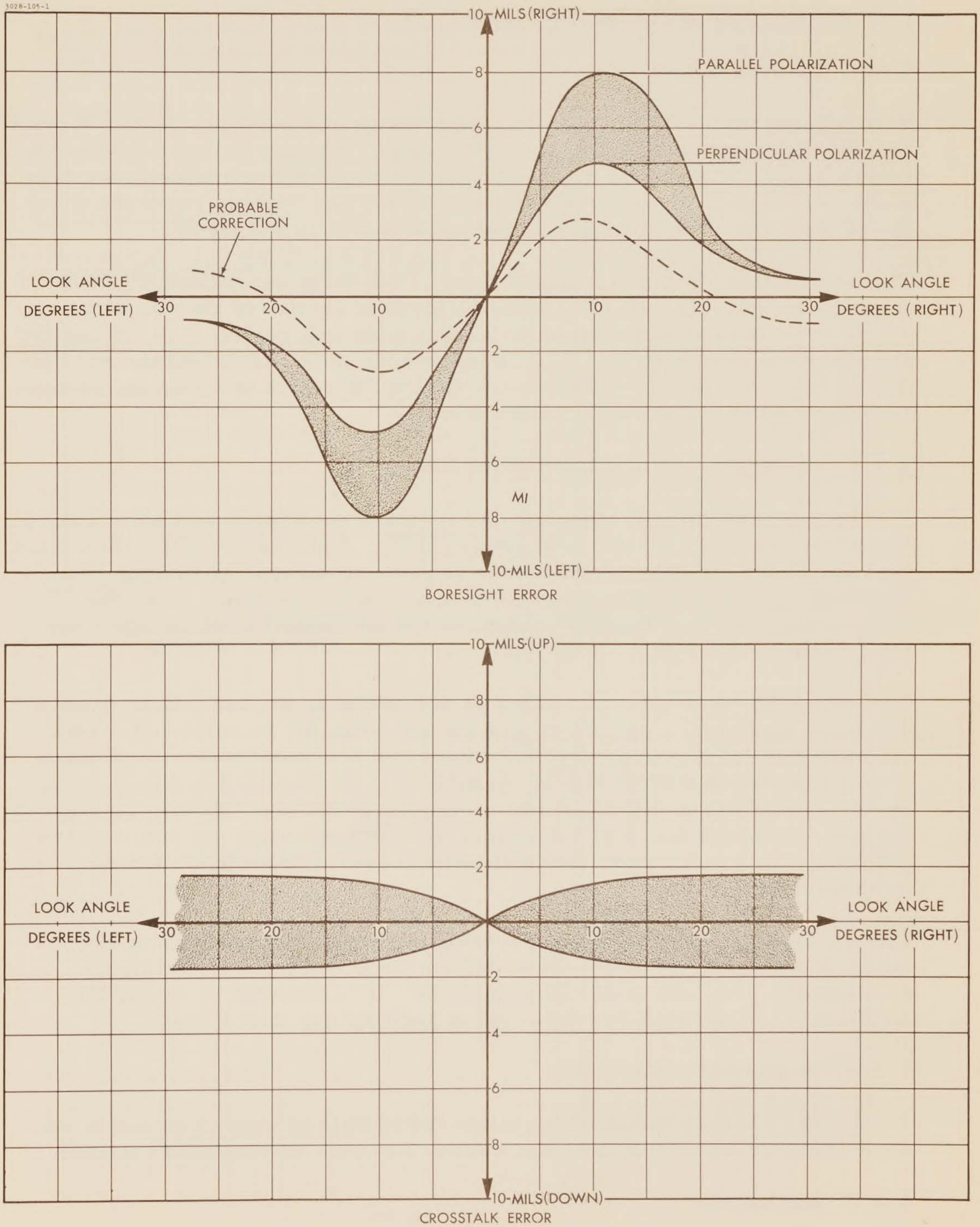


FIG. 19 RADOME ERROR PREDICTIONS

7.4 INFRA-RED SYSTEM INSTALLATION

7.4.1 IR INSTALLATION

Detailed investigations of possible configuration and locations for the IR seeker head pod have now reached a conclusive stage. Consideration of field of view, fin stiffness, aerodynamic drag, airframe changes and available space have led to the adoption of a cylindrically shaped fairing, located below the tip of the vertical fin. Configuration of the pod is shown in Figure 22.

The seeker head, complete with IR dome, will weigh approximately 35 lb. and together with its case will form the forward portion of the ten-inch diameter pod. It is estimated that the seeker head will require 1 lb. of cooling air per minute at 75-80^{o}F. RCA is requesting the RCAF to add the development and production of the IR dome to RCA's statement of work, as it forms an integral part of the seeker head assembly.

7.4.2 IR DETECTOR COOLING SYSTEM

The seeker head detector develops the required sensitivity only when cooled to the temperature of liquid nitrogen (-196^{o}C). Discussions have taken place between AVRO and RCA to assess the relative advantages of an open-loop type cooling system and a closed-loop recirculation system. Cool-down time has a bearing on the choice of system as the operating life of the open-loop type, before recharging, is limited.

Cool-down time has been calculated as two minutes, or less. Consequently, a storage type open-loop cooling system will probably be employed. This offers certain advantages; simplicity in design and maintenance, and maximum reliability and flexibility of installation. An insulated gaseous nitrogen, storage bottle will be used and will be located in the area between the aircraft engines. To meet the cool-down time, the bottle charging pressure will be 3000 psi. This will ensure that sufficient pressure is available for the mission duration.

The nitrogen storage bottle will probably have an initial capacity of 500 cubic inches. The location chosen and the environmental temperatures involved are presently under consideration. It is intended to charge the bottle in place, through an access panel provided for this purpose.

7.4.3 IR FLIGHT TESTING

A retrofit trial installation of the infra-red system is planned on either aircraft 25204 or aircraft 25205, the ASTRA I system development vehicles.

7.4.4 PROGRAM

Further investigations and liaison with RCA will be aimed at completing the

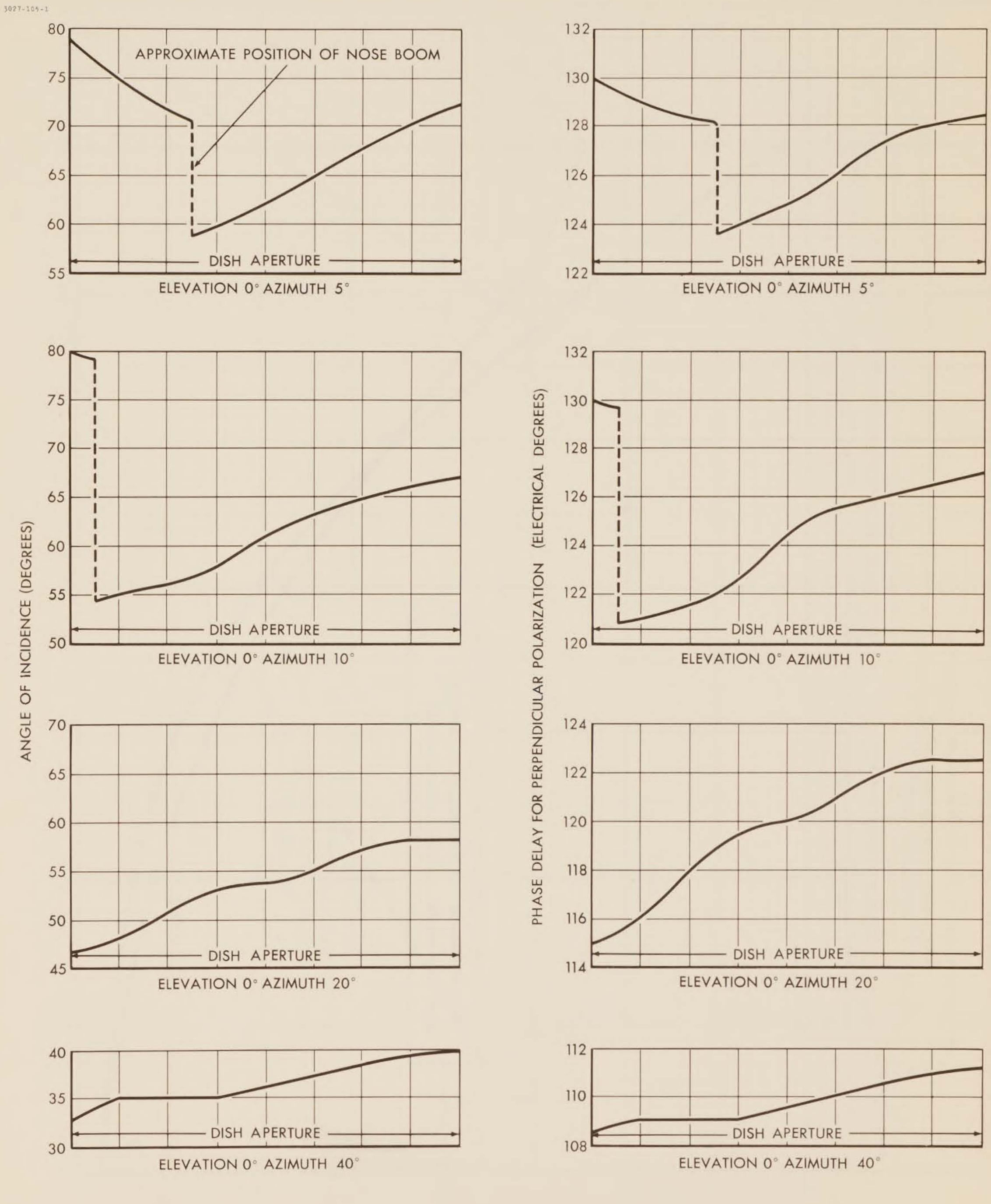


FIG. 20 PHASE DELAY & ANGLE OF INCIDENCE VS. POSITION OF RAY ALONG DISH

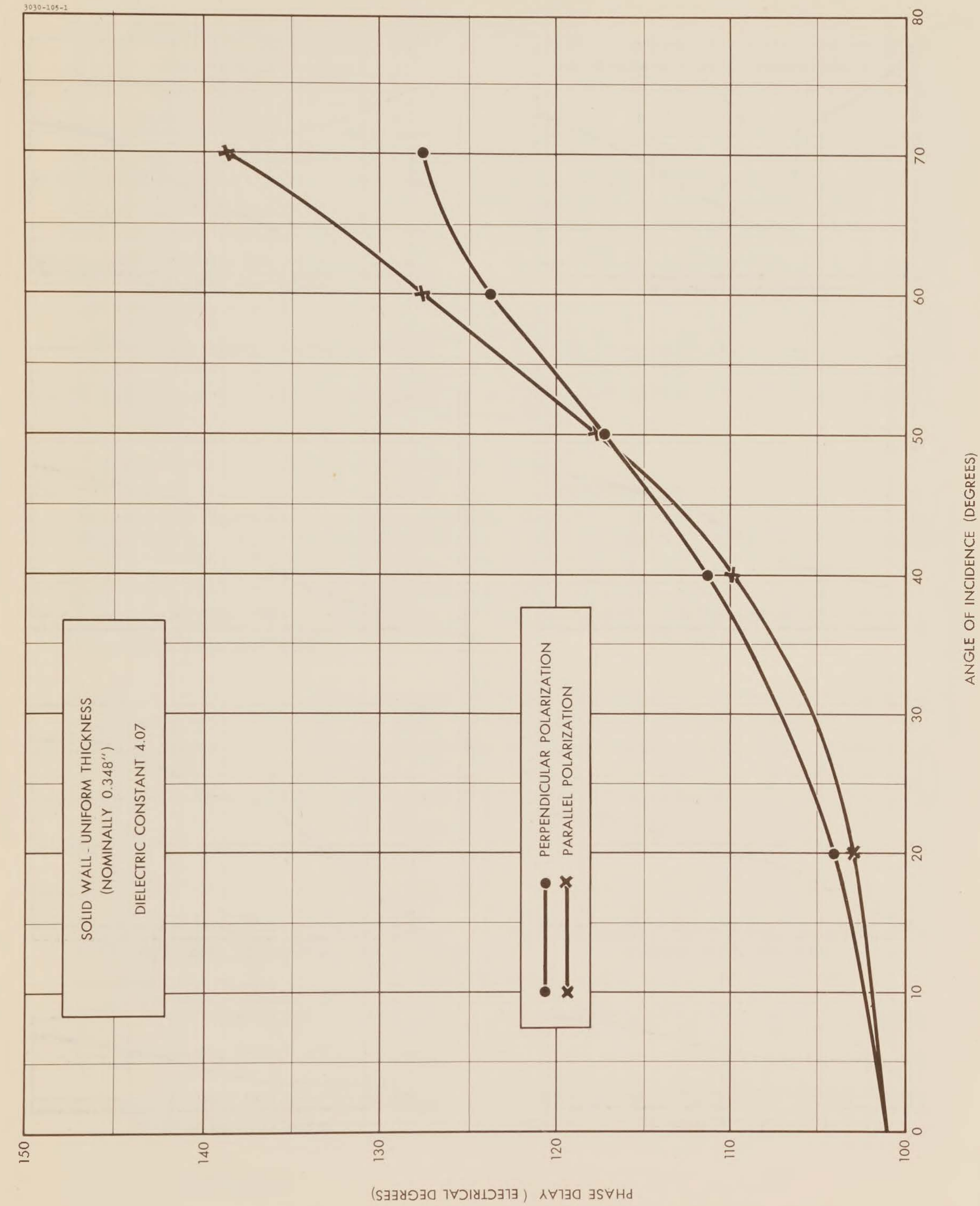


FIG. 21 RADOME WALL PHASE DELAY VS. ANGLE OF INCIDENCE

installation design as soon as possible. Detail problems of seeker head attachment, boresighting, cooling air, wiring requirements, electronic equipment location and structural design have still to be finally resolved.

7.5 ANTENNA DESIGN AND DEVELOPMENT

7.5.1 SUMMARY OF ANTENNA PROGRAM

A program of design and development for the ARROW UHF and L-band antenna is being conducted on behalf of AVRO by Sinclair Radio Laboratories Limited. This program includes ARROW 2 UHF belly antenna development, CF-100 model pattern studies to support the ARROW antenna evaluation program described in the previous Quarterly Technical Report, and ARROW model pattern studies.

7.5.2 ARROW 2 UHF BELLY ANTENNA

The UHF (annular slot type) belly antenna for the ARROW 2 retains the same envelope size as the ARROW 1 version, but will be modified in other respects to achieve a voltage standing wave ratio (VSWR) of less than 2.5 (the ARROW 1 antenna had a VSWR of 3.0). Studies are presently being conducted to design a satisfactory prototype antenna.

Experiments have been performed with printed circuits for the matching section, in an attempt to incorporate the matching section into the cone area of the antenna. However, it was decided that the antenna could be designed without a matching section, and that the omission would contribute to increased radiation efficiency. Measurements for the antenna without an external matching section showed a reduction in the VSWR to 2.4 for the high and low ends of the frequency band. This was reduced again to about 2.2 by further experiments with the size of the octagonal top loading plate, tuning inductors, trimmer capacitors and the capacitive slot. After completion of further work on the cone feed gap and feed section length, and after assembly with the dielectric foam, the VSWR was reduced to less than 2.0 over the frequency band. Trimming capacitors are to be incorporated on the antenna to peak the power output.

The dielectric foam used in the antenna is of a new type allowing a high degree of quality control during assembly. It is of lower density than previous foams used, absorbs less moisture and will withstand high temperatures. The main problem in this respect is one of shrinkage occurring after the foam solidifies, and experiments are being conducted on different types of foam, in an attempt to overcome this problem.

7.5.3 MODEL PATTERN STUDIES - CF-100

Pattern measurements for UHF and L-band antennas on a CF-100 model are being compared with single plane measurements obtained from the antenna evaluation flight test program on CF-100 No. 18186. Comparison of model range patterns and flight test patterns will establish the technique and verify model pattern results for full antenna coverage. Provided model and in-flight measurements agree to within 1 db, all model patterns will be accepted as representing full scale aircraft patterns.

A 1/10 scale CF-100 wind tunnel model, mounted on a model tower, is being used for the fin UHF antenna pattern measurement. The patterns measured are to include principal plane cuts, and conical cuts at 1, 2 and 3 degrees to the horizontal, for frequencies of 226.8, 324.3 and 384.3 mc/s respectively. The entire model is sprayed with silver paint to simulate the surface conductivity of the aircraft.

Three principal plane pattern records for one of the test frequencies have so far been obtained. These are presently being compared with records from the flight evaluation program and preliminary indications are that model and flight test patterns will show good agreement.

7.5.4 MODEL PATTERN STUDIES - ARROW

Model pattern studies for the UHF antennas will be undertaken on a 0.07 scale ARROW wind tunnel model, mounted on a model range tower.

Pattern measurements will be taken for the principal planes, and for 30 to 40 conical cuts, at frequencies of 226.8, 324.3 and 384.3 mc/s. L-band antenna patterns will be measured in the same manner for frequencies of 970 and 1060 mc/s.

These studies will permit comparison of flight and model antenna pattern measurements for the ARROW, to establish the accuracy of model pattern measurements for full antenna coverage. The effect of the infra-red seeker fairing on the performance of the UHF/L-band fin antenna will also be investigated.

7.5.5 ANTENNA EVALUATION PROGRAM

CF-100 Aircraft 18186 has been engaged in flight tests for the antenna evaluation program (Ref. Para 7.5.3) but the tests have now been discontinued until early spring 1958. A ground station has been set up at Penetanguishene, Ontario, to record field strength measurements for aircraft transmissions over the check point, Cape Rich, 25 miles distant from the ground station.

A total of five test flights have so far been completed and the patterns obtained are being compared with model pattern studies. Conclusive statements cannot yet be made but preliminary results are encouraging.

7.6 AUTOMATIC SELECTION OF UHF ANTENNA

7.6.1 REQUIREMENT FOR ANTENNA SELECTION

UHF antenna multiplexing will be necessary when UHF data link is introduced into the ARROW in addition to the present UHF communication. However, prior to this, it may be decided that an automatic method of UHF antenna selection is required to effect omnidirectional antenna coverage at all times. With this in mind, design studies have been carried out to formulate a method of automatically selecting either the fin antenna or the belly antenna for optimum UHF performance.

7.6.2 ANTENNA SELECTOR

As stated in the last ARROW Quarterly Technical Report the proposed installation would utilize an antenna selector unit (C-2193/ARC) manufactured by the Autonetics Division of North Amercian Aviation Inc. This device controls an antenna switching relay, and causes the input to the UHF communication set to switch between the fin and belly antennas at a rate of approximately 70 cycles per minute. When an RF signal is received, the selector causes the relay to lock the radio on to the antenna receiving the stronger signal. An override control would be incorporated, allowing the pilot to manually select either antenna at his discretion. Lights would indicate the antenna in use. During transmission, a memory relay in the selector automatically locks the communication set on to the antenna receiving the last signal.

A flight test installation to check the performance of the antenna selector system on an ARROW 1 aircraft is under consideration.

7.6.3 ALTERNATIVE PARALLELED ANTENNAS

Connecting the fin and belly UHF antennas in parallel would also provide omnidirectional coverage but it is estimated that mismatching would then result in a 30% reduction in operating range. In addition, phasing problems may become apparent in the region of antenna coverage overlap, resulting in poor coverage in this region. Model pattern studies are being carried out to estimate the effect of operating the antennas in parallel, and this method may also be flight tested in conjunction with the antenna selector.

7.7 RADIO COMPASS INSTALLATION

7.7.1 DESCRIPTION OF RADIO COMPASS INSTALLATION

The AN/ARN-6 radio compass installation in the ARROW employs certain advanced design characteristics which have necessitated an extensive development and evaluation program prior to its use. New techniques embodied in the system are listed as follows:

(a) The introduction of a zero-drag flush mounted magnetic loop antenna, in place of the conventional AS-313/ARN-6 loop antenna.

(b) The introduction of a new type sense antenna, located in the dorsal fairing.

(c) Matching of the abnormally long sense antenna cable (determined by the aircraft configuration) to the radio compass receiver.

In order that the radio compass would perform satisfactorily on the ARROW, it was essential to carry out flight tests on a CF-100 aircraft to check the functioning of the system and the effectiveness of the redesigned antennas.

7.7.1.1 Magnetic Loop Antenna

The loop antenna for the ARROW radio compass system is the Bendix Radio type LPA-6A, flush mounted on the electronics bay door. A feature of this type of loop antenna is the simplicity of the quadrantal error compensation method. Fore and aft and transverse ferrite compensating bars at the extremities of the antenna arms can be varied in length between two and ten inches to achieve correction.

7.7.1.2 Sense Antenna

The sense antenna consists of a curved copper sheet fastened to the inner surface of the fiberglass laminate dorsal fairing. This location was chosen as a result of a model test program using an electrostatic cage to determine the electrical centre of the aircraft. A sense antenna at this point exhibits optimum station passage characteristics. The cable to the receiver is 37 feet in length, and this fact, combined with the relatively high capacitance of coaxial cables suitable for the ARROW's environmental temperature, resulted in a sense antenna cable capacity several hundred percent higher than that for which the AN/ARN-6 receiver is designed. A study of optimum sense antenna parameters resulted in the choice of a capacity of 100 micro-microfarads and an effective height of 0.1 meter.

7.7.1.3 Susceptiformer

Matching of the sense antenna, with its abnormally long cable, to the radio compass receiver is achieved by means of a susceptiformer (RF transformer) assembly adjacent to the antenna, and a special L103 antenna coil assembly (replacing the original L103) in the receiver.

Included in the susceptiformer, which is manufactured by the Bendix Radio Division of Bendix Aviation Corporation, are four relays operated by receiver frequency selection. The special L103 antenna coil assembly which is also manufactured by Bendix Radio, is being used to match the receiver with the susceptiformer and 37-foot cable.

7.7.2 EVALUATION PROGRAM

Flight tests for evaluation of the ARROW radio compass installation have been carried out by means of an operating mock-up on CF-100 aircraft 18185. These tests were undertaken in three phases:

Phase 1. Magnetic loop antenna evaluation, using the conventional CF-100 sense antenna.

Phase 2. System evaluation using the magnetic loop antenna with the ARROW type dorsal sense antenna and a conventional 16-foot sense antenna cable.

Phase 3. System evaluation using the magnetic loop antenna with the ARROW type sense antenna, susceptiformer, redesigned antenna coil and 37 foot cable.

The object of the tests was to determine the radio compass deviation with a flush mounted magnetic loop antenna replacing the conventional type, and to assess the performance of an ARROW type sense antenna located in a mock-up fairing. Direct comparison with the CF-100 radio compass installation was considered the best method of evaluating the ARROW system, and tests were conducted to determine calibration accuracy, sensitivity and station passage characteristics for each phase of the program.

The magnetic loop antenna was installed in the gun bay fairing of the test aircraft, as shown in Figure 23. Installation of the ARROW type sense antenna was as shown in Figure 24. To simulate accurately the antenna capacitance and effective height, mock-ups of the electronic equipment which is housed in the ARROW dorsal were included within the mock-up dorsal fairing on the test aircraft. Relative location of both antennas is illustrated in Figure 25.

7.7.2.1 Phase 1

For phase 1 tests the installation was as shown in the block diagram, Figure 26. The AN/ARN-6 test receiver, which was operating with the magnetic loop antenna was connected to the existing CF-100 sense antenna in parallel with the aircraft receiver, the latter using the conventional loop antenna. This was found to be the best method of achieving loop antenna performance evaluation, as the advantages of using a common sense antenna for loop comparison outweighed any disadvantage resulting from the reduced sensitivity of the receivers.

The results showed that the range of the test installation was above the specified requirement. Quadrantal errors were found to be within three degrees of the true bearing.

3040-105-1

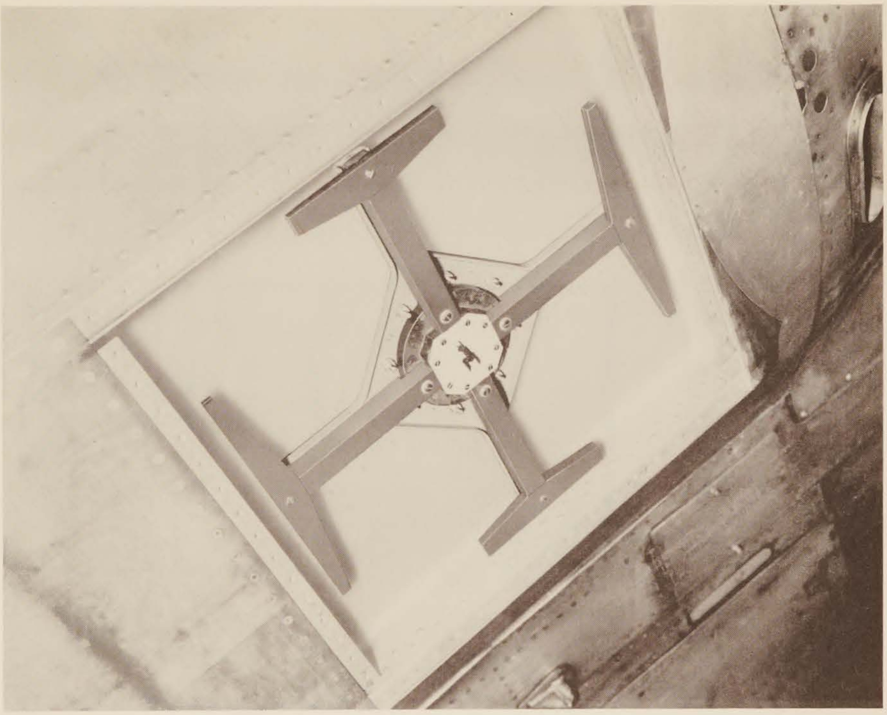


FIG. 23 MAGNETIC LOOP ANTENNA TEST INSTALLATION CF-100

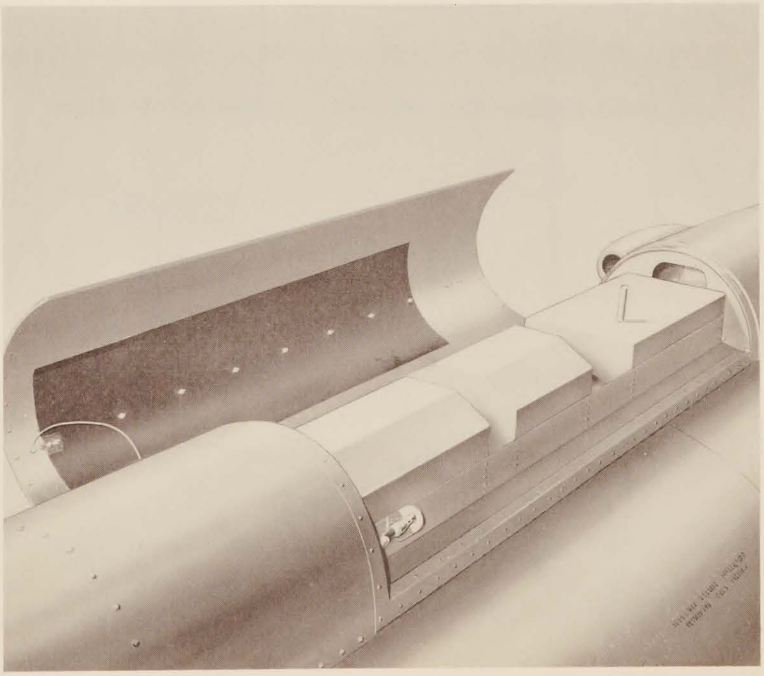


FIG. 24 ARROW TYPE SENSE ANTENNA INSTALLATION CF-100

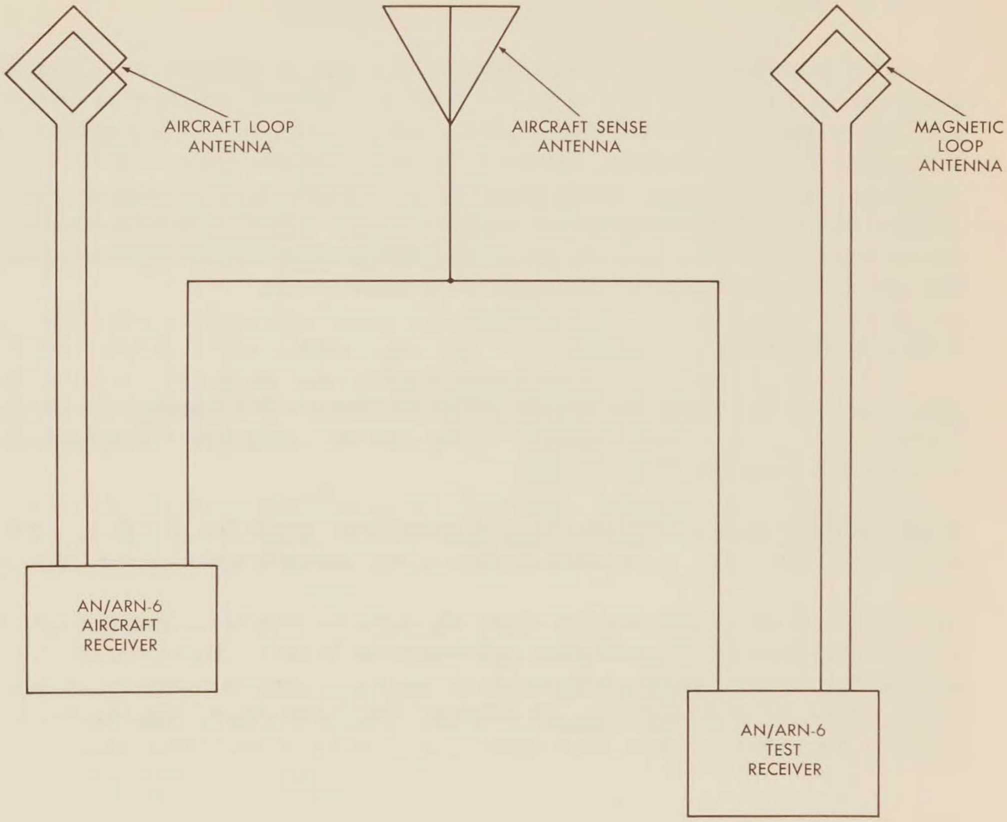


FIG. 26 BLOCK DIAGRAM – MAGNETIC LOOP ANTENNA EVALUATION

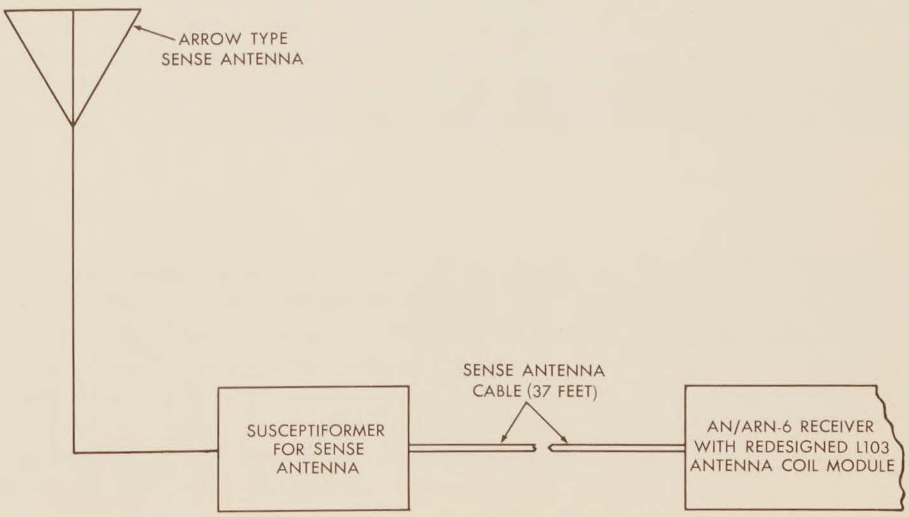


FIG. 27 BLOCK DIAGRAM – SUSCEPTIFORMER AND SENSE ANTENNA

7.7.2.2 Phase 2

To carry out phase 2 tests, the conventional sense antenna was disconnected from the test receiver and replaced with the ARROW type sense antenna, in the dorsal fairing. Sensitivity of this configuration was of the same order as that of the CF-100 system, range tests being satisfactory. Radio compass readings were in error, being generally too high, but the errors were constant, indicating the possibility of index error. Station passage characteristics were generally satisfactory, providing cone of silence angles less than the specified maximum of 90 degrees in most cases.

7.7.2.3 Phase 3

Phase 3 tests included the Bendix susceptiformer installed in the dorsal fairing, the 37 foot sense antenna cable and the redesigned L103 antenna coil for the receiver (Figure 27).

This configuration produced results similar to those for phase 2, with a slight improvement in sensitivity due to the use of the susceptiformer.

The sense antenna dimensions were obtained by trimming the antenna to achieve the correct capacitance and effective height. However, to improve sensitivity, it was decided to obtain a longer, narrower pattern by lengthening the antenna and trimming the sides. This was done, and the final antenna dimensions were 58 inches by 8.5 inches (flat layout size).

8.0 ENGINE INSTALLATION

8.1 ENGINES AND ENGINE ACCESSORIES

8.1.1 ARROW 1

Two J75P3 engines, complete with the engine built-up accessories and components, are installed in the first aircraft, serial no. 25201. Both engines have been run during the ground running tests.

The ground runs to date have been directed towards clearing the aircraft for first flight. Engine calibration runs will be conducted after the first flights of the aircraft. Proof running of the engines prior to first flight will be preceded by engine removal. This will permit inspection checks of engines, airframe, and accessories.

Two Pratt and Whitney J75P5 engines have been received at AVRO. These units have been calibrated by the manufacturer and the calibration data has been made available to AVRO.

8.1.2 ARROW 2

An engine performance indicator system has been designed by AVRO for use with the Iroquois engines in the ARROW 2 (Ref. para 5.3).

