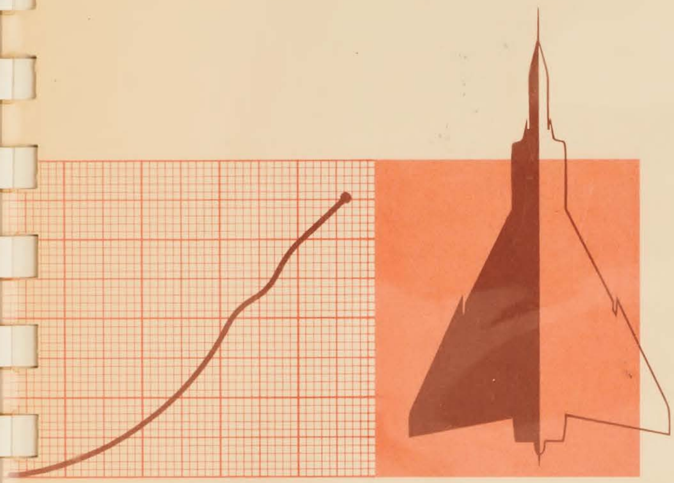

AVRO ARROW

quarterly technical report

FOR THE PERIOD ENDING

March 31 1958

AVRO AIRCRAFT

AVRO AIRCRAFT LIMITED

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ARROW

QUARTERLY TECHNICAL REPORT

70/ENG PUB/6

FOR PERIOD ENDING 31 MARCH 1958

Prepared by: PROJECT MANAGEMENT SERVICES
ENGINEERING DIVISION

AVRO AIRCRAFT LIMITED

MALTON - ONTARIO

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PART 1

GENERAL INFORMATION

1.0 INTRODUCTION

1.1 SCOPE OF QUARTERLY TECHNICAL REPORT

This is the third Quarterly Technical Report on the AVRO ARROW aircraft project. The object of the report is to inform the Canadian Government of technical development of the project during the three months period ending 31 March 1958.

The report presents a description of work performed and the results obtained in the design and development activities of the project; it summarizes technical progress, changes and problems, in all phases of the program during the report period. The text is divided into seven major sections which cover design, testing and development.

1.2 THE ARROW

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 Iss. 3.

There are two versions of the ARROW; the ARROW 1 powered by two Pratt & Whitney J75 turbojets and the ARROW 2 powered by two Orenda Iroquois turbojets. The ARROW 1 is normally considered as an unarmed aircraft used for development purposes, but one aircraft (25203) will be equipped with a weapon pack containing simulated air vehicle (SAV) missiles. The first five aircraft will be ARROW 1 and subsequent aircraft will be ARROW 2. The first fifteen ARROW aircraft have been allocated for contractor's testing and development, leading to the production of fully operational ARROW 2 aircraft which include the Sparrow 2D air-to-air guided missiles and ASTRA I electronic system.

Both ARROW 1 and ARROW 2 have essentially the same basic configuration but the more powerful engines of the ARROW 2 will give it superior performance. The ARROW 2 is designed to operate at altitudes up to 60,000 feet and 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.

1.2.1 ARROW 1 FIRST FLIGHT

The first flight of an ARROW aircraft was made from Malton on 25 March 1958. This flight was made at moderate speeds and altitudes, and no excessive manoeuvres were performed.

1.3 BRIEF DESCRIPTION OF ARROW AIRCRAFT

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 clamshell 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. The steerable nose gear retracts forward into the front fuselage.

The landing gear, wheel brakes, nosewheel steering and speed brakes are actuated by a 4,000 psi utility hydraulic system. A compressed air system is available for emergency lowering of the landing gear. 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 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.

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

1.3.2 ARROW 2

The 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 air-to-air guided missiles, installation of the ASTRA I electronic system, and replacement of the J75 engines with Orenda Iroquois engines. Provision is made for a jettisonable external fuel tank,and the mechanical proportioner type fuel system used for centre of gravity control on the ARROW 1 is replaced by an electrically controlled sequencing system.

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AVRO ARROW

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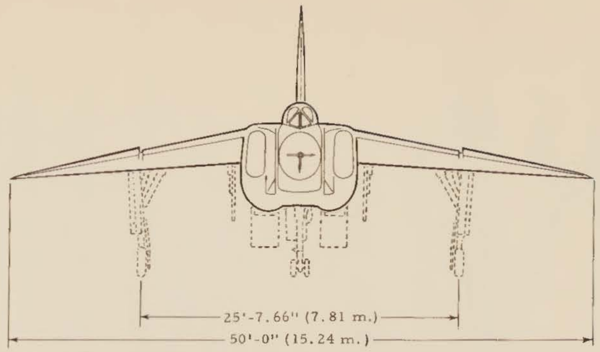
Tank	Location	Capacity Imp.Gal.	Capacity Litres
1	Fuselage	252	1145
2	Fuselage	254	1155
3	Wing	151 each	686
4	Wing	90 each	409
5	Wing (collector)	146 each	664
6	Wing	154 each	700
7	Wing	279 each	1268
8	Wing	173 each	787
9	External Tank (Long Range)	500	2273



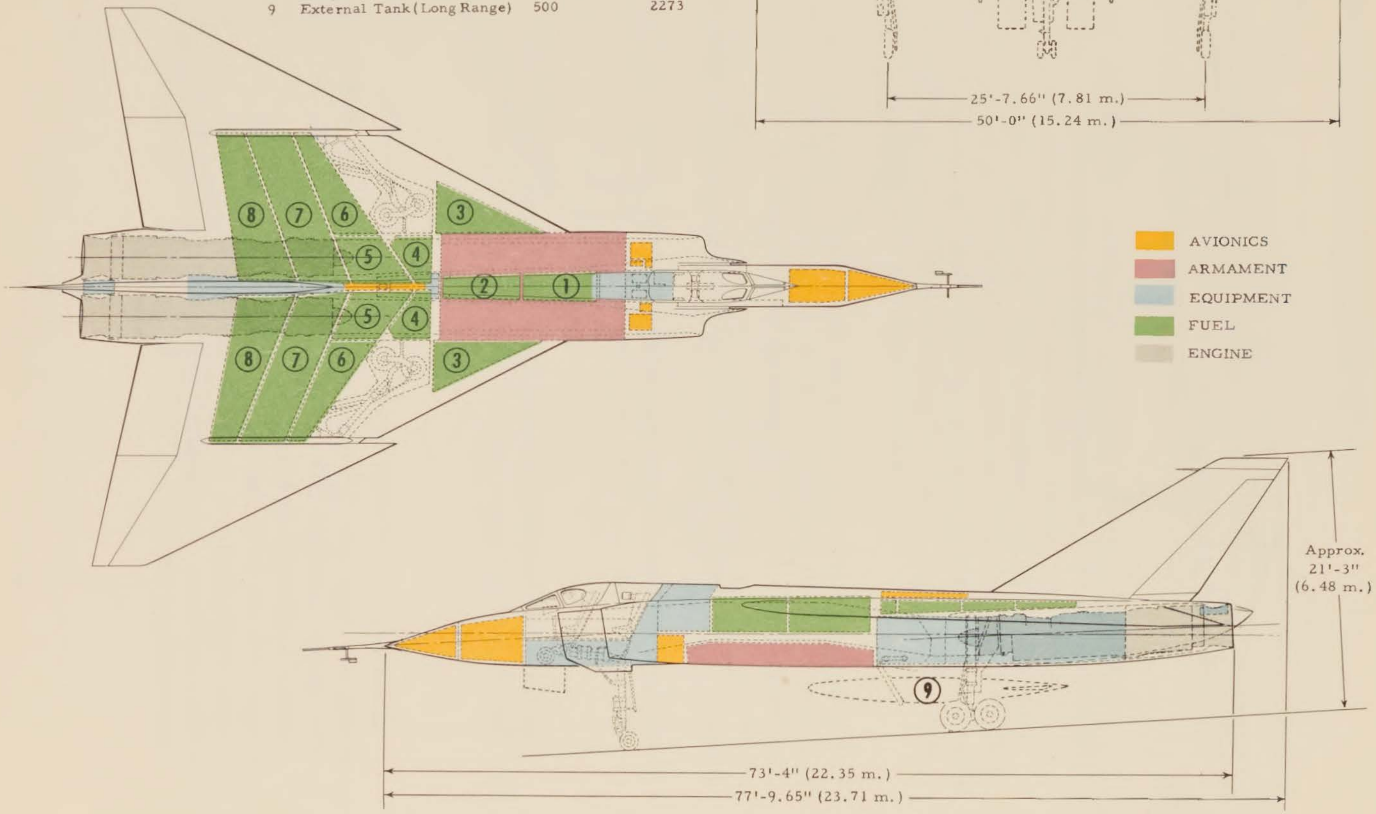


FIG.2 EQUIPMENT ZONES

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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 (See Note 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^{o}	4.98 in
Elevators:	Up	30^{o}	Aft. 6.63 in
	Down	20^{o}	Fwd. 4.37 in
Rudder:	Left	30^{o}	Fwd. 3.28 in
	Right	30^{o}	Aft. 3.03 in
Speed Brakes		60^{o}	-

Note 1. Aircraft 25201, 25202, 25203

Note 2. Aircraft 25204 and subsequent aircraft.

PART 2

TECHNICAL DESIGN

2.0 WEIGHT AND CENTRE OF GRAVITY

2.1 WEIGHTS

The ARROW 1 weight history (Figure 3) and weight summary (Table 1) are applicable to aircraft 25201. Since this aircraft is now actively engaged in a flight testing program, the applicable monthly weight report will be discontinued and will be replaced with periodic weight reports, based on actual weighings of the aircraft.

The ARROW 2 weight history (Figure 4) and weight summary (Table 2) are estimates, based on the fully operational aircraft.

Weight changes noted in the tables are explained by footnotes. Weight accounting, as used in the foot notes, refers to recorded weight changes arising from minor design changes, revised weights obtained from production drawings, and the incorporation of actual weights or vendor quoted weights. Significant accounting changes and weight changes occurring for reasons other than normal weight accounting, as defined above, are explained by more detailed footnotes.

2.2 CENTRE OF GRAVITY

2.2.1 ARROW 1 - AIRCRAFT 25201

Based on the data given in Table 1, 815 pounds of nose ballast is required to restrict centre of gravity (C.G.) travel to the extreme aft position of 31% MAC. Under these conditions, the extreme forward C.G. occurrs at approximately 29.90% MAC.

2.2.2 ARROW 2

The extreme points of C.G. travel, corresponding to the weight summary in Table 2, are as follows:

Extreme forward C.G.	27.00% MAC
Extreme aft C.G.	29.60% MAC

2.3 ACTUAL WEIGHING - AIRCRAFT 25201

Pre-flight and post-flight weighings of aircraft 25201 indicate that first flight weights were approximately as follows:

Engine start-up	64,500 lb
Take-off	62,500 lb

Landing	56, 000 lb
Fuel load	15, 300 lb
Operational weight empty	49, 200 lb

Centre of gravity throughout the flight varied from approximately 28.00% MAC at take-off (landing gear up) to 29.25% on landing (landing gear down).

The derivation of the weights stated above is dependent on fuel loading as determined from readings of the individual tank fuel gauges located in the instrument pack. There is, however, reason to doubt the accuracy of the fuel gauge readings. This is evident from the following:

	Weight - lb Pre-flight	Post-flight
Corrected weighed weight	64, 394	55, 929
Fuel load (from gauge readings)	15, 240	6, 120
Operational weight empty (zero fuel)	49, 154	49, 809

Although the discrepancy between the two figures for the zero fuel condition represents approximately 1.25% of the operational weight empty, which might be considered as acceptable accuracy, the numerical difference is considered to be greater than the known errors in the weighing equipment. The reliability of fuel gauge readings, however, will not be known until an actual weight of the aircraft with no fuel aboard is obtained. The aircraft has not yet been made available for weighing in its drained condition.

A true comparison of operational weight empty as given in Table 1 with a corresponding actual weight cannot be made until an actual dry weight is obtained.

TABLE 1 - STATEMENT OF WEIGHT

ARROW I - AIRCRAFT 25201

	Weight - Pounds			Notes
	Present	Previous	Change	
Structure	18642	18606	+ 36	(a) + 77 (b) - 41
Landing gear	2609	2601	+ 8	(a)
Power plant and services	14414	14392	+ 22	(a) + 12 (c) + 10
Flying controls group	2014	1853	+161	(a) + 11 (d) +150
Fixed and removable equipment	9625	9627	- 2	(a) + 55 (b) - 91 (e) + 34
BASIC WEIGHT	47304	47079	+225	
Useful load (less fuel)	921	921		
Ballast	815	905	- 90	(f)
OPERATIONAL WEIGHT EMPTY	49040	48905	+135	
Maximum internal fuel	19562	19843	-281	(g)
ALL UP WEIGHT - Maximum internal fuel	68602	68748	-146	

Notes: (a) Weight accounting
(b) Emergency ram air turbine system not required for aircraft 25201
(c) Modification of waste fuel drains
(d) Introduction of surge damping accumulators and hydraulic power boosters
(e) Introduction of sequence mechanism for closing nose landing gear door
(f) Ballast adjustment to limit aft C.G. position to 31% MAC
(g) Fuel capacity of fuselage tanks decreased 36 gallons due to an increased air space requirement.

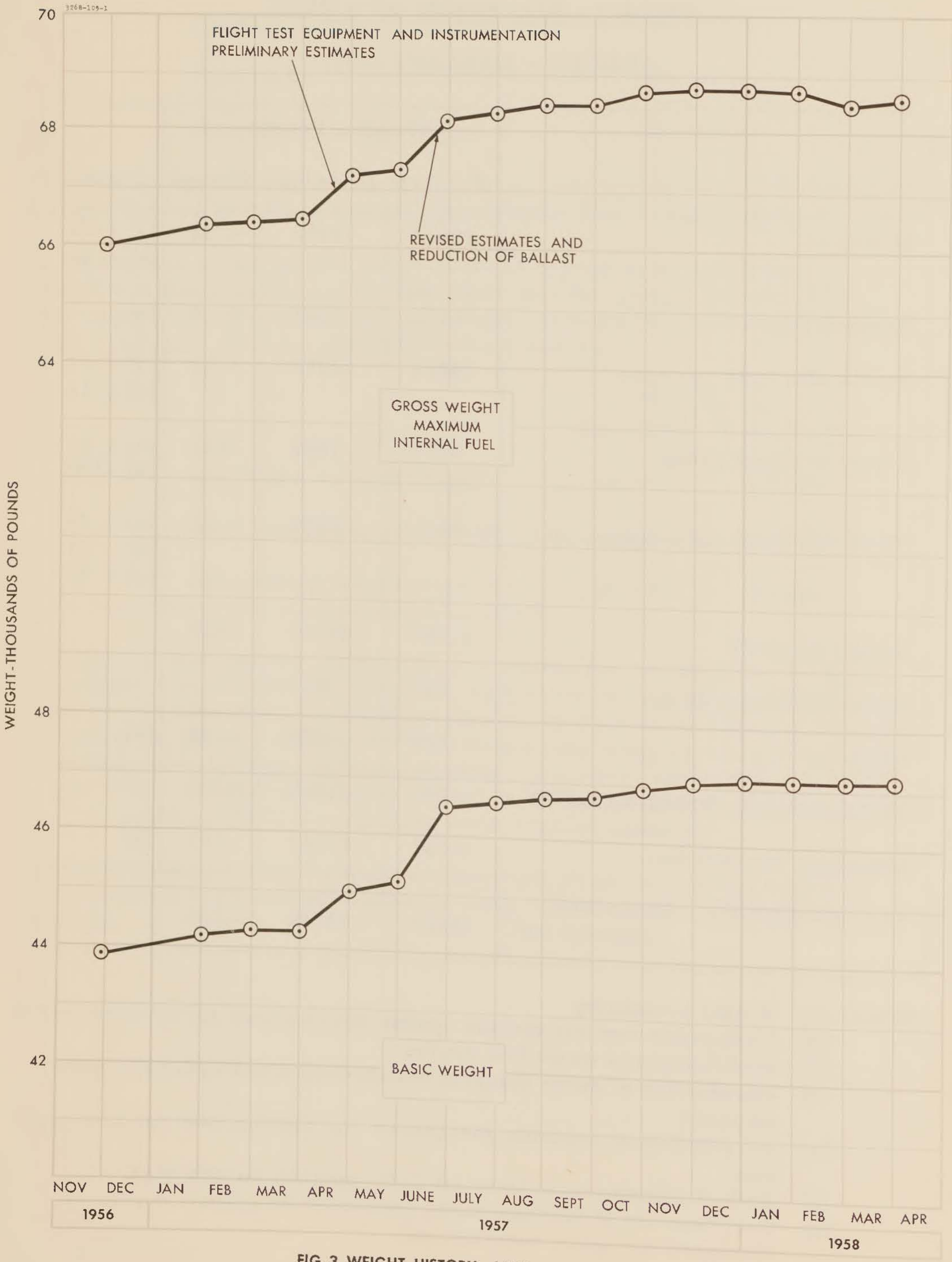

FIG. 3 WEIGHT HISTORY - ARROW 1, AIRCRAFT 25201

TABLE 2 - STATEMENT OF WEIGHT

ARROW 2 - OPERATIONAL AIRCRAFT

	Weight - Pounds			
	Present	Previous	Change	Notes
Structure	19161	19044	+117	(a) +106 (b) + 11
Landing gear	2584	2542	+ 42	(a) + 9 (c) + 33
Power plant and services	10800	10800	-	
Flying controls group	1927	1793	+134	(a) + 8 (d) +126
Fixed and removable equipment	8916	8641	+275	(a) + 90 (e) +121 (f) + 64
BASIC WEIGHT	43388	42820	+568	
Useful load (less fuel)	2799	2789	+ 10	(a) + 10
OPERATIONAL WEIGHT EMPTY	46187	45609	+578	
Normal combat mission fuel	17530	17370	+160	(g) +160
NORMAL COMBAT WEIGHT	63717	62979	+738	

Notes:
- (a) Weight accounting
- (b) Structural provision for fin IR seeker mounting
- (c) Anticipated Dowty weight saving in leg shortening mechanism cannot be achieved
- (d) Allowance for introduction of power boosters and surge dampers based on ARROW 1 installation
- (e) Revised and additional data from RCA on ASTRA radar equipment
- (f) Sparrow pack equipment weight based on production drawing estimates of ARROW 1 trial installation
- (g) Due to increase in operational weight empty

3.0 PERFORMANCE

3.1 ARROW 1

3.1.1 PREPARATION FOR FLIGHT TEST ANALYSIS

Proposed non-dimensional performance curves have been plotted to facilitate the immediate comparison of estimated performance results with the flight test data during the development period of the aircraft. Flight test data is recorded on datatape, which, after editing, is processed by the IBM 704 for automatic analysis. The following non-dimensional graphs have been prepared:

(a) Installed Engine Performance (all plots vs RPM)

1. Thrust
2. Intake pressure recovery
3. Specific fuel consumption
4. Others - academic

(b) Basic Engine Performance (all plots vs RPM)

1. Relative RPM's - N_1 vs N_2
2. Airweight flows
3. Fuel consumption
4. Jet pipe temperature
5. Pressure ratios

(c) Aircraft Performance

1. Drag
2. Rate of climb
3. Level speed
4. Time, fuel and distance to height
5. Take off and landing

Test data on the high speed taxi runs and first take-off and landing is being analyzed and compared with the estimated performance. Results to date indicate fair agreement with previous estimates, the estimated air runs being conservative.

3.1.2 TRIM DRAG CORRECTION TO FLIGHT TEST RESULTS

An investigation has been completed to determine the magnitude of trim drag correction to be applied to flight test results arising from changes in thrust, C of G and load factor. Under normal operating conditions, trim drag correction was found to be small. Reference: 71/PERF/10 - Trim Drag Correction to Flight Results - January 1958.

3.2 ARROW 2

Periodic Performance Report No. 13, January 1958, requested by the RCAF, has been issued. The primary purpose of report No. 13 is to show the effect of change in operational weight empty, on the pertinent performance characteristics. It is not anticipated that the performance parameters used in reports No. 12 and 13 will be changed until flight test data becomes available. However, additional data (e.g. zoom ceilings and means of improving overload range mission) will be added from time to time.

3.3 ARROW 2 OVERLOAD RANGE MISSION

A study has been made of the factors influencing aircraft range (i.e. drag, propulsion and fuel capacity) in order to establish the best means of meeting the 1500 NM range requirement specified in AIR 7-4.

It is proposed that the propulsion system be modified by the insertion of a jettisonable nozzle insert, optimized for the overload range mission. The estimated result of such a modification indicates a range of 1550 NM with the 500 gallon external tank jettisoned and 1490 NM with external tank retained. For flight at M = .92 and 40,000 feet, with the nozzle insert, the fuel flow rate, and therefore the specific range, is improved by 15%. Because of improvement in climb and loiter the overall gain in range is approximately 18%.

The nozzle insert improves thrust in the subsonic speed range, largely by reducing the mass flow and, therefore, the losses in the bypass system. Since the intake duct is not choked in the cruise condition, the reduced bypass flow results in less air being taken aboard. This in turn, results in reduced losses in the intake duct. The reduced bypass and duct losses, will allow a reduced engine RPM for the thrust required at the cruise condition, and therefore, a lower rate of fuel flow.

The nozzle insert is to be secured to the ejector by four explosive bolts connected electrically to the throttle lever. When the throttles are moved to the afterburner-on position, the insert nozzle is ejected (see figure 5).

With this system, the take-off distance will be increased since afterburners cannot be used without jettisoning the nozzle insert. On a standard hot day at sea level, the ground run is estimated to be 3,750 feet; and to clear a 50 foot obstacle a very conservative estimate is 9,500 feet.

Consideration was also given to the refinement of cruise speed and the estimate of the external fuel tank drag. Cruising at M = .90 and 40,000 feet, gives the maximum specific range and contributes about 4% to increased range estimates. A review of the drag coefficient for the external tank, led to a drag reduction which contributes less than 1% to range increase if the tank is jettisoned and about 3% if the tank is carried throughout the mission.

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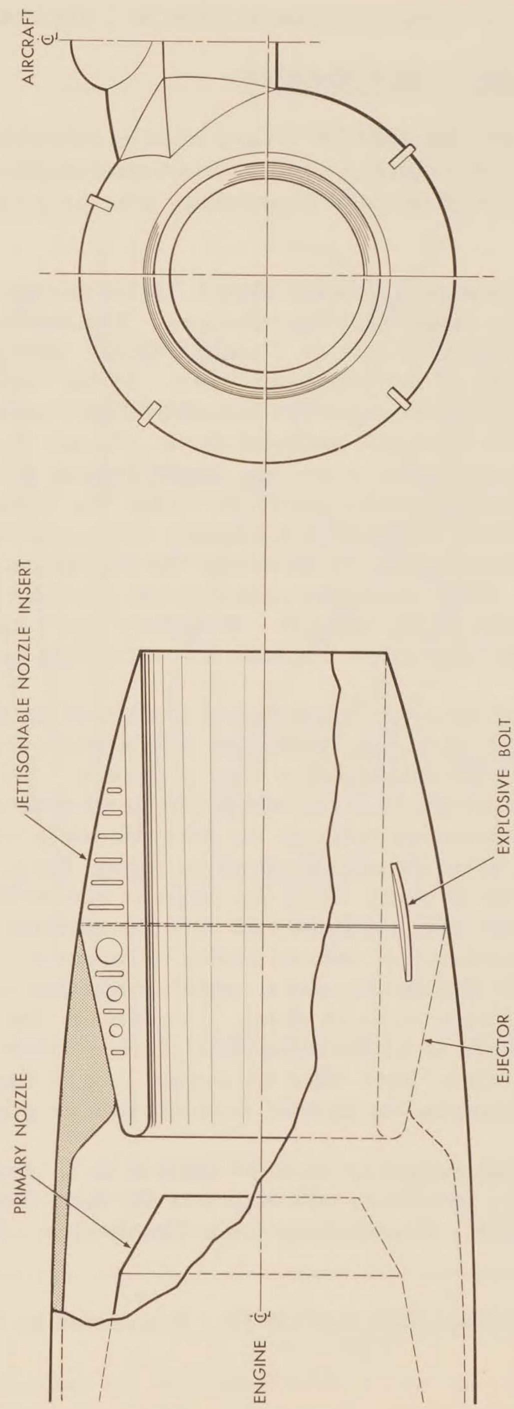

FIG. 5 NOZZLE INSERT

Reference 72/PERF/16 - Improvement of ARROW 2 Overload Range Mission.

3.4 WINDMILLING BUZZ BOUNDARIES

The buzz boundaries for the ARROW 2 have been recalculated due to changes in windmilling mass flow estimates and revised trim angles of attack. Since this subject has not been discussed in previous quarterly reports it is explained as follows:

At free stream Mach numbers greater than 1.76, the normal shock wave structure off the intake ramp inlet lip, changes. The normal shock wave alters to a bifurcated (forked) foot or "lambda wave" which results in a wedge shaped separation of the boundary layer. At the lower inlet mass flows, this lambda wave, and in particular the boundary layer wedge, become increasingly unstable and fluctuate back and forth. At low duct flows, this fluctuation finally reaches the point where the whole system fluctuates violently. This results in a fluctuating static pressure within the intake duct, which is defined as buzz. Because of the high frequency and pressure amplitude resulting from this phenomenon, it threatens the structural integrity of the duct and the engines. Buzz may also result in the periodic oscillation of thrust and even blow-out of the engines. Boundary layer bleed-off on the face of the intake ramp is used as a means of controlling buzz.

The buzz threshold has been ascertained and is defined as the engine mass flow, in subcritical operation, for which the duct static pressure amplitude of fluctuation is 20% of the free stream total pressure. This is synonymous with the final violent movement of the whole shock structure as described. The threshold is transposed to indicate the aircraft Mach numbers and altitudes above which the aircraft should never be flown, for a range of normal accelerations to be free of buzz. With the engines windmilling, the aircraft will not be clear of buzz while altitude and speed conditions are those to the right of the normal acceleration curves shown in figure 6. At the lower altitudes, the aircraft will decelerate more rapidly when one engine cuts, so that time in the buzz danger zone is short. In addition, the engine will take a short time to slow down to windmilling RPM, during which time buzz cannot occur. The pilot may avoid buzz when an engine cuts in flight, by entering some manoeuvre to increase the normal acceleration or g load.

Any increase in aircraft weight or forward shift of C.G. from 47,000 lb. and .31 $\bar{\bar{C}}$ respectively, generally will improve the buzz free range. Reference: 72/INT. AERO/8 - Windmilling Buzz Boundaries - ARROW 2 - March 1958.

3.5 SPECIAL PERFORMANCE REPORTS - Requested by RCAF

Performance estimates for use in SAGE studies, as requested by AAWS for the ARROW SAGE Co-ordinating Committee, have been issued. Reference: 72/PERF/13 - February 1958.

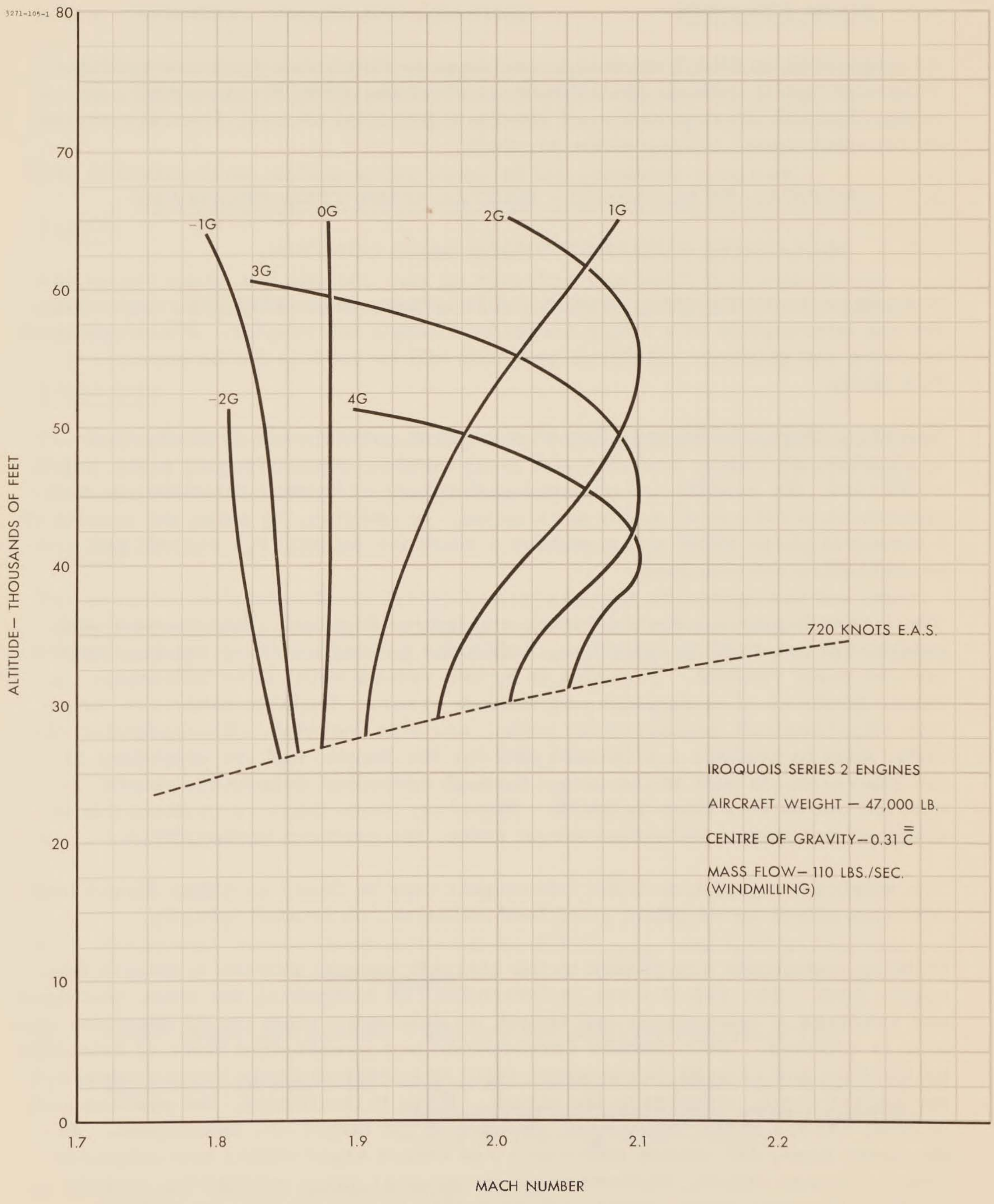

FIG. 6 WINDMILLING BUZZ BOUNDARY

3.6 ZOOM CEILINGS

As requested by RCAF, work is in progress to determine the zoom ceiling. This report will indicate the altitude to which the ARROW may climb, or zoom, through the conversion of kinetic to potential energy, from the normal or constant-speed ceiling of the aircraft.

3.7 TACTICAL EVALUATION - DIGITAL COMPUTER PROGRAMS

3.7.1 MIDCOURSE GUIDANCE UNDER SAGE CONTROL

A study is currently being written and programmed for IBM 704 computation, for the interception of a single target by a single interceptor. Philosophies of tracking and guidance employed by SAGE will be used as far as possible in this study.

So-called Monte Carlo sampling of statistical quantities is used wherever applicable, so that by analysis of a large number of runs for any given initial conditions, the results are obtained in the form of heading distributions and placement errors at AI radar acquisition. In addition, by using the results of a terminal phase study and assuming a value for missile P_k, overall kill probabilities are obtained.

The target flight path may be either straight or dog-leg. Target speed and manoeuvre are input parameters, and in the case of a weave, the leg-lengths may be made random. Tracking is by two radars which have individual characteristics of systematic and random errors. Suitable models to simulate blip/scan and random radar noise, are also included. The reduction of radar data to produce a predicted path for the target, and the smoothing of the interceptor's path to the target through additional data as it arrives, is carried out as it is done by SAGE. However, since there is no necessity to distinguish between individual target paths, the problem is simplified.

The midcourse guidance of the interceptor may be based on either direct lead collision attack or an offset point lead collision, as is used initially.

Heading commands are passed to the aircraft so as to provide a smooth interceptor path. The end of a run is determined by comparing the actual positions and headings of interceptor and target, to determine when the AI detection contour is crossed. This contour corresponds to a preselected value of detection probability, and is used as computer input in a table of range versus interceptor aspect angle, relative to the target. When it is crossed, the position and heading errors of the interceptor relative to the target are determined. At the same time, the course difference and aspect angle values are compared with a success region, determined by a terminal phase evaluation, to show whether the interceptor would have been able to correct its course for a successful attack. These results are stored, and after a number of runs for the same initial conditions, they are analyzed. Thus, the error distributions and probability of success are obtained.

3.7.2 TERMINAL PHASE EVALUATION

Work is proceeding on a digital computer program to simulate interceptor terminal phase attack in three dimensions with automatic steering (A similar study has been made by CARDE without considering the steering mode).

The following is an outline of the scope of the computer program.

Target

Any target speed and altitude may be specified and must then remain constant. The target may manoeuver at any time after detection by the interceptor.

Interceptor

The interceptor continues on a steady manoeuver, usually straight and level flight, after target detection, either for a fixed time or until a given range is reached, when automatic steering is introduced.

Steering

The program may consider a target having a height advantage and the interceptor manoeuvring in either "snap-up" or a climbing turn with missile or rocket armament. The steering command may be based on any of the following courses:

1. Pure pursuit
2. Pure collision
3. Lead Pursuit
4. Lead collision
5. Combined mode. (lead collision for $R > R_{max.}$ and lead pursuit for $R < R_{max.}$)
6. Fixed range lead pursuit - this has the property that $\sin \theta \propto \omega$
7. Modified lead collision - this is mechanized to include the acceleration capability of the interceptor when flying between its limiting Mach number.
8. Counter counter measure mode (CCM)

Methods of Solution

An iterative solution is used, and the time interval may be chosen as desired. The program may be used to initiate a single attack for one set of initial aircraft speeds and positions. Alternatively, for an assumed detection contour, it will attempt interception for one course difference from all aspect angles to find the marginal conversion boundaries, and repeat this for course differences through 360°. If desired, this detection contour can be automatically varied to cover all ranges up to 40 nautical miles. These boundaries also depend on the steering mode employed.

Discussion

The program may be used with heading and placement probabilities derived from a midcourse Automatic Ground Control Interception (AGCI) study to give overall conversion probability.

Alternatively, for a given AI detection contour, a conversion contour, drawn against course difference and aspect angle may be derived from the terminal phase study. Given a value for course difference or aspect angle, this will show the conversion bounds on the other angle and may be written into the midcourse guidance study to give overall conversion probability. By considering several manoeuvres, the conversion boundaries against any target at a certain speed may be found. Reference: Report No. 72/TACTICS/8 - Notes On Digital Computer Programs for Tactical Evaluation of the ARROW.

3.8 EFFECT OF STEERING LOOP GAINS ON INTERCEPTION CAPABILITIES

Analog computer studies have shown that radome errors are causing instability in the ASTRA I system steering loop. It is further shown that a reduction to one quarter of the present gain will completely stabilize the system at all ranges in the tail region. The tail aspect at long range is thought to impose the greatest restriction on the useable gain. A preliminary systems evaluation study shows that the reduction in gain gives an improvement in time in the firing zone for launching Sparrow 2 missiles, for the cases considered. However, the firing error at the mechanized firing time, is greater with the reduced gain. This is unimportant in missile attacks where the armament may be launched at any time, provided that steering error is less than the allowable tolerance. However, rockets may only be released at the mechanized firing point, and the error tolerance is comparatively small. Initial studies with a scheduled gain indicates that it should be possible to cure the stability problem without loss of accuracy for rocket firing.

The steering loop stability at other aspect angles is still under investigation. Reference: Report No. 72/TACTICS/5 - A Preliminary Study of the Effect of the Steering Loop Gains on the Interception Capabilities of the ARROW - Sparrow 2 Weapon System - March 1958.

3.9 EFFECT OF MAXIMUM g STEERING ON INTERCEPTION CAPABILITIES

A brief study has been made to examine the effect of maximum g steering on the interception capabilities of the ARROW aircraft. Manual control of the aircraft has been assumed with normal accelerations corresponding to maximum elevator deflection to correct for any off-course error. A range of co-altitude attacks at 90° and 180° initial course difference was studied. The target in all cases makes a constant speed, 1.8 g turn, away from the interceptor. Lead-pursuit and modified lead-collision trajectories were determined, and in all attacks the target and interceptor initial speed was M = 2.0. The change in interceptor speed was calculated as a function of the applied normal acceleration along the flight path.

The results of the study have shown that the rapid correction of steering errors early in the interception are not desirable, since the high normal accelerations that follow, result in large speed losses. This is particularly true in the lead collision mode where the large lead angles demanded result in sustained periods of flight at high normal accelerations, loss of speed, and consequent increase in the lead angle required. The end result in most of the cases considered is that the interceptor is unable to follow the target as it continues its evasive manoeuvre, and falls away behind the target before reaching the firing zone.

The conditions of high target speed and evasive manoeuvre are severe; however, it is thought that even with more reasonable parameters, qualitatively similar results would be found. Reference: Report No. 72/TACTICS/7 - Effect of Trim Limit Conversion Manoeuvres on Terminal Phase Capabilities of the ARROW when Operating Against Equal Speed Targets - March 1958.

4. 0 STABILITY AND CONTROL

4. 1 DAMPER SYSTEM PROBLEMS

An investigation has been made to eliminate the influence of structural oscillations on the damper system. These oscillations are picked up by accelerometers and transmitted to the flying controls as a false signal. The pertinent parts of the system were tested on the damper simulator and the following parameters were investigated to determine a solution.

1. Stiffening of the accelerometer mounting brackets

2. Relocation of accelerometers closer to the torsional axis of the aircraft.

3. Reduction of accelerometer frequency response characteristics.

4. Reduction of servo loop gain.

5. Reduction of damper loop static gains.

6. Rate limiting of the servo by hydraulic means (i. e. orifices)

7. Electrical filters.

For the initial flight of aircraft 25201, an interim solution was provided using parameters 1, 4 and 5, above. This resulted in some restriction to the flight envelope. In order to provide a permanent solution, and remove the limitations imposed on the damper, the other parameters will be considered. A filter has been designed and is at present undergoing tests before installation in the aircraft. This will eventually be introduced as a design change to the damper system.

4. 2 CALIBRATION OF DAMPER SYSTEM

Damper system checking prior to first flight was to have been conducted using go/no-go ground test equipment. However, considering the system state of development, it became apparent that checking by this method would be inadequate. As a result, ground calibration was undertaken by AVRO for a more complete investigation, during which the following faults were found:

1. The transverse accelerometers brackets were not of sufficient stiffness. This contributed to the magnitude of the structural oscillations (as described above) transmitted to the damper servos.

2. Accelerometers were not mounted level with the aircraft datum. This caused the rudder to be out of trim when the damper was switched on.

3. A prohibitive amount of cross-talk (one system influencing the other) existed between the elevators and rudder. This was the result of the servo loop gains required to attenuate the structural oscillations. By adjustment to the Moog valves, cross-talk was reduced to a temporarily acceptable level.

4. Small random oscillations of the rudder were experienced. This was traced to the Moog valves and corrected through further adjustment.

5. Wiring errors were found which were causing incorrect static gain setting of the damper.

6. Spurious electrical signals to the damper were being picked up, causing incorrect operation. These interferences were traced and corrected.

4.3 SIMULATION TESTS

Simulation tests were performed on the first aircraft to check out the operation of the damper and control systems. For these tests, aerodynamic simulation was accomplished on the analog computer with a presentation to the pilot by means of a special panel in the cockpit. These tests verified the functional operation of the damper for all flight cases expected in the initial flighttest program. The control operations were checked with reduced hydraulic systems operation, simulating engine-out or system-out conditions. The damper switching functions were also checked out at this time. The evaluations were made by AVRO experimental pilots.

4.4 STICK-FORCE MODE "GEAR DOWN" INSTABILITY

Further problems in the stick force transducer loop have been encountered since the previous report. A new Humphrey stick force transducer has been received and is being evaluated on the flight test simulator. In addition to the fixes proposed previously, it was found necessary to design and incorporate additional filters. These will be tested in the near future.

4.5 IR SEEKER

Airloads on the seeker have been calculated. Wind tunnel tests, at the NAE high speed tunnel, are to be made to determine the effect of the seeker on stability and control.

4.6 g LIMITER

Honeywell submitted two proposals to AVRO to overcome the g limiter problems. Of the two systems, a servo accelerometer system was chosen. This system has the advantage of providing a wider g range of normal performance, free of nuisance disengagements while still adequately protecting the aircraft for combat gross weight. The servo accelerometer system also affords a large part of the extra protection required for the 60,000 pounds gross weight configuration.

4.7 MISSILE CAPTIVE FLIGHT

The dynamic analysis and investigation is continuing. Difficulties were encountered in obtaining a satisfactory digital solution. As a result, a new mathematical formulation of the problem was required. Ref: ARROW Quarterly Technical Report Sept. 1957.