8.2 ENGINE INSTALLATION

8.2.1 ARROW 1

The ground running tests on aircraft 25201 have revealed some minor installation problems. Most difficulties have been experienced at the quick-disconnect points between the engine and the airframe systems and services. The majority of these problems have been eliminated by either closer adherence to, or modification of the engine installation procedures and techniques.

8.2.1.1 Fuel Drains

Provision was made in the basic design to drain the fuel from fuel lines on the engine. Upon shutting down an engine, fuel remains under pressure in these lines. The basic design provided three separate drains for each engine, but during ground running tests these drains were found to be inadequate to handle the flow. The waste fuel drain system has been redesigned to collect the fuel from the three drain points in a common sump and discharge it overboard through a single waste fuel discharge point. This modification will be applied to all ARROW 1 aircraft.

8.3 POWER CONTROL SYSTEM

8.3.1 ARROW 1

Some difficulties were experienced with the control system during the ground running tests of aircraft 25201.

A lag in engine response to power lever movement was observed. Backlash in the control system was found to be responsible and was traced to the universal coupling between the airframe system and the fuel flow control unit on the engine. (See Figure 28 and 29).

The universal coupling is being modified for the first flight of aircraft 25201 by introducing heavy duty universal fittings at each end of the coupling, and by modifying the drive couplings. For subsequent ARROW 1 aircraft, selective assembly of the telescopic portion of the coupling will further reduce backlash effects.

During the ground running tests, it was observed that throttle lever position did not correspond to engine rpm in the low operating range. The backlash in the control system would, of course, contribute to this lack of control. However, the main source of trouble was traced to the control cable quadrants operated by the throttle levers (Figure 28 and 30). A modification to the control quadrant, permitting increased travel of the control cable, has rectified the problem.

8.3.2 ARROW 2

The design of the ARROW 2 power control system is influenced by the experience gained with the operation of the ARROW 1 system. Design work is presently in progress on the universal coupling between the airframe system and the engine fuel flow control unit. In view of the recent difficulties with the corresponding ARROW 1 component, close attention is being given to the backlash problem.

The design of the throttle quadrant assembly has recently been completed. Provision has been made in this unit to accommodate the jettison control for the long range tank. The ARROW 2 design effort has been directed towards eliminating the difficulties experienced with the control cable quadrants in the corresponding ARROW 1 assembly.

3047-105-1

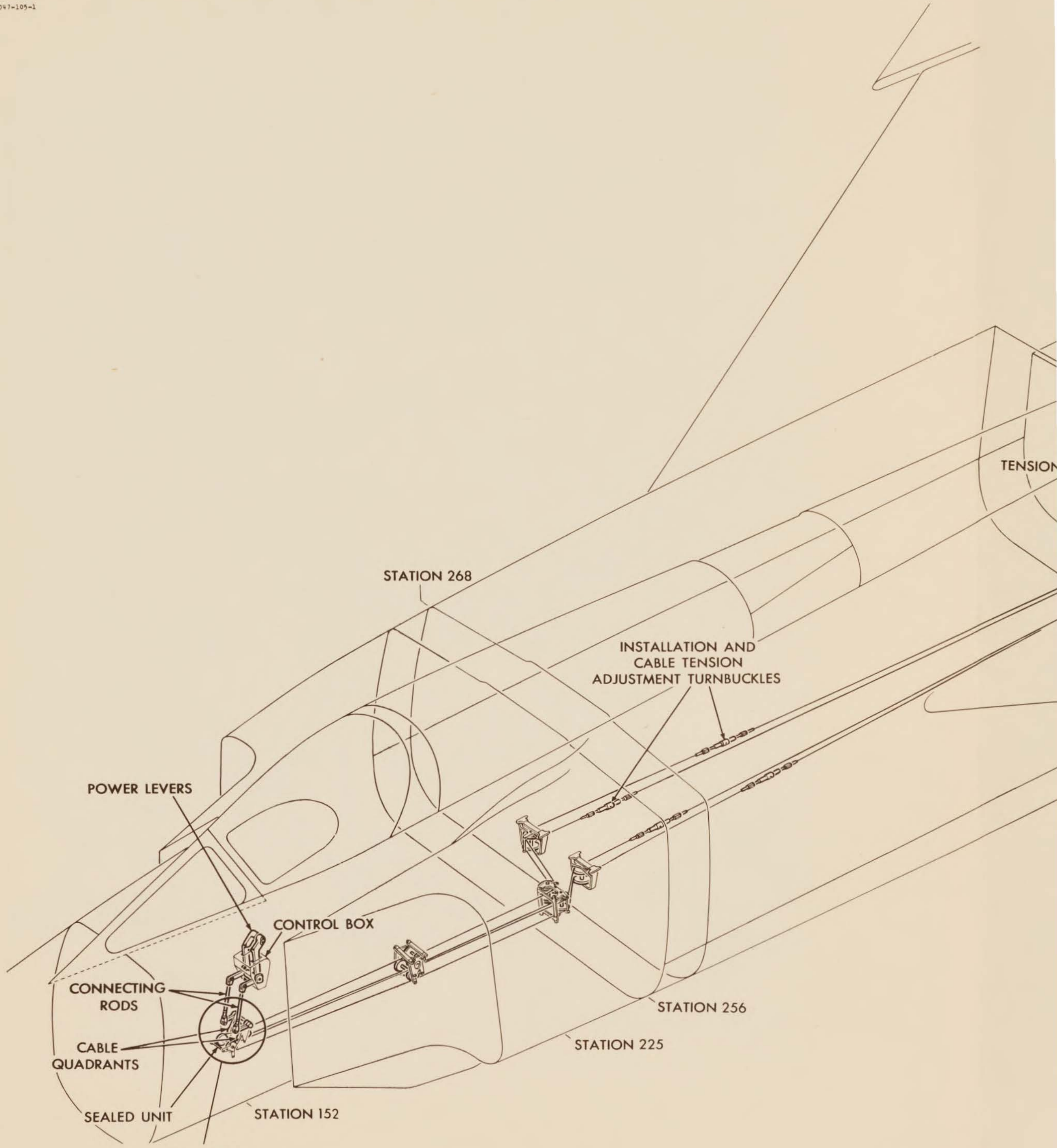


FIG. 28 ENGINE CONTROLS - SCHEMA

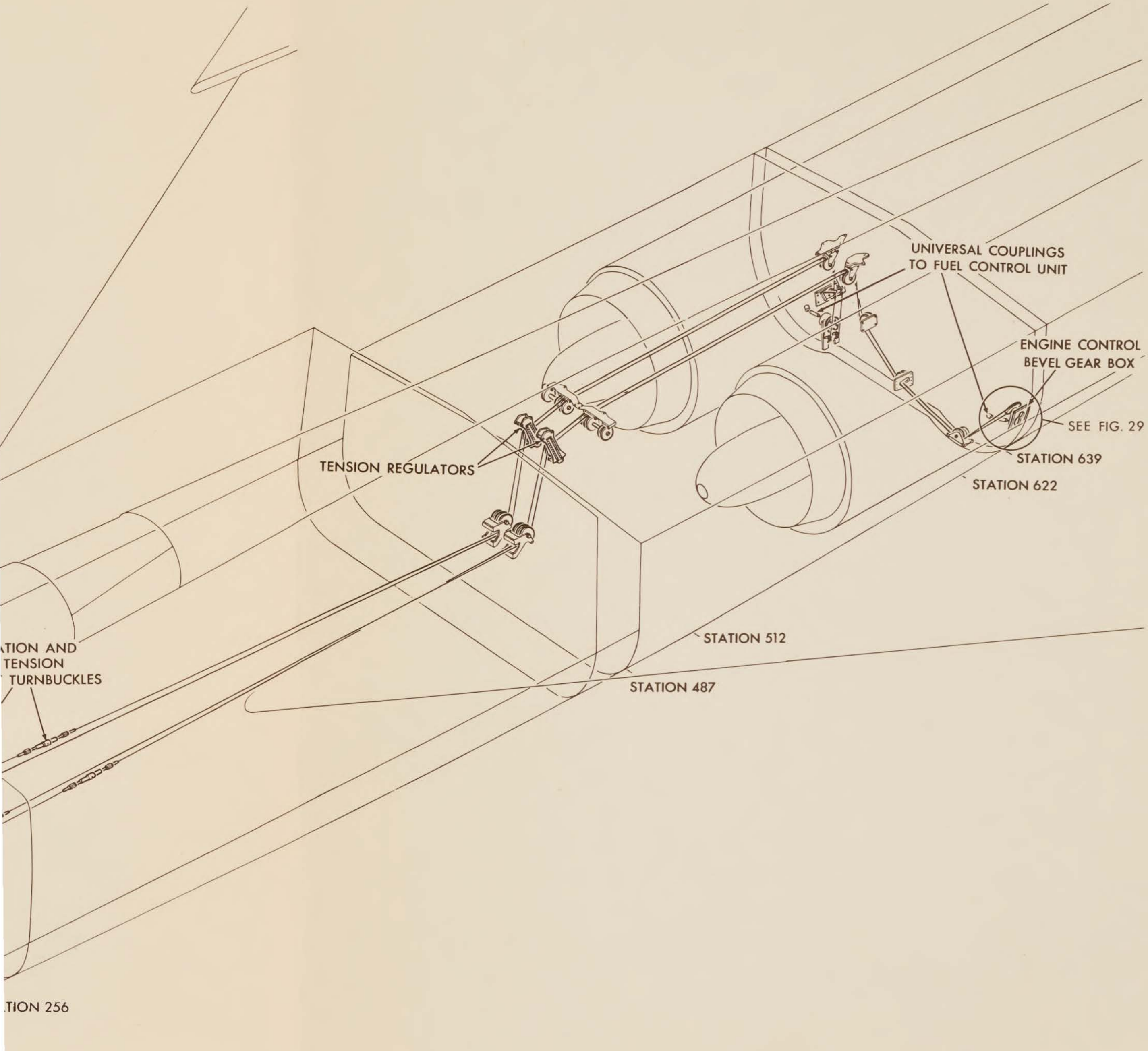


FIG. 28 ENGINE CONTROLS - SCHEMATIC

3048-105-1

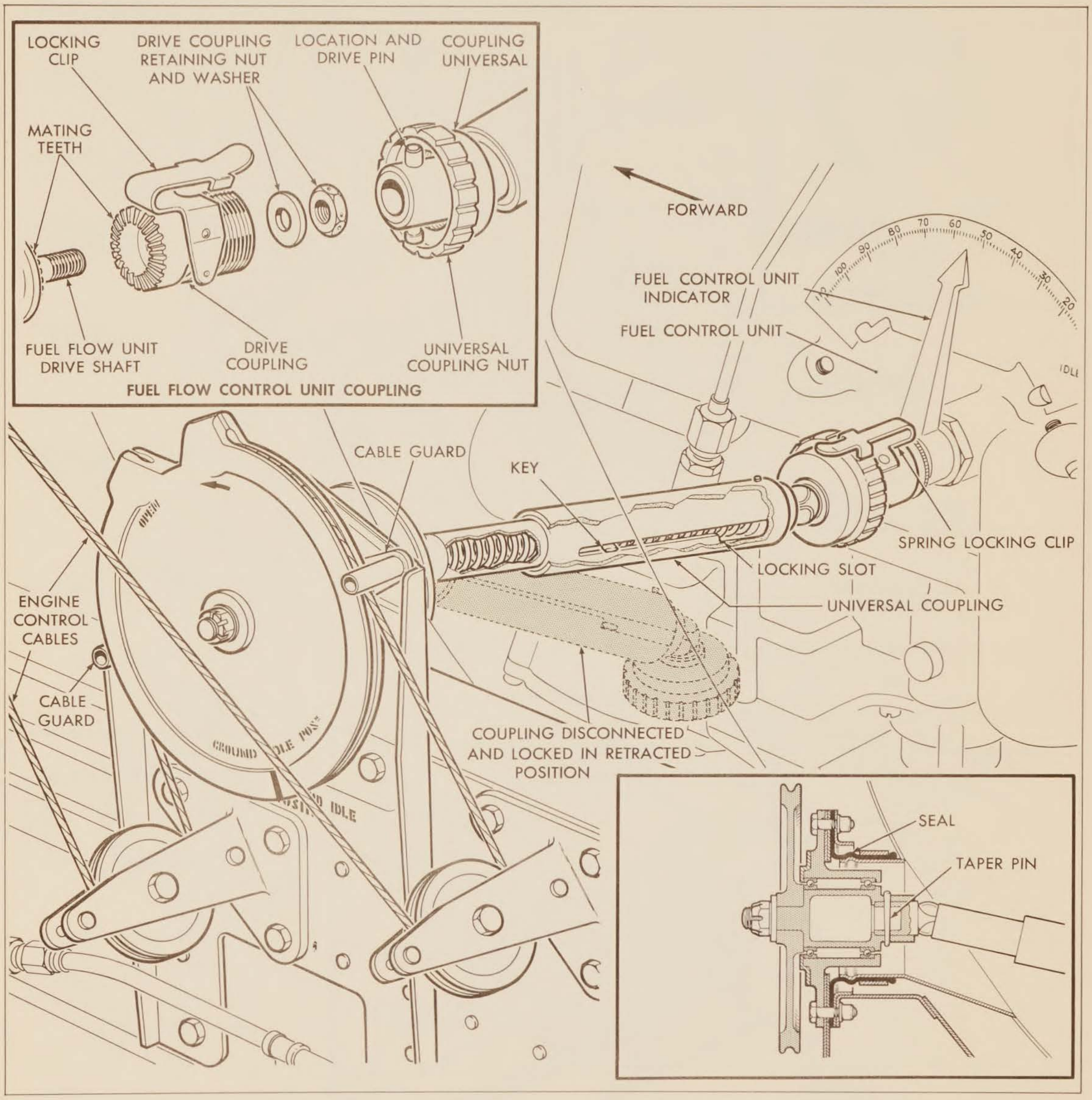


FIG. 29 ENGINE CONTROL DETAILS - R.H. ENGINE

UNCLASSIFIED

3046-103-1

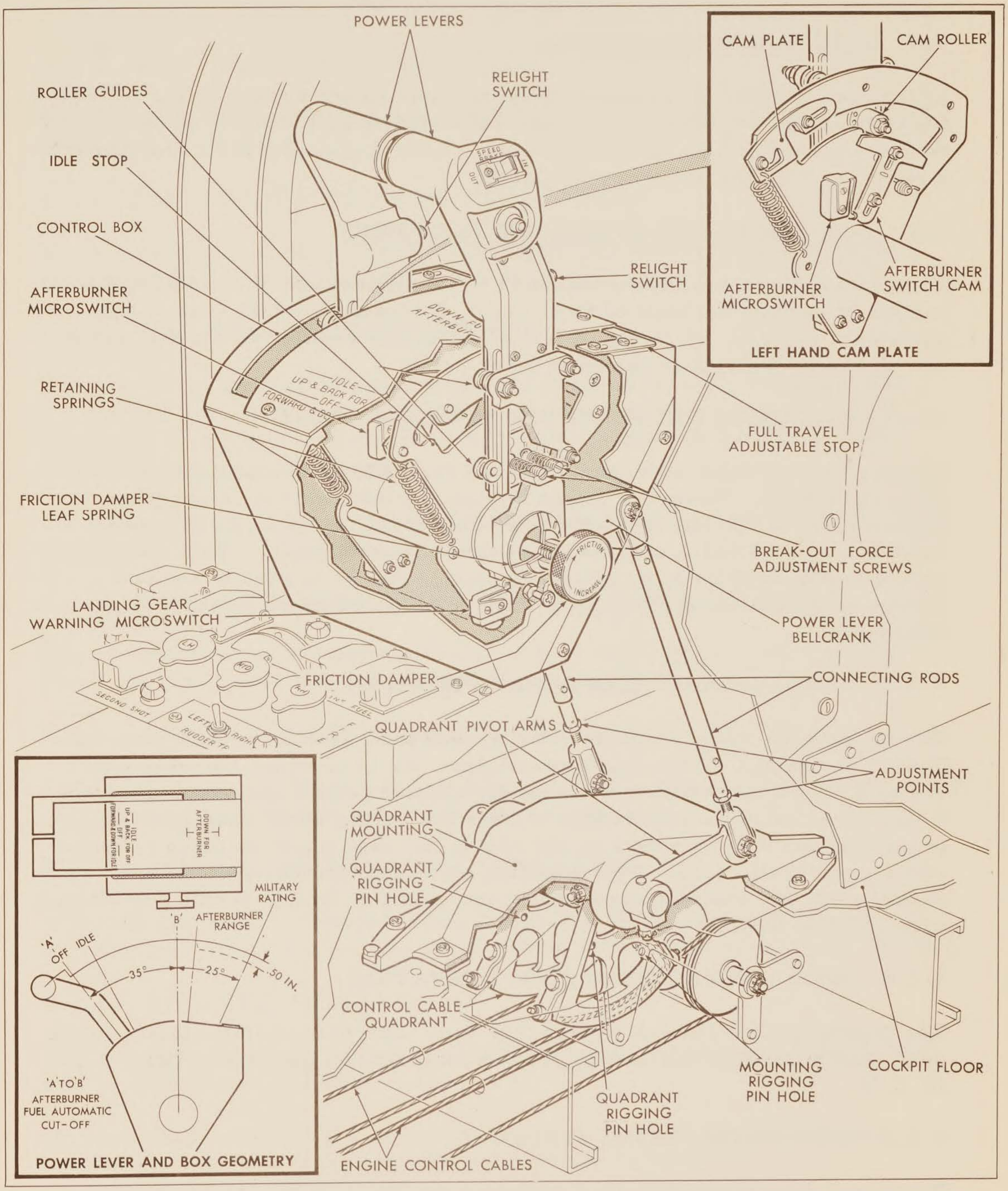


FIG. 30 POWER CONTROL BOX AND LINKAGE

9.0 ELECTRICAL SYSTEM

9.1 ELECTRIC POWER SYSTEM

It has been decided to install the Westinghouse power system in the ARROW 2. Circuits and installation details have been studied and most of the necessary design alterations have been completed. Test requirements for the ARROW 2 power system are being investigated

9.2 ELECTRICAL SUB-SYSTEM DEVELOPMENT

Important changes and developments in the electrical sub-systems are discussed in various sections of this report, for example, fuel system and armament system. However, a list of the major changes to the sub-system circuits over the report period is given below.

9.2.1 EMERGENCY ALTERNATOR SUPPLY - ARROW 1

Balanced phase loads have been achieved for ARROW 1 under normal operating conditions. However, it was discovered that when the emergency alternator is in use (feeding only essential loads), an unbalanced loading condition existed which created unbalanced phase voltages. This was corrected by the addition of a voltage dropping resistor in each of the two phases of the emergency supply to achieve balanced phase voltages at the equipment.

9.2.3 NOSE LANDING GEAR DOOR OPERATION

During the nose wheel steering valve ground test, it was found that due to a fault in the design of the clutch mechanism, the nose wheel steering was engaged before the wheel and the rudder control pedals were synchronized. It was found necessary therefore to alter the wiring of the nose gear system to incorporate a limit switch in the line to the nose wheel steering solenoid valve on aircraft 25201. This operates as an engagement switch, and ensures that the steering does not engage until the control pedals are aligned with the nose wheel.

9.2.3 NOSE LANDING GEAR DOOR OPERATION

The landing gear electrical system has been altered to permit closing of the nose gear door when the landing gear is extended during flight. (Ref. para 14.1.2).

9.2.4 ANTI-SKID BRAKING SYSTEM

Electrical provisions have been made for the Messier Ministop anti-skid system which it is expected, will be evaluated in ARROW 1. Lights will be incorporated to indicate to the pilot when the anti-skid valves are operating.

9.2.5 WESTINGHOUSE POWER SYSTEM

The Westinghouse alternator and control system circuitry has been completed for the ARROW 2.

9.2.6 MISSILE HYDRAULICS CONTROL SYSTEM

Missile hydraulic control circuits have been reissued to comply with the requirements for aircraft 25203 (Ref. para. 16.1.5).

9.2.7 FUEL PROPORTIONING SYSTEM

To prevent the ARROW 1's fuel proportioning system from locking in the bypass position when power is applied with the aircraft on the ground, it was found necessary to route the lock on feature through the landing gear scissors switch control.

The refuelling system wiring on ARROW 2 has been changed to that it is now operated from the emergency DC bus.

9.2.8 AIR CONDITIONING REQUIREMENTS - AIRCRAFT 25204-5

The air conditioning system electrical circuit requirements for aircraft 25204 and 25205 (ASTRA I development vehicles) have now been finalized.

9.2.9 OXYGEN CONTENTS CAPACITANCE GAUGE - ARROW 1

Oxygen contents indication has now been provided in the rear cockpit by means of a repeater indicator for the capacitance-type gauge.

9.2.10 JET PIPE TEMPERATURE INDICATION

A facility has been introduced which allows the utilization of the starting vehicle A.C. power for jet pipe temperature indication during engine starting.

9.2.11 FIRE PROTECTION

The fire protection system has been altered to incorporate overheat as well as fire indication. (ARROW 2 only).

Incorporation of a pilot's crash switch permits fire extinguishant to be supplied to the three fire zones when the switch is operated.

9.3 ARROW 2 ELECTRICAL SYSTEM STUDIES

The formulation of procedures for breadboard testing of the ARROW 2 electrical system is progressing, and load values on the buses are presently being determined.

Preliminary studies are proceeding on the design of circuits for level indication of the ASTRA I magnetron cooling and hydraulic system.

The ARROW 2 missile hydraulic control system (Ref. para 16.2.4) and engine performance indication system (Ref. para 5.3) are at present under design consideration.

10.0 AIR CONDITIONING

10.1 COCKPIT ENVIRONMENTAL TESTS

The cockpit environment tests were started during this quarterly period, using the aircraft metal mock-up. The tests have been concerned primarily with cockpit air distribution.

The initial test run with the original air distribution duct configuration indicated an unsatisfactory distribution of air, and excessive noise in the cockpit. For the second run, baffle plates were added to the air outlets to prevent direct drafts on the crew. The air distribution was improved but was not entirely satisfactory, and the cockpit sound level was in the region of 100 decibels. New outlets providing reduced outlet velocities were tried on the distribution system, and a test run on the rear cockpit yielded improved results. In a test run for both cockpits with the new outlets, the air distribution was considered satisfactory, but the noise level, at approximately 105 decibels, was still too high. Tests have shown that most of the noise is being generated in the ducting. To provide the necessary noise attenuation, the air outlets will be provided with silencers. Prototypes of these silencers are being manufactured and will be laboratory tested by AVRO. When satisfactory units have been developed, they will be installed in the test cockpit and the distribution tests will be repeated. A noise level of 80 to 90 db would be considered satisfactory. AVRO test pilots participated throughout this series of tests.

10.2 DISTRIBUTION OF EQUIPMENT COOLING AIR ON AIRCRAFT 25201

Tests to determine the distribution of equipment cooling air have been completed on aircraft 25201. Three runs were made in which duct restrictors were installed to adjust the distribution of cooling air. The final run indicated the following distribution which was considered satisfactory:

Equipment Group	% Total Equipment Cooling Air Flow
Rectifier Unit	43.5
Dorsal electronics (in the dorsal electronics compartment)	3.6
Fuselage electronics	20.1
Dorsal equipment (in the dorsal equipment compartment)	5.6
Windshield de-icing transformer	5.8
Battery	1.5
Radar nose	19.9
	100

10.3 ARROW 2 COCKPIT SYSTEM CONTROL (See Figure 32)

Flow control for the ARROW 2 system has been under study to establish suitable limits for mass flow, and to design or select equipment which will maintain the flow within these limits. The upper limit for cockpit mass flow is governed by the maximum air velocity as dictated by crew comfort. The lower limit is governed by the cooling capacity which is dictated by crew comfort and armament cooling requirements. The upper and lower limits have not, as yet, been definitely established. Cockpit environment tests and design studies on missile cooling are presently in progress, and when completed, the flow limits will be more clearly defined. System engineering to date has been based on a design flow of 27.5 ± 2.5 lb/min. The limits defined by this design flow are shown in Figure 31.

Total system flow is controlled by a variable area inlet nozzle on the refrigeration turbine. The method of controlling the turbine inlet nozzle area determines the variation in cockpit flow.

Existing equipment for nozzle area control consists of a pneumatic controller which senses turbine outlet static pressure. The pneumatic controller adjusts the nozzle area to alter the downstream static pressure so that it corresponds with a reference pressure, which is programmed against altitude. The programmed pressure schedule was chosen to give a reasonably constant cockpit flow. The tolerances in this control and in the cockpit pressure regulator result in a possible variation of cockpit flow shown in Figure 31. The possible cockpit flow variation of ± 33% is obviously well outside the design limits. However, this variation in mass flow may be acceptable, depending on the limits which will be established by the cockpit environment tests and the missile cooling studies.

The vendors of the pneumatic controller and the cockpit pressure regulator, AiResearch Manufacturing Company and Normalair (Canada) Ltd. respectively, were requested to investigate the possibility of reducing the controlling tolerances in their equipment. Neither vendor could guarantee this, however, and an investigation is under way to find an alternative means of controlling the mass flow, which would more closely approximate the design flow limits.

The use of a flow sensor in the cockpit inlet duct, as an alternative means of controlling the turbine inlet nozzle area, is now being considered. An AiResearch proposal indicates a cockpit flow variation as shown in Figure 31. This control system is obviously no improvement over the existing control system. A proposal is being submitted by Normalair which will be subjected to test by AVRO. In the meantime, AVRO is conducting studies and tests independently of potential vendors.

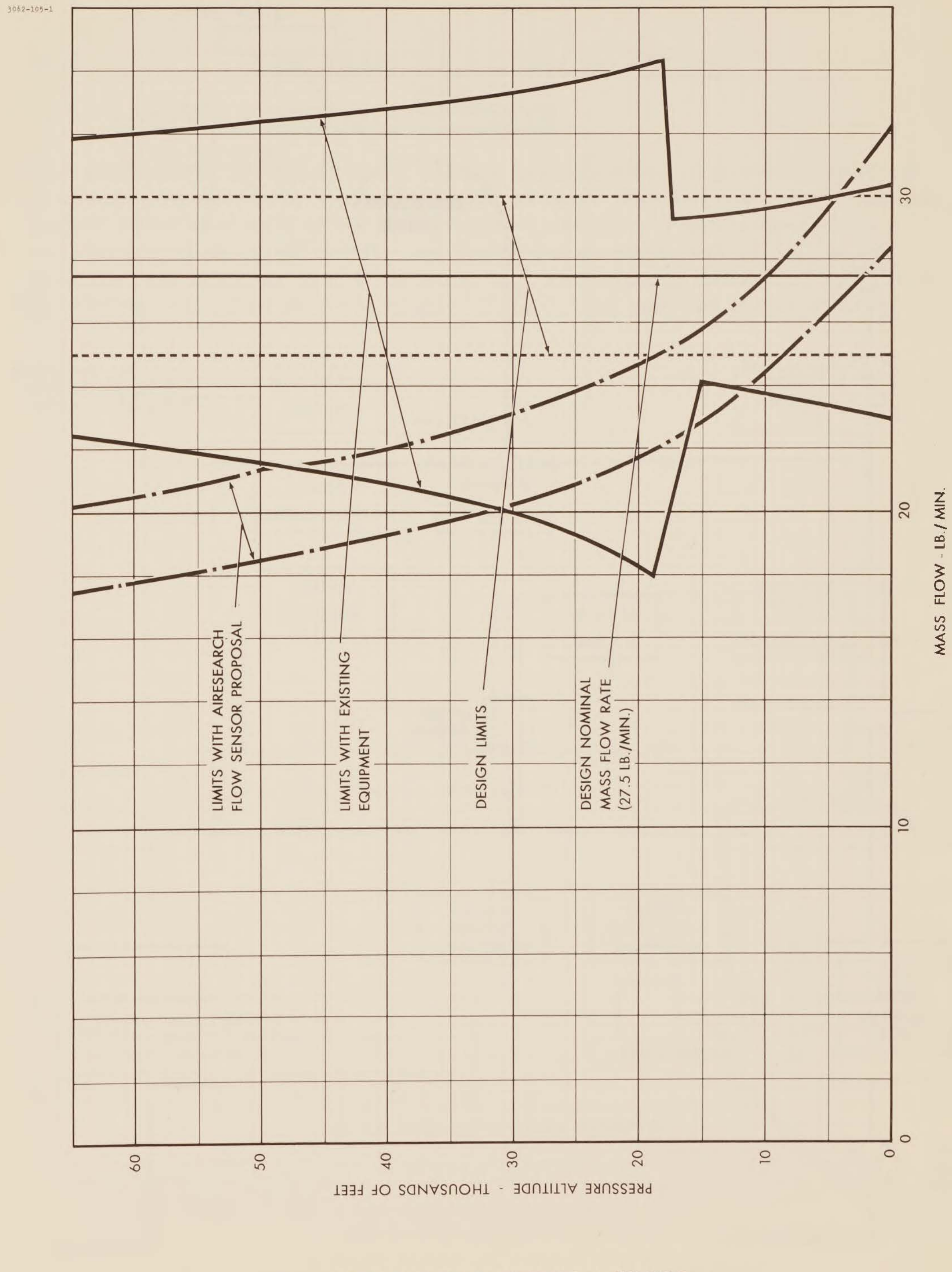


FIG. 31 COCKPIT AIR MASS FLOW - ARROW 2

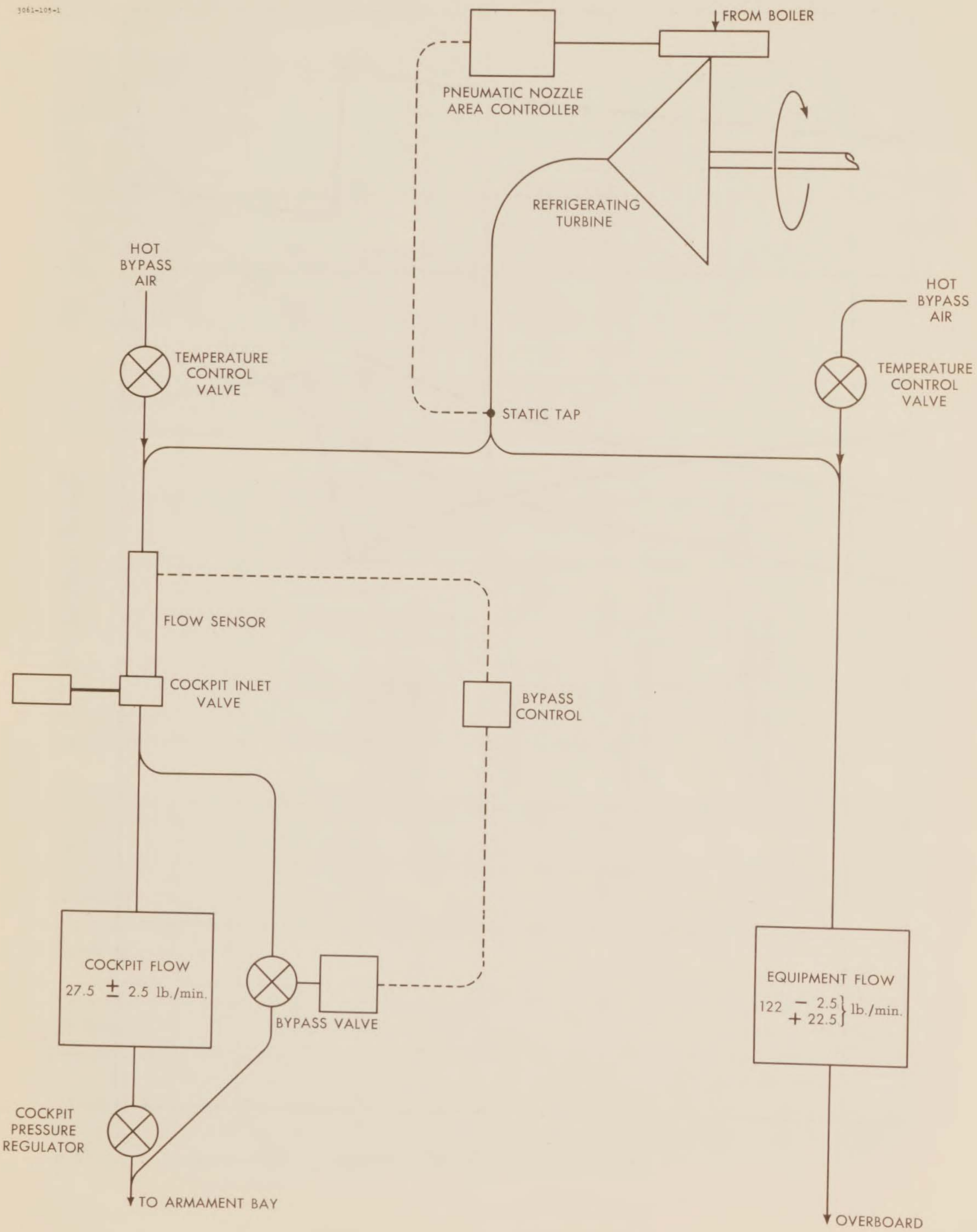


FIG. 32 SYSTEM FLOW CONTROL-ARROW 2 AIR CONDITIONING

11.0 LOW PRESSURE PNEUMATIC SYSTEMS

11.1 ARROW 2 - PITOT-STATIC SYSTEM

The pitot-static system (Figure 33) has been modified to provide static air pressure for an aileron control altitude switch. The aileron control altitude switch which is a part of the flying control system, is set to operate at a pressure altitude of 45,000 ft. and energizes an electrical circuit which deflects both ailerons to a 4° up attitude. This aileron deflection was incorporated to minimize elevator angle. This in turn reduced the trim drag.