4.8 WIND TUNNEL TESTING

4.8.1 AIR DATA NOSE BOOM

The nose boom has been under test at the Cornell Aeronautical Laboratories 8-foot transonic wind tunnel. Isolated testing (i.e. free from influence of aircraft body) has been conducted to check nose boom specifications and to provide data for other testing programs.

The alpha-beta vanes have been calibrated over a range of $\pm 15^{\circ}$ yaw and 0° to $+10^{\circ}$ angle of attack. The vane errors, measured simultaneously, are $\pm 1^{\circ}$ for the beta vane and from 0 to 2° for the alpha vane, measured over the whole range of angles.

The dynamic stability and damping of the vanes has been investigated and the resonance frequencies measured. The results are being analyzed.

The static and total pressure error of the nose boom pitot head has also been determined.

Icing characteristics of the releative wind sensors and the pitot section of the nose boom have been rescheduled for the early part of the next quarter when testing facilities are expected to become available. These tests will be performed in the full scale NAE icing tunnel.

4.8.2 SPIN AND RECOVERY CHARACTERISTICS

The basic program to determine spin and recovery characteristics of the aircraft has been outlined in the previous Quarterly Technical Report. Testing is approximately 75% complete. High altitude tests have not been conducted to date, because of the handling difficulties previously described.

4.8.3 POST STALL GYRATION TESTING

A description of these tests is contained in paragraph 4.10 of the previous Quarterly Technical Report. These tests are to follow completion of basic program, provided that facilities are available by that time.

Difficulties encountered with the first experimental catapult, have caused some delay. This device is being redesigned, and a stronger version is to be built.

5.0 THERMODYNAMICS

5.1 REVIEW OF NOISE PROBLEMS ON ARROW AIRCRAFT

The following is an abreviated discussion of the various noise-producing elements associated with the ARROW and the effect of these elements on personnel, equipment and structure. Several problems are shown to exist and suggestions for their solution are made. The overall problems appear soluble with little resulting penalty to the aircraft. Complete details of noise problems was contained in Report 7/ELASTICS/3 - Review of Noise Problems on ARROW Aircraft - February 1958.

5.1.1 NOISE OUTSIDE THE AIRCRAFT

5.1.1.1 Far Field Jet Noise

By a method of approximation, comparison to the known data for other jet engines, and available Iroquois data, the far field noise pattern for the Iroquois has been estimated. Figure 7 shows the total noise levels in terms of decibels (for definition of decibels see para. 5.5).

5.1.1.2 Near Field Jet Noise

The Iroquois near field noise, based on scaled up Orenda 11-R data, is shown in Figure 8. These total sound pressure levels correspond to maximum engine thrust conditions which produce the greatest amount of noise. The rudder, wing control and stinger are shown to be in sound pressure level fields of 148 to 155 db, 143 db and 165 to 170 db respectively.

5.1.1.3 Boundary Layer Noise

The pressure fluctuations in each eddy of a turbulent boundary layer are small. The total effect of a sheet of these eddies washing a skin panel can, however, induce pronounced affects both in transmitted sound to the interior and in direct effect to the skin structure. Since these pressure fluctuations are heard only as a result of the vibrations they induce in the structure, they are termed pseudo-sound. The motion of a panel under boundary layer excitation differs from panel flutter. The latter phenomenon arises from the interaction of aerodynamic forces and the elastic forces which causes the panel to move as a whole, while the boundary layer causes running ripples, which are independent of the panel motion.

The size and associated pressure of the boundary layer eddies vary with altitude, boundary layer thickness and traversing speed (approx. 0.7 of flight speed). Boundary layer fluctuating pressure levels are plotted against altitude and speed in Figure 9.

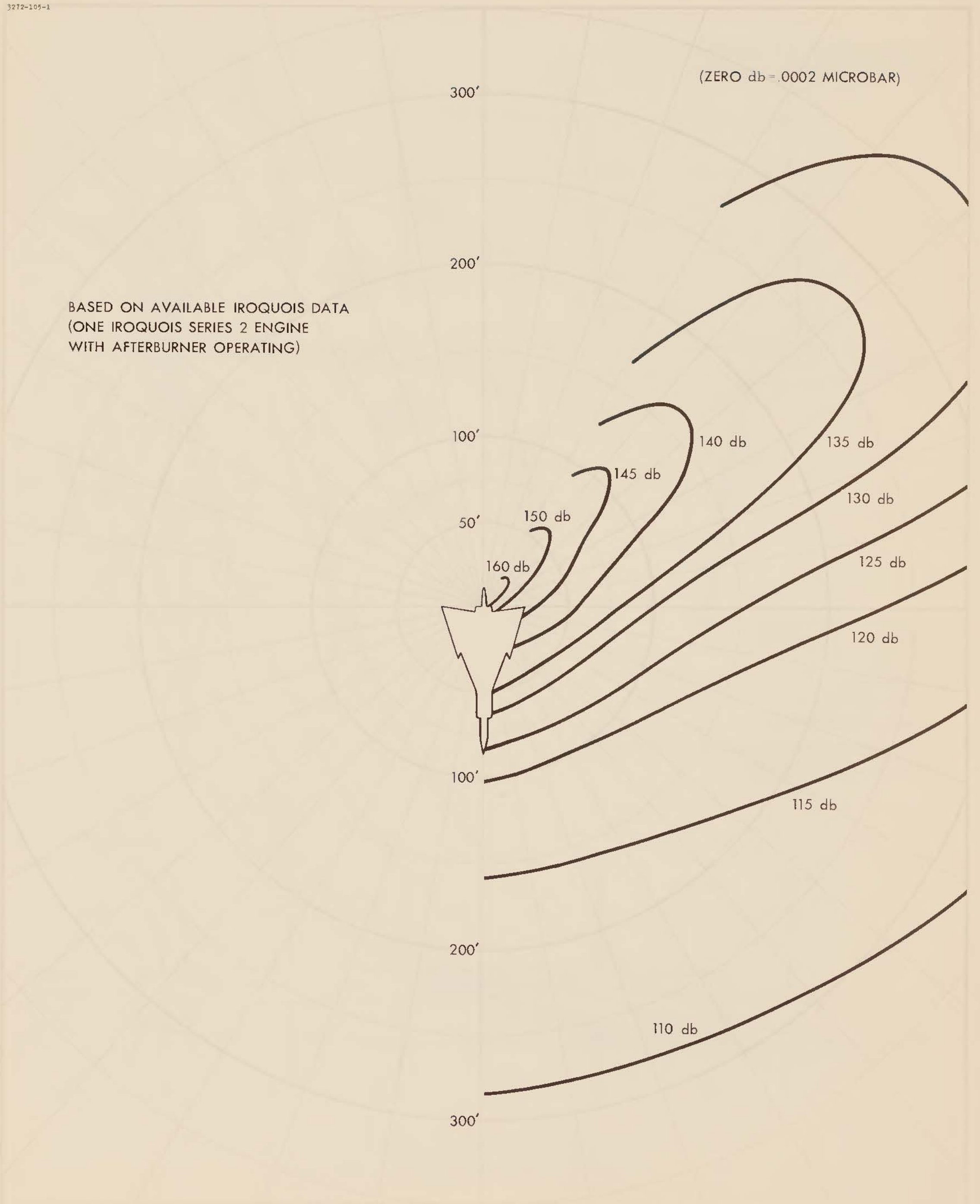

FIG. 7 ESTIMATED FAR FIELD NOISE

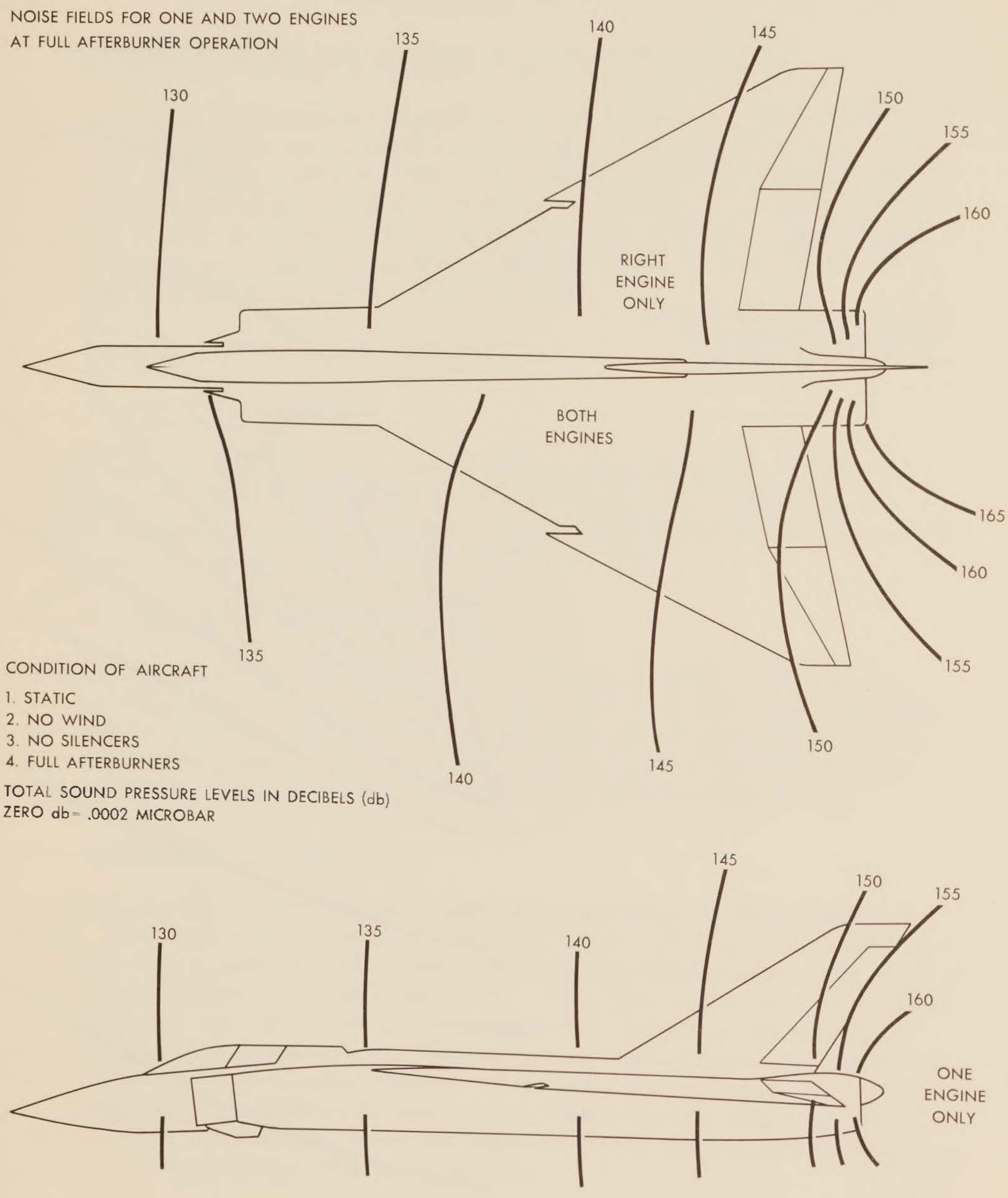

FIG. 8 NEAR FIELD NOISE

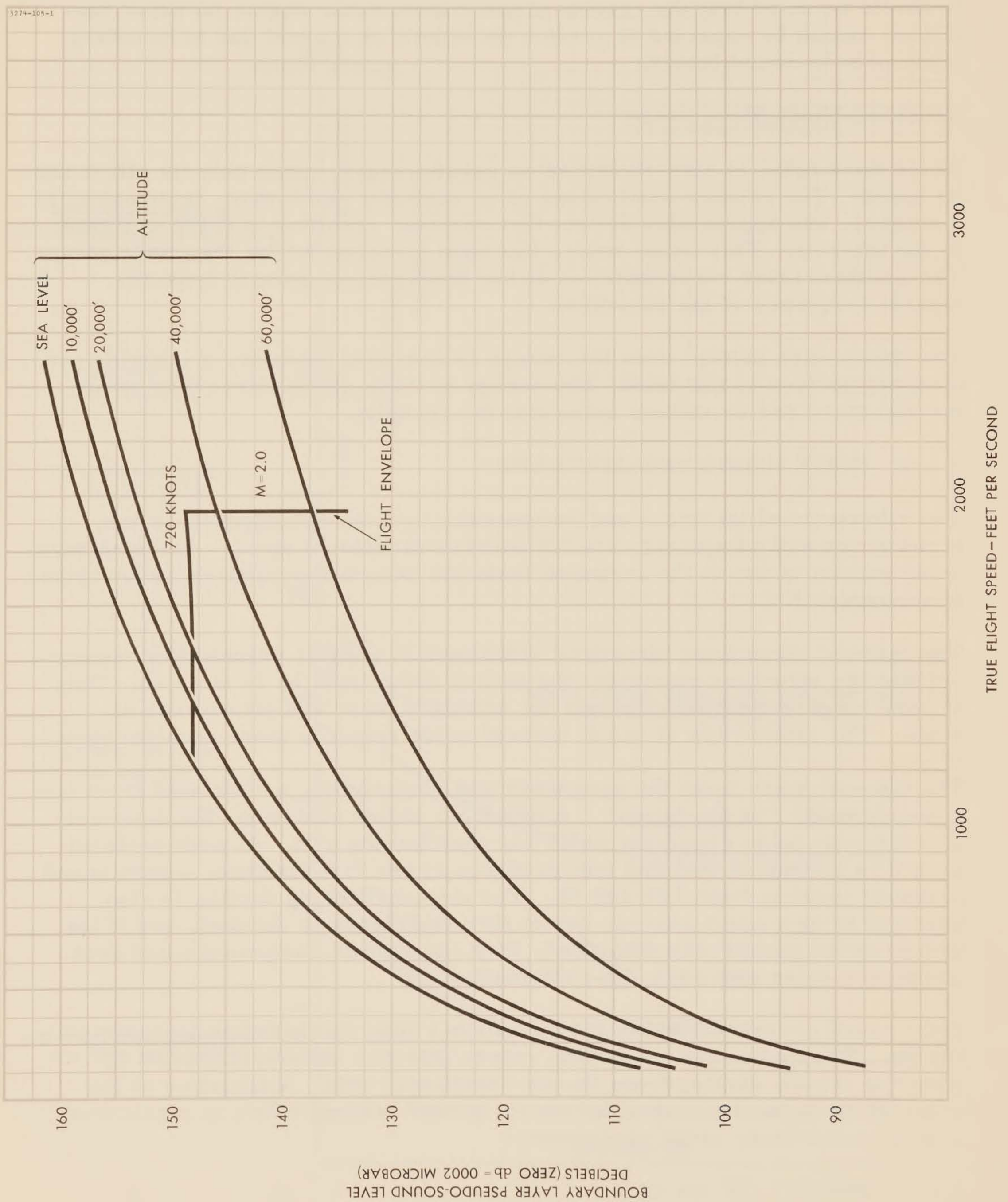

FIG. 9 BOUNDARY LAYER PSEUDO-SOUND LEVEL

It should be noted that the results of panel tests under engine jet noise conditions cannot be expected to predict the effects on a panel due to boundary layer pseudo-sound levels.

5.1.1.4 Shock Wave Noise

Shock wave noise is also classified as pseudo-sound, since only its effect can be heard. The random oscillations of an unsteady shock wave, terminating at a surface, can appreciably deflect that surface, resulting in noise radiation to the interior. This noise source has been investigated for the intake ramps. The greatest noise contribution is at low frequencies, remote from the natural frequency of the structure.

The shock wave pattern over the canopy is unknown; however, a complex and unsteady pattern is indicated. The energy of these shock waves may excite the structure and so radiate noise into the cockpits.

5.1.2 NOISE INSIDE THE AIRCRAFT

The vibrations of structure and components, and the operation of certain accessories and equipment, are the sources of internal noise. The structural noise source is due to the response from the external excitations as described in paragraph 5.1.

The internal noise from structural vibrations is roughly estimated by assuming that the structure attenuates the source noise level by 10 to 20 db. This is very roughly applicable to both the boundary layer and jet sources of noise. The attenuation of shock-induced noise requires more consideration.

The internal noise from the equipment and accessories is estimated from test measurements "on the bench". Aside from engine vibration, the most serious internal noise generator is the air conditioning unit. At peak performance it is estimated that this unit produces at one foot distance, a total sound pressure level of from 130-135 db. The air conditioning ducting is also expected to be a noise source, due to flow turbulence in the duct, and if the outlets to the cockpit are unmuffled, they can radiate compressor and duct turbulence noise with very little attenuation.

Total sound pressure level in the cockpit has been measured at approximately 100 db. Efforts have been made to reduce this to a satisfactory level by silencing the air ducting from the air conditioner unit. Attenuation of the order of 20 db has been obtained during ground testing. This is considered to be a satisfactory sound environment.

5.1.3 EFFECT OF NOISE

Figure 10 shows the estimated results of various sound pressure levels and suggested protection requirements.

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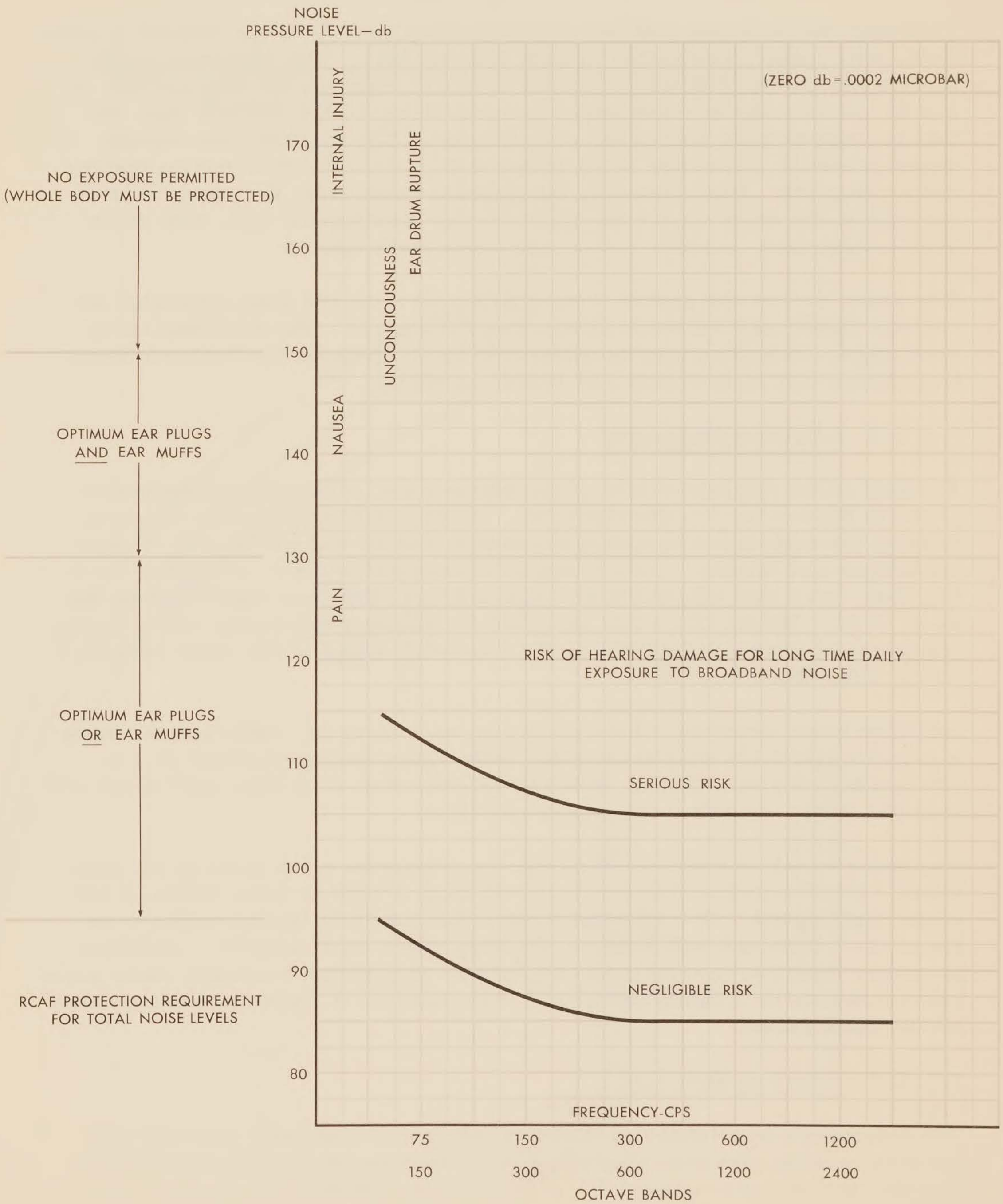

FIG. 10 PHYSIOLOGICAL ASPECTS OF NOISE

5.1.3.1 Effect on Personnel

Personnel directly exposed for relatively long periods to the ARROW's noise field, will require protection against hearing loss. The maximum sound level to which unprotected personnel should be exposed for long periods is 110 db; no one should repeatedly approach closer to the ARROW than 300 feet in front or 1200 feet to the rear quarter. The present rear cockpit sound pressure levels are being reduced to protect the crew during taxiing and at take-off. After take-off, the internal noise from the engine's jet is negligible; the chief source becomes the air conditioning unit, with some contribution from boundary layer and shock wave effects.

The noise level on the ground from a flying aircraft has been estimated for various speeds and altitudes (Reference Figure 11). The resulting noise level inside a building may be attenuated by 40 db under the best conditions (i.e. heavy wall with double and sealed openings).

5.1.3.2 Effect on Equipment

Because of the delicate nature of its components, electronic equipment is prone to failure from acute vibration. Grids and filaments may resonate, causing the tube characteristics to become altered and finally end in physical fatigue and failure. Few design recommendations are available, but it has been found that subminiature vacuum tubes have been unaffected by 140 db at 10 kc. Where electronic equipment contains components whose natural frequency of vibration lies within the frequency range of the noise environment, difficulties may arise.

Isolation from structure vibration is also necessary for subminiature tubes; maximum allowable acceleration for some tubes decreases from 10 g at 100 cps to 1.0 g at 350 cps. Relays and other fine-clearance equipment will also fail under conditions of severe vibration.

Assuming a wall attenuation of 20 db, the maximum noise level in the electronic compartments could reach 130 db from boundary layer effect at 720 knots flight speed. The effect of these noise levels is largely unknown at the present time and will require checking under representative conditions to ensure effective operation of equipment. It is planned to apply these noise considerations to AVRO'S equipment requirements if a practicable approach can be established.

5.1.3.3 Effect on Structure

The deflections of a loaded panel in its response to acoustic loading, will produce bending stresses which oscillate about the existing static stresses. In order to estimate fatigue life, it is necessary to know the stress history of a panel.

SOUND PRESSURE LEVEL—db (ZERO db = .0002 MICROBAR)

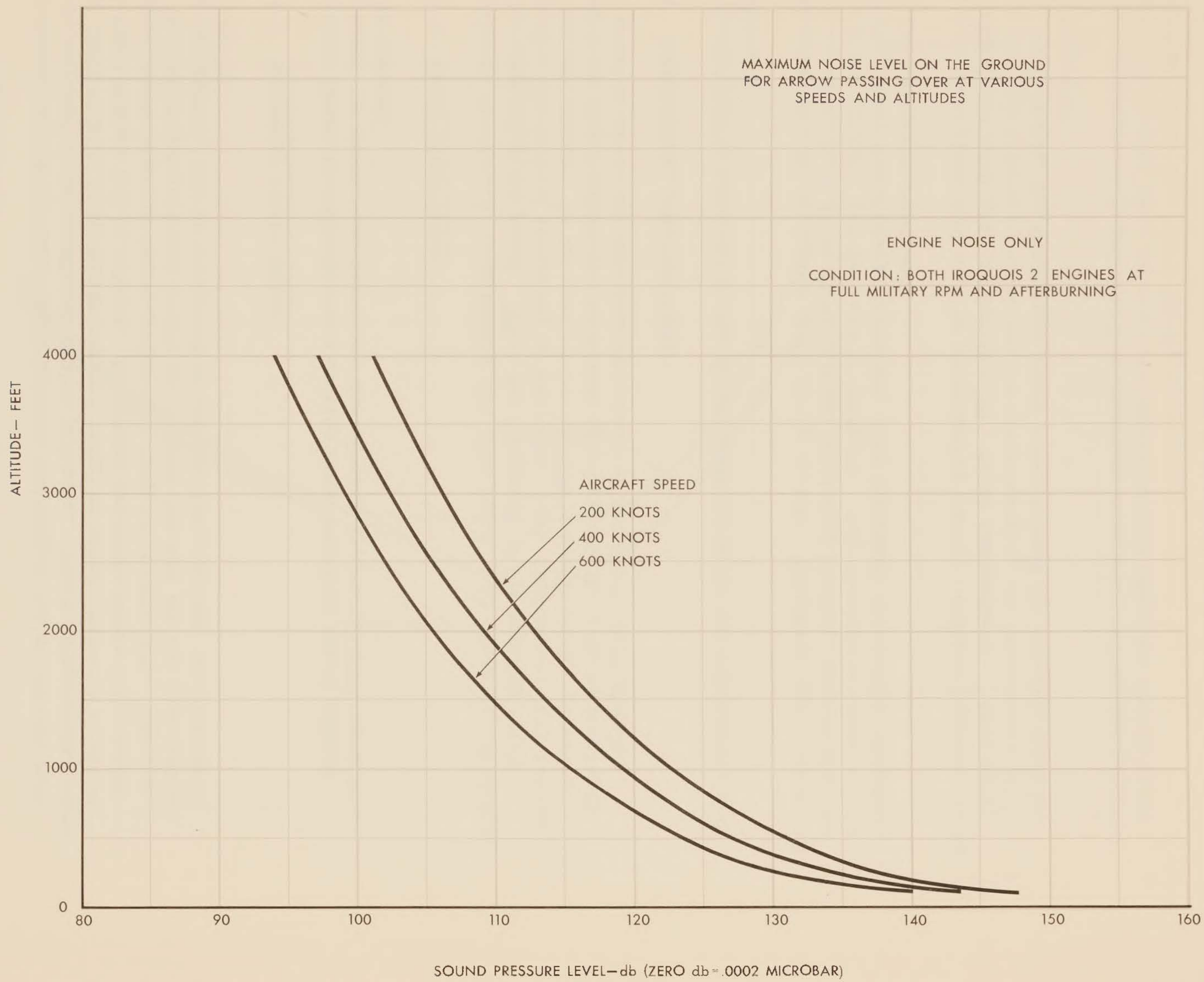

FIG. 11 GROUND NOISE LEVELS

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5.1.3.4 Jet Noise

Calculations have been made which indicate the fatigue life of various components exposed to the maximum engine jet noise. These components, vibrating at their natural frequencies, have a range of sound pressure levels in which their fatigue life is infinite. Continued exposure above these sound pressure levels leads to deterioration and reduced service life. The frequencies and sound pressure levels for infinite life and for the environment of the component, can be compared to indicate the problem, as illustrated in Figure 12.

5.1.3.5 Boundary Layer Noise

It has been shown that the stresses arising from boundary layer induced deflections are small, and therefore the effect on typical ARROW structure may be ignored.

5.1.3.6 Shock Wave Noise

An unsteady shockwave, terminating at a skin panel, will cause an oscillating bending moment in the panel. When the frequency of the exciting force approaches the natural frequency of the panel, (i.e. resonance) very severe stresses will result In the case of the ARROW's intake ramp, stresses arising from shock waves are moderate only because the natural frequency of the panel is above the dominant frequency of the shock-wave motion.

5.1.4 PERSONNEL PROTECTION

Crew protection is achieved by the direct muffling of the air conditioner and its outlets.

Engine noise will be negligible at all flying speeds. Boundary layer noise will be attenuated by from 40 to 75 db through insulating blanket, surrounding the inside of the cockpit, and the crew's headgear. Any shockwave noise will be of low frequency and although inaudible, may introduce unpleasant vibrations which are not easily removed.

Ground support personnel may be protected by run-up mufflers (approx. 35 db attenuation) and head gear protectors (20 - 40 db attenuation). These aids reduce the high frequency noise but the chest-surging low frequencies still exist. As a result, the dangerous sound areas around the aircraft must be clearly marked.

Community response to noise will be based on annoyance rather than physical damage. The noise level for vigorous legal action is about 40 - 50 db below the level for pain, i.e. about 84 db.

5.2 EFFECT OF TEMPERATURE ON STIFFNESS

This investigation is being conducted in order to determine the influence of a non-uniform heat distribution on the stiffness characteristics of the ARROW wing and the resulting deformation or warping of the wing. The non-uniform heat distribution is largely due to the irregular pattern of "heatsinks", such as structure and fuel tanks. A half-empty fuel tank for example, acts as a much greater heat sink to the lower surface of a wing than to the upper surface. This results in a greater degree of expansion to the upper surface than to the lower, which, in turn, results in a change of shape.

5.3 ALTERNATOR COOLING (ARROW 1)

A further investigation has been made to determine the adequacy of the alternator cooling, using more recent aerodynamic information. Cases have been considered for both standard and hot day atmospheres. The results of the investigation indicate that there may be problems at conditions of high altitude, and Mach numbers in excess of 1.7 on hot days. See Figure 13. The magnitude of this problem will be determined from flight test results.

5.4 FUEL TANK TEMPERATURES

Initial estimates of the temperature rise in the fuel tanks, due to aerodynamic heating, were made some time ago for the ARROW 1. See section 5.2.2, Fuel Temperatures Quarterly Report December/57. Since the geometry of the tanks and the sloshing of the fuel makes the problem difficult to solve accurately, an extreme type of mission was analyzed to provide a conservative answer. The maximum fuel temperature expected in the collector tank is 140°F. Fuel in the fuselage tanks remains cooler, heating up very slowly, while the fuel in the most exposed wing tanks may rise to 180°F.

As more detailed information has become available on the types of high speed missions likely to be flown, new estimates have been made which indicate that for the normal type of mission, maximum fuel temperatures will be approximately 25 to 50°F below the conservative estimates. This reduction is attributed to the fact that most missions will probably include a certain amount of subsonic cruising, which cools the fuel rapidly. In the extreme type of mission considered, a minimum of subsonic cruise was allowed for.

The high temperature estimates are used for the design of the fuel system, where high temperatures are critical, since the extreme mission is a possibility. The lower temperatures, on the other hand, are the more critical from the stressing point of view since large temperature differences between fuel and structure may result in large stresses.

5.5 TERMINOLOGY

Airborne sound is pressure variation in normal atmospheric pressure. The

variation in pressure is measured in terms of microbars. A microbar is equal to a pressure of one Dyne/square centimeter, or approximately one-millionth of normal atmospheric pressure. The starting point in the scale of noise levels is taken as 0.0002 microbars, also referred to as "o decibals". This is the sound pressure level of the weakest sound that can be heard by a person with very good hearing, under quiet conditions. A one decibal increase in sound intensity corresponds to a 1.259 increase in energy. It is convenient to use the decibel scale to express the ratio between any two sound pressure levels (SPL).

Then:

$$SPL = 20 \log_{10} \frac{P}{0.0002} \text{ db}$$

When P = rms of sound pressure in microbars, and 0 decibels = 0.0002 microbars.

6.0 AEROELASTICITY

6.1 THE AEROELASTIC CLEARANCE PROGRAM FOR THE ARROW

The aeroelastic clearance program has its origin in the first stages of design. Since that time, the program has been a continuous process of theoretical and experimental investigation to clear the aeroelastic behaviour of the aircraft over the entire envelope. Although all the unknowns related to the aeroelastic problem cannot be solved prior to first flight, the investigations up to that time will indicate the flight envelope area which may be flown with safety. This area then becomes the basis for the further investigation of the remaining unknowns. It is the continuity of the program that ensures that all the aerodynamic and structural considerations, for the control of the various aeroelastic effects, have been incorporated in the design at the most suitable times.

6.1.1 SEQUENCE OF INVESTIGATIONS

The first requirement was to develop an accurate method of calculating the flutter derivatives of a delta wing. This was followed by an analysis of the dynamic landing loads.

The aeroelastic work proper was started in 1954, with a study of the aileron reversal problem. With the resulting evaluation of wing stiffness, a preliminary calculation of the wing vibration modes was made and a simple strip theory flutter check conducted. The elastic effects on stability derivatives were then calculated. By the end of the year, stiffness, vibration and flutter work was begun on the vertical tail

As the structural design proceeded, more accurate stiffness calculations became available and, together with improved estimates of pressure distributions, were constantly fed into the continuing revision of the stability derivatives. The large stiffness influence coefficient matrices also allowed the proper calculation of the aircraft vibration modes to proceed, preparing the way for the large scale wing flutter clearance. The flutter analysis of the vertical tail was completed by the end of the year.

Subsonic wing flutter analyses were completed by early 1956, but due to some conflicting results, and on the advice of WADC, a low speed flutter model program was started. Supersonic flutter analyses were also started, with a survey of control surface buzz tendencies. Subsonic tunnel testing was delayed by the partial destruction of the model while being tested.

The low speed model work was expanded in 1957 and approximately six weeks tunnel time was used before the program was completed with the destruction of the model. Supersonic flutter analyses were completed on the wing and tail. To round out the experimental program, a low budget, transonic, wing flutter test was carried out at MIT. Work on the special non-classical component of this test program has been the development of improved aerodynamic theories. This work is currently proceeding.

6.1.2 AILERON REVERSAL STUDY

To establish an adequate aileron reversal speed, the structure was analyzed as swept box beams. A six-point antisymmetric load distribution matrix was developed to calculate the loading due to roll and twist. Rational distributions were used for the aileron loadings. Several different configurations were examined in order to determine the advantage of each. From the reversal aspect, the moving tip was best, although presenting many problems for other reasons. Of the flap type controls, the conventional outer wing type proved to be the most suitable design. The inboard types were eliminated because of the excessive torques they developed over the inner wing. As the design progressed, it became apparent that the provision of adequate stiffness in a 3% section would be inefficient. As a result, the section was increased to 3 1/2%. Based on the results of the ground resonance test, the outer wing torsional stiffness estimate appears to have been somewhat conservative. The aileron reversal speed, therefore, may be expected to be higher than that calculated.

On an aircraft of the tailless delta type, reversed elevators, as such, cannot occur, as there is no fixed frame of reference. The twisted wing, may always be adjusted to give a constant life, and this approach was adopted. The aeroelastic correction to elevator effectiveness was expressed as a shift of elevator C.P. at a constant lift.

The aeroelastic corrections to the rudder derivatives were calculated in a conventional way. A six point spanwise distribution matrix for fin loads was developed and modified to agree with results of wind tunnel tests. The corrections were presented as loss of side force and shift of C.P.

6.1.3 WING AND TAIL STABILITY CORRECTIONS

The six-point span loading matrix method was used for correcting the wing and tail stability derivatives. Corrections were expressed as a change in the shape of the lift curve and a shift in the aerodynamic centre. The beam type structural treatment was retained for both wing and tail. When more advanced structural and aerodynamic treatment became available, specific calculations conducted to provide detailed loadings for stress purposes, showed excellent agreement with the more simplified approach.

To modify the fin corrections, an attempt was made to design an aero-isoclinic structure, i.e., one whose stiffness distribution forces bending and torsion deflections to work against each other to allow no change in incidence under load. While a completely isoclinic structure was not possible, such measures as were taken undoubtedly saved considerable weight over a "brute force" approach to achieve the same overall result.

6.1.4 FLUTTER STUDY

6.1.4.1 Fin and Rudder

The theoretical approach to the fin flutter analysis was based on the concept of the fin as a swept beam. The structure was analyzed and vibration modes calculated on this basis. There is a choice of aerodynamic flutter theories for a surface of this shape. The first uses streamwise strips and conventional flutter derivatives, but requires an aspect ratio correction and a correction for the chord lines camber. The second splits the flow into components along and normal to the elastic axis and this avoids camber and aspect ratio corrections for the normal components. It does however, require additional derivatives to take the spanwise component into account. The second method was chosen for the ARROW fin, although as a check, the stream flow method was tried for the fin flutter case, with excellent agreement with component flow.

Three degrees of fin freedom were included in the analysis, together with the rudder fundamental mode. A wide range or rudder frequencies were covered to establish the stiffness required of the control circuit for flutter prevention. The results showed that flutter should be no problem on the fin, providing the rudder frequency was kept to twice the fundamental bending frequency. The streamwise strip method was used for the supersonic analysis, and no flutter speeds were found.

The low speed model program demonstrated that, except for very low rudder frequencies, the calculations were conservative. A very high margin was obtained and the flutter point agreed well with NACA data on similar planforms. In view of the high margin, it was considered worth while to proceed with a transonic model program.

The ground resonance test confirmed the fin vibration characteristics, but brought to light a new mode, rudder torsion, which was strongly coupled with the fin torsion mode. When this mode was included in the calculations, a flutter loop was obtained at low speed. However, when the theory was adjusted to give better agreement with the test data, the flutter loop was suppressed. The model was modified to provide this mode, and retested, showing that although there was a margin of low damping, no instability existed.

6.1.4.2 Wing

Flutter work on the wing was started by treating the wing structurally, as a cantilever beam. Vibration modes were calculated using the early beam

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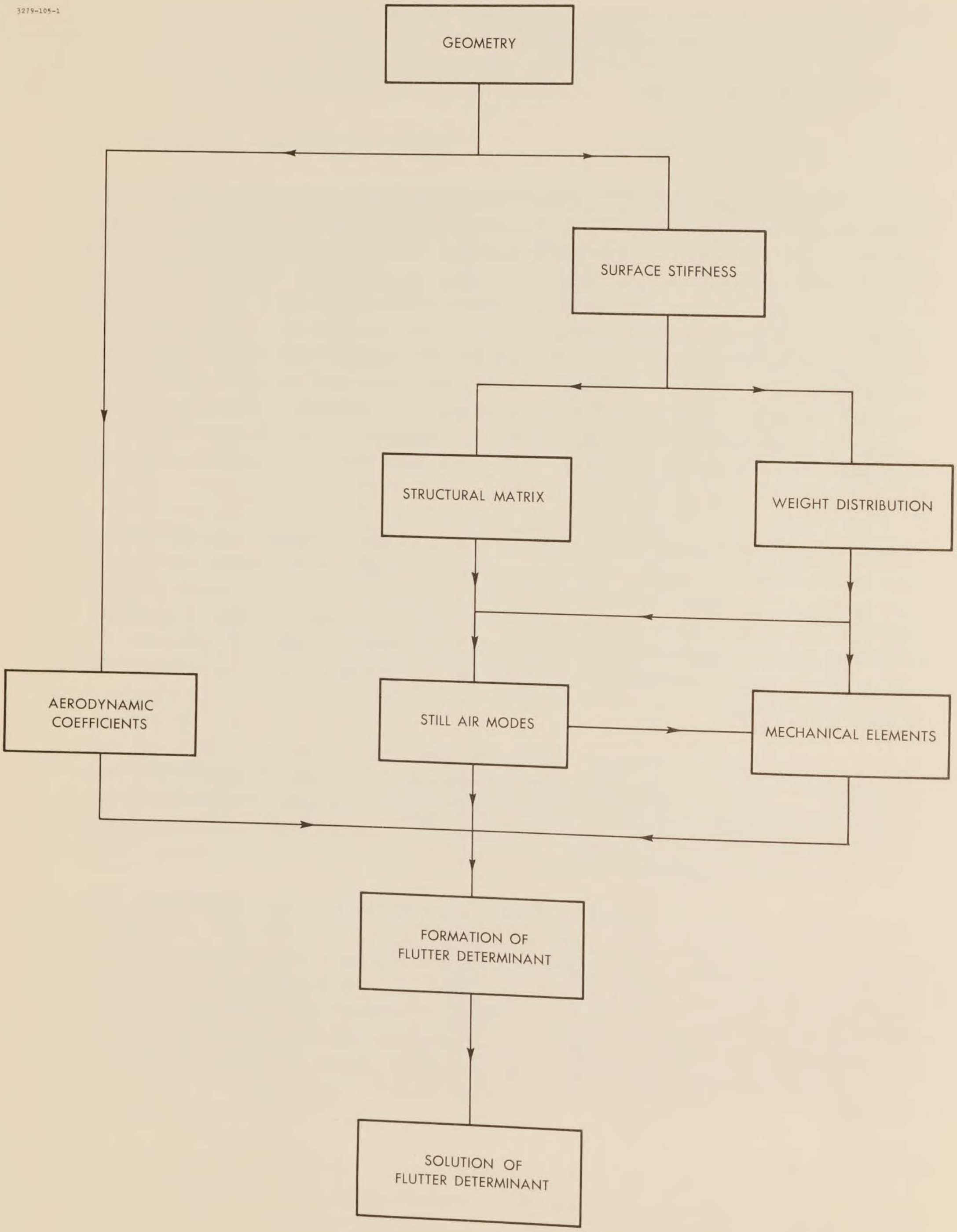


FIG. 14 STEPS IN A FLUTTER CALCULATION – ARROW 2

structural analysis. A conventional flutter analysis was performed, using two dimensional theory. The results were encouraging, but were considered to be optimistic due to the tendency of strip theory to underestimate the loads over the outer wing, where the model slopes are more important. The three dimensional theory mentioned earlier, proved very difficult to put into numerical form and approximately a year was spent in formulating the derivatives for use in a flutter calculation. As a plate type structural analysis became available for vibration calculations, the modes were calculated for the complete aircraft, as no rational attempt can be made to separate wing and fuselage in an aircraft of this type. Cantilever modes were also calculated to compare with previous work.