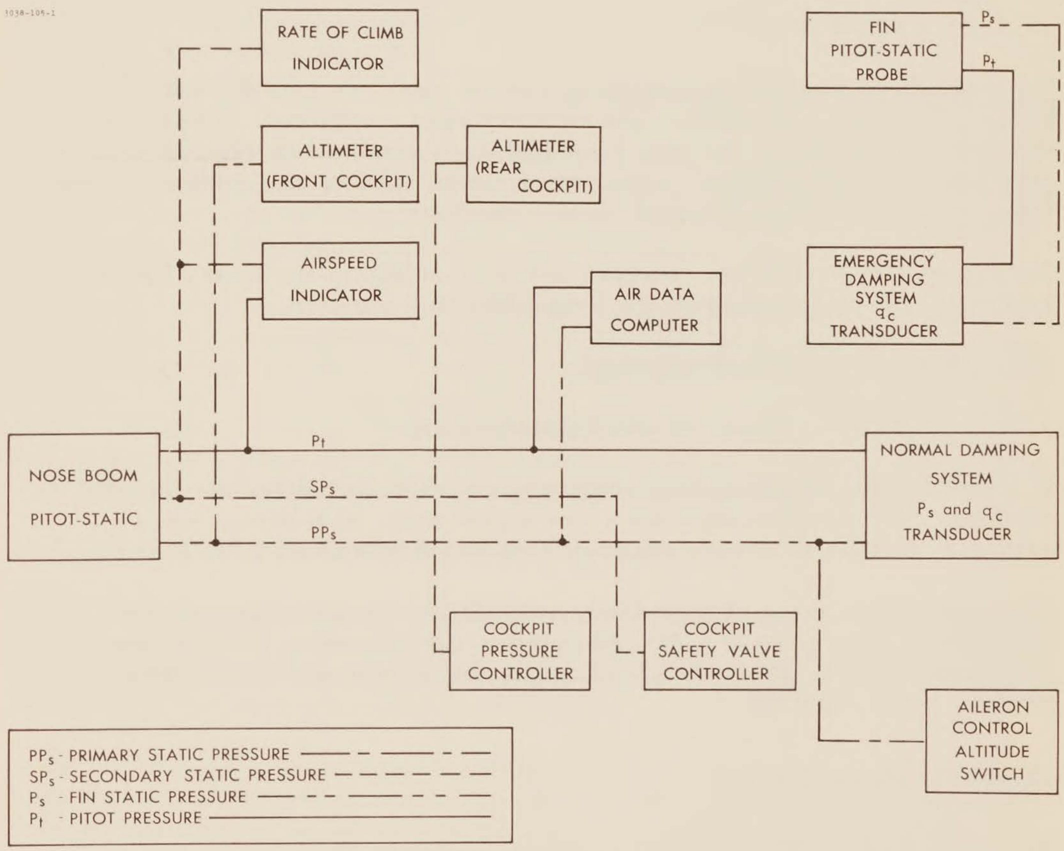


FIG. 33 PITOT STATIC SYSTEM SCHEMATIC

12.0 FIRE PROTECTION

12.1 FIRE EXTINGUISHING SYSTEM

12.1.1 ARROW 1

The distribution tests noted in the preceding ARROW Quarterly Technical Reportare now complete. The tests were conducted under normal ambient temperature conditions, using air and water separately.

Based on the experimental results, the discharge nozzles of the system installed in the first aircraft were equipped with orifice disks of a size which would give the predetermined distribution of extinguishing agent.

12.1.2 ARROW 2

An analysis of the fire extinguishing system requirements for the complete range of operating conditions has recently been completed. The location of discharge nozzles and the pipe runs and sizes required to give the desired distribution of extinguishing agent were established by the analysis. Installation design will also be based on this analysis.

Additional fire protection, similar to that provided in the first ARROW 1 aircraft, will be installed in the first ARROW 2 aircraft.

12.2 FIRE DETECTION SYSTEM

12.2.1 ARROW 1 FIRE DETECTION SYSTEM

A Water Kidde, fire detection system has been selected for the ARROW 1 aircraft. The system provides a fire signal only, and incorporates a rate-rise feature to ensure adequate fire warning without false alarms.

Pre-installation tests of the system installed in the first aircraft have been completed. (See para 2.1.2). As noted in the preceding ARROW Quarterly Technical Report, some modifications to the system as delivered by the vendor, were required.

12.2.1.1 Basic Circuit

The basic fire detection circuit is shown in Figure 34.

Three such circuits are installed in the ARROW 1 aircraft. A fire detection circuit is provided for each of the two engines, and a third independent circuit is provided for the hydraulic equipment bay.

The control unit in each case is designed to operate with a detector element resistance value R_T in the range 200 to 2000 ohms.

12.2.1.2 Results of Pre-Installation Tests

The two engine fire detection circuits gave fire warning signals at temperatures within the specified range of fire temperatures. The hydraulic equipment bay circuit, however, produced fire warning signals at a temperature below the lower limit of the specified temperature range.

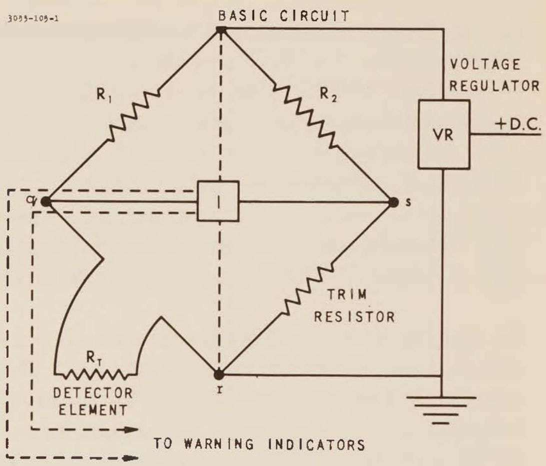


FIG.34 BASIC CIRCUIT - ARROW 1 FIRE DETECTION SYSTEM

The hydraulic bay fire detection circuit is required to give a fire warning signal at a temperature of 425°F ± 10%. On test, fire signals occurred at a temperature of approximately 350°F. This indicated that the detector loop resistance was improperly matched with the control unit.

Although a value of 63 ohms was specified by the manufacturer for the trim resistor R_V, a subsequent investigation by AVRO showed that a resistance of 17.5 ohms was required to produce a fire signal at 417°F. The detector loop resistance R_T corresponding to this condition was approximately 140 ohms. In order to more closely approximate the lower R_T design limit of 200 ohms, the manufacturer recommended a trim resistor of 22 ohms. Fire warnings are now obtained within the specified temperature range.

12.2.2 ARROW 2 FIRE DETECTION SYSTEM

The RCAF has requested that overheat warning be provided in the ARROW 2 fire detection system, in addition to the normal fire warning signal. Report 70/Systems 11/34, Fire and Overheat Detection, Sept. 1957, covering such a proposal, has been submitted to the RCAF.

A specification for the system has been submitted for tender to the various suppliers of fire detection systems. Walter Kidde and Company is the only manufacturer known to have a system available which meets the RCAF requirements and at the same time satisfies AVRO requirements for performance, weight, bulk and installation simplicity. Consequently, preliminary design work has been based on the Water Kidde system.

12.2.2.1 Operation of Fire Detection System

The following discussion is supplementary to the description of the operation of the Water Kidde continuous wire fire detection system, given in Report 70/Systems 11/73, Operation of Continuous Wire Fire Detection System, November 1957. The basic principles involved in system design are reviewed briefly and the principal features of the system are noted.

12.2.2.2 Basic Circuit

The Wheatstone bridge circuit forms the basis of the Water Kidde fire detection system. The bridge network in its simplest form is shown in Figure 35.

R_1 and R_2 are fixed resistances, R_V is a variable resistance, and R_T is a temperature sensitive resistance. Power for the network is provided by a supply E and bridge balance is monitored by an indicator circuit I. The bridge is balanced with no signal across I when $R_1/R_T = R_2/R_V$. A change in R_T due to a temperature change unbalances the bridge and produces a signal at I.

In the ARROW fire detection system, the temperature sensitive resistance R_T corresponds to the continuous wire detector, and the remainder of the network is incorporated in the control unit.

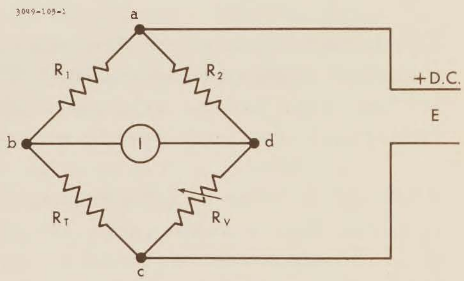


FIG.35 BASIC WHEATSTONE BRIDGE CIRCUIT

12.2.2.3 Resistance - Temperature Properties of Materials

In general, the resistance R_T of a given material at any temperature T may be expressed in terms of its resistance R_O at some reference temperature T_O by the following power series:

$$R_T = R_O \left(1 + at + bt^2 + ct^3 + \text{etc.}\right) \qquad \text{Equation (i)}$$

where a, b, c, etc. are constants and

t = the change in temperature

$= T - T_O$

For most metallic conductors, sufficient accuracy is obtained by simplifying equation (i) to

$$R_T = R_O \left(1 + \alpha t\right) \qquad \text{Equation (ii)}$$

Where

α = temperature coefficient of resistance in ohms/ohm/degree

The linear relationship expressed by Equation (ii) is not sufficiently accurate for the non-metallic materials and an expression of the form of Equation (i) must be used.

With reference to Equation (ii), the temperature coefficient is positive for most metallic conductors since resistance increases with temperature. For non-metallic materials, such as carbon, liquids, electrolytes, most dielectrics, and insulating materials, resistance decreases with increase in temperature. Consequently, the temperature coefficient is negative for the non-metallic materials.

12.2.2.4 Continuous Wire Detector

The detector element consists of two conducting wires embedded in a non-metallic material and contained within an Inconel tube. A typical cross-section of such an element is shown in Figure 36. One of the two conducting wires is connected to ground and the other is connected to the power source. Thus, to complete an electrical circuit, current must flow from the hot (power) wire to the cold (ground) wire. The only path available to current flow is through the non-metallic material isolating the two conductors from each other. The total current which can flow in this circuit is then dependent on the length and temperature of the detecting element.

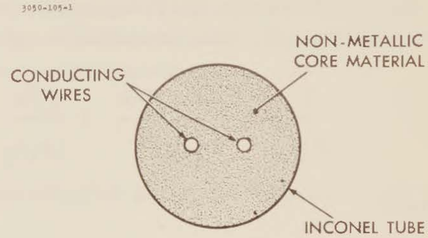


FIG. 36 CROSS SECTION OF FIRE DETECTOR ELEMENT

12.2.2.5 Resistance of Detector Element

The detector element may be considered to be composed of a number of resistances connected in parallel. This is represented in Figure 37 where the total detector length L is sub-divided into a number of equal lengths. For any typical element of length x, the core resistance to current flow between the two conducting wires may be represented by r_i. Thus, the total resistance R_T of the detector may be obtained from

$$\frac{1}{R_T} = \sum_{i=1}^{L/x} \frac{1}{r_i} \qquad \text{Equation (iii)}$$

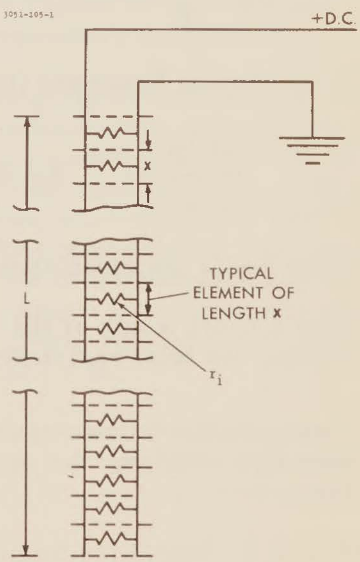


FIG. 37 REPRESENTATION OF DETECTOR ELEMENT RESISTANCE

If the full length of the detector is subjected to the same temperature, then the r_i elements are all equal and

$$\frac{1}{R_T} = \frac{L/x}{r_i} \qquad \text{Equation (iv)}$$

Resistance vs temperature curves for uniform temperature distribution over the full length of the detector are given in Figure 38. These are typical experimentally determined curves which show the exponential nature of the relationship and demonstrate the dependence of the value of R_T on element length and core material.

In the case where only a portion of the total detector length is exposed to temperature T, the resistance of the detector is obtained from

$$\frac{I}{R_T} = \frac{L/x - \ell/x}{(r_i)_o} + \frac{\ell/x}{(r_i)_T} \qquad \text{Equation (v)}$$

where

ℓ = the length of detector exposed to temperature T

$(r_i)_o$ = resistance of an element of length x when exposed to the normal ambient temperature T_o

$(r_i)_T$ = resistance of an element of length x when exposed to temperature T.

By comparing Equations (iv) and (v), the following inequality can be established

$$\frac{(r_i)_T}{L/x} < R_T < \frac{(r_i)_o}{L/x}$$

where R_T is the resistance value obtained from Equation (v).

When a short length of the detector is exposed to a fire for an extended period of time, the total resistance of the detector will approach $(r_i)_T / \frac{L}{x}$

Thus, a delayed fire warning signal would be given by the system. To overcome this problem, the system is made sensitive to the rate of temperature rise.

12.2.2.6 Rate-Rise Sensitivity

The bridge balance indicator circuit I (Figure 34) without rate sensitivity, monitors only the difference of potential across the indicator. To provide prompt fire warning where a fire is localized over a short length of detector, the bridge balance sensing circuit is modified to incorporate a sensitivity to the rate of change of potential across the detector element. The principles of operation of the indicator circuit and details of the circuit modification are beyond the scope of this report. They are discussed in Walter Kidde and Company report No. MR-600 "Discussion of the Rate Sensitive and Combination Features Used in the Kidde Detection Control Units", May, 1956.

The rate of change of potential across the detector is obviously dependent on the rate of change of the detector resistance R_T. The rate of change of (dR/dt) of R_T is in turn dependent on the rate of change (dT/dt) of the temperature T. The typical curves shown in Figure 39 illustrate the relationship of the rate of change of resistance to the rate of change of environmental temperature and the total element resistance.

12.2.2.7 Temperature Design Points

For a given installation, the following temperatures must be established:

(a) maximum ambient

(b) overheat trip setting

(c) fire trip setting

These temperature design points are illustrated in Figure 40 for a typical 20-foot detector element exposed to a uniform temperature distribution.

Below the maximum ambient temperature no warning signal is provided, regardless of the rate of temperature rise. At a temperature corresponding to the overheat trip setting, the system becomes sensitive to the rate of temperature rise and should the rate-rise be characteristic of a fire, a fire warning signal is given immediately. For temperatures between the overheat trip setting temperature and the fire trip setting temperature an overheat warning signal is established, provided the rate of temperature rise is below the characteristic rate-rise of a fire. For temperature in excess of the fire trip setting temperature, a fire signal is given, regardless of past temperature history or existing rates of temperature rise.

If the temperature distribution along the length of detector is not uniform, the total resistance of the detector corresponding to the temperature design points determines the localized temperatures at which overheat and fire signals are given.

12.2.2.8 Actual Bridge Circuits

Fire Warning

The fire warning circuit as used in the ARROW 1 system is shown in simplified form in Figure 34. The bridge balance sensing circuit is represented by I and incorporates the rate sensitive feature. Output signals from the transistorized circuits I operate the warning light relays. Bridge input voltage is regulated by a transistor type voltage regulator (VR) to make the rate sensitive components of I insensitive to variations of bridge input voltage. The trip point temperature is determined by selecting a proper resistance value for the trim resistor R_V.

Combination Overheat and Fire Warning

A simplified circuit diagram of the system proposed for the ARROW 2 is shown in Figure 41. Separate bridge balance sensing circuits are provided for overheat warning and fire warning. Since the trip point temperatures for overheat and fire are different, a separate trip point adjusting resistor is required for each sensing circuit.

The warning light circuit is wired so that the overheat warning lights are extinguished when the fire warning lights are illuminated.

The fire warning sensing circuit I_F incorporates the rate sensitive feature as in the ARROW 1 system. The overheat sensing circuit I_o however, is not sensitive to the rate of temperature rise in the detector element.

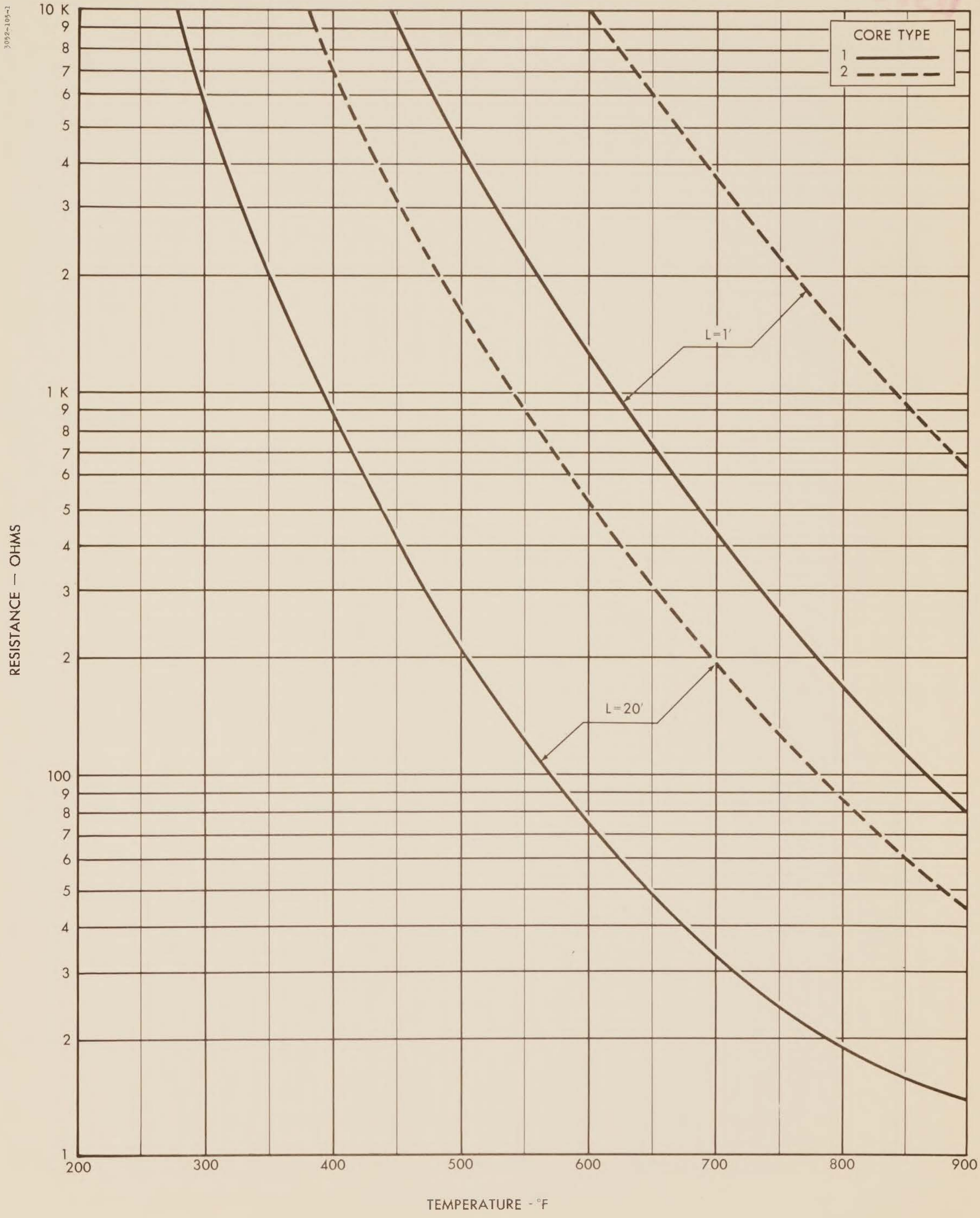


FIG. 38 TYPICAL RESISTANCE VS. TEMPERATURE CURVES FOR DETECTOR ELEMENT OF DIFFERENT LENGTHS AND CORE MATERIALS

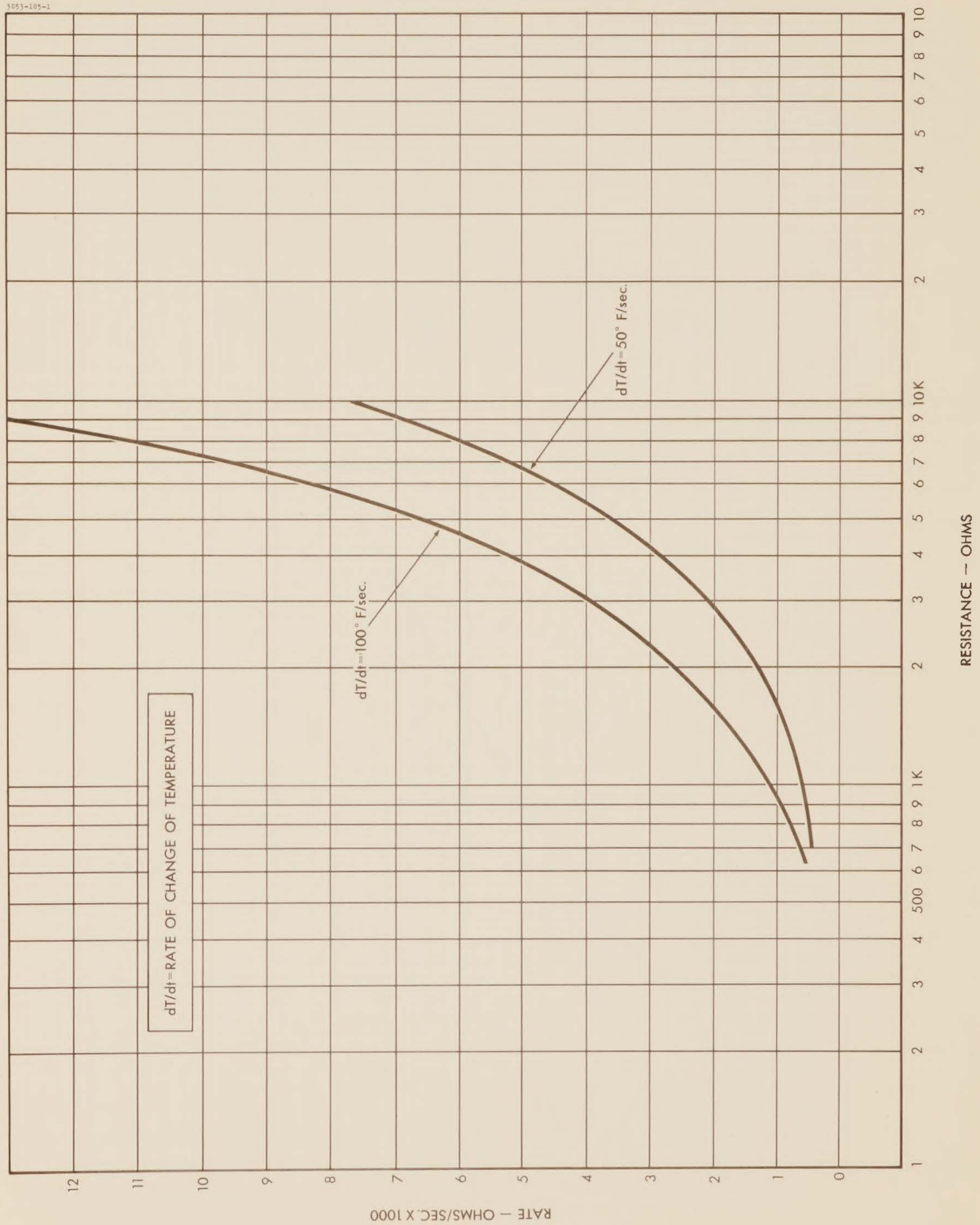


FIG. 39 RATE OF CHANGE OF RESISTANCE OF FIRE DETECTOR ELEMENT

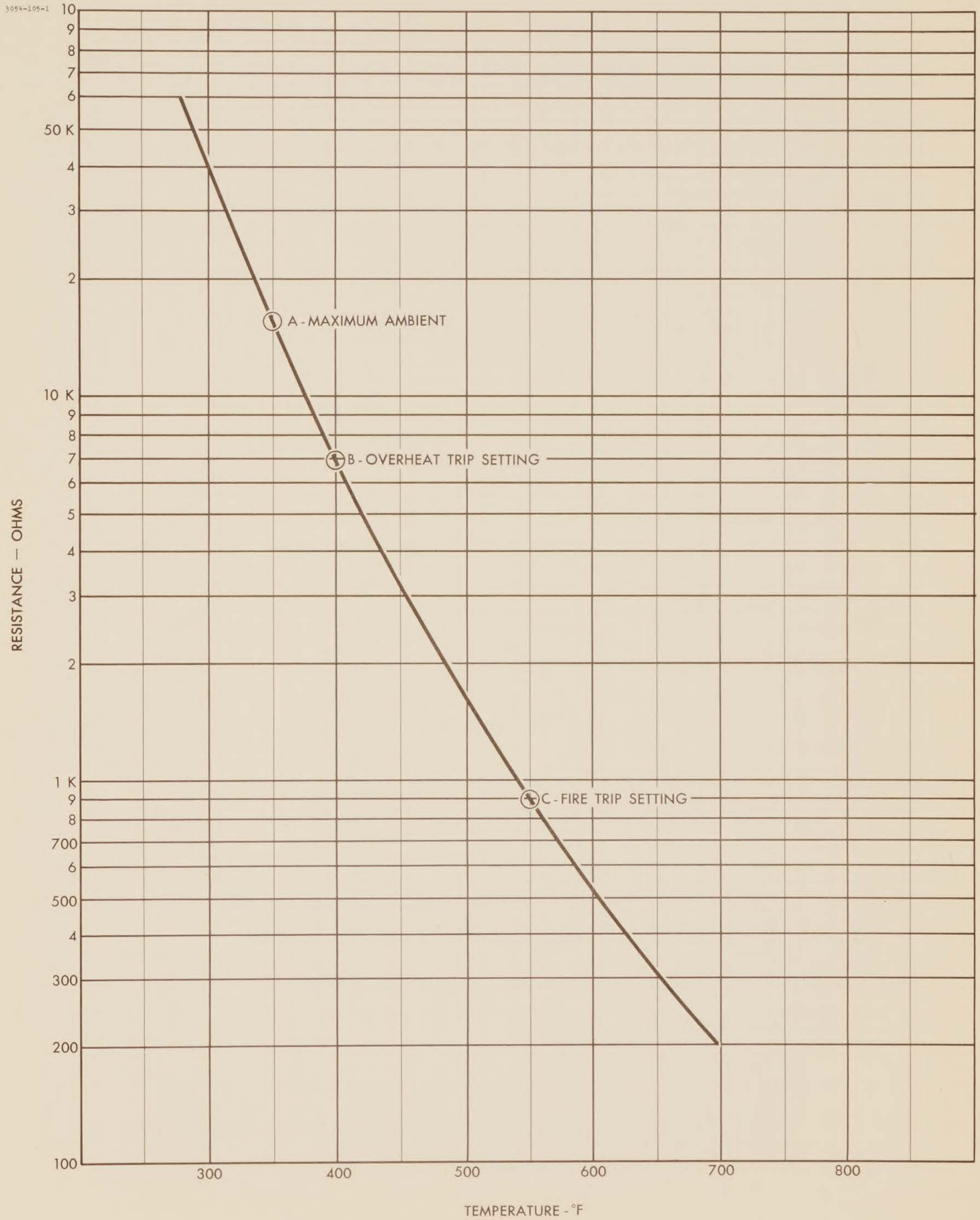


FIG. 40 TEMPERATURE DESIGN POINTS FOR A 20 FOOT LENGTH OF A TYPICAL FIRE DETECTOR ELEMENT

3056-105-1

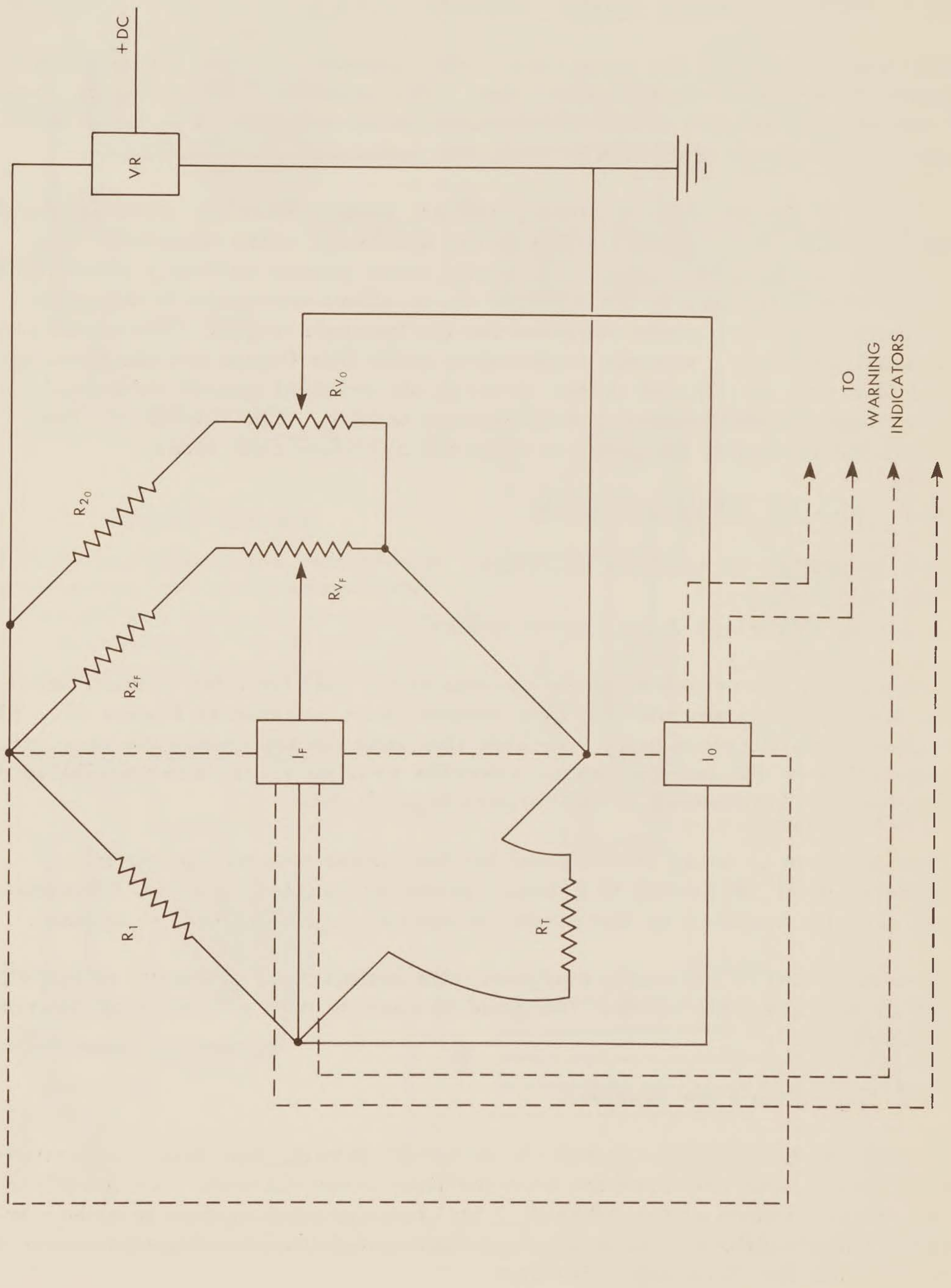


FIG. 41 BASIC FIRE DETECTOR BRIDGE CIRCUIT WITH OVERHEAT WARNING - ARROW 2

13.0 FUEL SYSTEM

13.1 FUEL BOOSTER PUMP - ARROW 1 AND 2

Fuel vaporization at the pump inlet, with consequent vapor choking of the pump has occurred at high pump rpm. The solution of this problem requires a redesign of the fuel intake and adjacent pump housing. The pump manufacturer has been requested to make the necessary modifications.

As noted in the previous quarterly report, pump discharge rates are below specification requirements although the discharge rates obtainable are adequate for the J75 engines. Since the same pumps with only minor modifications will be used in the ARROW 2, an effort was made to determine maximum fuel flow rates required for the Iroquois engine. Orenda Engines Limited, however, was not prepared to quote this figure and the maximum fuel flow rate of 100,000 lb/hr. given in the original specification will govern pump qualification and acceptance testing. The vendor has been requested to modify the pump to meet the specified flow rates.

13.2 ARROW 1 FUEL SYSTEM

13.2.1 FUEL TRANSFER SYSTEM

13.2.1.1 Fuselage Tank Fuel Transfer

A schematic presentation of the portion of the fuel transfer system between the fuselage tank and the fuel flow proportioner is given in Figure 42. The vendor's qualification tests indicated that both the fuel pressure regulating valve (Figure 43) and the bypass override sensing valve were unreliable due to improper functioning of the bellows in each valve.

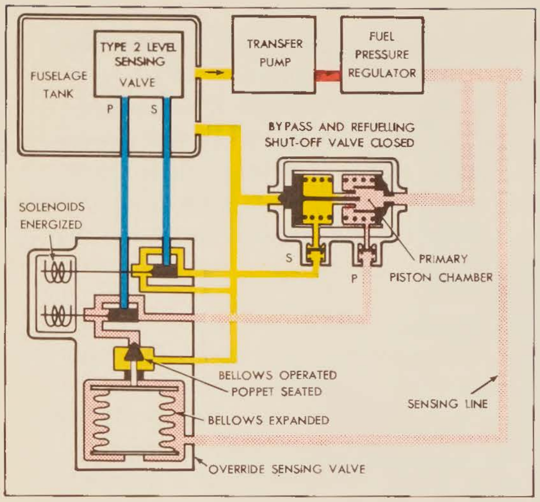
This problem is being investigated by the vendor who will perform any redesign work necessary to produce properly functioning units. These units will then be modified by the vendor to operate in the ARROW 1 system.

Aircraft 25201 is presently equipped with satisfactory pressure regulators and bypass override valves, designed to operate with a 25 psia pressurization system.

13.2.1.2 Fuel Flow Manifold

As noted in the previous quarterly technical report, fuel flow proportions have so far been procured for only the first three aircraft. To permit the use and evaluation of the ARROW 2 fuel management system in either or both of the other ARROW 1 aircraft, fuel flow manifolds have been designed to replace the fuel flow proportioners.