Large scale flutter work, on both free aircraft and cantilever modes followed two approaches. Strip theory calculations were maintained with modifications for span loading effects. In addition, the three dimensional theory was used. Results showed the three theories to give widely varying results. Compared with the results of model tests for the cantilever case, the theories ranged from 26% conservative to 22% unconservative. The best agreement was obtained with the modified strip theory at 6% conservative. However, both normal strip theory and the surface theory predicted a gentle approach to flutter and low subcritical damping, while the modified strip theory predicted high subcritical damping and a sharp flutter speed. The model showed a low subcritical damping and a mild onset of flutter. Five degrees of freedom plus controls were considered in the calculations for the cantilever case, but only the first and second modes had any significance.

Calculations involving the complete aircraft in the symmetric case involved six degrees of freedom plus controls. As the vibration modes were free body modes, the pitch and translation freedoms were not included in the flutter analysis. The same three theories were used as in the cantilever case. The results here did not show such a spread with conventional strip theory; only 3% conservative and the other theories about 24% unconservative compared to tests. The strip theory also showed the low damping and mild flutter onset that was, in fact, obtained.

Supersonic calculations were performed using two dimensional strip theory with the derivatives adjusted to give a constant reduced frequency across the wing. No Mach number alteration was made because of sweep. Comparison with tests showed the results to be about 15% conservative over the transonic regime in the cantilever case and to be very close for the complete aircraft.

The conclusion to be drawn from these results is that a conventional strip theory analyses, using two dimensional derivatives, has provided the best flutter picture. This theory is propounded in the literature to be about 15% conservative on speed and our experience would seem to bear this out.

It also appears from this work that the aircraft's flutter speed is dependent on the fuselage bending and torsional stiffness. (See paragraph 6.2). The fuselage bending mode gives the wing an apparent torsion mode which becomes the flutter mode. Thus, the wing torsional stiffness plays a secondary role in the flutter picture. It remains to be established whether local buckling at high stress levels will effectively reduce the stiffness and bring out a flutter condition at high g.

6.1.5 CONTROL SURFACE BUZZ

In addition to the types of flutter described above, there are other phenomena that fall under the heading of non classical. The first of these is control surface buzz.

Two types of control surface buzz exist. The first is a shockwave-boundary layer interaction problem which occurs at a speed slightly higher than the wing or tail critical Mach number. An oscillation arises from the shock waves jumping back and forth across the control surface hinge line. This problem is dependent on the shape of the pressure distribution over the control, and so is influenced by trailing edge angle and by flight manoeuvres. It is completely unpredictable, but seldom dangerous.

The other type of buzz arises from the reversal of sign at low supersonic speeds and reduced frequencies of the control damping derivative $C_{h\dot{\delta}}$. This gives what might be called single degree of freedom flutter. Calculations for the three ARROW controls showed that all should suffer from this trouble. However, when the other wing and tail modes were introduced into the calculation, flutter was suppressed, undoubtedly because of the large amount of damping arising from bending modes. As a precaution, however, provision was made in the design for buzz dampers to be fitted to all controls.

Another potential flutter condition arises from interaction between the structure and the high performance autopilot and fire control system. The accelerometer and gyro servos of the autopilot may sense airframe vibrations as well as rigid body motion. When the mode frequencies are within the range of the system a potential instability exists as the loop is closed through several paths in addition to the aircraft rigid body reaction. As frequencies and mode shapes alter with speed and Mach number, provision must be made in the system design to cope with these false signals.

This latter problem is the primary flutter analyses problem which remains to be investigated. In this connection, it is not enough to predict a flutter speed, rather, the subcritical frequency and phase changes become paramount. Thus the effort is going into the preparation of theories which provide better agreement with wind tunnel and flight results. Substantial progress has already been made, and this work will be more fully reported as it becomes available.

6.2 REVISIONS OF WING INFLUENCE COEFFICIENTS

As reported in the December 1957 issue of the Quarterly Technical Report, (section 6.3, Ground Resonance Tests) the stiffness values for the fuselage and wing, particularly in torsion, were shown to be higher than was originally estimated. In order to obtain agreement between test result and theory, the stiffness values of these portions, including the leading edge in chordwise bending, have been reviewed.

The previously low leading edge stiffness values are attributable to the greater effectiveness of a thicker skin and the addition of three stringers, since the original estimates were made.

The outer wing bending and torsional stiffness appears to be low since they were computed for use under conditions of restricted skin effectiveness in the control box region. The ground resonance tests, however, were made under a one g load condition, during which the skin in this area is fully effective.

The fuselage stiffness values were originally based on values for the fuselage frames bending out of the "U" shapes. It has been found that these frames are quite flexible in this form of deformation and hence the fuselage became flexible when the frames resisted torque. The present approach reacts fuselage torque by a shear flow in the outer skins of the fuselage. This method assumes the skin to be fully effective, (i.e. no buckle) resulting in a much higher stiffness value. The revised stiffness values will be checked in the theoretical ground resonance calculations by comparison to test results. If shown to be suitable, they will be used in revised flutter calculations. Ref. P/STRESS/X7/1060/8A27. December 1957.

7.0 ELECTRONIC SYSTEM

The ASTRA I electronic system for the ARROW 2 is being designed and developed by RCA. Certain supporting aspects of the system and associated studies are AVRO's responsibility and these aspects are discussed in this report. ASTRA I system progress is adequately reported by RCA.

7.1 SYSTEM APPLICABILITY

The electronic systems contemplated for the 37 aircraft program are in varying stages of development. The aircraft to which the different systems are applicable are as follows:

Aircraft	Electronic System
25201 to 25203 inclusive	Interim electronic system
25204 and 25205	Developmental ASTRA I
25206 to 25208 inclusive	Partial ASTRA I
25209	Developmental ASTRA I
25210	Developmental or Pre-production ASTRA I
25211	Partial ASTRA
25212 to 25215 inclusive	Pre-production ASTRA I
25216 and 25217	Partial ASTRA
25218 and 25219	Pre-production ASTRA I
25220 to 25225 inclusive	Partial ASTRA
25216 to 25237 inclusive	Pre-production ASTRA I

7.1.1 INTERIM ELECTRONIC SYSTEM

The interim electronic system provides communication, navigation and IFF equipment necessary for operating the aircraft during its development. It also includes features to enable the ARROW 1 to be used for the evaluation of telecommunication antennas.

7.1.2 PARTIAL ASTRA

The Partial ASTRA system includes the minimum necessary communication, navigation, IFF, flight instruments and air data equipment required to fly the aircraft during the research and development parts of the program, for which complete systems are not essential.

7.1.3 DEVELOPMENTAL ASTRA I SYSTEM

The developmental ASTRA I system includes all the telecommunication, navigation, fire control and automatic flight control for the ARROW 2. The developmental and preproduction systems are both complete ASTRA I systems. Variations from system to system will occur as the result of the normal course of development.

7.2 INSTALLATION DESIGN

7.2.1 ARROW 1

The ASTRA I developmental vehicles (aircraft No. 25204 and 25205), will be temporarily equipped with the interim electronic system for early flights. Subsequently, the interim system will be removed and replaced with ASTRA I for the development flight test program. The program proposal for these two aircraft is presently under revision to comply with latest requirements.

Internal wiring schematics of ASTRA equipment for aircraft 25204 and 25205 are not yet available to AVRO, but 85% of the point-to-point wiring information has been received. 35% of the information required for instrumentation is still outstanding. Layout of the aircraft cabling is in work for these two aircraft.

Unless complete ASTRA and instrumentation wiring information is received from RCA by early May 1958, the completion of the modified aircraft may be delayed accordingly.

Installation design for aircraft 25201, 2 and 3 is complete, with the exception of:

(a) Modifications to the damping system wiring due to relocation of the parallel servos and feel units.

(b) Revisions to the qc actuator system

(c) Installation of the g limiter.

(d) Installation of the automatic antenna selector.

7.2.2 ARROW 2

Wiring information for Partial ASTRA is virtually complete and production drawings for aircraft cabling have now been released. A number of accumulated modifications will be incorporated as soon as possible. Partial ASTRA system requirements for the research and development program were not included in the original RCA statement of work. These are presently being negotiated.

Full ASTRA I system requirements for ARROW 2 are now being investigated. It is expected that these will be an extension of the requirements for aircraft 25204 and 5 but the situation has not yet been clarified.

7.3 RADOME

7.3.1 SCOPE OF RADOME PROGRAM

AVRO has sub-contracted the design and construction of the radome to the Brunswick-Balke-Collender Company of Canada Limited (B. B. C.). Under the terms of the present contract, two developmental radomes, fully qualified to specification AVROCAN E-411, are being produced for the ASTRA I development vehicles, aircraft 25204 and 25205.

7.3.2 STATUS OF RADOME PROGRAM

During this reporting period, decisive efforts have been made to finally establish the electrical requirements of the radome. The electrical performance originally specified was considered unrealistic by the radome manufacturer and some relaxation of the specification was necessary. It was realized that a compromise would have to be accepted to achieve an optimum design for both linear and circular polarizations within the available time scale. Following discussions with RCA, the requirements for boresight accuracy and transmission efficiency have been stated, and AVRO is currently revising the specification accordingly.

The transmission study for solid wall construction has been completed. As a result it has been decided to use a wall thickness tapered along the longitudinal axis of the radome. Computations have indicated that the tapered wall will have an average power transmission 3% to 5% better than the uniform thickness wall. This will provide a very slight increase in radar range. Uniform wall thickness will be retained for any given radome station, as the requirement for variable polarization precludes the use of circumferential taper to favour linear polarizations.

The AVRO digital computer simulation of radome errors has been modified to handle the improved radome contour and the tapered wall. The computed range of boresight error was very small, (1-2 mils). However, previous experience indicates that the actual errors will be greater by a factor or 3 or 4 mils than those predicted by computation. Boresight shifts of 4 - 8 mils before correction, are expected when the radome is built and tested.

The design and computation of the radome was originally based on Bakelite Company BRSQ 142 resin. However this material is no longer available and the use of alternative resins is being investigated.

The radomes currently being developed for aircraft 25204 and 25205 represent the present state of the art and will probably define the radome for use with the preproduction ASTRA I system. AVRO is considering an additional development program if it should not prove possible to achieve a satisfactory result.

Radomes for testing and electrical correction are expected to be available by mid-1958. Two fully qualified radomes for aircraft 25204 and 25205 will be delivered to AVRO in December 1958.

7.3.3 SIMULATION OF RADOME ERRORS

AVRO has programmed a digital computer to assess percent transmission, boresight error and error slope of given radome designs. Radome errors predicted in this manner are not considered representative of practical measurements, but they are useful in providing an indication of performance trends. Results of this study are being transmitted to the radome manufacturer to assist in the electrical correction program.

The computing program is designed to produce far-field patterns for the radar antenna. Percent transmission through the radome is calculated by comparison of the field strength intensities with and without the effect of the radome. Displacement of the antenna pattern in the far-field provides an indication of the boresight error.

The program has been modified for the improved radome contour, and predictions of boresight error and transmission have been obtained for both the uniform and the tapered wall constructions. The results are not yet fully analyzed but the overall range of boresight error did not exceed 2 mils for linear and circular polarization. Transmission through the tapered wall showed a very small increase (approximately 1%) over the uniform wall. Since the results of this study are based on a purely theoretical approach to the problem,they cannot be taken as indicative of the performance to be expected from the actual radome, but they provide a basis for the correction program. Simulation precludes the effects of variations in electrical performance due to manufacturing tolerances and techniques. Therefore the results obtained by this method are more favourable than can be expected from tests on the actual radome.

7.3.4 TRANSMISSION STUDIES

Power transmission studies have been completed for both uniform and varying thickness, solid wall, radome constructions. From these, the optimum construction of the radome wall was determined.

7.3.4.1 Uniform Thickness Wall

Calculations of optimum wall thickness and average power transmission were completed for both plane and circular polarizations. The uniform wall thickness was determined by a ray analysis and from pomputations of the transmission and phase delay characteristics of a flat panel, having a dielectric constant of 4.07. Over the range of incidence angles found from the ray analysis, and for the 8,800 mc/s to 9,400 mc/s frequency range, the optimum thickness was determined at 0.350 inch. The average power transmission at this thickness was calculated as 89.5% for plane polarization and 90.7% for circular polarization.

7.3.4.2 Varying Thickness Wall

A similar study to that for the uniform wall was completed for a varying thickness wall. An analysis of incidence angles, transmission and phase delay, enabled an optimum construction to be determined. The results of the study indicated that the best construction for plane polarization would be a wall tapered both longitudinally and circumferentially. A wall thickness varying about the circumference is unacceptable for circular polarization, however, hence the optimum construction is a wall tapered in the longitudinal direction only. Figure 15 shows the computerd values of wall thickness Vs radome station for the tapered construction. The average power transmission for this tapered wall was calculated as 92.3% for plane polarization and 91.4% for circular polarization.

7.3.5 BORESIGHT ERRORS

RCA has produced information on permissible boresight errors, to enable the radome specification to be finalized. The maximum allowable errors and error slopes have not yet been agreed upon with B.B.C. and thus it remains to be seen if the correction program will reduce the present range of predicted errors to within acceptable limits. (Ref. para. 7.3.5.1 of the previous quarterly Technical Report). The correction of boresight errors is largely an empirical process and predictions serve mainly as an indication of error trends rather than magnitude.

The effect of the air data boom, on boresight accuracy, is somewhat uncertain. The boom will probably give a reduction in radome performance for small look angles. However, the resulting average reduction in radar range is not expected to exceed 10% for look angles up to 25 degrees off the

nose. Deletion of the boom from the nose location would provide a performance improvement of no more than 5%, or approximately 1 nautical mile in range.

7.3.6 PHYSICAL TEST PROGRAM

Two radomes are being fabricated by the radome sub-contractor for physical testing. One will be tested under environmental conditions, and the other under static test conditions. The environmental program will include weathering, low and high temperature tests, temperature shock tests, altitude cycling, vibration and rain erosion. The design requirements of the radome for structural strength and the structural test program have been formulated by AVRO. These requirements will be complied with in the sub-contractor's static test program.

7.3.7 ELECTRICAL TEST PROGRAM

Electrical tests on the radome will be performed using California Technical Industries (CTI) automatic boresight range equipment. In this equipment, a microwave signal is transmitted a minimum distance of 130 feet and received by the ASTR I radar antenna, over which the radome is mounted, at the radome test position. The effect of the radome on the signal received by the ASTRA I antenna is automatically assessed and recorded by the equipment.

When in operation, the equipment automatically follows and records the null position for every radar scanning position. Separate recorders continuously indicate the in-plane and cross-talk components of boresight error introduced by the radome. Variations in scanning position are obtained by gimballing the radome in azimuth and elevation and rotating it about its own axis, the test antenna being rigidly mounted in a holding fixture. Tests are conducted at frequencies of 8800, 9100 and 9400 mc/s respectively and radome error measurements made for horizontal, vertical and circular polarizations of the test antenna. The effect of the radome on power transmission can also be checked. Power reflection measurements will be measuring the voltage standing wave ratio (VSWR) of the radiating system before and after the radome is mounted over the radar antenna. From these measurements, the power reflection coefficient can be calculated. Pattern recordings will be taken, with and without the radome, to determine the antenna pattern distortion caused by the radome.

7.4 <u>INFRA-RED SUB-SYSTEM</u>

IR design and development, including procurement of the IR dome, is RCA"s responsibility. AVRO is directly responsible for the associated airframe and installation aspects.

It is intended to incorporate IR as a retrofit trial installation on aircraft 25204 and 25205, (the ASTRA 1 system development vehicles). Installation of the IR seeker pod on the fin, structural provisions, system plumbing and wiring will be provided on aircraft 25209 and subsequent. Installation of the seeker assembly, the nitrogen cooling system and the electronic equipment in the armament bay will be completed on all aircraft with developmental and pre-production ASTRA I systems. A dummy IR seeker pod will be fitted to aircraft with Partial ASTRA, to standardize their aerodynamic characteristics with the aircraft which have full ASTRA. Aircraft 25202 will also incorporate the dummy pod for fin performance assessment (ref. para. 7.5.6).

Design of structural modifications to the vertical fin, to facilitate retrofit of IR, is presently in work. The size and configuration of the seeker head installation on the fin was agreed with RCA, and AVRO has been able to establish the seeker envelope and define the structure in which it will be contained. RCA requirements for boresighting the IR seeker have been stated. It is proposed that the seeker mounting shall be adjustable to permit boresighting in azimuth, elevation and axial rotation. This will allow any misalignment between the IR seeker and the fire control radar to be eliminated, with the exception of that which occurs due to fin deflection.

Shapes and sizes for the electronic packages have been agreed and it has been decided to locate these units in the armament bay roof. The nitrogen bottle for the IR detector cooling system will be located in the engine bay. Wiring details and cooling air supplies for the electronic equipment are not expected to present difficulty.

7.5 ANTENNA DESIGN AND DEVELOPMENT

Design and development of ARROW telecommunication antennas is being conducted for AVRO by Sinclair Radio Laboratories Limited.

7.5.1 ARROW 2 UHF BELLY ANTENNA

Fabrication of the production prototype of the ARROW 2 UHF (annular slot type) belly antenna is nearing completion. Development has been delayed in order to procure suitable dielectric foam of the correct density, for use in the construction of the antenna. A new type of foam, with an 8 lb per cubic foot density will be used and tests are to be conducted to determine its resistance to moisture absorbtion. Impedence tests have been completed on the unfoamed prototype, but these results will be slightly modified after foaming. The VSWR is expected to remain less than 2.3 over the 225 - 400 mc/s frequency band, after assembly is complete.

7.5.2 MODEL PATTERN STUDIES CF-100

A 1/10 scale CF-100 model was used to measure ARROW UHF fin antenna patterns in the azimuth plane. These patterns were compared with actual in-flight measurements to establish the model range technique for ARROW antenna evaluation. Principal plane and conical cut patterns for the UHF fin antenna, at 1, 2 and 3 degrees below the horizontal, were completed at frequencies of 226.8, 324.3 and 384.3 mc/s. Gross polarization patterns were also measured, at the same frequencies, for the principal planes. The cross polarized power radiated by the antenna was 3 to 6 db less than the maximum radiated power, measured on the diagonals of the aircraft's azimuth plane.

7.5.3 MODEL PATTERN STUDIES - ARROW

A 0.07 scale ARROW wind tunnel model is in use for pattern studies of the UHF antennas.

7.5.4 UHF BELLY ANTENNA

Test pattern measurements were taken for the UHF belly antenna, with and without the vertical fin installed on the model. Measurements made at a frequency of 324.3 mc/s indicate that there is very little change in the antenna patterns when the fin is removed.

The effect of the external fuel tank on the antenna was also assessed. Principals plane pattern measurements were recorded for the clean aircraft and with the fuel tank installed. Both principal plane patterns and cross polarization patterns were measured at a frequency of 324.3 mc/s. Results show that the antenna pattern forward, both on the horizon and below the aircraft is virtually unchanged by the pressure of the fuel tank. The tank shielding causes the gain on the aft horizon to be decreased by approximately 7 db. This is not a critical area however and communication should not be seriously affected by the presence of the drop tank.

Conical cut pattern measurements have been completed for the belly antenna on a clean aircraft, at a frequency of 324.3 mc/s.

7.5.5 UHF FIN ANTENNA

Principal plane pattern measurements have been taken for the UHF fin cap antenna, at frequencies of 226.8, 324.3 and 384.5 mc/s. Cross polarization patterns were also measured. Patterns were measured with and without the infra-red seeker installed on the vertical fin. Comparison of the results showed that the effect of the IR seeker is to decrease the gain in the forward direction by about 1.5 db. These tests indicate that the IR seeker will not degrade the performance of the UHF fin antenna to any appreciable extent.

7.5.6 ANTENNA EVALUATION PROGRAM

The object of the antenna evaluation program is to verify that model antenna range patterns are essentially the same (within ± 1 db) as those obtained in actual flight. When the validity of the tequnique has been established on a CF-100, further single plane measurements will be made on the ARROW 1, and the results compared with ARROW model patterns to verify model results for full antenna coverage.

The flight test procedure will include pattern range measurements on three UHF frequencies and two L-band frequencies. Checks will be made to establish repeatability accuracy within ± 1 db. A dummy infra-red seeker pod is to be installed on the ARROW 1 for assessment of its effect on the fin antenna.

CF-100 antenna evaluation test flights began in late 1957, using aircraft 18186, as reported in the previous Quarterly Technical Report. However, after a total of five flights, these tests were discontinued until Spring 1958. The program will be resumed shortly, and it is hoped to present some conclusive results in the next report.

7.5.7 AUTOMATIC SELECTION OF UHF ANTENNA

Prior to the incorporation of data link, when antenna multiplexing will be necessary, an automatic method of UHF antenna selection is desirable to achieve omnidirectional coverage.

It has been decided to conduct trial installations of the Autonetics antenna selector described in the previous Quarterly Technical Report, on aircraft 25201, 25202 and 25203. The antenna-in-use indicator lights will not be employed, but otherwise the installation will operate as described. The pilot's selector switch will be labelled UPPER-AUTO-LOWER.

7.5.8 PARALLEL UHF ANTENNAS

Parelleled fin and belly UHF antennas have been investigated as a method of achieving omnidirectional antenna coverage without the use of an antenna selector. Measurements of the three principal plane patterns were made at a frequency of 300 mc/s for several phasing conditions. Patterns for the paralleled antennas were compared with principal plane reference patterns taken for the individual antennas. The results showed considerable degradation of antenna performance due to overlapping interference of the patterns in all three principal planes. The most critical distortion occurs in the region behind and below the aircraft. Consequently, it was considered inadvisable to adopt this method of obtaining full UHF coverage.

7.5.9 X-BAND ANTENNA

The RCAF has recently stated a requirement for full omnidirectional coverage of the AN/APX-27 (X-band) IFF transponder antenna. The present installation single X-band antenna in the ARROW cannot meet this requirement.

One method proposed to obtain spherical coverage is to employ two antennas connected in parallel. This may not be satisfactory, however, as pattern interference and power loss may result. A further solution would be to install two separate transponder units, each with its own antenna. This would present obvious weight and space penalties. Alternatively, the present system might be satisfactory if interrogation were continuous throughout an attack sequence.

It is evident that further study of this problem will be necessary to meet the spherical coverage requirement. This can be undertaken when RCAF authorization is received by AVRO.

8.0 ENGINE INSTALLATION

8.1 ENGINES

8.1.1 J75-P3 ENGINES IN AIRCRAFT 25201

During a recent engine run, a foreign object entered the right-hand engine air intake duct, and subsequent routine inspection revealed that damage had been caused to some compressor blades. The engine was removed from the aircraft and stripped of its build-up items and accessories. These were re-assembled to one of the two spare P3's available for aircraft 25201. The new power plant was installed in the aircraft, ground tested and has operated satisfactorily in flight. An RCAF decision is awaited on the action to be taken on the damaged engine. Since the P3 is a prototype engine, is is unlikely that the damaged unit will be replaced by another P3 engine. The intention is to replace the prototype P3 engines in aircraft 25201 with the production P5 engines at the earliest possible date.

8.2 ENGINE ACCESSORIES

8.2.1 NOSE BULLET - ARROW 1

Difficulty has been experienced in removing the engine nose bullets on aircraft 25201, without damaging the fairing skins. This is due to the stove-pipe nature of the assembly and the snug fit which made it difficult to separate mating parts. This problem has been solved by providing reinforced holes in the fairing, to permit the use of a pulling bar.

The double-walled anti-iced portion of the nose bullet (ref 70/ENG PUB/4 ARROW Quarterly Technical Report No. 1 section 9.3.4) will be available for installation in aircraft 25202.

8.3 INSTALLATION

8.3.1 WASTE FUEL DRAINS - ARROW 1

The fuel dump lines are being further modified to meet Pratt & Whitney requirements and to decrease fire hazards following engine shut-down. The fire hazard is caused by the high temperature of the engine and its surrounding structure. Consequently, spillage of fuel in these areas must be avoided. In addition, experience with aircraft 25201 has indicated that fuel should be dumped at some point well clear of the main landing gear. Should dumped fuel be carried by the wind on to the main wheels, following hard braked taxi runs or landings, there is a possibility of the fuel being ignited. Fuel drains are therefore being provided to carry the waste fuel from the three drain points on each engine to an overboard dump port, located at the aft end of the engine tunnel.

8.3.2 DEMONSTRATION OF ARROW 2 ENGINE INSTALLATION

A demonstration of the ARROW 2 engine installation was held 17 February 1958. Main points of concern to the RCAF personnel were (a) alignment of the floating duct with engine, and (b) accessibility to the engine relight oxygen bottle.

At a subsequent meeting between AVRO and the RCAF, the following recommendations were made:

(1) AVRO is to check the alignment provisions of the floating duct, articulated portion to ensure foolproof alignment and clamping to the incoming engine.

(2) Orenda is to be requested by AVRO to alter the position of the high altitude relight oxygen bottle, so that the bottle may be removed with the engine installed in the airframe.

(3) AVRO is to provide final recommendations regarding the proposed new engine reoiling system.

(4) AVRO is to make provisions for supporting the air conditioning system bleed duct elbow, to prevent obstruction of the tunnel to the incoming engine.

The RCAF will approve the engine installation subsequent to the receipt of the new oiling scheme recommendations from AVRO.

8.4 <u>POWER EXTRACTION</u>

8.4.1 ACCESSORIES DRIVE GEAR BOX SYSTEM

Minor modifications are being introduced in the ARROW 1 and ARROW 2 systems, based on experience gained with the system installed in aircraft 25201. Since the oil in the gear box reservoir is almost colorless, difficulty has been experienced in reading the oil level. In order to overcome this, a magnesium float is being introduced in the oil level sight glass.

Lubrication of the top output gearbox of the accessories drive gearbox is presently accomplished by grease packing. AVRO has requested the vendor to provide oil jet lubrication in order to improve gear box cooling.

Qualification testing of the ARROW 1 system is nearing completion and sufficient test results were available to permit first flights of aircraft 25201. Design of the ARROW 2 system is progressing, although mechanical problems have not been entirely resolved.

8.4.2 CONSTANT SPEED DRIVE SYSTEM

The previous Quarterly Technical Reports indicated that the General Electric constant speed drive unit for ARROW 1 did not meet AVRO specification requirements. AVRO has since reviewed the performance requirements of the unit and GE has incorporated minor modifications which have made the unit acceptable for installation.

Previously reported vibration failures have been overcome by installation of a reinforcing sleeve which attaches to existing CSD flanges, thus increasing the housing strength and rigidity against vibration.

Sundstand's design of the ARROW 2 CSD unit is well advanced and only minor modifications are required to meet AVRO space requirements.

8.4.3 LUBRICATING SYSTEMS

The lubricating systems provide circulation and cooling of the lubricating and working oil. In the ARROW 1, a single system serves the accessories drive gear box and constant speed drive systems of each engine. In the ARROW 2, separate oil systems are proviced for the two power extraction system of each engine.

8.4.3.1 Oil System - ARROW 1

A press-to-test dial gauge has been introduced in the system to permit checking of the system oil pressure. This provision is made for the convenience of test personnel and to facilitate servicing and maintenance.

Breakdown of the accessories drive gear box oil seals, with a resultant leakage and loss of oil, may occur with the large pressure differential existing between the gear box and its environment. Venting the gear box to an area of lower pressure than that presently provided by the engine oil breather is being considered as a solution to the problem.

A proposed relocation of the engine oil breather is presently dependent on further information from Pratt & Whitney regarding the permissible pressure differentials for the engine oil tank. A relocated breather pipe may be required for aircraft 25201. This problem however, does not exist with the P5 engine installation in remaining ARROW 1 aircraft.

8.4.3.2 Constant Speed Drive Oil System - ARROW 2

An oil system independent of the engine oil system is now being used for the ARROW 2 constant speed drive system. An integrated oil system had been under development and test until recently.

The integrated oil system was originally selected to avoid the weight penalty and eliminate the space problem involved in the selection of an independent system. Recent progress in the development of the oil system, combined with more recent information regarding equipment available and oil requirements of the CSD have removed any advantage previously held by the integrated system over the independent system. Furthermore, the reliability inherent in a separate system now outweighs all other arguments for the integrated system.

A preliminary schematic for the proposed separate CSD oil system is shown in Figure 16. Comparison with Figure 20 in 70/ENG PUB/4, shows that the main difference is in the oil supply and starting pump circuit. This is due to combining the oil supply tank with the deaerator.

8.4.3.3 Accessories Drive Gear Box System Oil System - ARROW 2

New information recently supplied by the power transmission system vendor has resulted in some changes to the accessories drive gear box oil system. Determination of new pipe sizes to handle an increased oil flow and rerouting of some oil lines has been necessary. A schematic view of the system is shown in Figure 17.

8.4.3.4 Oil Systems Air Supply - ARROW 2

The air supply requirements for the accessories drive gear box and constant speed drive oil systems have recently been established. The air supply system, shown schematically in Figure 18 provides protection against cavitation in the oil systems.

8.5 POWER CONTROL

8.5.1 ENGINE CONTROL SYSTEM

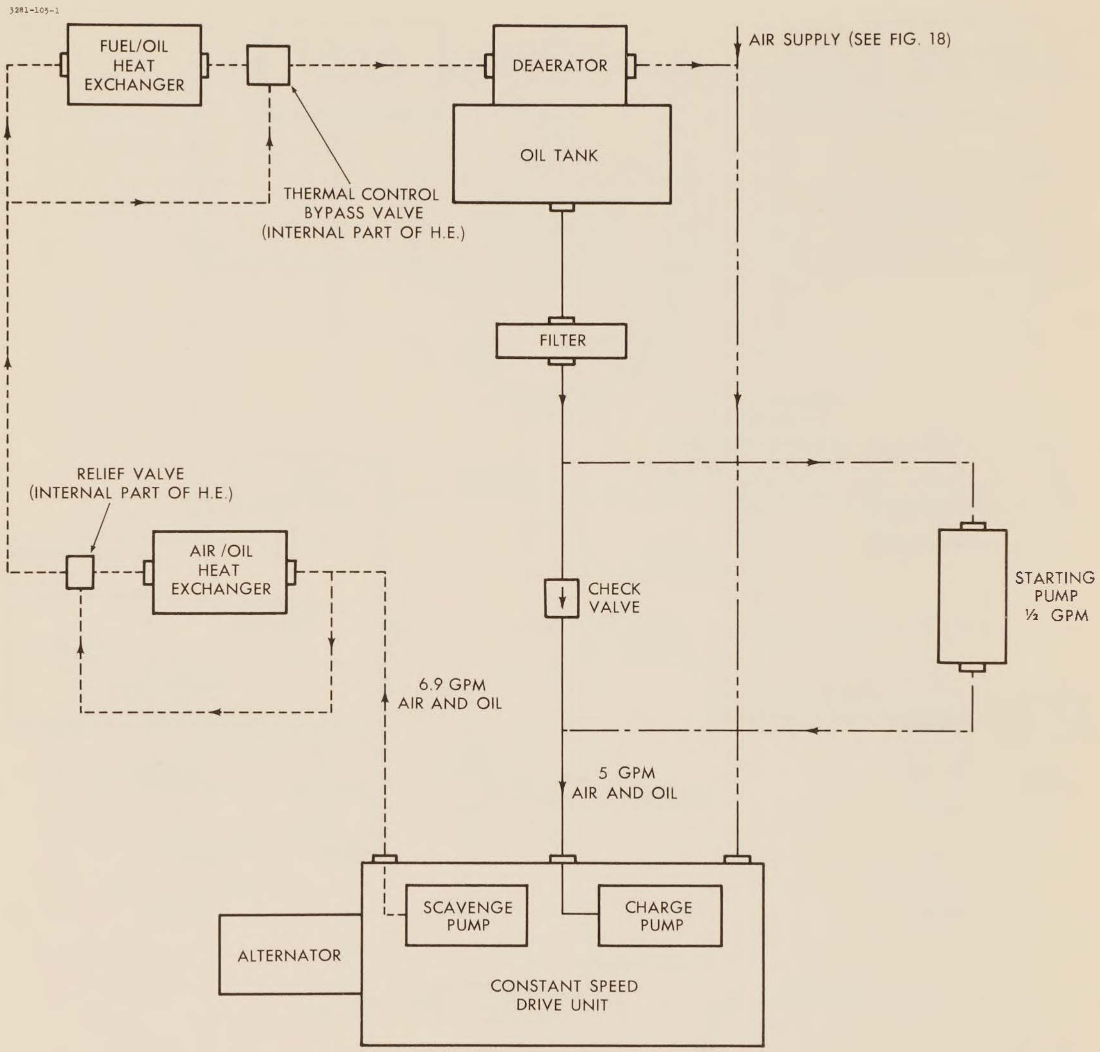
As noted in the previous Quarterly Technical Report, minor problems have been encountered in the engine control system. Elimination of backlash in the system has been the main area of activity. The ARROW 2 throttle drive shaft, which is presently being redesigned, should eliminate the backlash problem experienced with ARROW 1 system.

8.5.2 PERFORMANCE INDICATOR SYSTEM - ARROW 2

The AVRO designed performance indicator system simplifies presentation and display of engine performance data. Monitoring of engine performance data and pilot interpretation of this data is provided in a single direct reading indicator. The theoretical aspects of the system were discussed in section 5.3 of the Quarterly Technical Report for the period ending 31 December, 1957 (70/ENG PUB/5) (see also section 6.6 of the Quarterly

Technical Report for the period enging 30 September 1957 (70/ENG PUB/4).

The AVROCAN specification for the proposed system has been approved by the RCAF, and the AiResearch Manufacturing Division of the Garrett Corporation has been selected as the supplier.



NOTE: SYSTEM SHOWN IS FOR ONE CONSTANT SPEED DRIVE UNIT.
TWO SUCH SYSTEMS ARE INSTALLED IN EACH AIRCRAFT.

LEGEND:

————————	90% OIL, 10% AIR APPROX.
- - - - - - - -	OIL-AIR MIXTURE IN COOLING CIRCUIT
—— - - —— - - ——	AIR SUPPLY
—— - —— - ——	STARTING CIRCUIT

FIG. 16 CONSTANT SPEED DRIVE SEPARATE OIL SYSTEM—ARROW 2

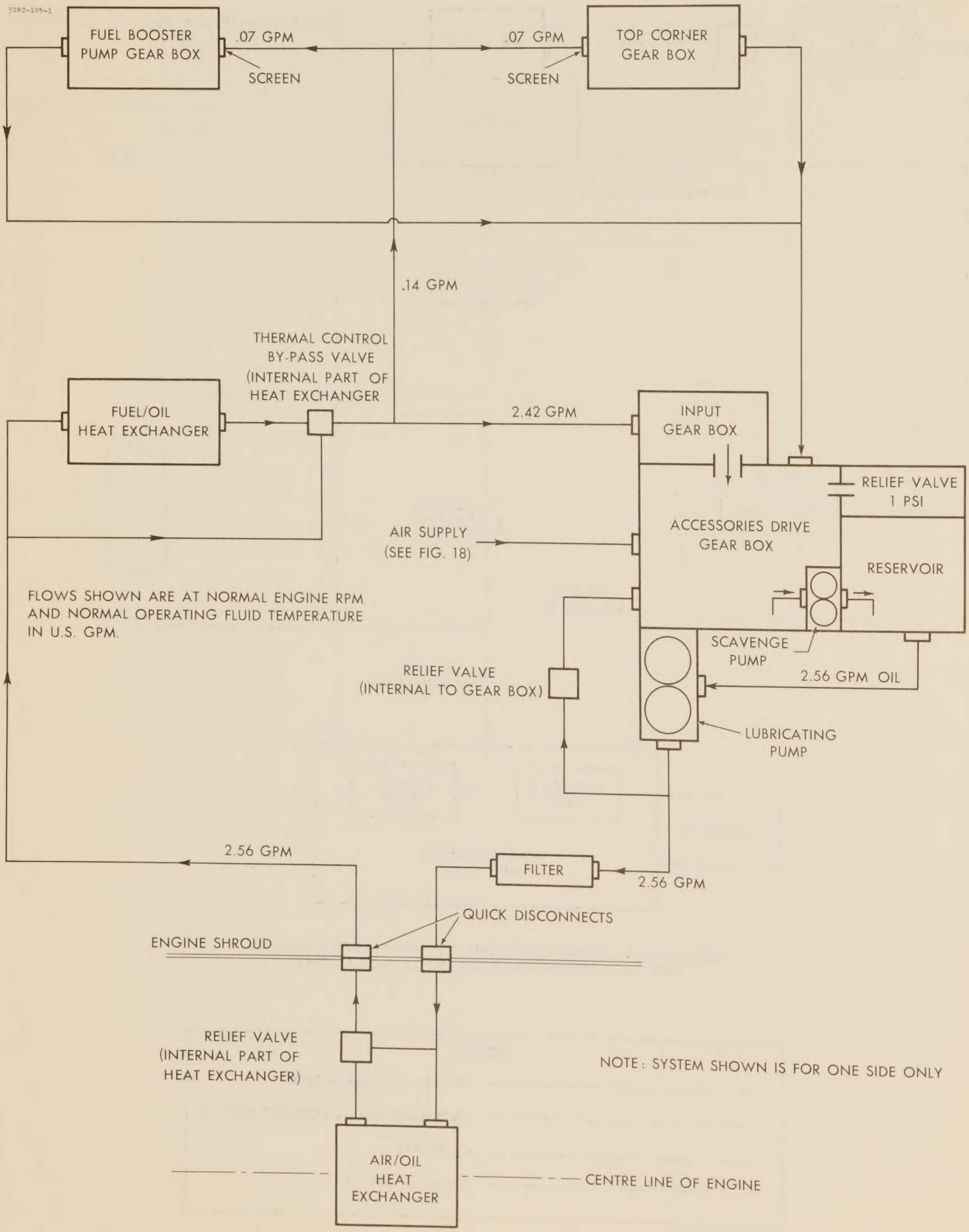


FIG. 17 ACCESSORIES DRIVE GEAR BOX OIL SYSTEM—ARROW 2

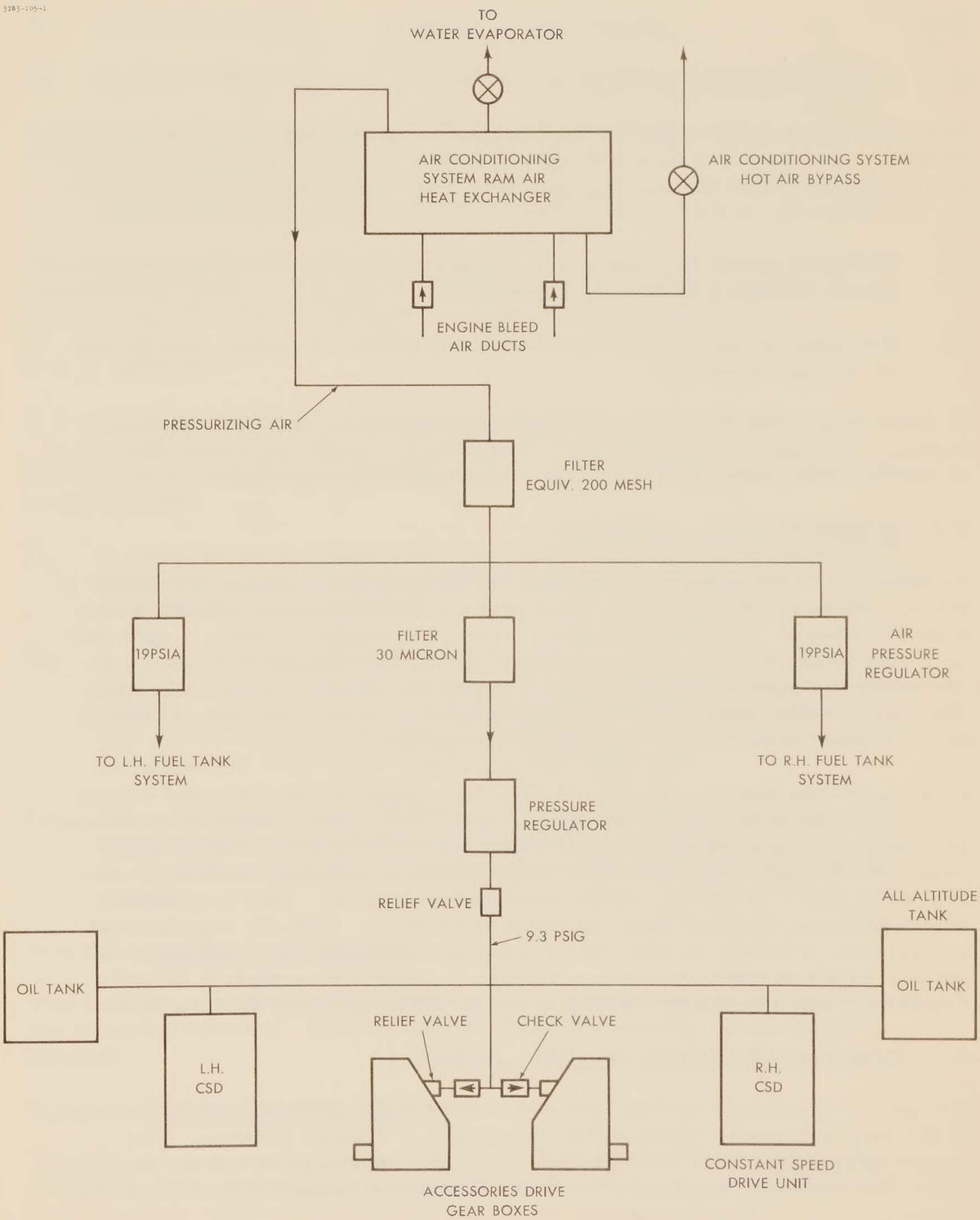


FIG. 18 OIL SYSTEMS AIR SUPPLY — ARROW 2

9.0 ELECTRICAL SYSTEM

9.1 POWER SYSTEM - ARROW 1

Barretter tubes were originally used to supply current reference for the ARROW 1 a-c power system voltage regulation, magnetic amplifier circuit. It was discovered on the first aircraft that these tubes were subject to failure for the following reasons:

(a) Hot spots on the filament caused a change in the performance characteristics, resulting in voltage fluctuation.