3064-105-1

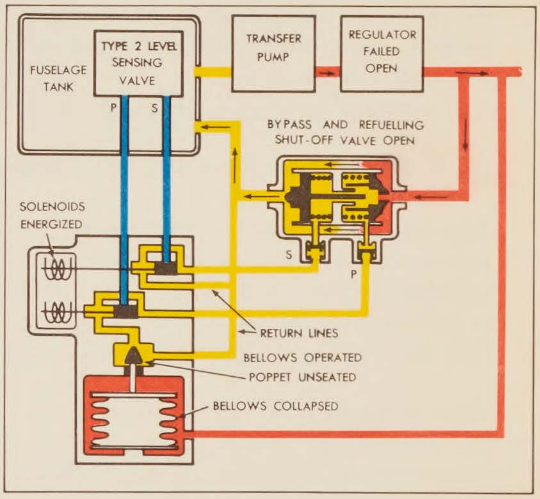


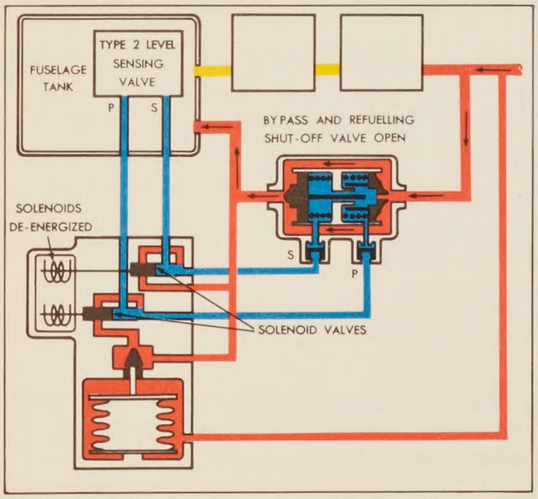
NORMAL

DURING NORMAL OPERATION, THE SOLENOIDS ARE ENERGIZED, AND THE SOLENOID VALVES CLOSE OFF THE PRIMARY AND SECONDARY SERVO LINES TO THE LEVEL SENSING VALVE, AND OPEN THE PRIMARY AND SECONDARY SERVO LINES FROM THE BYPASS AND REFUELLING SHUT-OFE VALVE. THE OUTLET PRESSURE OF 25 P.S.I.A. IS INSUFFICIENT TO CONTRACT THE BELLOWS, SO THE BELLOWS OPERATED VALVE REMAINS CLOSED. THIS BLANKS OFF THE PRIMARY SERVO LINE AND TRAPS FUEL IN THE PRIMARY CHAMBER OF THE BYPASS AND SHUT-OFF VALVE WHICH PREVENTS THE VALVE FROM OPENING AND FEEDING EUEL BACK TO THE TANK.

REGULATOR FAILED OPEN

IF THE REGULATOR FAILS, IT NORMALLY FAILS IN THE OPEN POSITION AND THE FUEL OUT-LET PRESSURE RISES. AT 28 P.S.I.A., THE PRESSURE BEGINS TO OPEN THE BELLOWS VALVE AND AT 30 P.S.I.A. IT IS FULLY OPEN. THIS ALLOWS THE FUEL TRAPPED IN THE PRIMARY CHAMBER OF THE BYPASS AND SHUT-OFF VALVE TC DISSIPATE, AND THE VALVE TO OPEN AND BYPASS EXCESS FUEL BACK TO THE TANK.





REFUELLING

DURING REFUELLING, THE TWO SOLENOID VALVES ARE DE-ENERGIZED WHEN THE MASTER REFUELLING SWITCH IS SWITCHED ON. THIS OPENS THE PRIMARY AND SECONDARY SERVO LINES TO THE LEVEL SENSING VALVE AND CLOSES THEM TO THE OVERRIDE SENSING VALVE. THE LEVEL SENSING AND THE BYPASS AND SHUT-OFE VALVE NOW CONTROL REEUELLING.

FUEL AT 25-28 P.S.I. | FUEL IN EXCESS OF 28-30 P.S.I. | FUEL AT REDUCED PRESSURE | REFUELLING SERVO LINES AT REDUCED PRESSURE

BLACK. Dr-No. 3064-105-1

FIG. 42 FUEL TRANSFER SCHEMATIC

13.2.1.3 Fuel Flow Proportioner Control

The fuel flow proportioner bypass valve is operated automatically by a low-level warning switch which is actuated by a liquid-level sensor. When the fuel level in the collector tank drops below the safe fuel level, the liquid-level sensor operates the low-level warning switch. This switch energizes the actuator to open the fuel proportioner bypass valve. A lock-on feature retains the bypass valve in the open position until after landing and shutdown of the aircraft power supply, when the bypass valve may again be closed by reenergizing the valve actuator.

On test, the proportioner bypass valve was opening whenever the aircraft electrical power system was initially switched on, even with the collector tanks filled. The liquid-level sensing system was giving an "out-of-fuel" signal which disappeared as soon as the controls warmed up. The fault has been remedied by altering the wiring to permit the proportioner bypass valve lock-on control to be disconnected from the circuit through a scissors switch when the landing gear is in the down position. In addition, the level sensing systems are now connected to the emergency DC supply so that they will remain powered throughout the flight.

13.2.2 SIMULATED MISSION TESTS - ARROW 1 FUEL SYSTEM TEST RIG

The ARROW 1 fuel system test rig duplicates the fuel tanks, transfer system and pressurization system for one side of the aircraft. The engine delivery system, which is downstream of the collector tank and fuel booster pumps, is not duplicated on the test rig. A collector tank drain pump, discharging to a fuel reservoir tank, replaces the booster pump of the aircraft system.

This system test rig was operated under simulated ARROW 1 flight conditions, the operating conditions being based on the flight cases specified in AIR 7-4. The original 25 psia pressurization system was used for these tests. Fuel temperatures, engine bleed air temperatures and pressures, and engine fuel flow requirements were determined over the flight envelopes of the selected missions and these conditions were simulated on the test rig. The fuel system operated satisfactorily throughout the tests.

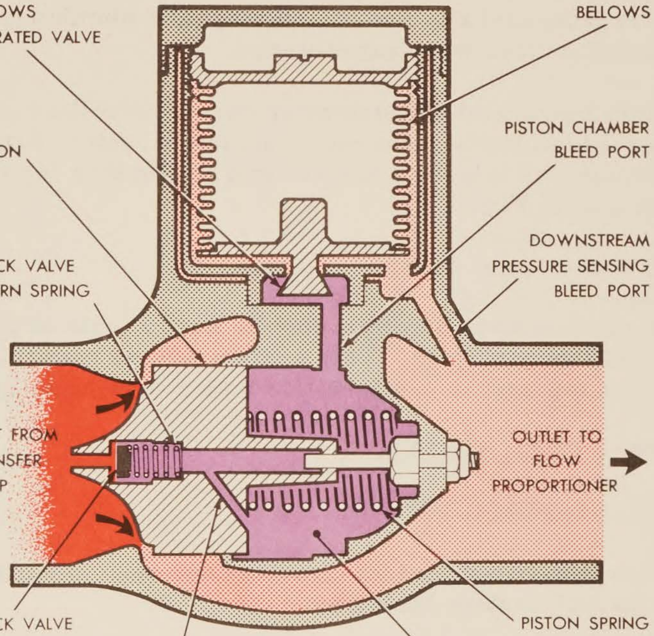
The system was also operated successfully at elevated fuel temperatures. The high temperature fuel tests were conducted with fuel heated to 185°F.

Operation of the fuel flow proportioner throughout these tests was encouraging. The difficulties previously experienced and noted in the last quarterly technical report appear to have been satisfactorily rectified.

13.2.3 AIRCRAFT 25201

13.2.3.1 Functional Ground Testing

Functional ground tests were completed during the first three weeks of



DESCRIPTION

FUEL DELIVERED BY THE TRANSFER PUMP OPENS THE PISTON AND CHECK VALVE AGAINST THE RETURN SPRINGS. THE FUEL THEN FLOWS PAST THE PISTON TO THE FLOW PROPORTIONER, AND ALSO PAST THE CHECK VALVE THROUGH THE BLEED PORT INTO THE PISTON CHAMBER.

OUTLET PRESSURE IS SENSED IN THE BELLOWS CHAMBER, AND WHEN THE PRESSURE RISES TO 25 P.S.I.A., THE BELLOWS START TO COLLAPSE AND CLOSE THE BELLOWS OPERATED VALVE. THE VALVE FULLY CLOSES AT 27 P.S.I.A.

THE BELLOWS OPERATED VALVE RESTRICTS THE BLEED FLOW FROM THE PISTON CHAMBER, AND THE RESULTANT BACK PRESSURE, ASSISTED BY THE RETURN SPRING MOVES THE PISTON TOWARDS ITS SEATING TO RESTRICT FUEL DELIVERY AND MAINTAIN THE OUTLET PRESSURE AT 25-27 P.S.I.A.

THE CHECK VALVE PREVENTS REVERSE FLOW SHOULD THE INLET PRESSURE FALL BELOW THE OUTLET PRESSURE.

(red)	FUEL AT TRANSFER PUMP DELIVERY PRESSURE
(pink stipple)	FUEL AT 25-27 P.S.I.A.
(mauve)	PRESSURE VARYING BETWEEN 25 P.S.I. AND INLET PRESSURE DEPENDING UPON PISTON POSITION

FIG. 43 FUEL PRESSURE REGULATOR

November and the results were generally satisfactory. Tests were conducted to check the operation of the tank pressurization system and the fuel transfer system. Apart from leaks in the transfer system plumbing, due to faulty assembly, no difficulties were experienced.

The fuel system functioned satisfactorily during refuelling operations, although several tank leaks occurred at structural joints during these tests. This was overcome by injecting sealant into the leaking joints,using the method illustrated in Figure 43.

13.2.3.2 Pressurization System

The fuel tank pressurization system for the first flights of aircraft 25201 will be the original 25 psia system. This system will be converted to the 19 psia system when the aircraft is grounded for major inspection and modification.

13.3 ARROW 2 FUEL SYSTEM

13.3.1 FUEL SYSTEM CONTROL

The fuel system electrical control circuits have been modified to accommodate changes in fuel system operation.

13.3.1.1 Refuelling Control Circuits

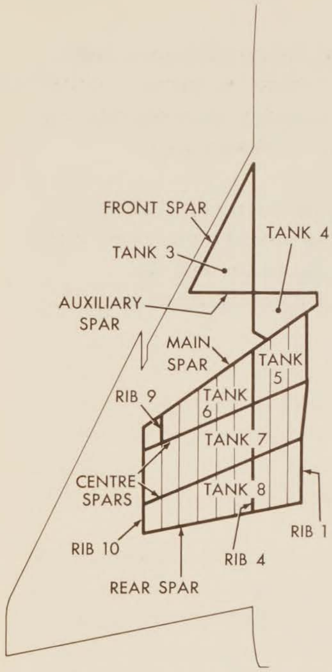
Electrical supply for the refuelling circuits has been changed from the main DC supply bus to the emergency DC bus. This permits operation of the fuel system in the refuelling mode when a secondary source of DC electrical power, such as the engine starting vehicle, is connected to the aircraft. A refuelling operation may be performed using the aircraft batteries, but to preserve battery charge an external source of DC power should be plugged in at the adaptor provided in the nose wheel well. Other circuit changes are due to the relocation of the collector tank override solenoid.

The override solenoid was previously an integral part of the collector tank level sensing valve but is now integral with the combined refuelling and shut-off valve. The solenoid permits the combined transfer and refuelling valve to operate in the transfer mode when energized, and in the refuelling mode when de-energized.

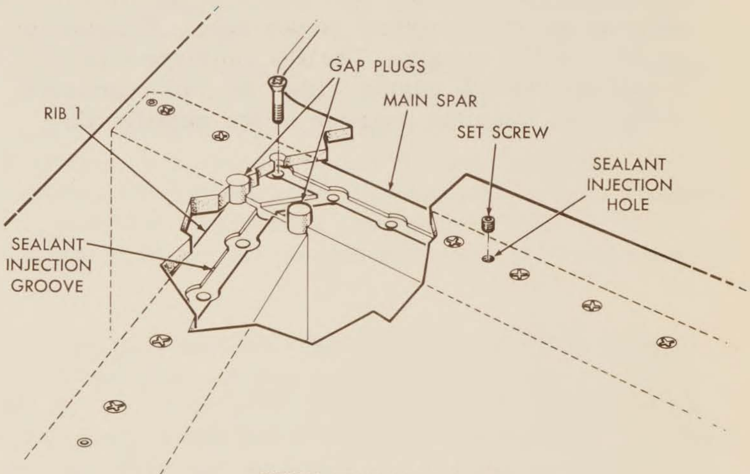
13.3.1.2 Fuel Management (Sequence Control) Circuits

Changes in these circuits are due to abandoning the use of the maximum shift sequence for reduction of trim drag. In addition to the deletion of the sequence switch, the connections to the sequence control unit have been changed. The existing bridge system and control relays within the sequence control unit are being used for external tank sequencing.

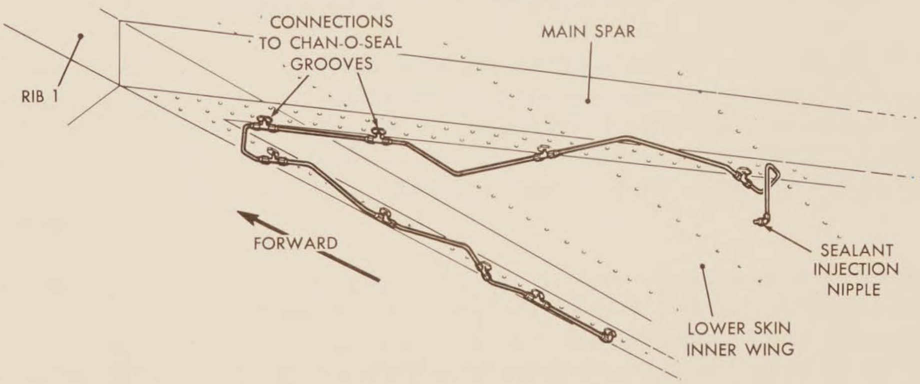
3063-105-1



CHAN-O-SEAL LOCATION DIAGRAM



TYPICAL CHAN-O-SEAL JOINT



TYPICAL SEALANT REPLENISHING PIPE (DUCT BAY AND ENGINE BAY)

FIG. 44 CHAN-O-SEAL METHOD OF SEALING INTEGRAL TANKS

13.3.2 FUEL QUANTITY INDICATING SYSTEM

The fuel quantity indicating system is being modified to incorporate a total fuel quantity indicator in the rear cockpit. The design work required to install the instrument and provide the necessary electrical circuitry is proceeding. This installation will be applicable to aircraft 25209 and subsequent.

The main fuel quantity indicators in the front cockpit are being modified to operate on AC electrical power only. Previously, these units required both an AC and DC supply. The DC requirements will now be obtained by rectifying the AC supply within the indicator unit. An improvement in reliability has been the purpose of the modification.

14.0 HYDRAULIC SYSTEMS

14.1 UTILITY HYDRAULIC SYSTEM

14.1.1 POWER SYSTEM

AVRO has investigated the possiblity of replacing the constant delivery pumps in the ARROW 2 utility hydraulics power system with the variable delivery type. A proposal for this pump replacement will be submitted to the RCAF as report No. 72/SYSTEMS 19/80. This change will allow the ASTRA I radar antenna to be driven from the utility hydraulic system, instead of from the flying controls hydraulic system, thereby improving the reliability of the latter. The present utility system is pressure-regulated and unsuitable for driving the continuously operating antenna because the regulator is not designed for a continuous duty cycle application. A further drawback to the use of a regulator on this type of load would be the increased frequency of surges due to rapid changes in demand.

It is proposed to introduce two 20 gpm variable delivery dual pressure range pumps, with a range from 1,000 psi (for antenna operation) to 4,000 psi (for landing gear, armament, etc.). An electrically operated pump control valve will be employed to switch the pumps from low to high pressure range, depending upon demand, and a reducing valve will maintain the correct antenna drive pressure.

Introduction of a variable delivery power system will present weight saving advantages and will reduce the amount of equipment involved. The utility system compensator volume will, however, have to be increased. The effect of the change on missile extension time will also require consideration.

14.1.2 NOSE LANDING GEAR DOOR OPERATION

In order to improve directional stability of the ARROW 1 during approach and landing it was considered desirable to close the nose gear door, following extension of the gear. A design investigation has produced a satisfactory hydraulic circuit to achieve this, and the revised system will be incorporated in aircraft 25201 (Figure 45). Requirements for the ARROW 2 are being investigated.

The additional equipment required for the system comprises a nose door selector valve with bypass control, a shuttle valve and two modified stop valves which are used as sequence valves. Sequencing of nose gear door operation is now achieved by the use of limit switches actuated by the nose gear in both the fully-up and fully-down positions to control the operation of the four-way, three-position, nose door selector valve. The original nose door closing sequence valve is deleted from the system. As before, the nose gear is sequenced by a door-actuated sequence valve. The modified stop valves are introduced into the nitrogen supply system to sequence nose door operation on emergency extension of the nose gear.

14.1.2.1 Sequence of Operation

(Refer to electrical circuit, Figure 46)

The sequence of operation is as follows:

1. On selection of gear down, the nose gear door opens and the gear is extended by the operation of the nose door actuated sequence valve.

2. When in the fully extended position the gear operates a limit switch to signal the nose door selector valve. This allows hydraulic pressure to the up side of the nose door jack, closing the door.

3. On touchdown, a switch operated by the nose gear scissors re-selects the door to the open position where it is retained by means of a latch relay until after the aircraft takes off again.

4. To make sure that the nose gear does not foul the door during retraction, the gear is interlocked with the "nose door fully open" limit switch which must be de-actuated for gear retraction to take place. After nose gear retraction, when the door starts to close, this limit switch is actuated.

5. If the aircraft overshoots without touching down, an UP selection signals the nose door selector valve, allowing hydraulic pressure to re-open the door.

   When the door is fully down, the "door fully open" limit switch is tripped, allowing the gear-up system to operate.

6. When fully up, the nose gear operates a limit switch to again close the door. The nose landing gear indicator shows UP only when both the gear and the door are locked up.

7. Emergency Extension (Refer to Figure 46).

   With the gear up, the emergency nitrogen pressure is supplied to the nose door via No. 2 valve, and No. 1 valve vents the nose door jack, UP chamber. When the gear is extended the valves are actuated, the nitrogen pressure through No. 1 valve raises the door. No. 2 valve then vents the DOWN side of the door jack.

3077-105-1

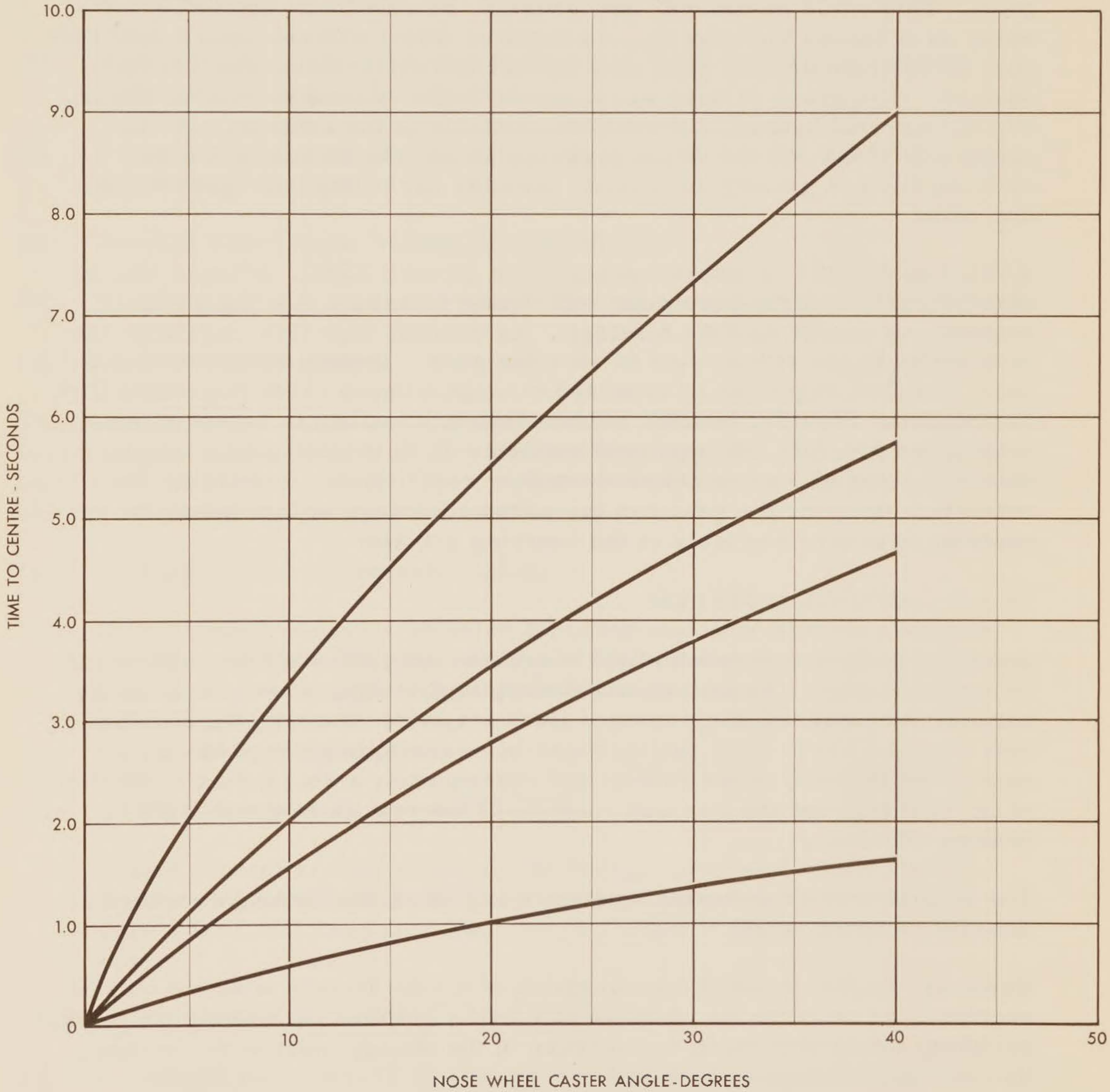


FIG. 47 NOSE WHEEL CENTRING TIME VS. NOSE WHEEL CASTER ANGLE

14.1.3 NOSE WHEEL CENTRING

There is a possibility that the nose wheel could be castered to an angle off the centre line immediately after take-off and prior to retraction of the gear. This could be caused, for instance, by side loads applied to the wheel as it leaves the runway. As the nose wheel will not centre until the gear is selected up, the gear may retract before the nose wheel is fully centred. The graph of nose wheel caster angle vs time to centre, Figure 47, for several hydraulic pressures available at the centring actuator, shows that if the full 4,000 psi pressure should not be available, the centring time is greatly increased, possibly exceeding nose gear retraction time.

AVRO has devised an interim scheme for aircraft 25201, (Figure 46), to slow retraction of the nose gear sufficiently to ensure that the wheel is centred and cannot foul the fuselage. An existing type flow regulator has been added to the retract line to slow the gear. During extension of the gear, the flow regulator is bypassed through a check valve to prevent flow restriction. Should it become evident that this facility is essential for subsequent aircraft, an improved method will be investigated, using a flow regulator which has not reverse flow restriction. Increasing the retraction time for the gear has the added advantage of increasing the centring pressure available at the centring actuator.

14.1.4 ANTI-SKID SYSTEM

Investigations have shown that the maximum drag effect of the brakes can be expected when a slight amount of slipping (rotating wheel skid) occurs between the tires and the runway. However, when the slipping develops into a locked wheel skid, the increase in tire temperature produces a rapid deterioration of the rubber and consequently a decreasing coefficient of friction between the tire and runway. This results in a reduction of braking efficiency.

The objective of an anti-skid system is to reduce the landing distance and the risk of blown tires.

However, further careful consideration of all the factors involved will be required, as an increase of brake efficiency (implied by a reduction in skidding) would also mean an increase in the energy input to the brakes, thereby aggravating problems of overheating.

14.1.4.1 System Requirements

An efficient anti-skid system should satisfy the following requirements:

(a) Detect incipient skids; prevent excessive sliding and the development of locked wheel conditions, by relasing the brakes until wheel torque and brake torque are balanced.

(b) Allow maximum braking and brake efficiency with a minimum cycling of the brakes.

(c) Provide a "fail-safe" feature for normal brake control in case of failure of the anti-skid system.

(d) Prevent engagement of the brakes prior to touchdown, and wheel locking during bouncing.

(e) Prevent swerving or yawing during landing.

(f) Be reliable, of a minimum weight and readily adaptable to the aircraft.

14.1.4.2 Choice of System

The decision to use an anti-skid system on the ARROW will not be made until taxi trials on the first aircraft permit a reliable appraisal of braking characteristics. However, various types of anti-skid equipment are being evaluated and the course of action is as follows:

(a) ARROW 1 - First Aircraft (25201)

It is intended to have a Messier Ministop anti-skid system available for aircraft 25201, and to install it if found necessary. This system has been chosen on the basis of satisfactory delivery dates, but is not necessarily considered the final answer to the requirement for maximum braking efficiency.

(b) ARROW 2 - Second Aircraft (25202)

A skid control system of improved design, proposed by the Pacific Division of Bendix Aviation Corporation, may be evaluated on aircraft 25202 and compared with the performance of the Messier Ministop.

14.2 <u>FLYING CONTROLS HYDRAULIC SYSTEM</u>

14.2.1 INTRODUCTION OF INPUT BOOSTERS (Ref. para. 15.2.2)

Investigations have shown that it will be desirable to provide flying control system input assistance to overcome the effects of flow forces in the surface actuator control valves, high breakout forces and inherent control system friction. This will be achieved hydraulically, by the use of aileron and elevator system input boosters powered from the flying controls hydraulic system. A system has been suggested for the ARROW 1, and the requirements for ARROW 2 are also under investigation.

It is proposed to connect the aileron and elevator input boosters directly to their respective control system forward quadrants, with their rams attached to the aircraft structure. Each booster is of the tandem piston type, with a dual control valve, powered from both the 'A' and the 'B' flying controls hydraulic systems. The booster control valves will be actuated by linkage from the control column, to supply hydraulic fluid to the booster rams. Output of the boosters will be applied directly to the forward control system quadrants.

14.2.2 RETURN LINE PRESSURE SURGE

A hydraulic pump case failed during flying controls pre-flight testing on aircraft 25201. This was attributed to excessive hydraulic pressures. Pressure transducers were installed in the pump casing to measure and record peak values during a series of control surface oscillations. Peak pressures up to a maximum of 414 psi were recorded at the pump case drain outlet. To reduce the pressure the case drain line was re-routed to enter the system return line downstream of the heat exchangers. This was successful in reducing the maximum case drain pressure to approximately 220 psi.

Further testing to determine peak pressures in the system return line showed surge values up to 670 psi, upstream of the heat exchangers. A 5 in. diameter spherical accumulator has been incorporated in each flying control system return line, and is effective in reducing surge pressures by as much as 250 psi. Dampers added to the surface actuator control valves are also expected to assist in reduction of return line pressure surges.

14.3 <u>DEVELOPMENT OF FLARELESS FITTINGS</u>

AVRO has been engaged for some time on a program of development of flareless hydraulic fittings to meet the hydraulic system requirements of 4,000 psi pressure over a temperature range of -65°F to+275°F. In addition, a relatively severe flexural vibration capability is being sought on a contingency basis. (Ground engine running has not shown a severe vibration spectrum, and, therefore the requirements for flexural vibration may not be as severe as anticipated).

The program comprises a series of tests on the following fittings:

1. Tube-to-connector joints (i.e. a combination of steel tube and a steel union to withstand high pressure).

2. Elbow-to-boss joints (i.e. a combination of a steel elbow and a boss).

This series of tests was considered representative of tube-to-boss joints also.

The tests are divided into three categories:

1. Extreme temperature, leakage and burst.

2. Flexural fatigue.

3. Impulse fatigue.

Each test category is sub-divided into a series of tests for different values, and methods of application of torque used for the assembly of the fitting.

Impulse tests were conducted using a machine developed by the National Research Council. Flexural vibration is being carried out on Sontag constant stress type test machine.

14.3.1 STATUS OF TEST PROGRAM

The majority of all leakage, burst and impulse tests have been completed for both tube-to-connector and elbow-to-boss joints, over a wide range of tightening procedures. Results were generally satisfactory and no failures or significant leaks occurred. Some flexural fatigue tests have been carried out and results have so far proved unsatisfactory. Fatigue life of the specimen tested was found to be considerably reduced at elevated temperatures. (Ref. para. 26.2.4.1).

Tests have indicated that the method of assembly of the fittings to the tubing has a considerable effect on flexural fatigue life. It is evident that shrunk-on sleeve assemblies have a markedly superior flexural performance over swaged on sleeves at elevated temperatures (up to +275°F). Accordingly, AVRO has decided to adopt the shrunk-on sleeve method of assembly, and tests are being continued. A major problem at this stage is to determine a satisfactory method of sleeve and tube assembly on a production basis.

14.4 <u>HIGH PRESSURE STEEL TUBING</u>

Welded steel tubing used in the ARROW hydraulic system has shown unsatisfactory results in impulse fatigue life due to failures occurring in the region of the weld.

This tubing is presently limited to a life of 50 hours in aircraft 25201 and will probably be replaced by seamless steel tubing in subsequent aircraft.

15.0 FLYING CONTROLS AND DAMPER SYSTEM

The ARROW's flying control system, while approaching its final form, will require several changes,as indicated by its operation on the flying controls test rig and flight simulator (Ref. para. 4.10). The correction of faults is essential to provide the reliability, efficient operation and ease of maintenance required of the flying control system. Several of these changes will be incorporated on aircraft 25201 before the first flight.

15.1 GENERAL CHANGES TO FLYING CONTROLS SYSTEM

15.1.1 CONTROL VALVES

The first control valves tested showed poor flow force characteristics. Honeywell later modified a control valve which, on test, showed freedom from binding under pressure, and had backlash in the rate spring to correct the flow force characteristics. Authorization has now been given to Honeywell to modify all control valves to the same extent. This problem still exists however and investigation will continue in the attempt to determine a satisfactory solution.

15.2 ELEVATOR CONTROL

15.2.1 RELOCATION OF PARALLEL SERVO AND FEEL TRIM UNIT

The elevator parallel servo and feel trim unit will be relocated. It is proposed that these units be moved to the front elevator quadrant. This change will improve control stability in the normal mode and is made necessary by the installation of the boosters.

15.2.2 ELEVATOR CONTROL BOOSTER (Ref Fig. 49)

Because of high frictional forces, the elevator control system has required modification. These frictional forces have been eliminated by the addition of an input booster attached to the front quadrant.

Formerly, the front quadrant was driven by a lever mounted on the torque tube and connected by a push rod to a crank on the control column. The torque tube has now been divided, and the front quadrant is driven by the input booster. The control valve of the input booster is actuated by a linkage connected to a lever mounted on the remaining portion of the torque tube. The geometry of the mechanism provides a follow-up system for control column movement. The input booster is powered by the dual hydraulic system to provide reliability. The feel unit is attached to the control column linkage and the bob weight location remains unchanged. The front quadrant can still be moved by the control column in the event of complete booster failure.

UNCLASSIFIED

FIG. 48 SCHEMATIC - FLYING CONTROLS

2646-105-2

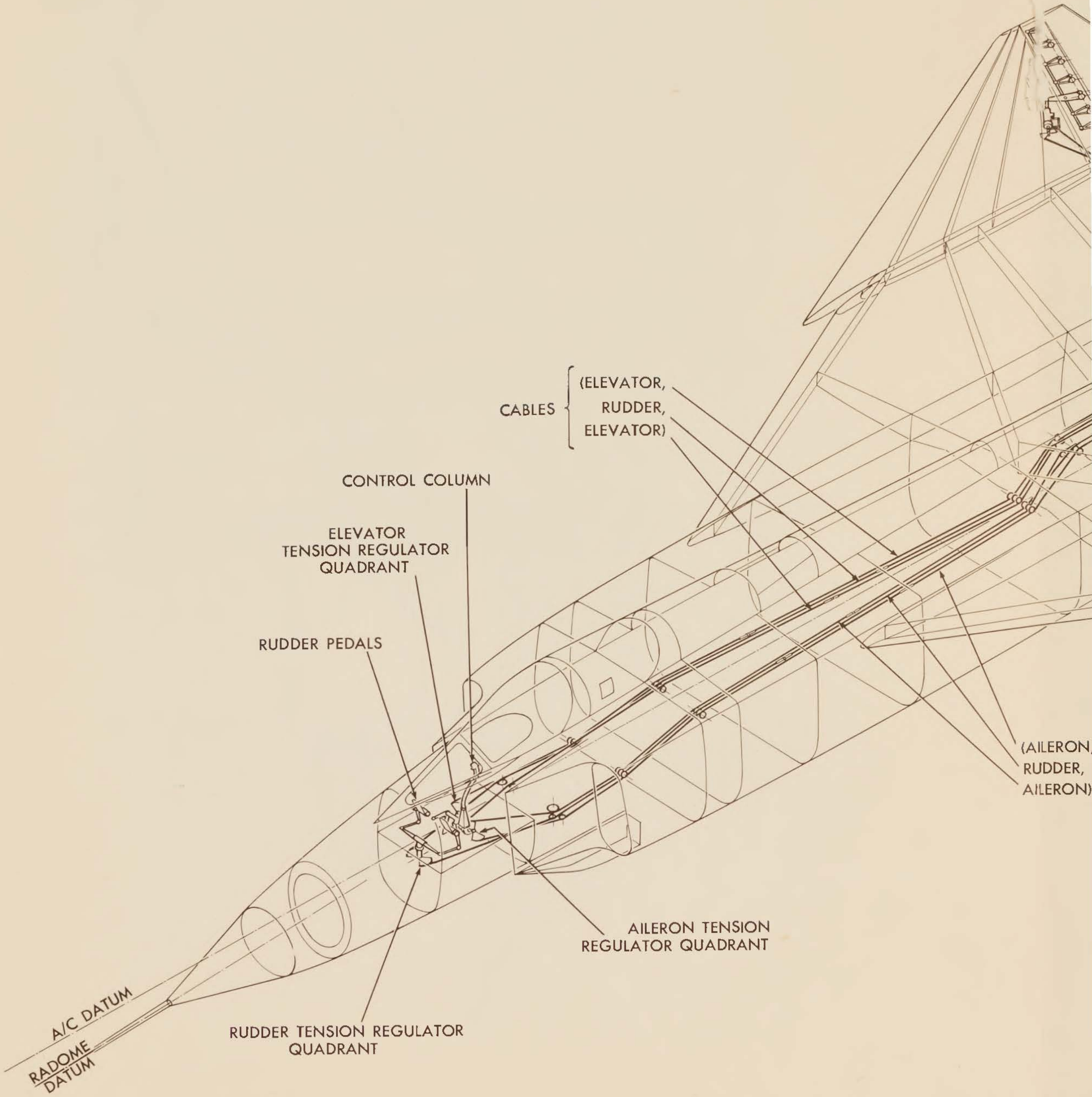


FIG. 48 SCHEMATIC - FLYING CO

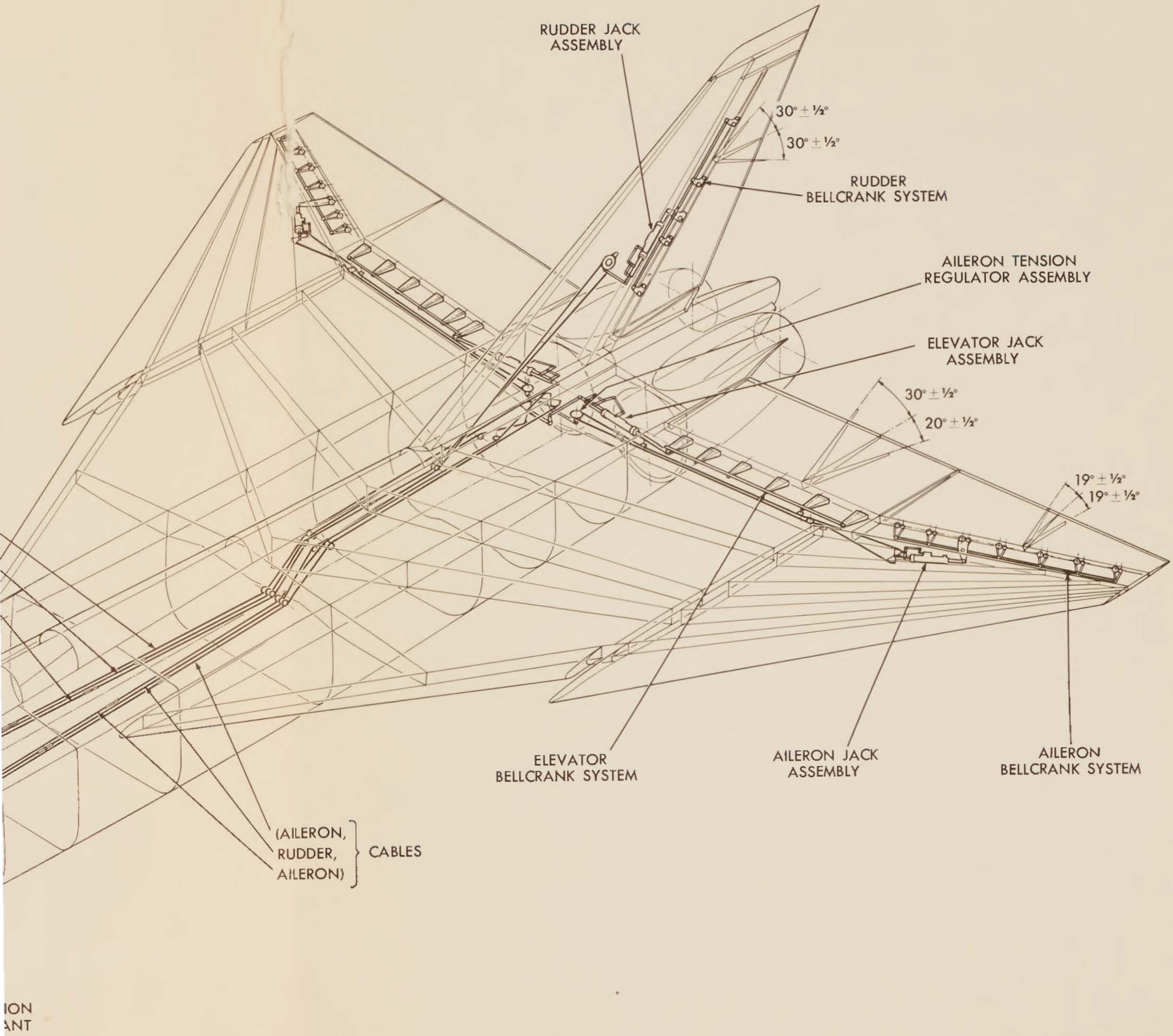


FIG. 48 SCHEMATIC - FLYING CONTROLS

15.3 AILERON CONTROL

15.3.1 PITCH TRIM BY AILERON DEFLECTION

The provision for pitch trim by aileron deflection, mentioned in the previous ARROW Quarterly Technical Report, will be incorporated on the ARROW 2. The control surface response indicator will be modified to show the position of each aileron independently.

15.3.2 AILERON FEEL UNITS

Tests have been performed on the aileron feel springs to determine the most suitable spring rate and break-out forces (Ref. para. 26.2.2) and to reduce friction.

15.3.3 RELOCATION OF THE AILERON PARALLEL SERVO

The aileron parallel servo will be moved to the front quadrant as required by the installation of the booster system to improve control stability in the normal mode.

15.3.4 AILERON CONTROL BOOSTER

An input booster will be added to the aileron control system. The booster will be arranged to provide the same operating functions as established for the elevator booster installation (Ref. para. 15.2.2).

15.4 RUDDER CONTROL

15.4.1 RUDDER HINGE MOMENT LIMITER

Results of the testing on the rudder hinge moment limiter has shown that the results are dependent upon the spring rate tolerances. Additional testing of this unit will be conducted on the flight simulator, and another analysis will be made when the results of these tests are available.

15.4.2 RUDDER PEDAL SELF-CENTRING

Tests of the flight simulator indicated the need for self-centring of the rudder pedals. The self-centring has been accomplished by adding a positive breakout spring at the front quadrant of the rudder control system. This spring reduces the affect of the hysteresis of the control system and hinge moment limiter.

15.4.3 PILOT'S AUTHORITY OVER RUDDER TRAVEL

Tests have shown that the pilot's authority over rudder travel at low values of q_c should be increased. This may involve changes to the rudder feed-back differential transformer. Changes will not be required, however, before the first flight of aircraft 25201 and no further modification action will be taken in this respect until testing has been completed.

15.5 DAMPER SYSTEM

In accordance with changes requested by the RCAF Damper Meeting, 3 October 1957) the following additions are being made to the damper system of ARROW 1 and 2:

(a) A spring-loaded switch to check the landing gear up mode while the damper system is in the landing gear down mode.

(b) A warning light which, when illuminated, will indicate malfunctioning of the landing gear down mode.

(c) A g trimming indicator. At the present time this indicator is planned for ARROW 2 but may be included on ARROW 1 as a retrofit.

Several problems of the damper system with respect to ASTRA I are being investigated.

The specification for the development of the damper system is being reviewed to bring it up to date, and the specification for the production damper system is being written.

16.0 ARMAMENT SYSTEM

16.1 ARROW ARMAMENT SYSTEM ACTIVITY

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.

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 circuits.

(5) Flight testing of weapon pack on ARROW 1 aircraft.

16.1.1 MISSILE PROTECTION

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

The missiles are reputed to have been designed to withstand temperatures that may be encountered during prolonged periods of flight at Mach 1.5, whereas the aircraft is capable of operating at Mach 2 for periods of 10 to 15 minutes, with a subsequent skin temperature rise to about 250°F. This temperature could be determined to the performance of the missile.

The RCAF has been considering the problem of missile compatibility with maximum aircraft performance, and a test program has been planned by Canadair Limited to ascertain the actual missile temperature tolerances. At the same time AVRO started an investigation into the possibility of protecting the missile from the ambient environment. This study has been delayed, however, due to lack of information on the internal construction and components of the missile.

The RCAF is in the process of compiling ARROW weapon system requirements. In the early stages of squadron service, it appears that the Sparrow 2 Mk 1 will be completely compatible with the overall weapon system requirements. Increasing system capability may be combined with an improved Sparrow Mk 2 missile.

The radome of the Sparrow missile will require some form of protection from mud and slush etc. which may be thrown up by the aircraft nose wheel, and also from the effect of rain erosion during flight. This protection problem is under investigation at AVRO, but the results obtained are still of a preliminary nature.

16.1.2 RIGIDITY EFFECT ON ANTENNA LINE OF SIGHT

Although the 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 structural deflections may be unacceptable during flight. An investigation into the magnitude of the deflections and their effect on the lines of sight is now underway. Flight tests will be made on aircraft 25203 to measure angular deflections between aircraft radar sight line and missile sight line.

16.1.3 DYNAMIC STUDY OF THE MISSILE LAUNCHER

A study has been undertaken to analyze the effect of the increased length of launcher rail from the 30 in. rail used by Douglas Aircraft Company to the 60 in. rail designed for use on the ARROW aircraft. The study will cover the missile trajectory and the loads induced during missile firing. The results of this study are expected to be available early in 1958, although the majority of the study has been completed for some time.

16.1.4 ELECTRICAL CIRCUITRY

The electrical circuits are divided into two parts:

(a) Actuating circuits

(b) Firing circuits

The actuating circuits are concerned with the missile lowering and door operation. The firing circuits are involved in missile extension sequencing, firing, jettison, and launcher retraction. RCA is responsible for the missile firing sequence, although the associated wiring will be installed by AVRO.

A redesign of missile firing and jettison circuits is being completed to allow hang-fire missiles to remain extended during a breakaway manoeuvre, since it was found impracticable to apply a jettison signal just prior to the breakaway. In conjunction with this redesign, a stress investigation was initiated and as a result, the aircraft will be cleared for breakaway manoeuvre with all missiles extended, or at any intermediate positions.

Following discussions between AVRO and RCA, the firing system for ARROW 2 aircraft is now envisaged as follows:

(1) Missile Lowering

(a) Automatic Firing Mode

The lowering of the missiles selected for firing will automatically be initiated at a time t seconds before computed R_{max} is reached. The value of t will be chosen so that lowering is completed and, in the case of a suitable target, missiles will have a short time to achieve lock-on before R_{max} is reached. The value currently assigned to t is 2 seconds.

Automatic lowering will take place whenever the automatic firing mode is selected, the AI radar is locked on to a target, the master arming switch is operated and a choice of missiles has been selected.

(b) Manual Firing Mode

The lowering of the selected missiles will be initiated by a switch in the pilot's cockpit. The pilot's choice of the moment to lower will be aided by the display of range-to-go on his attack display scope. The operation of the switch will lower the missiles at all times when the aircraft is airborne as its function is not interlocked with the AI radar.

The automatic and manual firing modes apply to both lead pursuit and lead collision attack modes.

(c) Optical and Spotlight Attack Modes

Optical and spotlight attack modes will utilize only the manual firing mode and therefore only the manual lowering system.

(2) The Arm-Ready Light

The arm-ready light (green) on the pilot's display will be illuminated when any one of the selected missiles locks on.

(3) Missile Firing

Firing pulses will be sent to the appropriate missiles only when the interlocks in the following modes are satisfied.

(a) Automatic Firing Mode

(1) Range to target lies between R_{max} and R_{min}

(2) Lead angle error lies within the computed limits for the particular attack condition.

(3) At least one missile has locked on to its target

(4) The pilot's firing trigger is depressed.

(b) Manual Firing Mode

(1) At least one missile has locked on

(2) The pilot's firing trigger is depressed

(c) Optical and Spotlight Attack Modes

(1) At least one missile has locked on.

(2) The pilot's firing trigger is depressed.

(4) Firing Order

Missiles will be fired in order of lock-on, except when two or more missiles achieve lock-on simultaneously. With simultaneous lock-on, the missiles will be fired in any order that will simplify the system design. This sequence has not yet been determined.

(5) Launchers

Empty launchers will be retracted automatically immediately after the successful firing or jettison of all missiles selected for firing.

Any missile which remains on a launcher at the end of an attack, and which has not had a firing pulse directed to it (for example due to its failure to lock on), will be retracted automatically at R_{min}. When computed range is not available such missiles will remain extended until retraction or jettison is initiated manually.

(6) Malfunction Light

The malfunction light (red) will be illuminated when any missile remains on its launcher after a firing pulse has been directed to that missile.

(7) Jettison

Manual jettison is a requirement. It is actuated by the pilot and effective at all times when the aircraft is airborne. The jettison selection will override all other conditions of the selecting and firing systems.

Facilities for automatic jettison, and jettison of missiles by firing as rockets, are not required.

Calculations indicate the breakaway manoeuvres can be performed with missiles fully or partially extended. The firing system will be designed on the assumption that it will not be necessary to jettison or retract misfired or unfired missiles before breakaway. After breakaway, the pilot can leave the missiles extended, retract, or jettison them.

(8) Breakaway Warning

Breakaway warning will be given to the pilot by means of a cross on his attack display scope and will be initiated when either firing pulses have been directed to all missiles selected for firing or at computed R_{min}.

16.1.5 WEAPON PACK FLIGHT TEST

Although the ARROW 1 aircraft is not considered as a missile carrying vehicle, aircraft 25203 will be equipped with a weapon pack which will be flight tested with dummy missiles in order to obtain advance flight data on the following:

(1) Weapon pack functioning during flight

(2) Aircraft handling, stability and buffeting characteristics when missiles are extended in various arrangements.

(3) Configuration that SAV missiles can be fired without hazard to the aircraft and without undue disturbance to the missile in its early flight stages.

(4) Confirmation that missiles can be jettisoned without hazard.

(5) Measurement of detailed characteristics of the armament system for comparison with design assumptions.

16.2 SPARROW 2D MISSILE - COMPATIBILITY WITH ARROW

Two alterations to the Sparrow 2D missiles are required for compatibility with the ARROW aircraft:

(1) Umbilical plug redesign

(2) Missile wing actuation time change

16.2.1 UMBILICAL PLUG

Because of the structural requirement for a semi-submerged missile installation, it is not possible to insert the umbilical plug by hand, as is the current practice on missile installation involving the use of a pylon-suspended missile. AVRO proposes to adapt a proposal originally presented by Douglas Aircraft Company, which involves the remotely-controlled engagement of the umbilical plug. The use of this scheme will require changes to the umbilical plug receptacle in the missile body. The RCAF was notified of this proposal on 25 February 1957 (AVRO letter Ref. 5746/03/J). While no reply has as yet been received by AVRO, work is proceeding on the assumption that the required alteration will be made to the missiles. All test work at AVRO involving the functioning of the umbilical plug will be conducted using the proposed AVRO scheme for remotely controlling the engagement of the umbilical plug, with a Deutsch electrical connector set at about 35° to the missile axis, and recessed in a well on the missile body.

16.2.2 MISSILE WING ACTUATION TIME

AVRO considers it is necessary to increase the time between firing and unlocking of the missile wings. This time increase is necessary to ensure aircraft safety, due to the unguided distance from launching position to the aircraft nose. The distance involved is relatively long compared with that normally associated with missile installations located under an aircraft wing. The RCAF was advised of this change request on 24 July 1957 (AVRO letter Ref. 9174/03/J) but a decision has not yet been received by AVRO.

17.0 ESCAPE SYSTEM

The escape system consists essentially of a firing unit and operating mechanism for emergency opening of the two split clam-shell type canopies, two Martin-Baker MK C-5 ejection seats, and the associated bail-out and distress warning arrangements The system as described is applicable to both the ARROW 1 and the ARROW 2.

17.1 ESCAPE DELAY - HUMAN FACTORS

A human factors engineering study has been completed to determine the nature of the delays in escape from tandem-crewed aircraft (Ref. 70/HUFAC/1 "Measurements of Delays in Escape from Tandem-crewed Aircraft". November 1957). Four sequences of escape warning and ejection were studied. Two sequences were conducted without oral instructions, using a visual/audio warning system, and two were conducted with oral instruction supported by a visual/audio warning system. The main finding of these studies was that the time taken to complete all sequences was excessive, although those without oral instructions took significantly less time than sequences using oral instructions.

As a result of these studies it was recommended that:

(a) Consideration should be given to a proposal to link the seats, so that the pilot would eject both crew members by operating one control. An override would be provided which would permit the observer/AI to eject independently. It is considered that a linked seat system would reduce the total escape time for both occupants to approximately 2.5 seconds.

(b) An attempt should be made to establish a brief and effective verbal warning

(c) The significance of short periods of time in escape should be studied more closely during the compilation of accident reports, in particular for aircraft carrying more than one occupant. Valuable information could thus be obtained which, in time, would provide more details about crew behaviour during actual escapes.

17.2 SLED TESTING OF ESCAPE SYSTEM

Following the issue of Avro Aircraft Limited reports RD84A and RD84B (Specification for Rocket Sled Testing of the ARROW Escape System) three proposals have been received from potential sub-contractors. These proposals are now being evaluated by AVRO, but the selection of sub-contractor and test site will not be made until contractual authority is received for the continuation of the ARROW escape system sled test program.

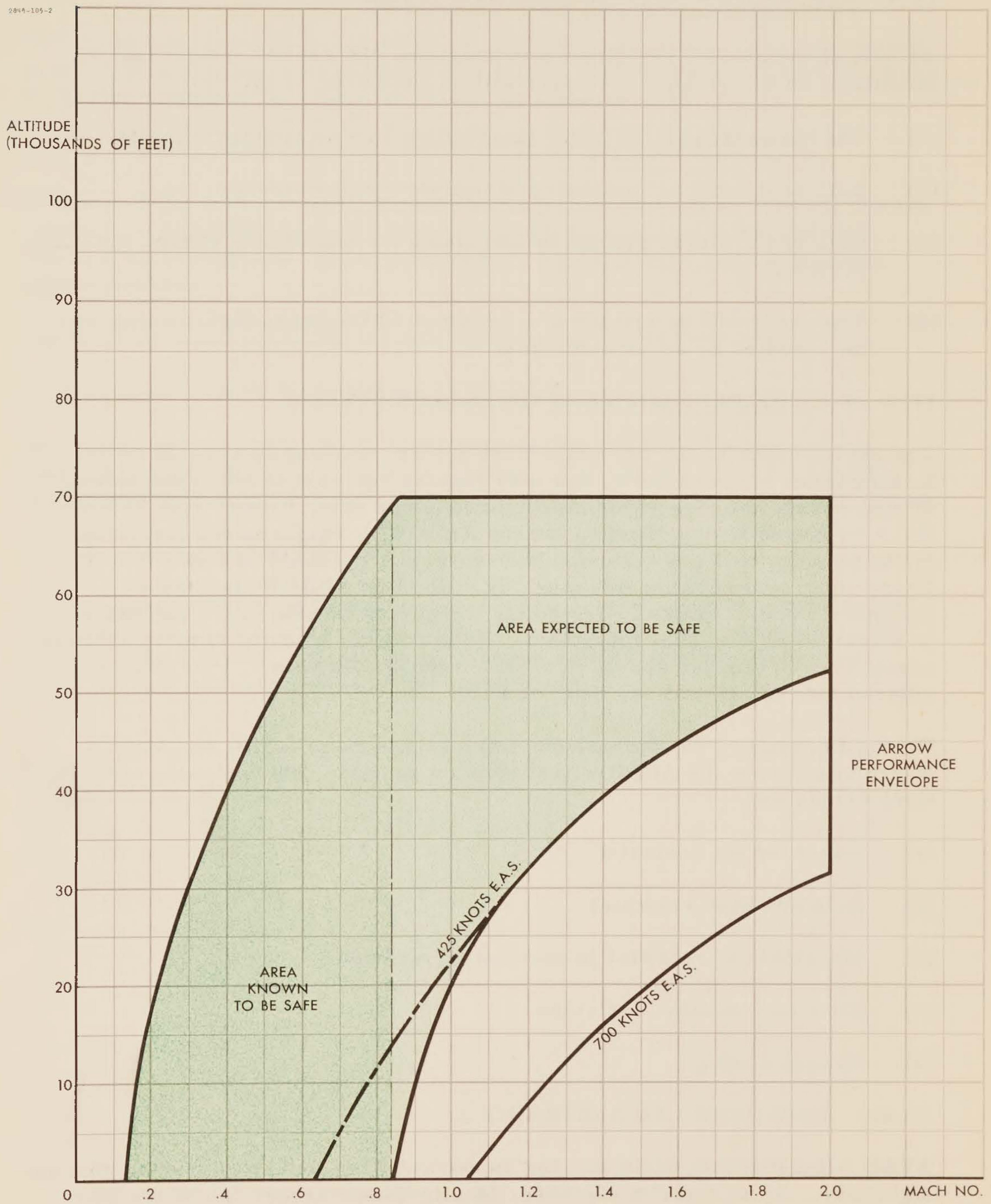


FIG. 50 PERFORMANCE LIMITATIONS OF MARTIN-BAKER C5 SEAT

17.3 PRE-FLIGHT TEST OF ESCAPE SYSTEM

As part of the overall pre-flight test program, the escape system will be functioned on the ground. The test will be conducted as follows:

(1) The static test aircraft will be positioned on an appropriate site.

(2) Both seats will be installed and equipped with dummies.

(3) Both seats will be ejected in turn using the emergency canopy opening system.

(4) The seats will be ejected to a height of 60 ft. where their ascent will be arrested by an arrester gear.

17.4 FUTURE DEVELOPMENT OF ESCAPE SYSTEM

The performance of the present escape system may not be entirely satisfactory under all conditions, and may require revision to attain the objective of safe escape over the complete flight envelope (Ref. Report P/SYSTEMS/45, a Proposed Escape System for the ARROW). Performance limitations of the existing seat are indicated in Figure 50. It should be noted that this figure supersedes Figure 40 (page 179 of the last ARROW Quarterly Technical Report, which indicated the "known to be safe area" as limited to Mach .64 at sea level conditions. Following the study of reports on tests carried out in similar escape systems in the United States, it is now considered safe to eject at sea level at Mach .84.

During December 1957 discussions were held between AVRO and Martin-Baker to improve the ARROW seat ejection system. The following subjects were discussed:

(a) Improved leg restraint

(b) Arm and head restraint

(c) Movement of occupant in seat during ejection

(d) Dual seat ejection cartridge

(e) Drogue chutes

17.4.1 IMPROVED LEG RESTRAINT

AVRO suggested that additional leg support was necessary to prevent possible leg injury. As an interim measure, Martin-Baker agreed to pad the front face of the seat, but would investigate the design of a suitable plate over which the leg load would be more evenly distributed.

17.4.2 ARM AND HEAD RESTRAINT

In view of the proposed actuation of crew ejection sequence, it is considered necessary to provide arm and head restraint. Martin-Baker agreed to supply a working model of the restraint system by January 1958.

17.4 3 MOVEMENT OF OCCUPANT IN SEAT DURING EJECTION

Because of the present seat geometry and ejection path, it is possible for the occupant to slump down on ejection, which would result in corresponding forward leg movement. This would reduce leg clearance and could cause spinal injuries.

Martin-Baker will fabricate a redesigned model for AVRO's evaluation.

17.4.4 DUAL SEAT EJECTION CARTRIDGE

Martin-Baker is conducting an investigation into the RCAF's request for duplicate cartridges in the ejection system. The system will be actuated by the existing alternative firing handle located between the occupant's knees.

17.4.5 DROGUE CHUTES

Martin-Baker intends to redesign the existing drogue chute to eliminate the high shock load. When the necessary alterations have been completed, the drogue chute and seat will be returned to the SMART track at Hurricane Mesa for further sled testing by the manufacturer.

201-CA-2

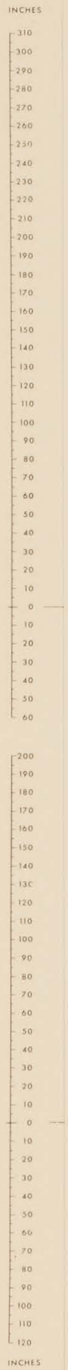


FIG. 51 ARROW STATION & DATUM LINES

201-CA-2

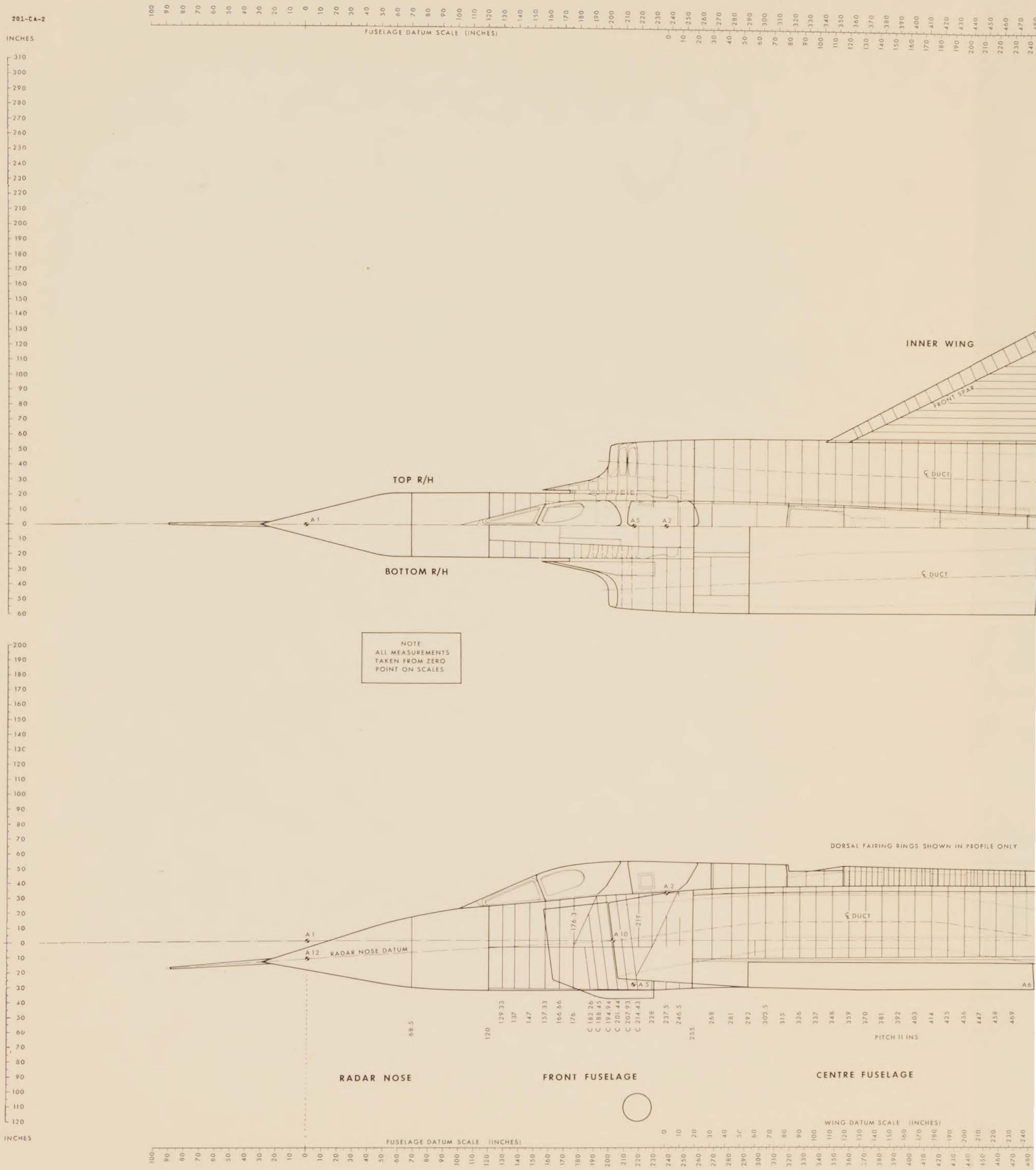


FIG. 51 ARROW STATION & DATU

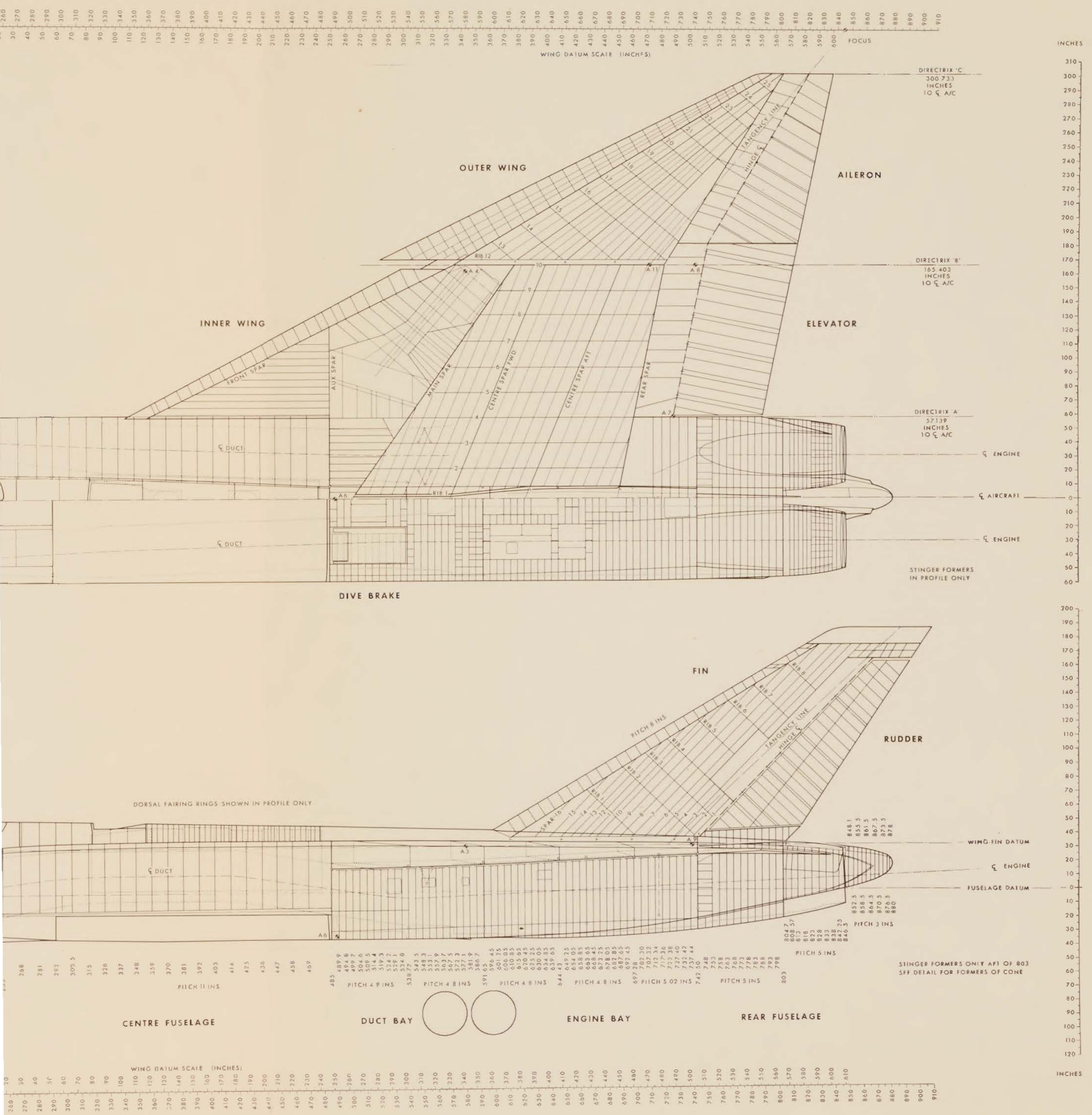


FIG. 51 ARROW STATION & DATUM LINES

18.0 STRESSING

18.1 THERMAL ANALYSIS

Heat introduced to the skin from the boundary layer is transferred into the structure. An efficient method of estimating the resulting temperature distribution is therefore required. A method has been devised which is a numerical analysis, where the usual differential equations are replaced by finite difference equations. In this method, the structure is divided into small elements and a heat balance is set up for each element over a small interval of time.

With this method, several assumptions are made:

(a) The time interval is so small that only adjacent elements need be considered.

(b) The temperature at the centre of any element is the average temperature of that element.

(c) The rate of change of temperature remains constant over any chosen interval.

(d) The problem is one of heat conduction and convection.

The effects of fuel acting as a heat sink are then considered. A test specimen serves as an example problem, the final equations being fed into a digital computer. A comparison has been established for free-flight and test heating conditions.

The results of calculations show several interesting factors with regard to constant heating in testing, compared with the actual heating for the free-flight case:

(a) The use of infra-red heating to simulate kinetic heating of aircraft structures gives results which are slightly in error. This error can be reduced by using more control points and controlling the heat over small surface areas.

(b) The maximum skin and web stresses for both free-flight and radiant heating cases occur at the same time. For short heating periods, the web stresses are nearly equal for both testing methods. In all other instances however, the radiant heating produces greater stresses than for the free-flight cases.

(c) Increased time allowed for the temperature of the boundary layer to reach maximum, gives the specimen time to heat up more uniformly.

This results in less temperature variation than for the case where the heating time is short, as in the case of the ARROW.