(b) Breaking of the filament, due to heating and/or vibration, caused loss of voltage regulation.

In order to prevent these failures, the tubes have been replaced with Zener diodes, a voltage reference device. Satisfactory results have been obtained to date with two units in which zener diodes have been installed.

9.2 POWER SYSTEM - ARROW 2

As reported in the previous Quarterly Technical Report, design alterations to accommodate the Westinghouse alternator and control system circuitry have been completed. The Westinghouse system is adaptable to either a split-bus or a concentrated bus d-c power system, AVRO has decided to adopt a concentrated bus system. This system will provide greater overall reliability, although in some ways it may be slightly inferior to the split-bus system as an individual method of d-c control and protection.

In the split-bus system, should one TRU fail, supply to the main and emergency buses would be maintained by the other TRU. There would be a momentary d-c power interruption of about 40 millisceonds however, which would cause the aircraft damping system to go into the emergency mode (i.e. yaw axis only) and in an attack, could cause loss of lock-on. This is obviously unacceptable as it would make the mission abortive. With the concentrated bus system, a failure of both TRU's would be necessary for interruption of the d-c supply to these services. The concentrated bus system is directly compatible with the present wiring of the aircraft electrical system.

9.3 EMERGENCY POWER SYSTEM - ARROW 2

Revision of electronic system load requirements has necessitated an increase in the specified power output of the emergency a-c power package. The requirements have now been doubled, i.e. 1800 VA at a power factor of 0.85 - 0.9. The following methods are presently being investigated to increase the power capacity of the generators.

(a) A larger generator

(b) Special winding of the existing generator, with additional control.

A proposal is being prepared by the equipment vendor for submission to AVRO.

9.4 ELECTRICAL SUB-SYSTEMS DEVELOPMENT

The changes to the electrical sub-systems during this reporting period are summarized below. More detailed description of the major modifications may be obtained from the paragraphs referenced in the summary.

9.4.1 ARROW 1

9.4.1.1 Nose Landing Gear Door Operation (Ref. para. 14.2)

The following alterations have been made to the nose landing gear door electrical circuits:

(a) A relay has been added to the door reclosing circuitry to prevent electrical power being disconnected from the landing gear up solenoid valve, until all doors are locked up.

(b) A ground servicing switch has been added to allow opening of the door after landing and reclosing before take-off. Circuitry has also been added to prevent the nose door reclosing until after engine start-up. The ground servicing switch is preselected to close, but the door remains open until either engine reaches 3020 rpm.

(c) The nose door circuity has been altered so that electrical power is supplied to the door OPEN solenoid valve, when the nose gear begins to retract, irrespective of the position of the door-fully-open limit switch.

9.4.1.2 Landing Gear Warning

The landing gear UP warning flasher, which operated below 10,000 feet with the throttles retarded, has been deleted from the system, at the RCAF's request.

9.4.1.3 Standby Compass Lighting

The 400 cps instrument lighting supply has been deleted from the standby compass, as it was found to cause compass interference. An instrument post light has been added to the panel to illuminate the compass.

9.4.1.4 Radar Antenna Drive - Aircraft 25204 and 25205

A solenoid-operated hydraulic control valve and a limit switch have been added to the radar antenna hydraulic control circuit. The control valve is energized from the main d-c bus through the contacts of the limit switch. When the electrical system is operating in the emergency condition, the hydraulic control valve is de-energized, stopping the supply of hydraulic fluid to the antenna drive system. The limit switch is broken when a ground hydraulic supply is connected to the aircraft. This also de-energizes the hydraulic control valve, ensuring that the 1000 psi antenna drive system is not supplied with fluid at 4000 psi.

Wiring for magnetron coolant and antenna drive hydraulic reservoir level-indication has also been added for these two aircraft, and for all ARROW 2 aircraft. (Note: The associated equipment is supplied by RCA; AVRO is responsible for the aircraft wiring only).

9.4.1.5 Wheel Brakes

The anti-spin solenoids have been deleted from the wheel brake hydraulic control valves. Anti-spin operation of the breakes will now be controlled hydraulically (Ref. para. 14.7).

9.4.1.6 Fuel System

Transfer pump air bleed solenoids with associated relays and wiring have been added to the fuel control system. These are energized by a low pressure warning signal from number 1 and number 2 fuel transfer pumps (Ref. para. 13.2).

9.4.1.7 Air Conditioning

(a) A temperature control bypass shut-off valve and pilot's temperature control "Emergency off" switch have been added to the system (Ref. para. 10.3).

(b) The electrically-operated air conditioning system throttling valve upstream of the turbine, together with its related controller, sensor and condenser, has been replaced by a pneumatically modulated type valve (Ref. para. 10.4.1).

9.4.1.8 Engine RPM Indication

Wiring for engine RPM indication has been added for aircraft 25202 to 25205 inclusive.

9.4.2 ARROW 2

9.4.2.1 Damping System

A "landing gear down mode" failure warning light has been added to the master warning system for connection to the damping system circuitry. The light will be illuminated if the damping system does not go into the low speed configuration (i.e. landing gear down mode) when the landing gear is extended.

9.4.2.2 Forward De-Icing System

A d-c supply was originally provisioned for the windshield and canopy anti-icing controller. This has now been delted, since no d-c is required for operation of the controller being supplied to AVRO.

9.4.2.3 Engine Hydraulics

Left and right-hand engine hydraulic low level warning lights have been added to the master warning system.

9.4.2.4 Cockpit Instrument Lighting

Alterations have been made to the cockpit lighting on aircraft 25212 and subsequent to provide integrally lighted instruments. Aircraft 25206 to 25211 inclusive will embody ARROW 1 type instrument post lighting.

9.4.2.5 Fuel System

(a) The "fuel transfer off" warning light has been deleted, together with the associated wiring for the manually operated gate valve limit switches and the wiring from the master refuelling switch. The mechanical design of the gate valve has rendered the warning light unnecessary.

(b) Circuitry has been added to the ARROW 2 for the totalizer in the rear cockpit.

9.4.2.6 Armament Control

Changes have been incorporated to coordinate and simplify the wiring connections between RCA supplied units and AVRO circuitry. These changes are also applicable in principle to aircraft 25203.

10.0 AIR CONDITIONING SYSTEM

10.1 ARROW 1 AIR CONDITIONING SYSTEM

The air conditioning system for the first three aircraft is shown schematically in Figure 19. The diagram is a revision of Figure 22 shown in the first Quarterly Technical Report, 70/ENG PUB/4.

The revisions are as follows:

1. Deletion of rain repellant provisions.
2. Corrected representation of hot air bypass line.
3. Inclusion of bleed air shut-off valves previously omitted.

Additions and modifications discussed in the following text are indicated on the schematic. Theoretical circuits for the associated electrical control of the system are shown in Figure 20.

10.2 COCKPIT TEMPERATURE CONTROL - ARROW 1

Cockpit temperature is controlled by regulating the quantity of hot bypass air mixing with the cold air from the turbine. In the first Quarterly Technical Report (70/ENG PUB/4) it was noted that the original electronic circuit was unstable, permitting large fluctuations in cockpit temperature. To improve temperature control characteristics, the circuit was modified by introducing a group of resistors and capacitors combined into a package referred to as the AVRO E50 unit. The nature of these modifications was described in paragraph 11.2.1.2 of 70/ENG PUB/4. However, cockpit temperature cycling was still evident with the E50 unit installed.

Since the operating characteristics of the original Hamilton-Standard control unit are apparently incompatible with the thermal properties of the cockpit, AVRO has undertaken to design and develop a control unit.

An interim workable system, using the original circuit with the Hamilton-Standard controller, has been obtained by disconnecting the over-temperature thermostat and connecting the control valve directly to the temperature controller. The thermostat was originally provided as an automatic override of the hot bypass air flow, closing the bypass valve when cockpit inlet air temperature exceeded 240°F. Instead, the thermostat interrupted controller signals to the valve, thus contributing to circuit instability. This interim system is now being used on aircraft 25201 for initial flights.

The disconnected thermostat in 25201 is now being utilized to operate a cockpit warning light, indicating high cockpit inlet temperatures. Thus, at the

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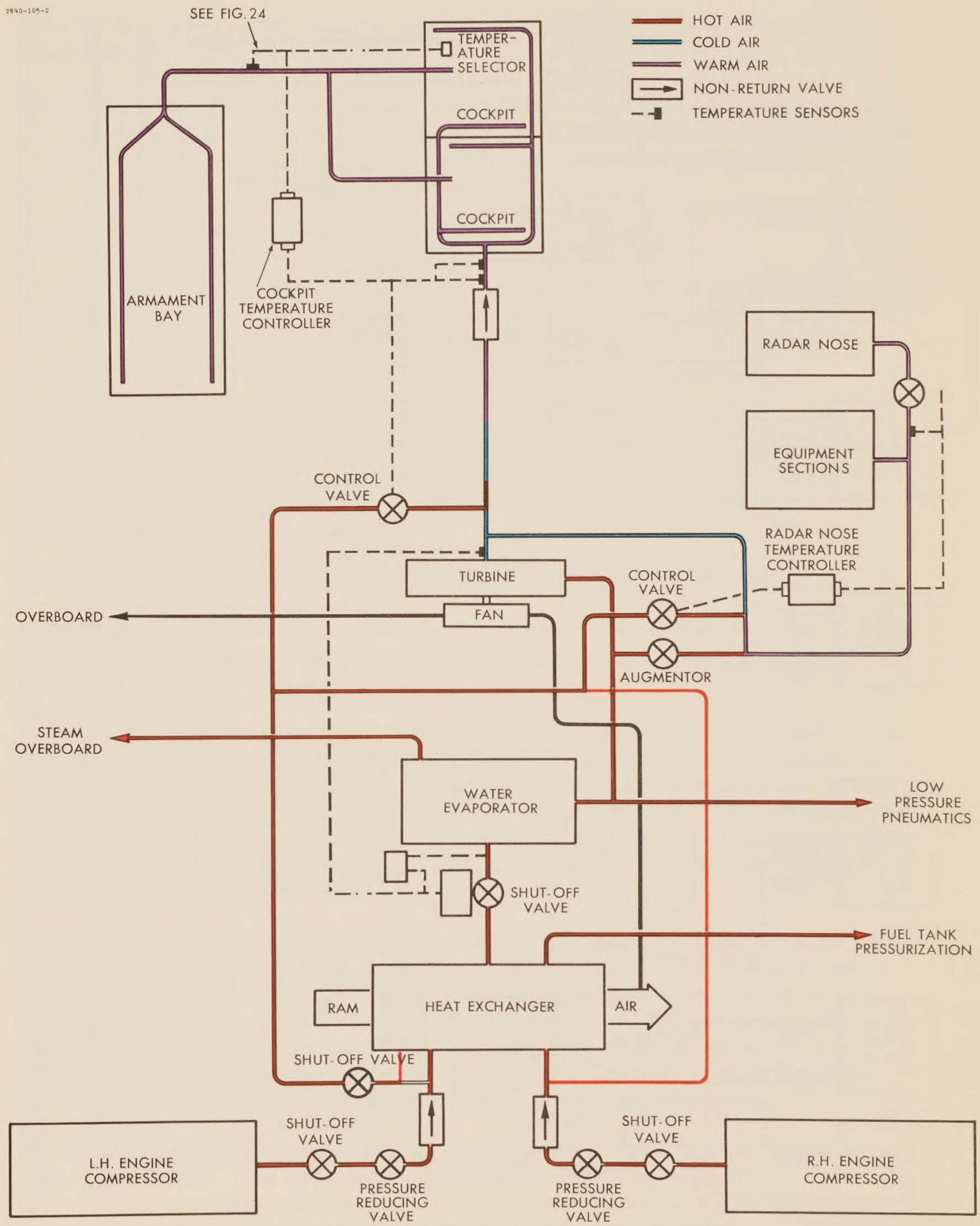

FIG. 19 SCHEMATIC - ARROW 1 AIR CONDITIONING SYSTEM

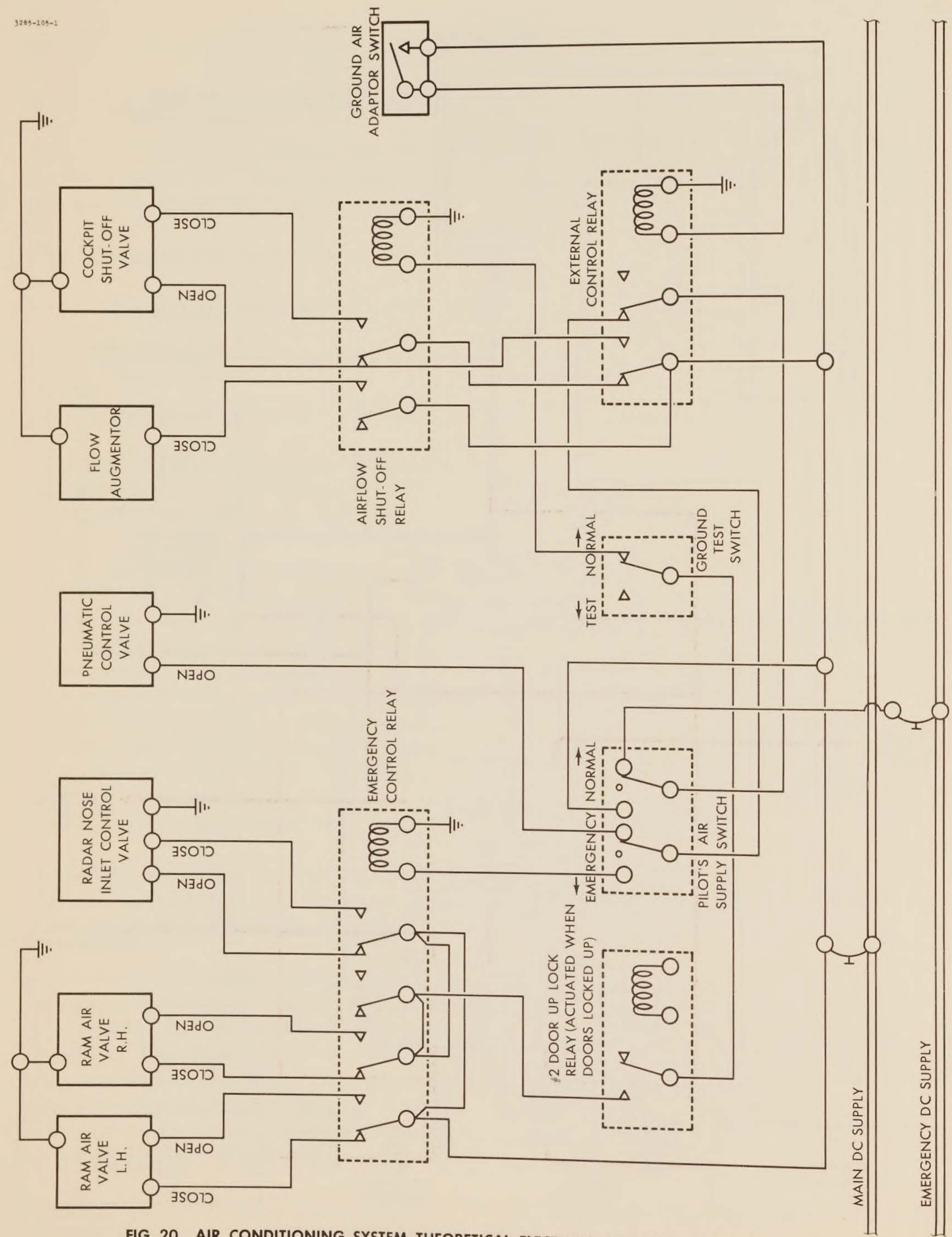

FIG. 20 AIR CONDITIONING SYSTEM THEORETICAL ELECTRICAL CONTROL CIRCUITS - ARROW 1

pilot's discretion, the recently introduced emergency hot air shut-off valve (see section 10.3) may be closed.

With the introduction of the AVRO controller, a new type of thermostat will be installed in the control circuit to provide automatic shut-off of the bypass hot air control valve, as originally intended. The emergency shut-off valve, will be retained as an added precaution.

10.3 EMERGENCY HOT AIR SHUT-OFF - ARROW 1

A shut-off valve has been introduced in the hot air bypass line, immediately downstream of the heat exchanger. The valve provides positive shut-off of the hot air flow to the cockpit and the equipment sections, should the automatic temperature control circuits fail to function properly.

The shut-off valve is electrically actuated through a manually operated two-position switch. The switch is normally in the TEMP CONTROL position, the valve thus being open. Selection of the HEAT OFF position closes the valve. Power to energize the actuator is taken from the emergency d-c supply bus.

10.4 TURBINE OUTLET TEMPERATURE CONTROL - ARROW 1

Due to the unsatisfactory operation of the turbine outlet temperature circuit, a pneumatic system using fully qualified components is being developed for installation on ARROW 1 aircraft.

Instability in the turbine outlet temperature control circuit was reported in the first Quarterly Technical Report (70/ENG PUB/4). With constant turbine inlet air temperature and pressure, turbine outlet temperature varied over considerably wider limits than the specified 20°F ± 5°F. Preliminary attempts to obtain a more stable control of the discharge temperature indicated that the throttling valve operating characteristics were not suitably matched with controller characteristics.

As an interim measure, the control circuit was modified in an attempt to improve the throttling valve operating characteristics. As noted in paragraph 11.2.1.3 of 70/ENG PUB/4, acceptable temperature limits were obtained by adjusting the internal resistance of the AiResearch control unit and grounding the valve CLOSE signal (from controller to valve) through a 10 microforad capacitor. However, circuit instability was still evident at high pressure and low temperature turbine inlet conditions.

To further improve temperature control stability, modification of the valve was undertaken. The non-linear relationship of valve angle to the time duration of the opening signal was considered a contributing factor to circuit instability. To obtain linear opening characteristics, a differential linkage was introduced between the electrical actuator and the valve butterfly. The extent of linearity achieved was not enough to entirely eliminate the regions of instability previously experienced. Stability could only be achieved by increasing the gear ratio on the motor drive which then resulted in a significant lag of valve response to controller signals.

Evidently, stable control of turbine outlet temperature cannot be achieved through minor modification of existing circuit components. The solution of the problem requires the introduction of a throttling valve with linear opening characteristics and providing the controller with some means of anticipating turbine inlet conditions.

10.4.1 PNEUMATIC CONTROL SYSTEM

The pneumatic control system (shown schematically in Figure 21, consists of a pneumatic thermistor, and a pressure rate control unit. The test data presently available on the pressure rate control unit will be supplemented by tests being conducted at AiResearch and at AVRO. Early clearance of the unit for installation in the ARROW 1 is expected.

An understanding of the manner in which the system functions to control turbine outlet temperature can be best obtained by recalling how the turbine produces cooling and noting the effect of throttle valve opening on turbine discharge temperature.

Cooling is achieved in the turbine by expanding air from a high inlet pressure to a low exhaust pressure. The resultant temperature drop of the air passing through the turbine can be related to turbine speed, which is in turn related to the pressure drop through the turbine. Thus, the higher the pressure ratio across the turbine, the higher the turbine speed; and the higher the turbine speed, the greater the temperature drop. In terms of temperature and pressure, this may be stated as

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

where k is the ratio of specific heats, C_p C_v, and has the value of 1.4 for air. Temperatures (T) and pressures (P) are as defined in Figure 21. From this expression for the ideal adiabatic expansion, it is evident that turbine outlet temperature (T_2) can be controlled by regulating turbine inlet pressure (P_1).

Regulation of turbine inlet pressure is obtained by a control valve located on the high pressure side of the turbine. The throttling effect of the valve is

used to vary turbine inlet pressure. Pressure on the downstream side of the valve butterfly is dependent on valve opening, and increases as the valve opens. The change in charge air temperature in the throttling process is generally insignificant.