A further study of ARROW thermal problems will be conducted and will include temperature distributions for a number of structural sections. The study will consider the variations of temperature with time in various missions and will take into account such factors as temperature variation with altitude, Mach number, normal acceleration, aircraft weight, fuel conditions, etc. In addition, the study will consider changes in structural stiffness due to temperature, and will include thermal stresses, thermal deflections, creep, thermal buckling and thermal fatigue.

Tests will be undertaken to determine heat transfer coefficients and to verify theoretical predictions.

18.2 ACCELERATED FATIGUE

Testing of structural panels in a noise field is continuing on ARROW 1 and ARROW 2 components. As reported in the previous issue of the ARROW Quarterly Technical Report, the magnesium rudder skin panels produced satisfactory fatigue life but further work will be required on engine nacelle panels, stinger panels and fuselage side panels. Naturally the test conditions are as severe as practicable, in order to promote early failure. The sound levels used for testing, therefore, are those which could be expected on the aircraft with afterburner operating, and cannot be guaranteed until a complete noise spectrum is established. This will involve a thorough noise survey on the actual aircraft.

In determining the satisfactory fatigue life of a structural panel, the number of cycles to failure is considered. When a specific number of cycles has been reached without failure, the panel can generally be considered capable of continuing for a much longer period. On this basis, the engine nacelle and stinger panels are considered satisfactory if 12 hours of testing produces no failure. The rudder and fuselage side panel, however, being constructed of magnesium, must withstand a greater number of cycles before they can be considered satisfactory, as the natural frequencies of magnesium panels are lower then those for stainless steel or iron base alloys used in the engine nacelle and stinger panels.

The following is a brief summary of testing.

Engine nacelle panels - The ARROW 2 engine nacelle panels are constructed of iron base alloy N155 material. These panels will be tested in the near future.

Further testing will be carried out on ARROW 1 panels constructed of stainless steel. These tests will serve mainly to check spot welding techniques which were a source of weakness in panels previously tested. Testing will be carried out at 164 db.

Stinger panels - The ARROW 2 revised stinger panels will be tested at 164 db.

Fuselage side panels - These panels comprise an .051 thick skin, .040 stringers and .032 stringer cleats and are tested at 140 db. One such panel has been satisfactorily tested.

Rudder panels - A double rib rudder panel was tested at 140 db and the results show a satisfactory fatigue life. A second double rib rudder panel will be tested at 140 db in order to check previous results.

A theoretical acoustic fatigue investigation is proceeding in an attempt to study various means of increasing the life of a structure in a noise field. The program includes the determination of a theoretical method for predicting the life of structure subjected to acoustic loading. In addition, the study will include the effects of damping in reducing the vibration amplitude, and increasing the fatigue life of a structure. Acoustic testing will be carried out on simple panels to substantiate the theoretical analysis.

19.0 ARROW 2 MOCK-UP

19.1 SUMMARY OF MOCK-UP ACTIVITIES

As a result of the ARROW 2 mock-up conference in September 1957, a total of 252 change requests required investigation. An investigation into the effect of these requests is being conducted by AVRO, RCA, Martin-Baker Limited, and the RCAF.

Paragraph 2.1.3 of the first ARROW Quarterly Technical Report indicated that nine items were not evaluated. This should have shown a total of ten, with two under structure and nil for air conditioning, in place of nil under structure and one for air conditioning.

The final totals are: 10 not evaluated
5 withdrawn
252 change requests for investigation
Total 267

19.2 STATUS OF MOCK-UP CHANGE REQUESTS

The current status of the 252 requests are listed below

Subject	Code	Items Not Evaluated	Change Req'st	Status: Under Initial Investigation	Status: Under going Corrective Action	Status: Completed	Status: Demonstration Req'd
Cockpit	A	1	62	24	25	13	5
Structure	B	2	51	16	19	16	2 *
Engine Installation	C	3	18	1	4	13	4
Electrical	D	-	22	5	8	9	-
Air Cond'g	E	-	6	2	0	4	-
Low Press. Pneumatics	F	-	1	-	1	0	-
Fire Exting. System	G	-	3	-	3	0	-
De-Icing	H	-	2	-	2	0	-
Fuel System	I	-	11	2	4	5	1
Hydraulics	K	-	15	7	4	4	-
Oxygen System	L	2	5	1	4	0	2
Instruments	M	-	7	4	3	0	-
ASTRA 1	N	1	34	24	5	5	1 *
Armament	O	1	15	4	5	6	1
		Total	252				

* Explanatory data will be provided in lieu of demonstration for these items.

19.3 FUTURE DEMONSTRATIONS

The ten items not evaluated during the mock-up conference will be demonstrated at future dates. Some items demonstrated but subsequently changed will be re-demonstrated. Certain items, although altered, will not require further demonstration but will be covered by letter to the RCAF.

The following items will be either demonstrated or covered by letter(s) to the RCAF.

Description	Ref. Code	Form of Demonstration
Pilot's and observer/AI's seat	A24	Aircraft
Seat oxygen equipment	L5	"
Console lights	A8 (Cat. 1)	Mock-up
Cockpit lights	A12 (Cat. 1)	"
Map lights	A37 (Cat. 1)	"
Intensity of light	A38 (Cat. 1)	"
Engine ducting	C8	"
Engine ducting	C9	"
Engine removal	C13	"
Engine removal	C21 (Cat. 3)	"
Facilities to remove fuel booster pump gear box	I1 (Cat. 1)	"
Liquid oxygen converter	L3	"
Missile umbilical plug and rail	O7	Test rig
Facilities for bladder tank removal	B11 Pt. 2	Letter
Emergency lowering of electronic bay access door	B25	Letter
Antenna multiplexing	N18	RCA to clear by letter

Items in the above list which have their categories marked in brackets e.g., (Cat. 1) were changed as a result of the mock-up conference, and will be re-demonstrated.

Items C8, C13 and C21 are near completion in their mock-up stage and are expected to be available for demonstration early in 1958.

The equipment associated with cockpit layout and lighting will be demonstrated using the ARROW mock-up. The date of this demonstration has not yet been established.

Although an actual aircraft seat was shown during the mock-up conference, it was not demonstrated. It is now proposed to demonstrate the latest type seats installed in an aircraft (Martin-Baker Mk C5) and equipped with seat oxygen equipment.

20.0 COMPONENT DESIGN

20.1 WING DESIGN - ARROW 2

The following design work has been completed on the ARROW 2 wings:

(a) A portion of the elevator was removed at the inboard end. This was necessitated by the redesign of the rear fuselage to accommodate the Iroquois engine. The portion removed was triangular in shape, and measured approximately four inches along the trailing edge and 29.5 inches along the inboard end, measured from the trailing edge.

(b) The main landing gear pivot door was partly redesigned to accommodate new linkage and to provide for changed leg dimensions.

(c) The joint between rib 24 and the main spar was redesigned to improve the structure.

(d) The spring boxes on the main landing gear leg were redesigned. This will provide improved adjustment of the fairing.

20.2 WING STRESS ANALYSIS ARROW 1 AND 2

Tests carried out on a section of the inner wing main torque box indicated that excessive deflection occurred on the machined skins. This problem was corrected by the addition of vertical posts placed throughout the main torque box area.

Further tests have since been completed on a section of the inner wing posted box. Several specimens were tested. Internal pressure using air and water inside the wing box was applied and the external load was increased on each specimen until failure occurred. Figure 52 shows the external load at failure for the various specimens tested, plotted against the internal pressure applied in each case.

The last specimen tested failed at 94.5% ultimate design load with an internal pressure of 41.0 psig. The failure of this specimen was similar to previous failures, so the description of failures of this panel may be taken as being representative.

The top (compression) skin, which had shown buckling at lower loadings, failed at the skin to rib attachment and intermediate spar intersection, and at the adjacent post-to-skin attachment on the same rib(Fig. 54). Several skins cracks radiated from these failures, the primary cracks being longitudinal, and approximately 18 inches long on either side of the failure. The stringers on both skins, and the web-to-skin attachment flange, were torn

away over a length extending from the single machined post to the inboard end of the box.

The centre spar web cracked as shown in Figure 55. The outboard spar web also failed with a large diagonal crack extending from the lower end of the second outboard vertical stiffener to the upper end of the collapsed box corner.

Figure 53 shows the average permanent set existing at the conclusion of each test run at various external load levels with 30.0 psig internal pressure. From these results it was concluded that the limit load condition of the box is dictated by the behaviour of the skin stringer combination, and occurs at 70.25% ultimate load.

The test specimens were not completely representative of the actual torque box area on the aircraft. The skin thicknesses and other discrepancies, however, were sufficiently representative to permit an accurate approximation of the actual strength of the torque box structure.

20.3 RADAR NOSE DESIGN - ARROW 2

All production drawings for the basic structure of the radar nose have now been issued.

A ballast rack is to be provided in the radar nose and will be used when the aircraft is flown without ASTRA I. The rack is designed to fit the mountings provided for radar equipment.

Several minor installation changes requested by RCA are presently being incorporated.

Modified drawings for the redesigned radome for use with ASTRA I have been issued.

A preliminary design study has been started for the installation of cameras in the radar nose of aircraft 25204 and 25205. A servo camera is to be mounted on the lower boom member, and a strike camera on the upper boom member.

20.4 RADAR NOSE STRESS ANALYSIS

All drawings for the radar nose basic structure have been stress approved and issued.

20.5 FRONT FUSELAGE DESIGN - ARROW 2

All production drawings for the front fuselage basic structure have been

3068-105-1

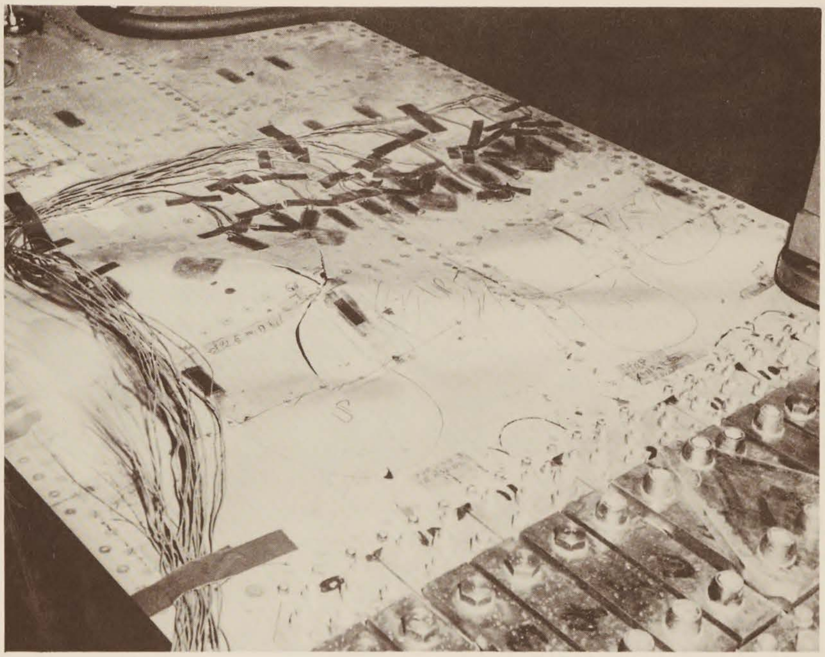


FIG. 54 FAILURE OF TOP (COMPRESSION) SKIN OF INNER WING POSTED BOX

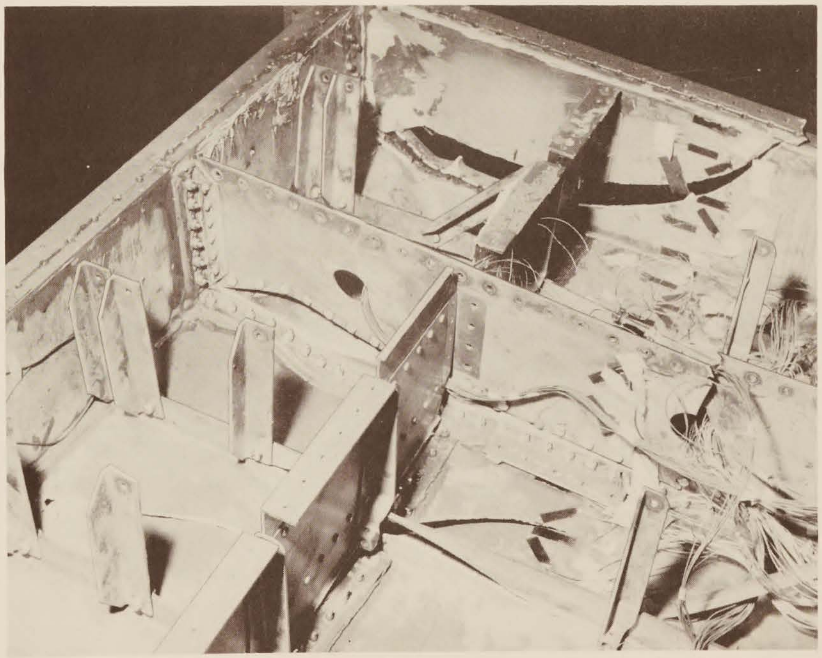


FIG. 55 INTERIOR STRUCTURE OF INNER WING POSTED BOX

issued. Some frames are being reworked to accommodate various services for which information was not available at the time of the first drawing issue.

Several changes were requested by the Manufacturing Division. Some of these have been completed and the remainder are either in stages of completion or are still under investigation.

20.6 FRONT FUSELAGE STRESS ANALYSIS - ARROW 1 AND 2

The possibility of omitting the engine air intake bleed holes in the fairing area with the consequent increase in hole diameter in the flat area of the ramp is being investigated to simplify alignment of holes in deicing boot and metallic structure.

Cockpit pressure tests to a proof pressure of 7.65 psig will begin during January. These tests will employ the data processing method which is to be used on the complete aircraft static test. This method involves coding of strain and deflection gauges, punching of IBM cards, and the use of a Kelk automatic plotter.

20.7 CENTRE FUSELAGE DESIGN - ARROW 2

The schemes have been completed for the electronic equipment bay, and production drawings are in progress.

Specification control drawings have been issued for the design investigation of a plastic heat exchanger outlet duct. The plastic duct would replace the existing stainless steel duct, thus simplifying the manufacturing operation.

20.7.1 WEAPON PACK

The design work involved to accommodate the electrical equipment in the weapon pack is virtually complete.

Due to the static deflection of the weapon pack, the forward end of the pack will be lifted to maintain the aircraft profiles. This will be accomplished by shortening the forward pickup fittings by .090 in.

The installation of the transverse cable duct and the missile seals will not be required for the first pack as it is used exclusively for functioning tests. The seals and the transverse cable duct will be incorporated in the second pack. All production drawings for the launchers are complete and issued.

An assignment has been raised for the design of a weapon/instrument pack which will be interchangeable with the weapon pack on the ARROW 2. The weapon/instrument pack will accommodate two Sparrow 2D missiles and the missile auxiliaries on the right hand side, and the two Datatape recording

systems, plus fuel system instrumentation and signal conditioners on the left hand side. The weapon/instrument pack will make use of as many parts as possible from the existing pack design. The design of this pack is expected to begin in March 1958, when the necessary information on instrumentation requirements is received.

20.8 CENTRE FUSELAGE STRESS ANALYSIS - ARROW 2

The new requirement for the retraction of missiles in flight is being investigated. The primary consideration of this investigation is the deflection of the retraction mechanism.

A report has been prepared for the static and flight test requirements of the weapon pack (Report No. 7/0500/38, Sparrow 2 missile pack, January 1958).

20.9 DUCT BAY DESIGN

As mentioned in the previous issue of the ARROW Quarterly Technical Report, an assembly problem was experienced with the heat exchanger. The investigation on this problem has led to a redesign of the heat exchanger and production drawings are now being issued.

20.10 DUCT BAY STRESS ANALYSIS

20.10.1 ARROW 1

A study has been completed to determine the design revisions required to permit the use of speed brakes over the entire flight envelope. The results of this study indicate that by using ARROW 2 speed brakes and with a minor modification to the duct bay structure, the speed brakes can be operated up to maximum aircraft speeds.

20.10.2 ENGINE INTAKE DUCT (FLOATING ASSEMBLY) PRESSURE TESTS (Figure 56)

Pressure, leak and suction tests were carried out on the engine intake duct (floating assembly). At 48.30 psig internal pressure (10 psig above 38.30 psig ultimate pressure) failure occurred in the main shell area, forward of the gill hinges. The lower longitudinal joint strip parted along the rivet attachment lines between stations 530.37 and 540.21. The duct skin sheared around its circumference at the edge of the stiffening member at station 540.21, and spot welds parted on the lower portion of the stiffening members at station 530.37. The test was considered satisfactory as the failing load was approximately 126% ultimate.

The internal pressures for the tests in the forward and aft position of the duct were varied in the ratio of the ultimate pressures in those duct portions

(38.30 psig for the forward portion and 28.20 psig for the aft portion), the pressure change being at the gill hinges. During test, a leak developed between the forward and aft duct portions through the rig seal at the gill hinge. Consequently, no leak rate records were obtained above 10 psig.

20.10.3 DUCT BAY STRESS ANALYSIS - ARROW 2

The drawings detailing the structural changes to the speed brakes and surrounding structure have been stress approved. The speed restriction on the ARROW 2 speed brakes has now been lifted to allow operation at speeds up to M= 2.0.

20.11 ENGINE BAY DESIGN

Work is continuing on the design of the ARROW 2 engine doors.

20.12 ENGINE BAY STRESS ANALYSIS

20.12.1 ARROW 1

A specimen representing the upper engine shroud and inboard shroud beam has been successfully tested. The specimen withstood the limit suction of 1.5 psig and the limit pressure of 18.0 psig. During the pressure case, the inboard beam was deflected to represent the deflection of the supporting structure under a 7.33g load. The specimen was finally subjected to a repeated pressure of 18.0 psig with the representative deflection for the 7.33g load. The specimen failed after 3700 applications of this load. This was considered satisfactory.

Tests will be carried out on the engine door and latches. Previous tests have shown that the door structure and latches, when tested individually, were satisfactory up to limit load but did not withstand the ultimate loads. A slight modification is being made to the latches and it is expected that this will enable the assembly to withstand loads above the ultimate requirement. This modification should also improve the fatigue life of the door.

20.12.2 ARROW 2

The lengthening of the number 1 engine door and subsequent shortening of the forward false door has received preliminary design approval. Production drawings are now being approved.

20.13 REAR FUSELAGE DESIGN

The design of the removable portion of the rear fuselage for ARROW 2 has been completed. The design of the remainder of the rear fuselage is complete, as noted in the previous Quarterly Technical Report.

20.14 REAR FUSELAGE STRESS ANALYSIS

The redesigned tailcones and centre structure for ARROW 1 were approved and the drawings issued in November.

20.15 FIN AND RUDDER DESIGN

Changes in geometry of the ARROW 2 rudder hinge moment limitation system required modification to the skins, the hydraulic actuator access door, and the relocation of the modified link assembly. An additional access door will be provided on the right-hand side of the fin for access to the rudder jack.

The lower pitot mast has been deleted and the original fin structure replaced.

The attachment of the leading edge skin to the front spar beam will employ smaller rivets from station 240 to the fin tip. This will fulfill the stress requirement by eliminating the need for dimpling of the skins.

20.16 LONG RANGE TANK

The design of the long range tank for the ARROW 2 is still held in abeyance due to higher priority work.

20.17 LANDING GEAR

No further design problems have been experienced on the ARROW 1 landing gear. The faults mentioned in the previous ARROW Quarterly Technical Report were eliminated before taxi tests started.

200-CA-2

FIG. 57 STRUCTURE - ARROW 1

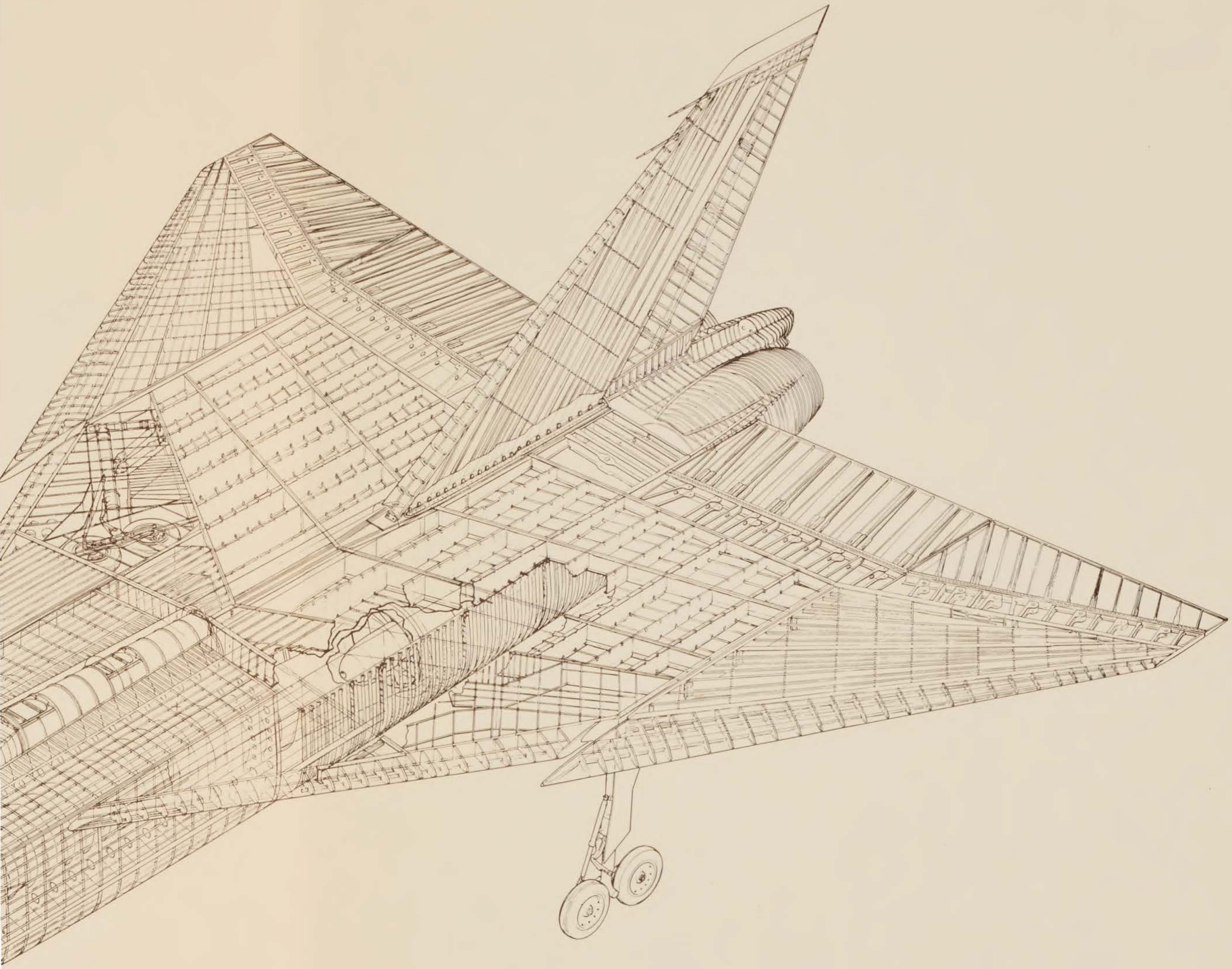


FIG. 57 STRUCTURE - ARROW 1

NOTE: THIS DRAWING IS INCOMPLETE

21.0 MAINTENANCE AND RELIABILITY

21.1 MAINTENANCE ENGINEERING

Comparatively little work has been done in the past quarter on issuing new maintenance data records, the main emphasis being on the preparation of preliminary maintenance instructions for ARROW 1. Existing maintenance data records have been reissued to meet the maintenance requirements of aircraft 25201. Approximately 100 maintenance data records and 26 maintenance instructions have been issued since the closing date of the last quarterly technical report. Eight of these instructions have been revised to include more up to date information, based on experience gained on aircraft 25201.

To facilitate aircraft ground servicing and to improve the safety aspects and operational efficiency of the ground support equipment, the maintenance group recommended modifications to the mobile engine starter, air conditioner and AC-DC generator, hydraulic test machine trainer, test cables for the Minneapolis Honeywell HT109 variable capacitance testers, and the 1 ft. by 4 ft. servicing stand. During the past quarter, 44 maintenance change requests have been incorporated in the aircraft design, leaving a balance of 70 change requests outstanding.

Commencing 11 December 1957, maintenance engineering personnel have also provided a 24-hour coverage for aircraft 25201 in order to report malfunctions on the aircraft and ground support equipment, and to monitor the maintenance procedures previously issued by the maintenance group. They are responsible for ensuring that the maintenance procedures are adequate, accurate and practicable, and for recommending any changes or modifications considered necessary.

21.1.1 MAINTENANCE AND OVERHAUL OF THE ASTRA I SYSTEM

A "Preliminary Report on Maintenance and Overhaul of the ASTRA I Electronics System" prepared jointly by AVRO and RCA was submitted to the RCAF in October 1957. RCA requested that comments be submitted by 15 December 1957, in order that they may be included in the next issue of the ASTRA I quarterly report.

21.1.2 PERSONNEL REQUIREMENTS DATA

W/C Magnussen DAPC

W/C Grant OCT

Due to diversion of maintenance engineering personnel to the intensive coverage of aircraft 25201, Personnel Requirements Data activity during the reporting period was limited to the preparation and submission to the RCAF of a sample PRD report conforming to the latest RCAF requirements and to discussions with RCA and Orenda. A DECP covering the expanded scope of PRD work, requested by the RCAF which necessitates additional staff, is being processed.

21.1.3 EQUIPMENT QUALIFICATION TESTS

During the past quarter, the total number of equipment items which require qualification for the ARROW 1 has increased from 916 to 927. Of these, 520 have been fully qualified, 176 items have received limited flight approval (LFA) based on a partial qualification program and 180 items have limited flight approval based on engineering assessment. The remaining 48 outstanding items lacking qualification are not required for the first flight.

The equipment qualification status is under continuous review by the Reliability Engineering group, and items of equipment are up graded in status as test information is received from the equipment vendors. A particularly close watch is kept on those items on which little test information is available, other than functional system testing on rigs and first aircraft. The reliability of those items is assessed against the planned flight envelope of initial flight tests.

22.0 GROUND SUPPORT EQUIPMENT

The essential ground support equipment required for the first flight of the ARROW 1 aircraft has been delivered to AVRO. During the past quarter, the engineering for the liquid oxygen converter trailer and the probe cover has been completed. Work is proceeding on the fuselage maintenance stand. The engine exhaust cover is being redesigned to provide for the increased diameter of the aircraft tailcone, and the engineering for the aircraft portable boarding ladder is being finalized. The sling for the J75P5 afterburner is in abeyance pending information from the engine manufacturer. Two new items of equipment, a tool for removing the canopy pip pin and a winch to raise the fire extinguisher bottles into the aircraft, have been added to the equipment list for the ARROW 1. These items are not yet in work.

The following items of ARROW 2 ground support equipment are in work: The armament pack test stand, Iroquois engine change stand, Iroquois maintenance trailer, and the radome and probe maintenance trailer.

As a result of the RCAF evaluation of the engine change equipment, work is proceeding on the Iroquois change crane.

The fuselage sling required for Station 255 is complete.

Engineering on the following items required for the ARROW 2 is still outstanding:

Engine starting truck - specification for
Power and air conditioning truck - specification for
Multiple missile trailer
Armament harmonization stand
Armament pack test console
Engine intake cover
Engine exhausts cover
Air conditioning outlet cover
Radome cover
Auxiliary external fuel tanks trailer
Fuel tank test intercooler unit
Aircraft component slings for:
- Rudder control box
- Rudder
- Elevator control box
- Elevator
- Aileron control box
- Aileron
- Air conditioning pack
- Tailcone

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 stand
Main landing gear tie-down

22.1 GSE QUANTITY REQUIREMENTS STUDY

The RCAF has requested AVRO to revise the ARROW GSE list to include all GSE for the weapons system and recommended quantities for RCA, Cold Lake, a staging base and an operational air base. The list would be based on a total of 37 aircraft. Because of contractual limitations which preclude direct procurement of the necessary information by AVRO from the other contractors involved, it was decided that the RCAF would ask the associate contractors to supply lists of GSE items, and that the RCAF would provide AVRO with information regarding the operations planned at the various bases. Upon receipt of this information, AVRO is recommending quantities for each base.

22.2 MOBILE GROUND POWER UNITS

Report LOG/105/24 issued by AVRO in 1956 evaluated various types of equipment to supply the required services for engine starting, electrical power and cooling air. However, in view of changes to the requirements for the ground equipment supplying power and cooling air, the conclusions in that report are no longer applicable. Consequently, in December 1957, AVRO issued report 72/GEQ/1 (ARROW 2 Design Study on Mobile Ground Power Units). The purpose of report 72/GEQ/1 is to present the RCAF with the results of a further study aimed at determining the optimum way to meet the new requirements for electrical power and cold air, based on environmental atmospheric conditions as laid down by the RCAF in letter S36-38-105-13 (ACE 1), dated 21 Auguest 1957. The following recommendations were made:

22.2.1 ENGINE STARTING UNIT:

The report recommends that the ARROW 2 engine starting unit use the AiResearch GTC 85-20 gas turbine compressor, provided that the engine starting times under extreme environmental conditions are considered acceptable by the RCAF. Using this unit, the ARROW 2 scramble time as defined in AIR 7-4, issue 3, para, 3.4.1 is calculated to be 1 minute and 4 seconds at 3,500 ft. with 120°F ambient temperature. If this time is not acceptable to the RCAF, the Blackburn Palouste 500 engine is recommended, which would reduce the time to approximately 44 seconds. It is recommended that the engine starting unit be designed as a self-contained package incorporating the following features:

(a) Fuel for one hour continuous running at maximum power.

(b) Two 3 1/2 in. diameter flexible hoses, 35 ft. long, terminating in automatic quick disconnect couplings.

(c) An electrical cable 50 ft. long to supply the following services to the aircraft:

(1) 50 amperes at 27.5 V. DC.

(2) 500 V.A. at 115 V 400 cps, 1 phase AC.

(3) Interphone between cockpit and ground crew

(4) 27.5 V. DC signals to control the air flow valves on the starting unit from within the aircraft.

Power for these services is to be obtained from the gas turbine engine on the starter.

(d) Batteries for starting the gas turbine compressor.

(e) Storage for air hoses and electrical cables.

22.2.2 MOBILE POWER/AIR CONDITIONING UNIT

It has been proposed that the ARROW 2 mobile power/air conditioning unit be a trailer powered by two Continental "Packette" gasoline engines, one driving a 60 K.V.A. 400 cps, 3 phase AC generator and a 28 V DC generator and the other driving air blowers and a Freon refrigeration compressor. It is recommended that the mobile power/air conditioning unit incorporate the following features:

(a) Tandem Miechle-Dexter 5516 air blowers, or equivalent.

(b) Self-mobility at approximately 1 mph on level ground for a distance of 100 ft. on battery power, or for an indefinite distance provided that the engine which drives the generators is running.

(c) Height not to exceed 5 ft. 6 in.

(d) Two 3 1/2 in. diameter air delivery hoses, 45 ft long, terminating in automatic quick disconnect couplings.

(e) A cable, 50 ft. long, to supply 400 cps, 3-phase AC power to the aircraft.

(f) Protective circuits to prevent electrical power from being supplied to the aircraft if the delivery air temperature exceeds 55°F, or if the delivery air pressure is below 6.3 psig at the outlet of the air conditioning unit.

(g) A convenient control panel for all controls and instruments.

(h) Convenient storage for air hoses and electrical cables.

(i) Light alloy construction wherever possible, in order to keep the weight to a minimum.

(j) Cooling air flow velocities up to 1500 ft/min. in order to keep the size of the heat exchangers to a minimum.

(k) Interphone between the cockpit and the ground crew.

22.2.3 PROCUREMENT RECOMMENDATION

It is recommended that AVRO be authorized to procure an adequate quantity of ARROW 2 engine starting units and mobile power/air conditioning units so that the RCAF development program will not be delayed.

22.3 PROVISION OF GOVERNMENT-FURNISHED TEST EQUIPMENT

The immediate problem of provision of government-furnished test equipment required for ground servicing the aircraft has been overcome. A revised procedure has been established by the RCAF and DDP to avoid future problems of this nature.

23.0 AIR BASE FACILITIES

In compliance with RCAF instructions, a report on the ARROW 2 first line maintenance and turnaround facilities (No. 72/GOPS/2) is being prepared and will be issued shortly to the RCAF. The report outlines the requirements for equipment, procedures, personnel and maintenance facilities for turnaround and first line maintenance. The report also reviews the ground servicing equipment for main and forward base activities.

A turnaround hangar consisting of four separate bays is recommended for bases from which one squadron will operate. A similar hangar with eight bays is recommended for two squadrons. Refuelling, rearming and between flight inspections would be carried out in this hangar. It is recommended that the turnaround hangar be located near the readiness facility to isolate the high noise level activities from other areas of the base.

The advantages and disadvantages of combining first line maintenance with the turnaround function are discussed.

A preliminary study indicates that the airframe inspection may be completed in approximately 1 hour 15 minutes, and the primary inspection of the ASTRA I sub-system may occupy 2 hours 40 minutes. However, a demonstration will be required to establish realistic times for primary inspections.

The report further indicates that 14 men will be required to complete a turnaround within 15 minutes at a forward base, using mobile equipment with no prepared facilities. The same number of personnel could complete two turnarounds within 15 minutes in a prepared turnaround hangar, under all weather conditions.

The equipment for field use is entirely mobile and is air-transportable in C-119 aircraft. A demonstration to establish the number of C-119 aircraft necessary to support a forward base is desirable.

Estimates are given of power requirements for the turnaround hangars and the first line hangar. Provision of emergency power supplies for both hangars is essential. Lighting and electrical equipment should be explosion proof. Underwing lighting will be required because of the aircraft's high wing configuration.

Heating in the turnaround hangar should be capable of maintaining a temperature above freezing, to ensure a supply of water to replenish the heat exchanger in the air conditioning system.

Hydrant refuelling is recommended for efficient handling of the large quantities of fuel involved. Tests are required in order to establish the fuel temperatures attainable in the fuel tanks while the aircraft is parked in the open, under tropical conditions and while parked in hangars during the summer. Tests are also necessary to establish refuelling tanker and hydrant system delivery temperatures under similar circumstances, to determine if cooling of the refuelling equipment is necessary.

A portable three-gallon-capacity oil dispenser would be adequate for each turnaround bay, for topping up engine oil tanks.

A vehicle suitably adapted to carry drag chutes is recommended for use in collecting drag chutes from a central parachute packing facility and delivering them to the turnaround hangar.

The most efficient form of cockpit access for the turnaround hangar appears to be the catwalk which not only eliminates a cumbersome ladder from the turnaround area, but facilitates rapid and safe access to the cockpit and dorsal area for replenishment of oxygen and water. It is recommended that the catwalk be provided as part of the readiness and turnaround hangar furnishings.

The "UNITOW" D-8 tractor is suitable for flight line towing of the ARROW but it is recommended that each base be equipped with other tractors with a 10,000 lb. draw bar pull, for winter use. The tractor used for towing the armament hoist trolley should not exceed five feet in height. This will permit the tractor to pass under the fuselage when towing a trolley.

A mobile water tanker is required at forward bases, for replenishing the heat exchanger boiler. This should incorporate a heater unit and hopper for feeding in snow to ensure a supply of water in sub-zero weather.

It is recommended that instruction on maintenance and operation of the ground equipment be incorporated in the RCAF training program, as the major items of new equipment, such as the gas turbine compressors for engine starting, the ground electric power and air conditioning truck, the hydraulic test rig, the liquid oxygen storage tanks and the hydrant refuelling system. These units are complex and will require comprehensive understanding for efficient operation.

24.0 AIRCRAFT SYSTEMS TRAINER

24.1 AST ACTIVITY

The past quarter has been devoted to the preparation of a Technical Proposal for an Aircraft Systems Trainer (AST), requested by the RCAF under the authority of the AD-44 Statement of Work. The proposal will be submitted to the RCAF in January 1958.

24.2 SUMMARY OF ARROW 2 AST PROPOSAL

Some systems, which are of the relatively conventional type and do not require special maintenance techniques and special test equipment, have been presented as illuminated schematic panels. These panels are supplemented where necessary by the use of sectioned components and projection slides. Training panels in this category are:

Hydraulic systems
Electrical systems
Air conditioning system
Airframe fuel system
Oxygen system
Flight instruments

For the illuminated schematic panels, it is considered that one basic panel design can be adapted to provide for any panel configuration. The design of all frames is identical, with the exception of the length which is determined by the layout of the system presented.

Where more complex systems are involved, dynamic trainers are proposed. These dynamic trainers, together with the test equipment, will permit the physcial operation of the relevant aircraft system to be demonstrated, and allow various maintenance procedures to be performed. By the use of the test equipment provided, regular first and second line checks can be made. Training rigs in this category are:

Flying controls system
Damping system
Escape system and ejection seat
Drag chute
Armament system

It is not at present intended to demonstrate the function of the flying controls in conjunction with the ASTRA I system outputs, as the complication caused by the integration of these systems would add little to the training value of the flying controls demonstration.

The effect of aerodynamic damping is simulated by the interconnection of the flying control trainer and the damping system trainer. Signals from the sensors mounted on the damping trainer, and signals from the pilot's controls on the flying control trainer are fed into the electronic damping system. The outputs are visually indicated by the control surfaces on the flying control trainer.

Projection slides are proposed, not only to support the previously mentioned training panels, but to adequately train personnel in the operation of various systems which are not sufficiently complex to require a training panel.

In addition to a complete description of the aircraft systems, and the training panels for those systems, the Technical Proposal for the Aircraft Systems Trainer includes relevant information on the trainer weights, dimensions, power requirements, and also shows a proposed layout of the entire training facilities.

24.3 GSE TRAINERS AND OTHER ASSOCIATE CONTRACTOR'S AST PROPOSALS

Ultimately the RCAF wants the AST proposals for the various sub-systems of the ARROW Weapon System to be coordinated with the airframe study. This task will have to follow the issue of the airframe AST proposal because two relevant RCA documents will not be issued until the first and third quarter of 1958, and dates for Orenda and Canadair proposals are not yet available.

Subject to receipt of contractual authority, a proposal for trainers for ground support equipment will be submitted when the RCAF has decided on the configuration of the more complicated units, such as the engine starter unit and the air conditioning and electrical power supply units.

25.0 STRUCTURAL GROUND TEST PROGRAM

25.1 SCOPE OF STRUCTURAL GROUND TEST PROGRAM

The structural test program includes the testing to be undertaken on a complete airframe and the detailed testing of components required for structural development. In addition, the program includes detailed tests to investigate fatigue life and high temperature properties, which will not be covered in the complete airframe test.

25.2 STATIC TESTING OF THE COMPLETE AIRCRAFT

The test program established for static testing of the complete aircraft is as follows:

1. Design and construction of the test rig, 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 the installation of external strain gauge.

5. Landing gear springback case to limit load.

6. Rolling pullout case to limit load.

7. Integrity test of main landing gear uplocks and doors.

8. The symmetric case with no pitch to limit load: test on front fuselage.

9. The symmetric case with no pitch to limit load; test on rear fuselage.

25.3 PROGRESS ON STATIC TESTING OF COMPLETE AIRCRAFT

Construction of the test rig tests applicable to the landing gear springback case to limit load and the rolling pullout case to limit load was complete by December 1957, although assembly of the loading structure cannot be completed until the static test aircraft is installed in the rig.

The installation of the internal strain gauges in areas of the specimen which are inaccessible after assembly was completed by mid-December. Installation of the external strain gauges cannot be completed until the test aircraft is installed in the test rig. It is anticipated that the installation of the aircraft in the rig, and completion of the external strain gauging will occupy eight weeks.

When the test program was originally set up, it was known that the installation of strain gauges and the testing of the static aircraft would take place over a relatively long period. Since no information was available on the reliability of strain gauges over similar periods of time, specimens bearing suitable strain gauges were built concurrently with the installation of gauges on the aircraft. These specimens have been tested every month, and will continue to be tested as long as the gauges on the aircraft are in use. Results so far obtained show that the strain gauges used retain their accuracy to within ±2%, over a two-year period.

25.4 TESTING OF MINOR COMPONENTS

Structural strength testing of a wide range of components has been undertaken under loading conditions covering the critical areas of the flight envelope. Many of these tests are continuing, but all tests considered necessary for the first flight will be completed by the time the flight takes place.

25.5 ENGINE INTAKE DUCT (FLOATING ASSEMBLY)

Preliminary air pressure tests showed that sealing procedures between the two portions of the floating ducts has not been incorporated in the production drawings for the extreme aft end of the engine intake duct in the gill area. The test specimen was modified and a repair scheme applied to the aircraft Limit pressure tests of the duct were then carried out using water at the pressurizing medium The duct withstood a pressure of 60 psi, which was considered satisfactory.

25.6 FATIGUE TEST - TYPICAL SKIN SPLICE

An extensive fatigue test and development program has been concluded on a series of typical skin splice specimens. As a result, a satisfactory type of splice has been developed.

25.7 RUDDER STIFFNESS AND LIMIT LOAD TEST

Strength and stiffness tests of the rudder were temporarily discontinued, after 60% of the work was completed, due to the extreme urgency of the flying controls functional tests now in progress on the test rig. Further tests are to be conducted which will consist of moving the rudder through its full range of travel under air loads up to limit values when the fin is in its deflected shape.

25.8 ENGINE SHROUD STRENGTH TESTS

The engine shroud strength tests have been completed and portions of the specimen have been sent to the metallurgical laboratory for further examination of the spot welding The shroud was subjected to internal pressure loading while side loading was imposed The specimen withstood 4800 cycles of load application before failure occurred The test was considered satisfactory.

During the engine shroud strength test, results were recorded by the IBM punch card system in order to test this system, prior to using it for the complete aircraft static tests. The results were unsatisfactory due to the card punch being incorrectly centered. Card sorting difficulties were also encountered. These faults can be remedied however, and no major problems are foreseen in using this system.

25.9 COMPLETE AIRCRAFT VIBRATION TEST

The complete aircraft vibration test was completed. The results of the modes demonstrated were generally in agreement with earlier calculations, although some differences were encountered on fin modes and some wing anti-symmetric modes. Vibration testing will be continued to confirm the fin modes, while the final preflight instrumentation is being completed.

25.10 COMBINED LOADING AND TRANSIENT HEATING OF WING BOX

Temperature-compensated strain gauges are required for this test, the earliest delivery date quoted for the required type of gauges being April 1958. The specimen is nearing completion and the loading and heating arrangements are almost ready. Testing will be started as soon as the strain gauges have been received and installed. The possibility of NAE producing temperature-compensated strain gauges is under active investigation.

25.11 TEMPERATURE DISTRIBUTION THROUGH TYPICAL STRUCTURAL SECTIONS

A limited number of temperature-compensated strain gauges have been available for this test, and it has been possible to carry out a restricted test program on typical joint specimens. This program is continuing.

25.12 FATIGUE TESTS OF MODEL FUEL TANK NO. 4

Tank number 4 was considered critical as its skin gauge and rib construction differs from the other fuselage tanks. After 706 cycles of fatigue testing on the specimen tank, failure occurred in the vicinity of the door edge stringer. This was considered unsatisfactory and the specimen was returned for modification and repair. Further testing will be undertaken in January with the improved design. The specimen must withstand at least 2000 cycles before being considered satisfactory.

25.13 ENGINE DOOR STRENGTH TEST

Tests performed on the original door specimen were unsatisfactory. The latches were found to jam easily and any distortion of the door made their operation extremely difficult. During the strength tests, failure of the latches occurred at 85% design ultimate load. Further tests will be made when the specimen has been modified and repaired.

25.14 WINDSHIELD - COMBINED FRAME AND GLASS TEST

These tests have been concluded satisfactorily. The windshield casting proved satisfactory up to class 1A ultimate loading, and the glass successfully withstood a pressure of 70 psi.

25.15 PANEL RESPONSE TO SOUND PRESSURE AND FREQUENCY

Representative panels from critical areas of the aircraft structure have undergone tests. Magnesium alloy rudder and fuselage side skin panels and stainless steel stinger panels have been tested in the acoustic chamber. The results of the tests on the stinger and side panels were not satisfactory and further development testing is being undertaken.

25.16 FATIGUE TEST OF ELEVATOR LINKS

Fatigue tests on the elevator links were to be carried out by Krouse Testing Machine Inc., at Columbus, Ohio. Test specimens and a test rig were despatched to Columbus and tests were started. The first three specimens failed prematurely, and Krouse was requested to return the specimen and rig. Investigation established that failure was caused by the use of incorrect adaptor fittings. Testing will continue with the correct adaptor fittings.

25.17 PRESSURE TEST OF ARROW FUEL TANKS - AIRCRAFT 25201

The wing fuel tanks were subjected to an internal pressure of approximately 24 psi and the fuselage tanks to approximately 10 psi. Minor leaks occurred but these were sealed during testing and the tanks were ultimately considered satisfactory.

25.18 DEVELOPMENT OF HIGH TEMPERATURE STRUCTURAL TEST TECHNIQUES

The metallurgical laboratory is engaged in a development program of high temperature strain gauge testing. In addition, as part of this program, an analysis of the thermal characteristics of test substitutes for JP-4 fuel has been completed. Further investigations are continuing into suitable methods of developing a kinetic heat simulator.

3066-105-1

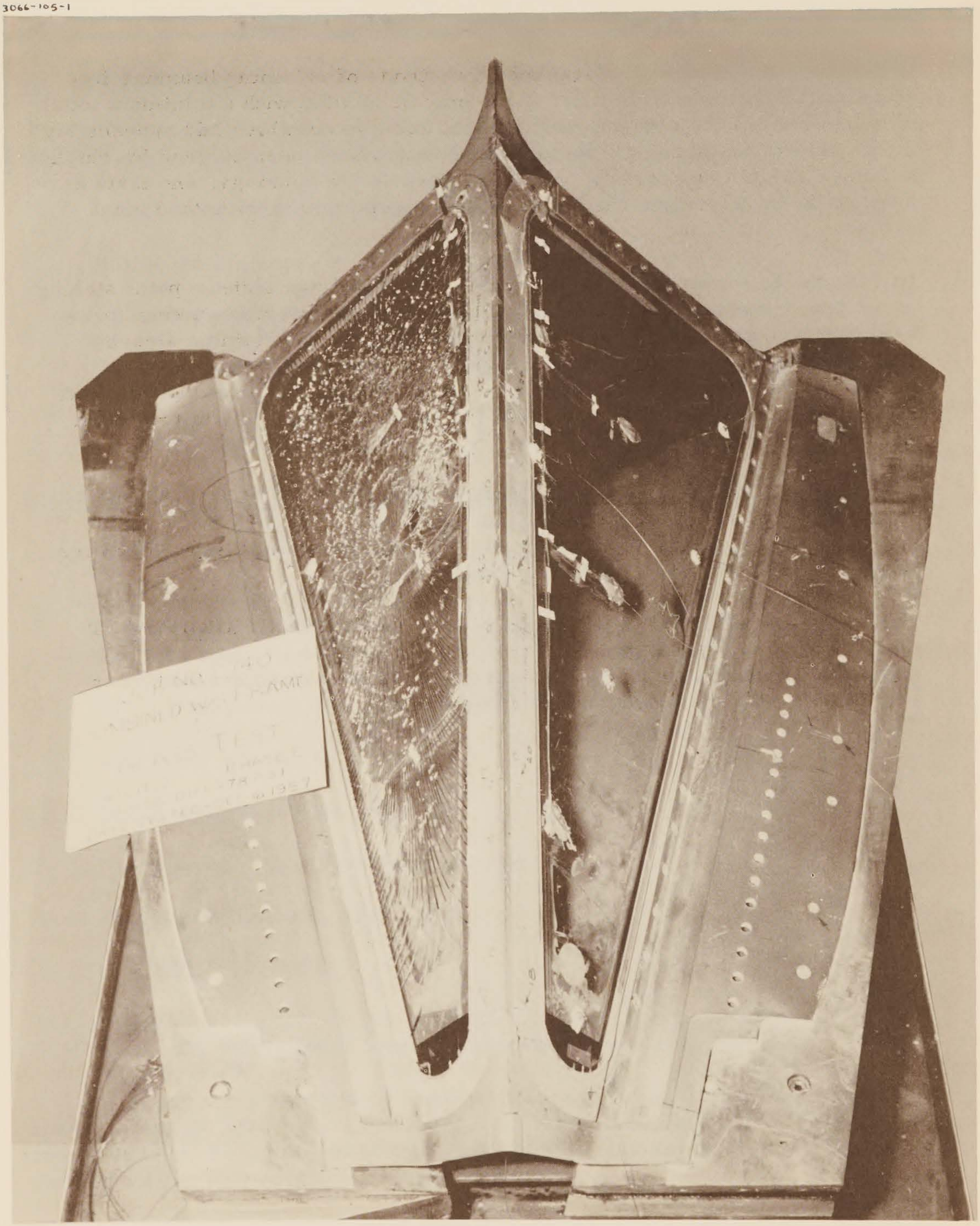


FIG. 58 WINDSHIELD GLASS AND CASTING TEST

25.19 BEARING RETENTION METHOD DEVELOPMENT

Development testing has proceeded on methods of retaining bearings in housings so that specified axial loads may be applied with a minimum axial displacement of the bearing, and without using parts which add appreciably to the bulk of the housing. Suitable techniques have been evolved by which bearings can be successfully locked in place in the housings, and tests have been made to determine the ability of the assemblies to withstand axial loads.

So far, the following methods have been developed and tested: point staking using press tools of various shapes, segment staking, ring staking, press swaging, and roll swaging with and without the use of retaining sleeves Results are available for a wide range of bearing diameters, and with housings and sleeves of different materials. It is anticipated that the roll swaging of corrosion-resistant steel retention sleeves, now in progress, will terminate this program.

25.20 STRENGTH TESTS OF LANDING GEAR FRONT PIVOT BEARING

No testing has as yet been performed. The design of the test rig has been completed and the rig is now being manufactured.

25.21 STRENGTH TEST OF ARMAMENT PACK REAR PICKUP BOLT

This was a combined shear and bending test of the armament pack rear pickup bolt. The test proved satisfactory.

26.0 SYSTEMS GROUND TEST PROGRAM

26.1 FUEL SYSTEM

The fuel system tests have been divided into seven main sections, and the tests are to be conducted in approximately the following order:

(1) Calibration and start-up.

(2) Initial testing under static conditions:

- (a) Pressure refuelling and defuelling.
- (b) Pressurization system check on pressure build-up and relief valve.
- (c) In-flight fuel transfer.

(3) Complete investigations of all operating aspects for full ranges of parameters under static conditions. This includes all tests listed in (2) plus:

- (a) Collector tank pressurization.
- (b) In-flight fuel transfer tests (inclined attitudes)
- (c) Fuel pressure regulator tests.

(4) Simulated mission tests.

(5) Changeover to low pressure test.

(6) Changeover to fuel management system.

(7) Fuel management system tests.

26.1.1 TEST PROGRESS

The in-flight fuel transfer tests at inclined attitudes were completed by mid-October and the preliminary results were considered satisfactory.

Loss of prime was experienced in a fuel transfer pump, and this has necessitated additional tests of the pump at ground level and extreme altitude conditions. It has been established that the fuel transfer pumps are satisfactory for the first flight of aircraft 25201.

A pre-installation test of the fuel bypass override valve proved unsatisfactory and the valve was returned to the manufacturer for adjustment. The valve was returned to AVRO, but subsequent tests performed were still unsatisfactory. Tests are still in progress. A preliminary simulated mission test was conducted, but the results indicated that the test conditions had not been clearly established. The missions were more clearly defined and the test continued.

Fuel system testing was suspended for ten days while the rig was insulated for hot fuel tests. Insulation of the rig has allowed a wide range of tests to be performed. Six critical simulated mission cases have now been completed, and preliminary assessment indicates that results are generally satisfactory, although mechanical operation of the fuel-no-air valves became difficult when hot fuel was used. These valves are deemed satisfactory for first flight of aircraft 25201, but further testing will be undertaken. Tests are being conducted to establish the most suitable form of air ejector operation for the fuel collection tank. Two designs were tested at extremes of the operational temperature range. The first design satisfies the requirements for the ARROW 1 fuel system. The second design is undergoing further development testing to meet the ARROW 2 fuel system requirements.

26.1.2 FUEL SYSTEM PRE-FLIGHT TESTS

Pre-flight testing of the fuel system on aircraft 25201 has been ~~completed~~ completely satisfactorily. The pressure regulation, pressure build-up and leakage tests were satisfactorily completed by mid-November and were followed by the refuelling and defuelling tests.

During these tests, several microswitches associated with the fuel valves were found to be incorrectly adjusted. Pressure leakage from an air line in fuselage tank No. 5 caused excessive pressure in that tank. Leaks from the left hand sub-system were traced to a Wig-O-Flex coupling from which the "O" rings had been omitted.

Fuel transfer tests and tests of the engine feed system followed. During these tests, it was found that when power was first applied to the electrical circuits, the proportioner bypass valve moved into bypass operation, and could not be diverted without turning the power off. This fault was caused by the slowness of the level sensing response switch in tank No. 5, giving rise to a spurious low level signal. This signal caused the relay on the bypass valve to operate, locking the valve in the bypass position. This trouble has been rectified by incorporating the bypass valve locking relay into the circuit of the nose landing gear scissors switch. Thus, by the time power is fed to this relay, the level sensing switch has reached a stable condition, and the bypass locking relay will then operate only on receipt of a true low level indication from the switch. (See also Electrical System Pre-Flight tests para. 26.4.1).

Electrical faults were also located in the isolating cocks downstream of the oil-to-fuel heat exchanger. The isolating cocks remained closed during the first attempts to transfer fuel. These faults were rectified, and the normal transfer of fuel was successfully performed.

Fuel transfer pre-flight tests, using power from the aircraft generators, were performed during the engine run on aircraft 25201.

26.2 FLYING CONTROLS SYSTEM

The main test of the flying controls will consist of an evaluation of the complete system from the cockpit controls to the control surfaces (See Fig 48). Individual tests on hydraulic and mechanical components will also be performed.

The program for the flying controls system testing is as follows:

(1) Elevator frequency response tests without hinge moment.

(2) Elevator frequency response tests with hinge moment.

(3) Duty cycling of elevator system at room temperature.

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

(5) Hydraulic system investigations.

(6) Aileron and rudder frequency response tests without hinge moment.

(7) Aileron and rudder frequency response tests with hinge moment.

(8) Combined system tests with hinge moments at room temperature.

(9) Simulated flight tests with full fuel flying controls and damping system and analog computers at normal, high and low temperatures.

Sections (1) (2) and (3) of this program were completed prior to the period covered by this report.

26.2.1 FLYING CONTROLS TEST PROGRESS

Development of the elevator input circuit for manual and stick force modes was completed by November. The elevator system was given qualified approval by the test pilots, although stick breakout forces were considered high. The control valves were modified by introducing backlash in the rate spring, but further tests showed that stick breakout forces were still too high.

Efforts have been made to reduce stick breakout forces by reducing system frictional forces. A test program was instituted to test combinations of a variety of cables of different sizes and materials, in conjunction with pulleys of different materials and sizes. The frictional forces of each combination were measured. Tests were also performed using cables impregnated with suitable lubricants, but the effects of this have been negligible. Results so far indicate that a 3/32 in. diameter cable of stainless steel running on a standard AN220/2 aluminum pulley has given the most satisfactory results.

During tests of the aileron circuit, using the complete mechanical system flying controls test rig, it became evident that there were self-oscillatory conditions in the aileron and elevator circuits. This condition caused excessive vibrations in the control surfaces.

Intensive efforts have been directed towards removing or reducing these oscillatory conditions. Hydraulic accumulators were installed in the hydraulic circuit, close to the control valves. This appreciably reduced the pressure surges which were causing the vibrations. Tests were then performed with the accumulators removed from the circuit, and with Humphrey and AVRO designed dampers fitted to the control valve spindles. The merits of the various methods of reducing oscillatory conditions are presently being assessed.

A test program has been performed to assess damper characteristics, using a pendulum test rig operated by a motor crank arrangement. Initial tests on a 3/8 x 5/8 Humphrey damper showed that the damping force level diminishes during endurance cycling, and the damping constant varies considerably during tests at high and low ambient temperatures. Tests of an AVRO designed damper are now in progress. The results so far indicate no reduction in damping force after 38,500 cycles of operation.

Simulated flight tests at 20,000 feet and at Mach .7 were conducted with the control valve dampers off and with the yaw damping system operative. During these tests, a number of elevator and aileron feel units were assessed in order to select the most satisfactory unit for flight. A production elevator feel unit spring with a rate of 75 lb/in. was selected for the elevator and aileron. Low speed tests to simulate normal landing were also conducted. For these low speed tests, it was arranged to produce elevator surface loads of 240 - 300 ft. lb. per degree of movement. The hydraulic pump speed control was connected to the throttle box mounted in the cockpit.

Simulated flight tests were also conducted with a Greer hydraulic pump rig to simulate the ram air turbine hydraulic supply. The control surfaces were loaded as above. The results indicate that the ram air turbine cannot supply sufficient hydraulic power to operate the control surfaces as required.

Tests have also been made to determine the stick forces for various rates of elevator motions. The test results are now being studied.

During tests to determine the frictional forces in the aileron control system, the breakout force at the control stick was found to be 8 lb. This was considered to be excessive and the system was modified by replacing the existing 1/8 in. diameter cables in the outboard cable runs with 3/32 in. diameter cables. The aileron control cable pulleys in the fuselage were remachined to reduce any ovality in them. These changes have further reduced aileron control system friction.

The installation of 3/8 x 5/8 Humphrey dampers on the control valve spindles has eliminated aileron oscillations. Tests have shown that these oscillations originate in the hydraulic system. Tests are now in progress to determine the critical damping range of the ailerons.

Frictional forces in the rudder control system were evaluated and were found to be acceptable.

Oscillations in the control surface have been eliminated by installing a dash-pot type damper on the control valve spindle.

Pedal forces, to produce known surface deflections, have been measured at differing trim positions.

Frequency response tests are continuing using the differential servo as the point of application for the electrical input.

Tests will be made to evaluate the hinge moment limitation system. With reference to para. 14.2.1, a program is proceeding on the flying controls test rig, to test and evaluate the control boosters. Preliminary results confirm that the boosters overcame the control friction break-out force.

26.2.2 GENERAL FLYING CONTROLS SYSTEM TESTS

26.2.2.1 Hydraulic Connections for 4,000 psi

The initial airworthiness tests of the connections are continuing. The results of a program to develop hydraulic connections to withstand high temperature flexural fatigue have not been satisfactory. An improved method of swaging the connection of the sleeve to the tubing, by expanding the tubing into the sleeve, has been developed and tested. Although the first of these specimens gave improved results they were still not satisfactory. Other specimens were manufactured to closer tolerances and tested, but the results have not yet proved satisfactory. The test program continues. (Ref. para. 14.3).

26.2.2.2 Wig-O-Flex Couplings for the Low Pressure Hydraulics

All the specimens required for these tests have been tested, and satisfactory results were obtained. A further range of specimens is to be tested.

26.2.2.3 Rudder Jack and Valve Test

Testing has been conducted on several types of damper to determine their effectiveness in reducing jack instability. The dampers were mounted on the rudder jack and oscillations in the system were induced by a vibrator striking the follow-up linkage. A Houdaille damper and an AVRO-designed damper had no appreciable effect in reducing the jack instability. An AVRO-designed dashpot type damper was then tried, and eliminated the induced oscillations. This damper has now been incorporated in the rudder control system.

26.2.2.4 Fatigue Tests of Aileron and Rudder Pressure Lines

A series of flexible pipes was tested, with varying degrees of success. Testing is being continued using chrome molybdenum steel and stainless steel tube specimens.

26.2.2.5 Flexural Endurance of Parallel Servo Pipes

Six specimens have been tested. Four failed at the brazed portion of the fixed end fittings after approximately 900,000 cycles. Two specimens have achieved 2-1/2 million cycles without failure. The results are being assessed.

26.2.2.6 Flying Control Hydraulic System Pre-Flight Tests

Pre-flight testing of the flying control hydraulic system began in November and continued through the engine running trials. Initial tests revealed faulty operation of the relief valve in the hydraulic "A" system. The valve was adjusted and reassembled in the aircraft and further tests will be made prior to first flight. The accumulator seals were also found to be defective, allowing hydraulic fluid to enter the nitrogen portion of the accumulators. Replacement accumulators will incorporate improved seals.

26.3 AIR CONDITIONING SYSTEM

Development testing of the air conditioning system has been in progress since mid-1956. Recently the test program has been directed towards ensuring system efficiency for the first flight tests.

26.3.1 AIR CONDITIONING SYSTEM TESTS

The test program established for the air conditioning system is as follows:

(1) Start up and calibration.

(2) System trials, adjustments and equipment evaluation at moderate inlet conditions, using some restricted performance equipment.

(3) Incorporation of fully operational equipment and ducting, and recalibration.

(4) Finalized system trials and adjustments, and equipment evaluation over full range of inlet conditions.

(5) Cockpit noise measurements.

(6) Detailed system performance tests.

(7) Ground operation tests.

Parts (1) (2) (3) and part of (4) have been completed and the results were noted in the previous quarterly technical report. The remainder of this test program is now almost complete.

26.3.2 TEST PROGRESS

The major problem on the air conditioning system tests has been the stabilization of the turbine outlet temperature control system. The control valve is very sensitive and oscillated continuously about its optimum setting. In an attempt to improve this situation, the operating time of the valve actuator was altered from 10 seconds to 20 seconds. This alteration has not produced completely satisfactory results, and with controller settings adjusted for minimum flight requirements, the control valve is only marginally acceptable for aircraft first flights. Tests results have been forwarded to the valve manufacturer for further action. During this period, the incorporation of the bleed air ducting in the test rig was completed. Insulation of the rig for high temperature testing was also completed.

Tests have been satisfactorily completed to check the expansion of the duct in relation to the aircraft structure, at high temperature. Future work will cover testing of the ground test panel, and a repetition of the tests of the overheat thermostats.

26.3.3 AIR CONDITIONING SYSTEM PRE-FLIGHT TESTS

Initial tests were conducted on the first flying aircraft using the air conditioning ground test vehicle as an external source of air supply. This is a mobile compressor unit, which can be regulated to supply a flow of air which is representative of the aircraft air supply.

In the tests performed, restrictors with various orifice sizes were added to the air conditioning ducts until satisfactory air flows were attained in all critical areas.

Further tests were made during engine running trials. During these tests, a cockpit temperature controller unit failed. It was replaced and results were satisfactory. All flows and temperatures were to specified levels for the taxiing trials and first flight.

26.3.4 ARROW 1 AND ARROW 2 COCKPIT ENVIRONMENT TESTS

The cockpit environmental tests are being conducted on the metal mockup aircraft. Testing has so far been directed towards reducing the noise level in the cockpits. This high noise level became apparent after adjustments had been made to bring the air flow within specifications. The effect of different duct outlet configurations has been investigated, and further tests incorporating silencing devices in the outlets are to be made.

Preparations for heating the cockpit area are currently in progress, and when complete, temperature distribution in the front and rear cockpits will be assessed.

26.3.5 PRESSURE DROP ON VANED ELBOW

Duct elbows equipped with splitter vanes to direct the airflow have been tested to determine the pressure drop caused by the change in flow direction and the constriction caused by the elbow. The results have agreed closely with calculated predictions of pressure drop.

26.3.6 AIR CONDITIONING SYSTEM - ARROW 2

Design work for the ARROW 2 air conditioning test rig has been in progress for some time and is nearing completion. All instrumentation requirements, including twenty-six thermocouple probes, have been investigated, and most of the equipment required is now available. Arrangements are under way to extend the external air supply to the ARROW 2 rig location.

The cockpit environment tests currently in progress for the ARROW 1 on the metal mockup are considered applicable to the ARROW 2 and no further testing is envisaged in this respect.

26.4 ELECTRICAL SYSTEM

The only tests which remained to be performed since the last quarterly technical report was issued were the pre-flight tests necessary to the first flight of aircraft 25201.

26.4.1 ARROW 1 ELECTRICAL SYSTEM PRE-FLIGHT TESTS

Pre-flight testing of the ARROW 1 electrical system began in October. External AC power from a ground servicing unit was applied to the aircraft electrical system, and tests were made to ensure that all electrical equipment functioned correctly.

In December 1957 the engine running trials of 25201 began. With these trials, the last phase of the electrical system pre-flight tests was started. All electrical equipment essential to the first flight was checked for functional integrity.

Alternator voltages and frequencies were checked over the full range of engine speed, and the voltage was checked at significant points in the system.

During these tests it was discovered that the alternator voltages fluctuated over a small range. While this is not considered detrimental to the system, the matter is being referred to the equipment manufacturers.

It also became evident that the fuel low level switch unit required a warm-up period, otherwise the bypass valve for the fuel proportioners operated and locked when the switch was cycled. (See para. 26.1.2 Fuel System Pre-flight Tests).

Further power checks are to be made during the next engine run.

26.4.2 ARROW 2 ELECTRICAL SYSTEM TESTING

26.4.2.1. Test Progress

The electrical load analysis for the ARROW 2 was completed some time ago. It was decided that the breadboard of the ARROW 1 electrical system could be modified and used for testing the ARROW 2 system. A study of the modifications required was begun in October 1957 and was largely complete by mid-November. Modifications to the existing ARROW 1 breadboard are in progress. Work on the necessary load banks to simulate the loads on the system will commence shortly.

The armament system breadboard test has been the subject of continual revision because of changing requirements resulting from system analysis. The embodying of current modifications in these circuits is continuing and testing will resume when this work is completed to the latest design standard.

26.5 LANDING GEAR SYSTEM

Information so far obtained in the landing gear test program is sufficient to permit initial flight trials of the aircraft.

The test program laid down for testing the landing gear is as follows:

(1) Test rig design and manufacture.

(2) Nose gear functional tests.

(3) Main gear functional tests.

(4) Tests on full landing gear system with loadings to suit various flight conditions.

(5) Full system tests at low and high temperatures.

Test rigs have been made for the left-hand main landing gear and for the nose landing gear. Work in nearing completion on the rig for the right-hand main landing gear.

26.5.1 TEST PROGRESS

26.5.1.1 Nose Landing Gear

The nose wheel steering valve and control cables were installed on the landing gear test rig in October 1957. Tests conducted at that time indicated that spool forces in the steering valve were excessive. At the same time tests were made on the steering valve for aircraft 25201. The spool forces in this valve were also excessive.

The first valve was returned to the manufacturer for modification, but subsequent tests indicated that the valve was still unsatisfactory. The valve was again returned to the manufacturer for further modification. On its return, further tests showed that a pedal force of 30 lb. was required to operate the nose wheel steering. This was considered acceptable by the test pilots. Nose wheel steering tests also disclosed that the steering could be engaged with the steering valve out of neutral. The defect has been remedied by the installation of a microswitch. The switch is actuated by the valve spindle and connected with the steering selection valve, and ensures that the rudder pedals are centred when steering is engaged.

Tests have been conducted and are continuing on the nose wheel retraction mechanism with attention directed mainly to the self-centring characteristics of the nose wheel during retraction. Some difficulty has been experienced due to the steering control cable developing excessive slack when the wheel is retracted. This is currently under investigation. The retraction time of the nose wheel has been regulated to 8.5 seconds, to allow time for the wheel to centre from its extreme limit of turn.

The nose wheel steering system pre-flight tests have been incorporated into the utility hydraulics pre-flight tests. Some testing has been conducted, and adjustment of the nose wheel steering control was required. This test will be repeated. Further testing is still required on wheel brakes and landing gear operation.

26.5.1.2 Main Landing Gear

Preliminary retraction tests of the main landing gear showed that damping characteristics were unsatisfactory. The jack was returned to the manufacturer for modification, but subsequent tests showed no improvement. The damper was finally removed and this improved the operation of the jack. Testing was performed with critical loads applied to the landing gear and a total of 25 retractions was completed under these conditions. The retractions were generally satisfactory. However, subsequent inspection revealed that the retraction jack head end bearing and the retraction jack pick-up bracket bushing had become partially extracted from their housings. It is proposed to repeat these tests when repairs to the bearing and bracket bushing have been completed. The back stay has been returned to the manufacturer to have a 1 inch diameter bearing fitted at the retraction jack pick-up.

26.5.1.3 Utility Hydraulics System Pre-Flight Tests

The tests on the system leakage relief valve operation and speed brake operation have been satisfactorily completed. Some testing in connection with the nose landing gear has also been performed (see Nose Landing Gear Para. 26.5.1.1).

26.5.1.4 Drag Chute Pre-Flight Tests

Preliminary pre-flight tests of the drag chute were performed on aircraft 25201. Additional tests were conducted during low speed taxiing trials in December. The tests were considered satisfactory after modifications had been incorporated to the pull-out wire for the drogue release mechanism.

26.6 <u>CANOPY AND ESCAPE SYSTEM</u>

The program for the canopy and escape system testing is as follows:

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

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

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

(4) Rocket sled tests with dummy ejections.

Section (1) was completed prior to the period covered by this report. All functioning tests connected with part (2) have been completed since November.

26.6.1 TESTS PROGRESS

Construction of the test rig for part (2) was completed by November 1957. Canopy functioning tests were then conducted as follows: three under atmospheric pressure conditions, three at -2.8 psi differential pressure and two at 5.75 psi differential pressure. These tests were satisfactory.

A stiffness test was performed by applying a single concentrated load midway along the canopies. Results from this test were also satisfactory.

Preparations are now in progress to test the emergency jack and recuperator in order to determine the damping constant. A rig has been designed and constructed to test the seat ejection system from a stationary cockpit. These tests will take place using the static test aircraft, after the aircraft has been subjected to cockpit pressure testing, and before it is installed into the static test rig.

Escape system pre-flight tests have been satisfactorily completed on the pilot's cockpit. Further pre-flight tests will be made on the observer/AI's cockpit at a later date.

There have been no further developments on the rocket sled testing of the escape system. This program is still subject to contractual negotation.

26.7 SPARROW MISSILE PACKAGE

The program for testing the Sparrow missile package has been defined as follows:

(1) Mechanism functioning tests with single missile.

(2) Missile door functioning tests.

(3) Package functioning and firing tests.

(4) Ground firing tests.

26.7.1 TEST PROGRESS

26.7.1.1 Missile Package Door Tests

Preparations for tests to investigate the operation of the roller-type doors closing about the missile body were begun in October 1957. The object of these tests was to check the mechanical function of the doors and link mechanism at the design speeds of door operation. Preliminary operation of the door disclosed that the diaphragm supporting the actuating jack required further stiffening. This was done, and testing continued. The door operating torque tube was calibrated to indicate the operating jack loads. The cycling tests were satisfactorily completed at normal room temperatures and at high temperatures. The specimen is to be dismantled and inspected, prior to further testing.

26.7.1.2 Missile Package Seal Test

The original seal developed to close the space between the forward edge of the weapon pack and the adjacent airframe structure consists of a strip of stainless steel fingers covered with Neoprene rubber and then Teflon sheet. This sandwich construction bridges the gap between the forward edge of the weapon pack and the airframe, with one end attached to the airframe, and the other end resting on an aluminum block in the weapon pack. 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.

It was found that the fingers overlapped each other on the curved portions of the bulkhead. To eliminate this, the seal was redesigned, with curved fingers. Tests were conducted on this form of strip but the results were unsatisfactory. Further tests on the original form of straight fingered strip but with tapered gaps between the fingers showed more acceptable characteristics. This concluded these tests.

26.7.1.3 Missile Lowering and Retraction System

The initial tests of the missile lowering and retraction system were conducted using parts which were representative of aircraft parts (Ref. previous Quarterly Technical Report). Since aircraft parts are becoming available, it is planned to repeat these tests, using parts designed for the aircraft installation.

A further test rig has been under investigation for testing the missile lowering and retraction system under the influence of dynamic loads, and under firing conditions. This rig will be designed when loads requirements are available.

26.7.1.4 Mechanical Wear Tests

In an attempt to reduce the wear of the magnesium side load reacting link of the missile lowering mechanism, a series of tests has been conducted in order to obtain a suitable liner. This liner will reduce the coefficient of friction and wear factor. The side load reacting link acts in a similar manner to a cylinder and piston, with two sets of rings and a side force acting on the piston rod.

Initial tests, using Teflon tape as the liner material, were unsatisfactory. After 102 cycles of operation the liner wore through to the metal surface, indicating that the liner material is not suitable for this purpose. Further tests have been conducted using a strip of .032 in thick Polyamide (nylon) plastic as the liner material. This is cemented in place using a metalbond process. With the liner cemented into position, the cylinder bore was machined out to the correct diameter. The surface finish of the bore was between 80 and 100 micro-inches. The test specimen was subjected to cycling tests at temperatures ranging from 0°F to 130°F with varying humidity, and with and without sideloads. On the completion of 1000 cycles, the specimen was examined and found serviceable. Although a slight indication of wear was evident on the nylon liner, no measurable increase in cylinder bore was detected.

It is considered that the results of these tests were satisfactory and Polyamide (nylon) plastic will be used as the liner material for the magnesium cylinders in the missile lowering mechanism.

26.7.1.5 Shear Pin Test

It is necessary to determine the shear strength of the shear pins used to retain the Sparrow missile in its launching position before firing.

Twenty-four pins of the same diameter have been tested. Twelve pins were loaded to their approximate yield point. The load was then reduced to zero and the pins were removed and examined for distortion. The pins were replaced and reloaded to their ultimate shear strength. The remaining twelve pins were loaded to their ultimate shear strength without removing the load at the yield point. The results of the two tests were compared to determine if any reduction in ultimate shear strength had occurred. The ultimate shear strength of each group of pins varied between 4300 lbs. and 4600 lbs. Neither form of loading showed any marked difference in ultimate shear strength. A shear force of 4500 lbs. was equivalent to approximately 10 1/2 g acting on the pin. This force is considered excessive and further tests are planned to produce a pin with a yield point in the region of 2600 lbs., and with an ultimate shear strength equivalent to between 9 and 9 1/2 g.

3074-105-1

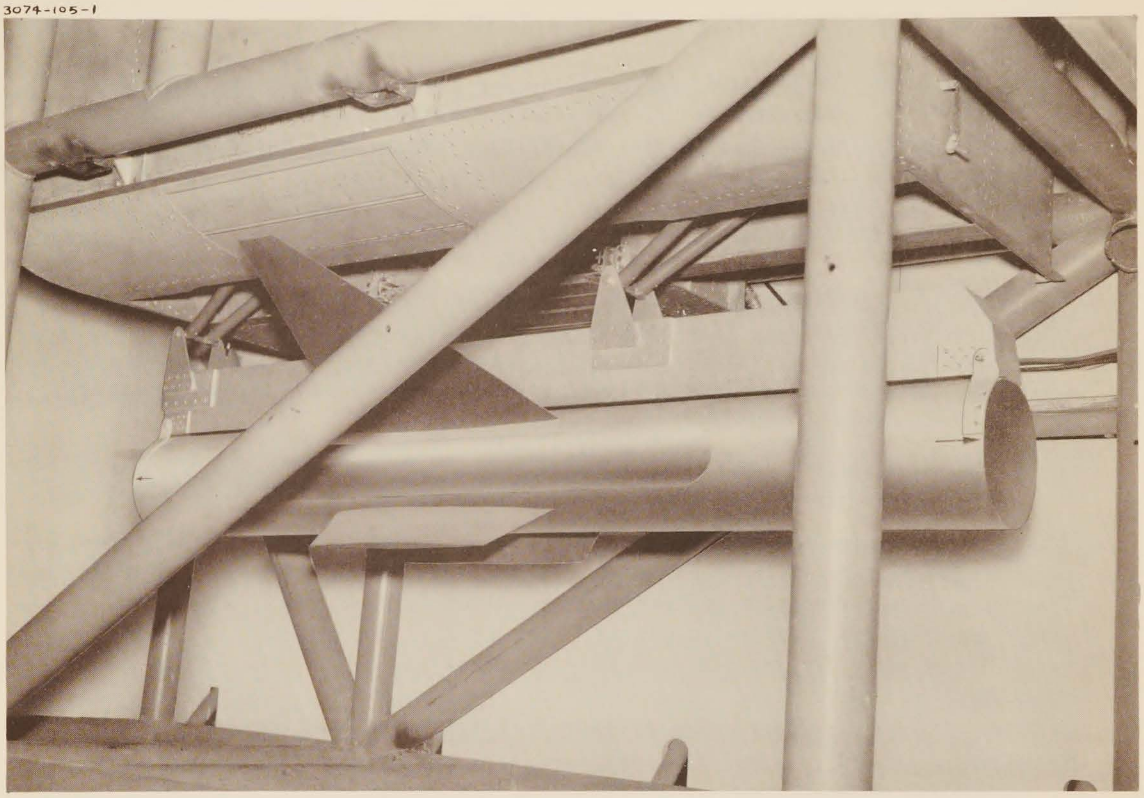


FIG. 59 MISSILE DOORS TEST RIG - MISSILE EXTENDED

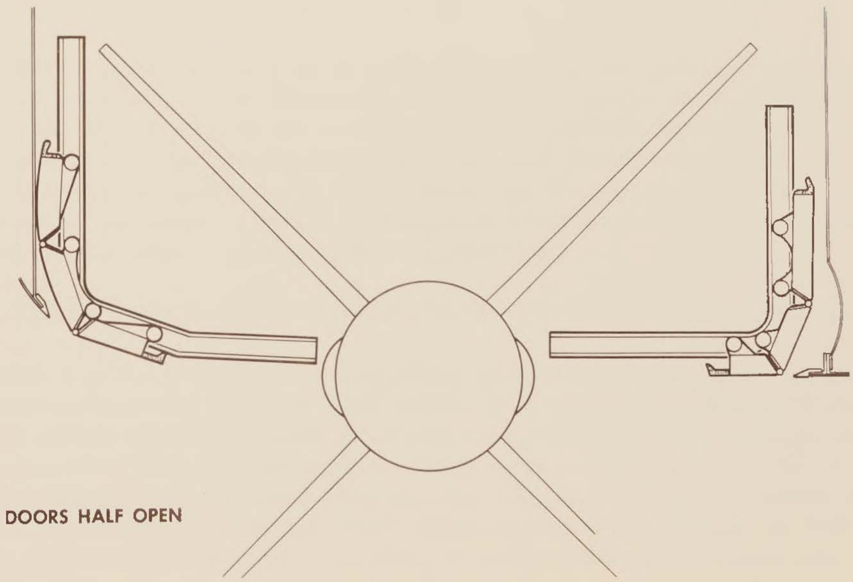


FIG. 60 MISSILE WING DOORS

26.7.1.6 Simulated Firing of Missile Tests

Tests are being performed to determine the wear effects on the missile launcher rails. Preliminary firing with a simulated missile has been undertaken to determine the functioning of the rig, but no data has been recorded as yet.

26.7.1.7 Explosive Charge - Missile Jettison

Tests will be conducted to develop a Canadian-manufactured explosive charge for missile jettisoning. It is desired that these charges be less sensitive to temperature and static electricity than the existing U.S.-manufactured charges. No contractor has yet been decided upon to develop the proposed Canadian-developed explosive charge.

26.8 PRE-INSTALLATION TESTING OF BOUGHT-OUT EQUIPMENT

Bought-out equipment is subjected to functional tests before it is installed in the aircraft, or as part of a test rig. Components subject to pre-installation tests are fuel, hydraulic and pneumatic valves and switches, electrical switches, servos, hydraulic pumps, fuel pumps, thermostats and temperature controllers etc. During the past three months pre-installation tests have been conducted on thirty-eight items of bought-out equipment. Thirty-three items proved acceptable, four received qualified approval, and one item was rejected.

27.0 FLIGHT TESTING

27.1 SUMMARY OF ARROW FLIGHT TEST ACTIVITY

The final selection and evaluation of airborne and ground instrumentation for aircraft 25201 through 25203 has been completed. The aircraft installation for phase 1 has been designed and is now being installed in the aircraft and in the instrumentation packs.

Ground station arrangements are progressing. Equipment is being installed and tested for the reception of radio transmitted data from the test aircraft. Testing of both ground and airborne instrumentation circuits, in breadboard form, is continuing.

The flight testing of airborne recording and telemetry facilities is continuing. CF-100 aircraft have been used as the test vehicles.

The testing of a simulated ARROW flying control system in a CF-100 aircraft was partially completed when problems were encountered involving severe stick oscillations in the elevator control system. A similar condition was encountered while conducting flying control system tests in the mechanical test rig. A complete investigation of this problem is being conducted. This phenomenon involves only the control stick steering mode using stick force transducers and the command channel of the damping system. It is not intended to operate in this mode on early flights of 25201. The early flight testing will consequently not be dependent on the solution of this problem.

27.2 FLIGHT TEST INSTRUMENTATION

Three ARROW 1 aircraft are to be fully instrumented for airframe development. Each aircraft will be equipped with an instrument pack designed to carry both the airborne magnetic recording tape and the telemetering systems. With the addition of the ground telemetry and data processing units, facilities for the automatic collection, storage, recovery, correction and presentation of test data are provided. Flight testing of the airborne instrumentation equipment is continuing, using a CF-100 aircraft.

27.2.1 TELECOMMUNICATION AND NAVIGATION SYSTEM TESTS

A preliminary assessment was made of the proposed method for evaluating the radiation pattern of ARROW UHF and L band antennas. Measurements were made of the signal strength received at a mobile ground station from a CF-100 aircraft flying on different headings over a fixed point about 30 miles away. The height of the ground antenna was adjusted to give a region of constant signal strength over the fixed check point at an altitude of 4000 ft., (the height selected for the test) enabling a comparison to be made between the results obtained and radiation patterns obtained by Sinclair Radio Laboratories using a 1/10 scale model of a CF-100 and 10 times the transmitted frequency. If reasonable agreement can be obtained it is intended to use this method for full scale checking of similar radiation patterns obtained from ARROW models.

27.2.2 STRAIN GAUGE INSTRUMENTATION IN ARROW AIRCRAFT

In-flight structural loads are to be measured by means of strain gauges. The strain gauge installation is complete in aircraft 25201 although a complete inspection of the wiring is still required before first flight. The work of installing strain gauges in subsequent test aircraft is progressing with the manufacture of the aircraft.

27.2.3 DATATAPE DEVELOPMENT FOR THE ARROW

Pre-installation tests on the compound modulation (CM) amplifiers required for the Datatape development system revealed that 15 amplifiers were unserviceable. These units have been repaired and modifications have been incorporated in all CM units for connection to the CM/telemetry signal conditioners. Initial tests made on Datatape equipment installed in a CF-100 were unacceptable, as pulse width modulation (PWM) signals were badly distorted and the records obtained were unintelligible. Investigations showed that the signal from the tape had poor pulse characteristics. By increasing the record current above the level recommended by the equipment manufacturer, the fault was eliminated. Complete waveform records are being taken throughout the system, to assist in future fault finding.

Initially, the ASCOP commutators became unserviceable after running only one hour. The commutators also caused considerable electrical interference which was evident as distortion on the recorded signal. The commutator contacts have been lapped to a mirror finish, and lubricated with molybdenum sulphide. Since this treatment, one test commutator has run 16 1/2 hours with signal distortion from 1/2 to 1% of the full scale signal. This is regarded as a satisfactory degree of accuracy.

27.2.4 AIRBORNE TELEMETRY DEVELOPMENT ON CF-100 AIRCRAFT

When the 50 watt telemetry transmitter was first used in conjunction with PWM/FM on the CF-100 aircraft the RF energy was found to interfere with the operation of the low level Ascop equipment. This problem was eliminated by the repositioning of the telemetry antenna on the tail cone of the aircraft. The system was then flight tested satisfactorily.

Some signal flutter was experienced on the ground and in flight. This was caused by the commutator. The amplifier and keyer controls were readjusted and the signal flutter was reduced to 2% of the full scale signal. Further improvements have been made, as described in the section above on Datatape development for the ARROW.

27.2.5 GROUND STATION TELEMETRY ARRANGEMENTS

A seven-turn helical antenna has been installed on the roof of the Flight Test Department hangar. The AN/GRR-7 ground station UHF transmitter-receiver has not yet been received. Until it is delivered and installed the telecommunications test trailer unit is being used. A control unit is being manufactured to operate this system from the flight controller's desk in the operations room.

27.2.6 DESIGN AND ASSEMBLY OF INSTRUMENTATION PACKS FOR ARROW 25201 - 25203

The instrumentation packs for aircraft 25201 and 25202 were received in October 1957, and the installation of components and wiring proceeded immediately. Some equipment had to be removed to permit the installation of air vents in the packs. This equipment has now been reinstalled. Assembly of the second pack continues, and proving of the electrical circuits is in progress.

The fuel contents measuring system which was used on the engine ground runs is now being modified for incorporation into the instrument packs.

27.2.7 DATA PROCESSING UNIT ASSEMBLY

The range-time decoder unit was completed. Manufacture of a unit to provide continuous range-time information is in progress.

27.3 GROUND ENGINE RUNNING PROGRAM

The engine ground running program began on 4 December 1957, but was temporarily suspended on 21 December, so that low-speed taxi trials could be performed. Testing began with engine runs of short duration, (approximately 3 3/4 minutes) during which starting and idling characteristics were assessed, and the engines were run to 70% rated power. Jet pipe temperatures and temperatures at critical structural points of the airframe were recorded and found satisfactory. Malfunctions of heat exchanger gill position indicator lights and bleed valves were remedied. Engine speeds were increased as testing progressed, until full military power with afterburner was attained.

Further engine running embodied tests of the aircraft systems which could be performed at this stage. These tests are outlined in the system test section. A temperature distribution survey was made of the airframe structure around the engine and jet pipe. The structure was painted with Thermindex paint, and colour changes of the paint could be interpreted as temperature changes in the particular area according to a colour coded reference chart. No excessive temperatures were noted.

A survey was made of the sound levels produced by the engines in the vicinity of the aircraft. They were satisfactory at all engines speeds and with after-burners in operation. Tail cone movements were measured under various engine running conditions. It was found that the tie rods which hold the aircraft in a steady position permit very little aircraft motion. Failure of a heat exchanger unit in the hydraulic system occurred during engine running. The heat exchanger was replaced and testing was resumed. Further engine running tests will be made prior to flight, in order to establish the power rating available from each engine.

27.3.1 PRE-FLIGHT LOW SPEED TAXI TESTS

Engine power conditions at which the aircraft started to 'creep' were measured with brakes applied by foot pedals and with handbrake. The aircraft was then allowed to move forward. As speed increased, the general behaviour of the aircraft was studied as it moved ahead in a straight line. The effect of the wheel brakes was noted as they were applied at various speeds. Turns of varying severity were then performed to assess the ground handling capabilities of the aircraft.

The maximum speed attained during low speed taxi tests was 100 knots. The aircraft steering was tested, using the nosewheel steering system, with a combination of braking and flying control assistance. A preliminary assessment of the flying control system was made during the low speed taxi tests. The drag chute was operated several times, and its effectiveness assessed.

28.0 SPECIFICATIONS ISSUED

28.1 MODEL SPECIFICATIONS

No model specifications have been issued during the period covered by this report.

28.2 AVROCAN SPECIFICATIONS

To date approximately 380 AVROCAN equipment specifications have been issued for the ARROW. An index of these specifications, dated 30 November 1957, has been issued to the RCAF.

In addition, to the above, the following indexes of standards and specifications have been issued to the RCAF during the period covered by this Quarterly Report.

(1) GEN/STDS/5 List of ARROW Proprietary Hardware Dec. 1957

(2) GEN/STDS/6 Index of Company Standard (CS) Hardware Oct. 1957

(3) GEN/STDS/7 Specifications and Standards, use of Nov. 1957

(4) GEN/STDS/10 Index of ARROW Material Specifications Oct. 1957

29.0 REPORTS ISSUED

29.1 PRELIMINARY DESIGN PROPOSAL

No preliminary design proposals were completed during the period covered by this report.

29.2 WIND TUNNEL DATA

70/W. TUNN/3	Missile Cross Plots CAL W.T. Tests	Oct. 1957

29.3 WEIGHT AND BALANCE REPORTS

Weight and Balance 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.

29.4 PERFORMANCE REPORTS

Report #	Description	Issued
11	Periodic Performance Report (ARROW 1)	Oct. 1957
12	" " " (ARROW 2)	Nov. 1957
72/POWER/3	An Engine and Afterburner Performance Indicator for Iroquois Engine	Nov. 1957
71/PERF/2	Programming for Performance Data from ARROW 1 Flight Tests	Dec. 1957
71/PERF/5	J75 Engine Trimming for ARROW 1 Flight Tests	Dec. 1957
70/PERF/1 APP2	Engine thrust from ARROW 1 Flight Tests	Dec. 1957

29.5 STRESS ANALYSIS REPORTS

ARROW 1	Iss.	Description	Issued
7/0500/7	3	Main Aircraft Static Test	Oct. 1957
7/0500/29	2	Thermal Analysis	Nov. 1957
7/0500/32	1	Thermal Distribution using I.B.M. 704	Dec. 1957
7/0500/39	1	Thermal Analysis on a Specimen of the Duct Bay	Dec. 1957
7/0510/9	2	Centre Fuselage Analysis	Dec. 1957
7/0553/3	5	Canopy Arch.	Dec. 1957
7/0554/6	2	Bottom Longeron - Centre Fuselage	Oct. 1957
7/0555/12	1	Redesign Ramp Structure	Dec. 1957
7/0562/20	1	Wing over Fuselage Stresses and Deflections with J75 Engines	Dec. 1957

ARROW 1	Iss.	Description	Issued
7/0562/21	1	Auxiliary Spar	Dec. 1957
7/0562/41	1	Outboard Engine Mounting J75	Dec. 1957
7/0562/43	1	Rib 4 - Main Torque Box	Dec. 1957
*7/0562/67	1	Fuselage Side Rib and Front Spar to Auxiliary Spar	Dec. 1957
7/0583/7	1	Rudder Controls in Fin-Load Analysis, Rudder Jack P/U Bracket	Dec. 1957
7/0583/9	1	Plastic Fin Tip and Fairing	Dec. 1957
7/0583/12	2	Detail Stressing of Fin Hinge and Adjacent Structure	Dec. 1957
*7/0583/18	1	Fin and Rudder Loads for Design Cases	Dec. 1957
7/0584/3	1	Rudder - Detail Stressing Hinge Ribs and Fittings	Dec. 1957
7/0584/4	1	Rudder - Detail Stressing Skins Spars and Air Load Ribs	Dec. 1957
7/0584/6	1	Analysis and Stressing of Buzz-Damper Installation	Dec. 1957
7/0590/8	1	Main Undercarriage - Forward Pivot Fitting	Dec. 1957

*These reports were erronously listed as issued in the Arrow Quarterly Technical Report for the period ending September 30, 1957

ARROW 2			
7/0554/208	1	Sparrow Pack Pick-up Structure	Dec. 1957
7/0554/212	1	Load and Stress Distributions from General Aircraft Analysis	Dec. 1957
7/0556/200	1	External Drop Tank Loads	Dec. 1957

29.6 STRUCTURAL STRENGTH TESTS

No formal structural strength test reports have been issued during the period covered by this report.

29.7 ELECTRICAL LOAD ANALYSIS

No electrical load analysis reports were issued during the period covered by this report.

29.8 WEAPON SYSTEM ANALYSIS

No reports available.

29.9 EQUIPMENT DESIGN (Airborne and Ground Equipment, Maintenance, Reliability)

Report #	Description	Issued
Status Report	AVRO ARROW Ground Handling Equipment	Nov. 1957
70/GEQ/3	AVRO ARROW Ground Handling Equipment Use of Giraffe Model 1G. 40	Nov. 1957
71/GEQ/4	Servicing Unit, Multi-Purpose, Aircraft Ground AVROCAN Specification No. E-452	Nov. 1957
72/GEQ/1	ARROW 2 Mobile Ground Power Equipment	Nov. 1957
72/GEQ/1	Methods of Reloading the Armament. Preliminary Time Study	Nov. 1957
70/AIREQ92/I	Investigation of Elimination of Main Landing Gear Twisting Mechanism	Nov. 1957
72/AIREQ13/2	ASTRA Minus Equipment (ARROW 2) (Partial ASTRA)	Nov. 1957
72/AIREQ13/4	Comparison of AVROCAN SPEC E-4II Iss. I "Radome" and Convair Report 2N-326 "Modification to AVROCAN Spec. E-4II for CF-105 Radome".	Nov. 1957
71/REL00/1	Qualification Status ARROW 1 (1st Aircraft)	Dec. 1957
71/RELI9/6	ARROW Charging valve and pressure gauge Utility and Flying Controls Hydraulics	Dec. 1957
70/MAINT00/2	Aircraft Towing	Dec. 1957
7I/MAINT00/3	Maintenance Instructions - Structure Lubrication	Oct. 1957
71/MAINT11/1	Maintenance Instructions Fire Detection and Protection Electrics	Oct. 1957
7I/MAINT11/2	Maintenance Instructions - Lighting Electrics	Oct. 1957
71/MAINT11/3	Maintenance Instructions - Master Warning System - Electrics	Nov. 1957
7I/MAINTI1/4	Maintenance Instructions - Landing Gear Electrics	Dec. 1957
71/MAINT1I/5	Maintenance Instructions - Canopy Actuation Electrics	Oct. 1957
71/MAINT11/9	Maintenance Instructions Engine Services - Electrics	Dec. 1957
71/MAINT1I/10	Maintenance Instructions - Power Supply - Electrics	Nov. 1957
*7I/MAINT1I/13	Maintenance Instructions - Windscreen and Canopy De-Icing	Oct. 1957
7I/MAINT12/2	Maintenance Instructions - Instruments Pressure Ratio Indication	Nov. 1957

Report #	Description	Issued
71/MAINT12/3	Maintenance Instructions - Instruments Fuel Contents Indication	Nov. 1957
71/MAINT12/4	Maintenance Instructions - Instruments Turbine Exhaust Temperature Indication	Dec. 1957
71/MAINT13/6	Maintenance Instructions - Instruments UHF Communication System AN/ARC-34	Dec. 1957
71/MAINT16/1	Maintenance Instructions - Fuel System - Bladder Cell	Oct. 1957
71/MAINT16/2	Maintenance Instructions - Fuel System Pressurization Testing	Nov. 1957
71/MAINT16/5	Maintenance Instructions - Integral Tank Sealing	Oct. 1957
71/MAINT18/1	Maintenance Instructions - Low Pressure Pneumatics	Dec. 1957
71/MAINT21/2	Maintenance Instructions - Oxygen - Ground Servicing Equipment	Nov. 1957
71/MAINT23/1	Maintenance Instructions - Fire - Extinguishing System	Dec. 1957
71/MAINT25/1	ARROW 1 - J75 Engine Installation and Removal	Nov. 1957
71/MAINT25/2	ARROW 1 - J75 Engine Pre-Run Checks	Nov. 1957
71/MAINT25/3	ARROW I - J75 Engine Running	Nov. 1957
71/MAINT33/1	Nose Wheel Steering - Mechanical	Dec. 1957
71/MAINT92/1	Maintenance Instructions - Main Landing Gear	Oct. 1957

*This report was listed incorrectly in the previous ARROW Quarterly Report as 71/MAINT11/3

The following reports were available but not listed in the previous Quarterly Report.

Report #	Description	Issued
MAINT. 105-15-4	Maintenance Instructions - Flying Controls Mechanical	June 1957
MAINT. 105-15-0	Maintenance Instructions - Electrics Fuel System	June 1957
MAINT. 105-19-7	Maintenance Instructions - Utility Hydraulics Power Circuits	Apr. 1957
MAINT. 105-19-8	Maintenance Instructions - Utility Hydraulics Speed Brakes	Apr. 1957
MAINT. 105-19-9	Maintenance Instructions - Utility Hydraulics Landing Gear	May 1957

Report #	Description	Issued
MAINT. 105-19-10	Maintenance Instructions - Wheel Brakes	Aug. 1957
MAINT. 105-20-3	Maintenance Instructions - Electrics Duct De-Icing	May 1957
MAINT. 105-21-2	Maintenance Instructions - Oxygen - Aircraft System	Feb. 1957
MAINT. 105-22-4	Maintenance Instructions - Air Conditioning System	Apr. 1957
MAINT. 105-32-4	Maintenance Instructions - Flying Controls Hydraulics	Apr. 1957
MAINT. 105-91-3	Maintenance Instructions - Nose Landing Gear	May 1957

29.10 AIRCRAFT GROUND AND FLIGHT TEST

Report #	Description	Issued
70/FAR/4	Control Mass Contribution to Hinge Moment	Nov. 1957
70/FAR/5	Control Surface Hinge Moment Derivatives from Flight Test	Nov. 1957
72/FAR/6	Aircraft 25206 Instrumentation	Dec. 1957
71/FAR/7	Preliminary Program - ARROW First Flight	Dec. 1957

No ground test reports are available.

29.11 FUNCTIONAL TYPE TESTS

Each item of equipment procured to an AVROCAN 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.

29.12 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.

29.13 ASTRA I SYSTEM

No Avro reports available.

29.14 SYSTEMS

Report #	Description	Issued
70/SYSTEM11/73	Operation of Fire Detection System	Nov. 1957
70/SYSTEM13/77	Tests Performed on Humphrey Stick-Force Transducer	Dec. 1957
71/SYSTEM13/54	Production Test Procedures	Oct. 1957
71/SYSTEM13/66	Post Installation Check of AN/AIC-10 in ARROW 1	Dec. 1957
71/SYSTEM13/70	UHF L-Band Antenna work on the ARROW by Sinclair Radio Ltd.	Dec. 1957
71/SYSTEM15/69	Flying Control System Data	Nov. 1957
71/SYSTEM15/82	Investigation into Failure of ARROW Flying Control Hydraulic Pipes on Fatigue Test	Dec. 1957
71/SYSTEM16/53	ARROW Defuelling Tender Requirements	Oct. 1957
71/SYSTEM16/55	ARROW Fuel System Ground Test Air Requirements	Oct. 1957
71/SYSTEM21/32	Schematic Oxygen System	Oct. 1957
71/SYSTEM22/65	Phase 2 Air Conditioning System Test Results	Oct. 1957
71/SYSTEM26/79	ARROW Air Trials Program - Sparrow 2 Weapon Launching System	Dec. 1957
71/SYSTEM32/56	Flying Control Hydraulics, Functional Test Procedure	Oct. 1957
71/SYSTEM32/74	Ram Air Turbine Power Requirements	Nov. 1957
72/SYSTEM11/34	Fire and Overheat Detection Proposals ARROW 2	Oct. 1957
72/SYSTEM13/33	Progress Report for UHF Annular Slot Antenna	Oct. 1957
72/SYSTEM13/59	Monthly Antenna Progress Report	Oct. 1957
72/SYSTEM13/76	Monthly Antenna Progress Report	Dec. 1957
72/SYSTEM24/68	ARROW 2 Ram Air Turbine Installation	Nov. 1957
72/SYSTEM29/45	Accessories Gear Box Oil Cooling System	Oct. 1957
72/SYSTEM29/46	Constant Speed Drive - Integrated Oil System	Oct. 1957
72/SYSTEM32/50	Effect of Failure in ASTRA Scanner Drive Hydraulic System	Oct. 1957

29.15 GENERAL TECHNICAL DESIGN

72/COMP A/1	Dynamic Manoeuvres Inducing Fuel Flow	Oct. 1957
72/COMP A/2	Stability of Unsymmetrically Loaded Aircraft	Oct. 1957
72/COMP A/3	Effect of Unsteady Lift and Moment on Aircraft Response	Oct. 1957
72/COMP A/4	Rolling Pull-out Traces at about n=4g	Oct. 1957

Report #	Description	Issued
72/COMP A/5	Frequency Respone of Damped Aircraft	Oct. 1957
72/COMP A/7	Effect of Missile Lowering on Pitch Axis	Dec. 1957
73/COMP A/6	Thermal Distribution in Insulated Skins	Oct. 1957
70/STAB/2	Available Rudder Angle for 10°/Sec and 30°/Sec Control Application	Oct. 1957
71/STAB/5	Digital Computer Determination of Lateral Derivatives from Oscillatory Flight Test	Nov. 1957
71/STAB/6	Digital Computer Determination of Longitudinal Derivatives from Oscillatory Flight Test	Nov. 1957
72/STAB/8	Dynamic Analysis of Fin Mounted IR Seeker	Oct. 1957
71/STAB/9	Digital Computation of Response using an Approximation to Lateral Damper System	Nov. 1957
71/STAB/10	Digital Computation of Response using an Approximation to Pitch Damper System	Nov. 1957
70/THERMO/8	Radiation Effect on Skin Temperatures	Oct. 1957
70/THERMO/9	Temperature History Throughout Life of the ARROW	Nov. 1957
72/THERMO/10	Estimation of Fatigue Life Due to Flight Missions and Jet Noise	Nov. 1957
71/ELASTICS/1	Influence of Fuel Distribution on Stresses Full Fuel vs No Fuel in Wings	Oct. 1957
71/ELASTICS/2	Influence of Fuel Distribution on Stresses Fuel Sequencing	Oct. 1957
71/AERO-ELAS/2	Ground Resonance Calculations for ARROW 1	Oct. 1957
70/HUFAC/1	Measurement of Delay in Escape from Tandem Crewed Aircraft	Nov. 1957