Throttle valve operation is governed by a pneumatic controller which reacts to pressure signals transmitted by the pneumatic temperature sensor and pressure rate control unit into the butterfly opening or closing signals. The manner in which this control is achieved is evident from Figure 21, and the brief description of system components given in the following paragraphs:

The system will be adjusted to maintain turbine discharge temperatures ranging from 0° to 23°F over the full range of bleed air conditions.

10.4.1.1 Pneumatic Control Valve

The control valve consists of the valve controller, a pressure regulator and an electrical pilot shut-off valve. The valve butterfly is contained in the air duct portion of the valve body. A mechanical linkage, driven by the controller diaphragm against spring pressure, controls butterfly angular position.

Air from the high pressure side (upstream) of the air duct is admitted to the controller pressure chamber through the shut-off valve and pressure regulator. The regulator limits air pressure in the controller to a maximum value of 15 psig. This control pressure is reduced by the air leakage permitted by the pneumatic temperature sensor and pressure rate control unit, and acts upon the controller diaphragm, tending to open the valve butterfly, against the closing force of a return spring. Thus, in the event of uncontrolled leakage from the pressure chamber, positive closing of the valve is ensured.

The electrical shut-off valve is energized during normal operation of the valve connecting the controller with upstream pressure by means of internal passages. De-energising the shut-off valve closes the supply port and exhausts the controller to atmosphere, allowing the spring fail-safe feature to close the valve.

10.4.1.2 Pneumatic Temperature Sensor

A temperature sensitive rod submerged in the turbine discharge air duct constitutes the temperature sensing device. The contraction and expansion of the rod with changes in the duct air temperature is used to control air bleed from an air release valve. Increase of turbine discharge temperature temperature expands the rod, decreasing the air release opening and reducing air leakage from the valve controller. The converse applies for decrease of turbine discharge temperature.

10.4.1.3 Pneumatic Pressure Ratio Control Unit

The pressure rate control unit senses excessive rates of pressure change downstream of the throttling valve. A rapid pressure increase in the sensing chamber deflects the diaphragm to increase the opening at the air release valve, thus decreasing the pressure signal from the pneumatic temperature sensor and tending to close the valve. When the pressure being sensed stabilizes, the pressures in the two chambers equalize and the valve again fails under the independent control of the pneumatic temperature sensor.

10.5 BLEED DUCT LEAK DETECTION - ARROW 2

Investigations have been progressing to develop a reliable system for detecting leaks in the air conditioning system bleed ducts. Leaks could occur at joints in the duct system or through failure of a section of the duct itself. In addition to creating an emergency situation insofar as air conditioning system operation is concerned, excessive loss of the high temperature, high pressure air could seriously impair engine operation or create a fire hazard in the airframe.

The routing of the bleed ducts is such that in certain regions it becomes difficult for a conventional temperature sensing system to discriminate between right and left-hand duct leaks. A pressure sensing system using double-walled ducts is the most positive method of detecting leaks, but lack of adequate space throughout the length of the bleed duct complicates installation and maintenance of the system. Consequently, a leak detection system combining pressure sensing and temperature sensing devices is now being considered.

The proposed system, combining pressure and temperature sensing devices, is shown schematically in Figure 22. The pressure sensing part of the system would be used in the dorsal area where the right and left-hand bleed ducts are close together and where a temperature sensing system would not be reliable. The temperature sensing portion of the system would be used in the duct bay region where the right and left-hand bleed ducts are far enough apart to present a leak in the left-hand duct being detected by the right-hand leak detector.

Bleed duct leaks are indicated by warning lights. Failure of the pressure reducing valves to limit bleed air pressure to 60 psig maximum is indicated by the same warning lights.

A double-walled duct combined with a pressure switch constitutes the pressure sensing device. The annular space between the inner and outer duct is vented through a controlled leakage area to local ambient air. The pneumatic line to the pressure switch is connected directly to this annular space.

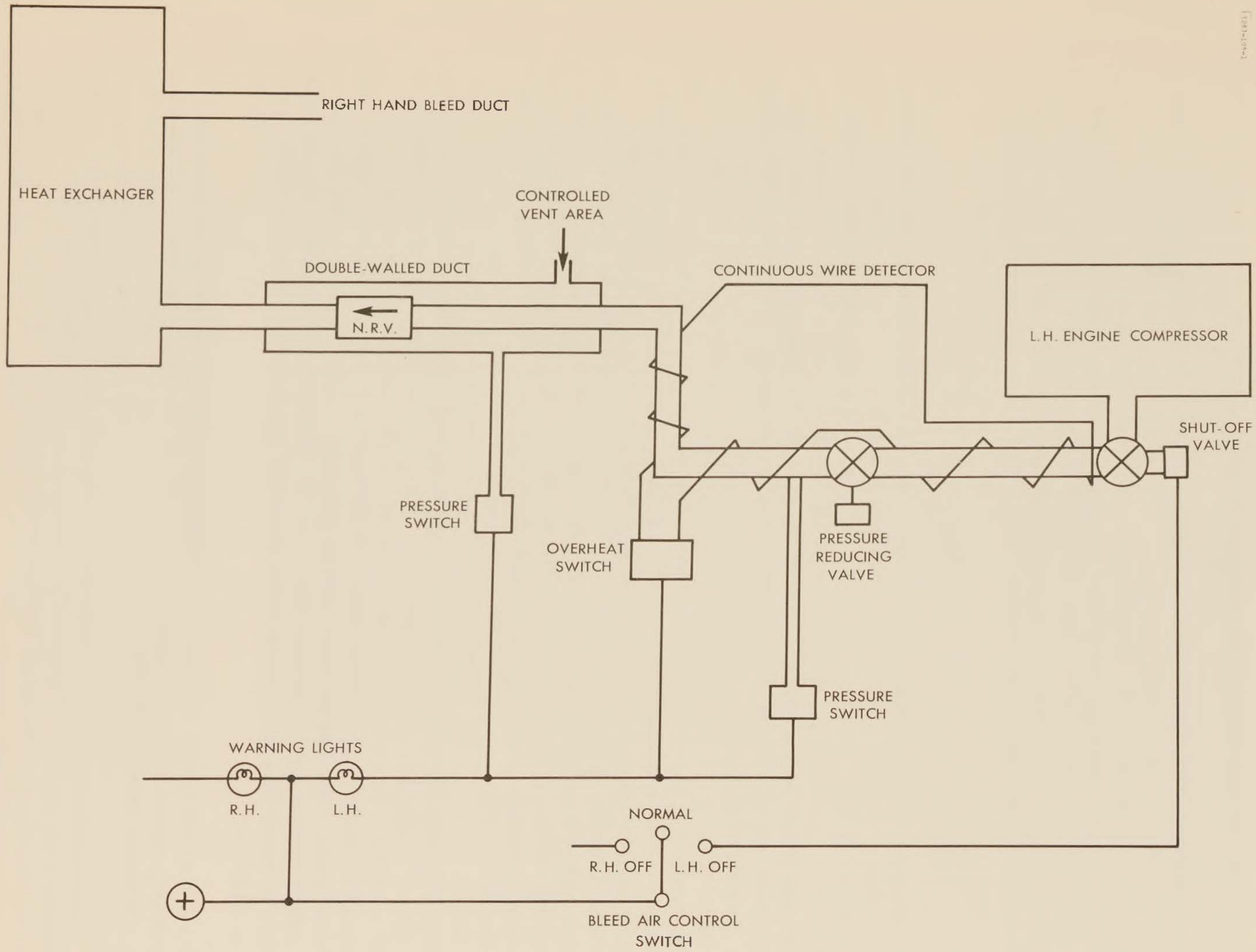

FIG. 22 SCHEMATIC - BLEED DUCT LEAK DETECTION - L.H. SYSTEM SHOWN - ARROW 2 AIR CONDITIONING SYSTEM

A leak in the bleed duct increases the pressure in the annular space, thus actuating the pressure switch to close the warning light circuit.

The temperature sensing device is a continuous wire detector in an electrical circuit similar to a fire warning system. Should a leak of high temperature air from the bleed duct occur, the section of the detector exposed to the escaping air is heated to a temperature above local ambient. The resultant change in detector resistance is sensed by the control unit which in turn closes the circuit to illuminate the warning lamp. A Fenwal continuous wire type system is being tested for this portion of the leak detection system. A Walter Kidde continuous wire system has been tested but proved to be ineffective.

The major objection to the Walter Kidde detector is that the total resistance of the detector is dependent on the length which is heated and the temperature to which it is heated. The characteristics of the detector reduces its sensitivity to local temperature increases. The Fenwal detector, however, displays more favourable characteristics in this respect.

Comparison of the resistance temperature characteristics of the Kidde and Fenwal detectors is illustrated in Figure 23. As shown, the Fenwal detector exhibits a large and abrupt decrease in resistance near its trip temperatures. The corresponding trip resistance is practically independent of the length of detector exposed to the trip temperature. On the other hand, the Kidde detector exhibits a gradual decrease in resistance according to an exponential law. The Kidde curve shown is based on a uniform temperature distribution over the full length of detector. As shown in the diagram, to obtain a given trip resistance with a shorter length of the same Kidde detector requires heating to a higher temperature. From this, it is evident that the Fenwal detector is more suitable for this particular application of overheat detection.

The construction of the Fenwal detector is similar to the Kidde detector described in section 12.2.2.4 of the previous Quarterly Technical Report (70/ENG PUB/5). The primary difference is that the Fenwal detector has a single wire embedded in the core material, with the Inconnel tube serving as the ground wire. The different resistance temperature characteristics are due to the use of different core materials; the Fenwal detector uses a salt whereas the Kidde detector uses a ceramic core material.

The major drawback to continuous wire systems is that the trip point temperature must be set substantially higher than the maximum possible ambient temperature, if false warnings are to be prevented.

Single point overheat detectors have been considered and tested. Although testing has shown that the single point detector gives adequate warning if properly located, the greater coverage afforded by the continuous wire system was considered to be a decisive advantage.

A system using a tracer gas, such as Freon, in the bleed air was also investigated but discarded due to its extreme sensitivity, complexity and lack of adequate airborne equipment.

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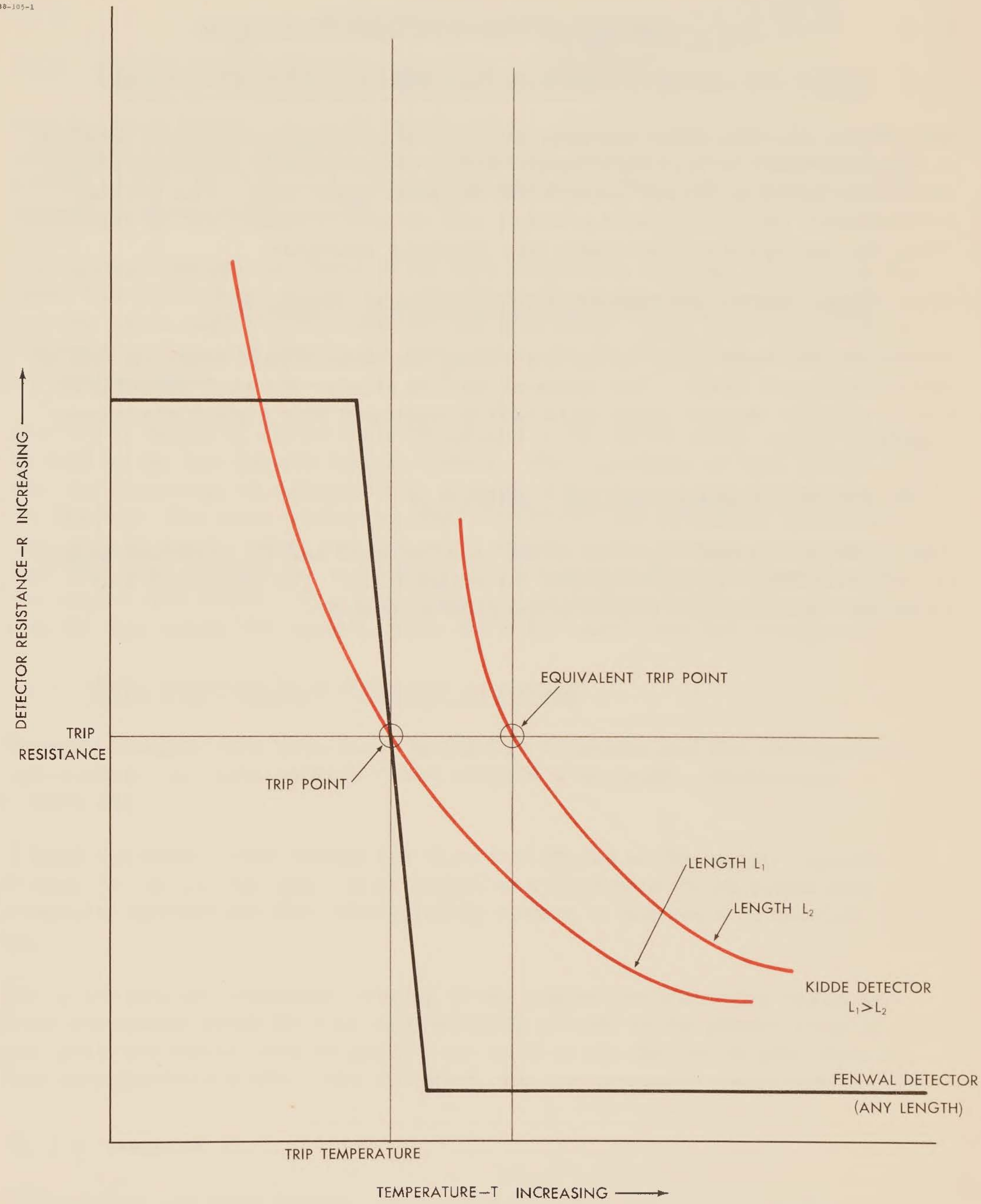

FIG. 23 COMPARISON OF RESISTANCE - TEMPERATURE CHARACTERISTICS OF WALTER KIDDE AND FENWAL HEAT DETECTORS

11.0 LOW PRESSURE PNEUMATIC SYSTEM

11.1 PITOT-STATIC SYSTEM FOR THE FIRST AIRCRAFT (25201)

In order to accommodate the pitot-static tube installed on the first ARROW, it was necessary to join the primary and secondary static pressure line to the single static pressure outlet from the pitot static tube. The normal arrangement (with separated primary and secondary lines) will be installed when the appropriate pitot static tube becomes available.

11.2 EQUIPMENT PRESSURIZATION SYSTEM - ARROW 1

Provision has been made for pressurizing the weapon pack seals as well as instrument pack seals. The systems will be similar for each installation, except that the weapon pack seals will be equipped with a quick disconnect coupling.

11.3 PITOT HEATER SWITCH - ARROW 1

The heated pitot-static portion of the relative wind and air pressure sensor for aircraft 25201 is not available for installation. The substitute unit requires a pitot-heater switch in the pilot's cockpit.

12.0 FIRE PROTECTION SYSTEM

12.1 EXTRA FIRE PROTECTION FOR AIRCRAFT 25206

For the first flights, aircraft 25206 will be equipped with extra fire protection consisting of three additional fire extinguisher bottles, each containing 20 pounds of CF_2 Br_2 (Freon), charged to 400 psi. The distribution of extinguishant is shown in Figure 24.

For normal inflight operation of the fire protection system, employing the extra fire protection, it will be possible to discharge one normal supply bottle and one extra supply bottle into any one fire area. This first shot may be followed by a second shot to the same area, by the discharge of the remaining normal bottle.

The crash switch is electrically connected to the three extra supply bottles as well as the two normal supply bottles. The operation of this switch initiates the discharge of extinguishant to each of the two engine fire areas and the fuselage fire area containing fuel, electrical and hydraulic equipment. One normal supply bottle and one extra supply bottle discharge into the fuselage fire area and the two remaining extra supply bottles discharge to the two engine fire areas. The remaining normal supply bottle also discharges into the two engine fire areas, when the outlet solenoids are energized.

12.2 FIRE PROTECTION CIRCUIT CHANGES

Several changes have been made to the fire detection and fire extinguishing sub-systems for both ARROW 1 and ARROW 2 aircraft. (Reference Figure 25).

A pilot-operated crash switch has been introduced at the pilot's request. Formerly, an inertia type crash switch was employed which would automatically operate the fire extinguishing system in the event of a crash landing.

The provision for automatic closing of the engine low pressure cocks has been eliminated from the fire extinguishing circuit at the pilot's request. The low pressure cocks were originally included in the circuit so that when the fire extinguishing system was operated, the low pressure cocks would close.

12.2.2 ARROW 2

The overheat warning system, described in paragraph 12.3.3 of the previous Quarterly Technical Report, has been included in the ARROW 2 fire protection system.

The system provides overheat warning in all three potential fire areas by means of three warning lights, each light operating from the continuous wire detector and bridge balance sensing circuit.

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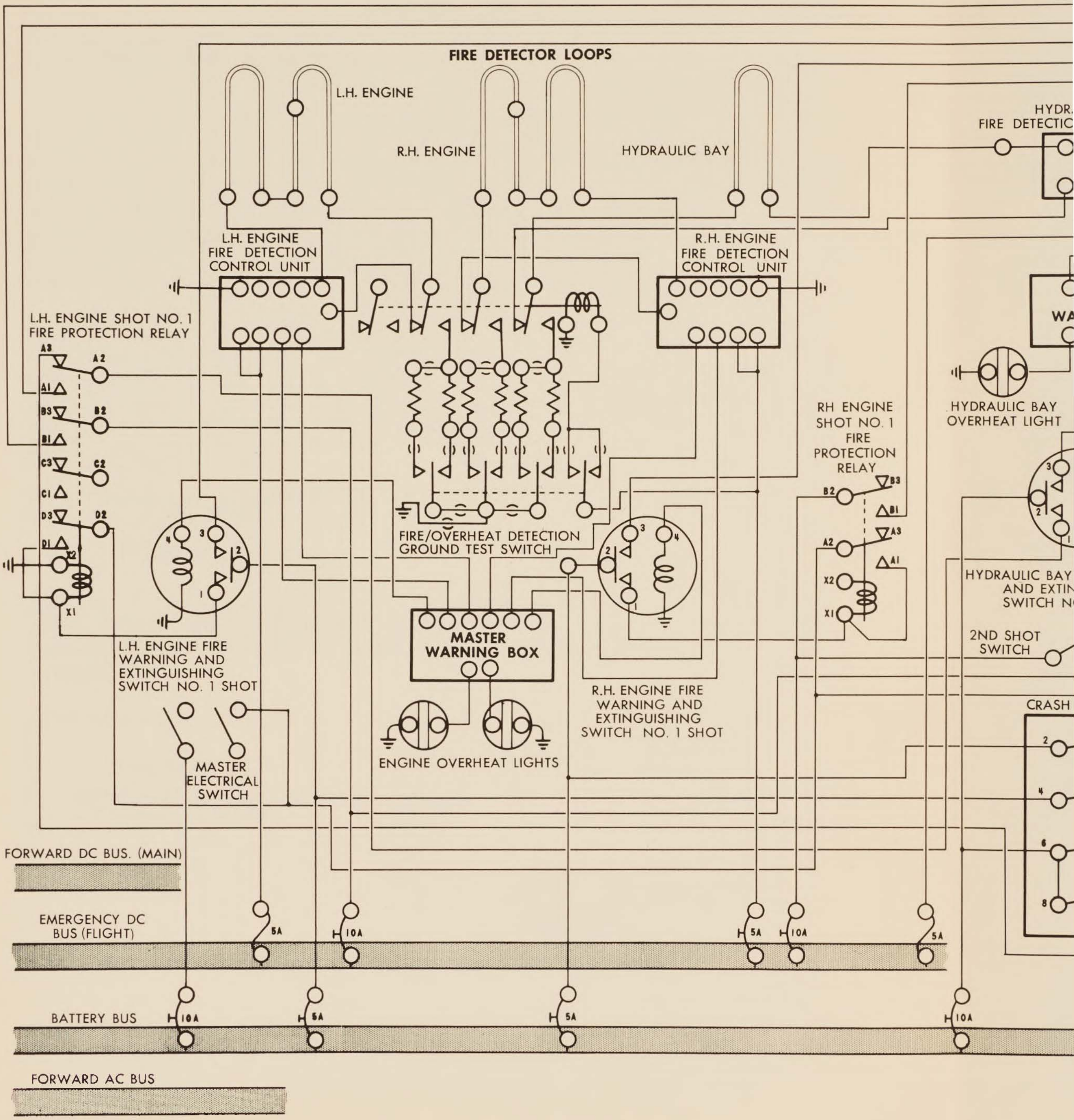
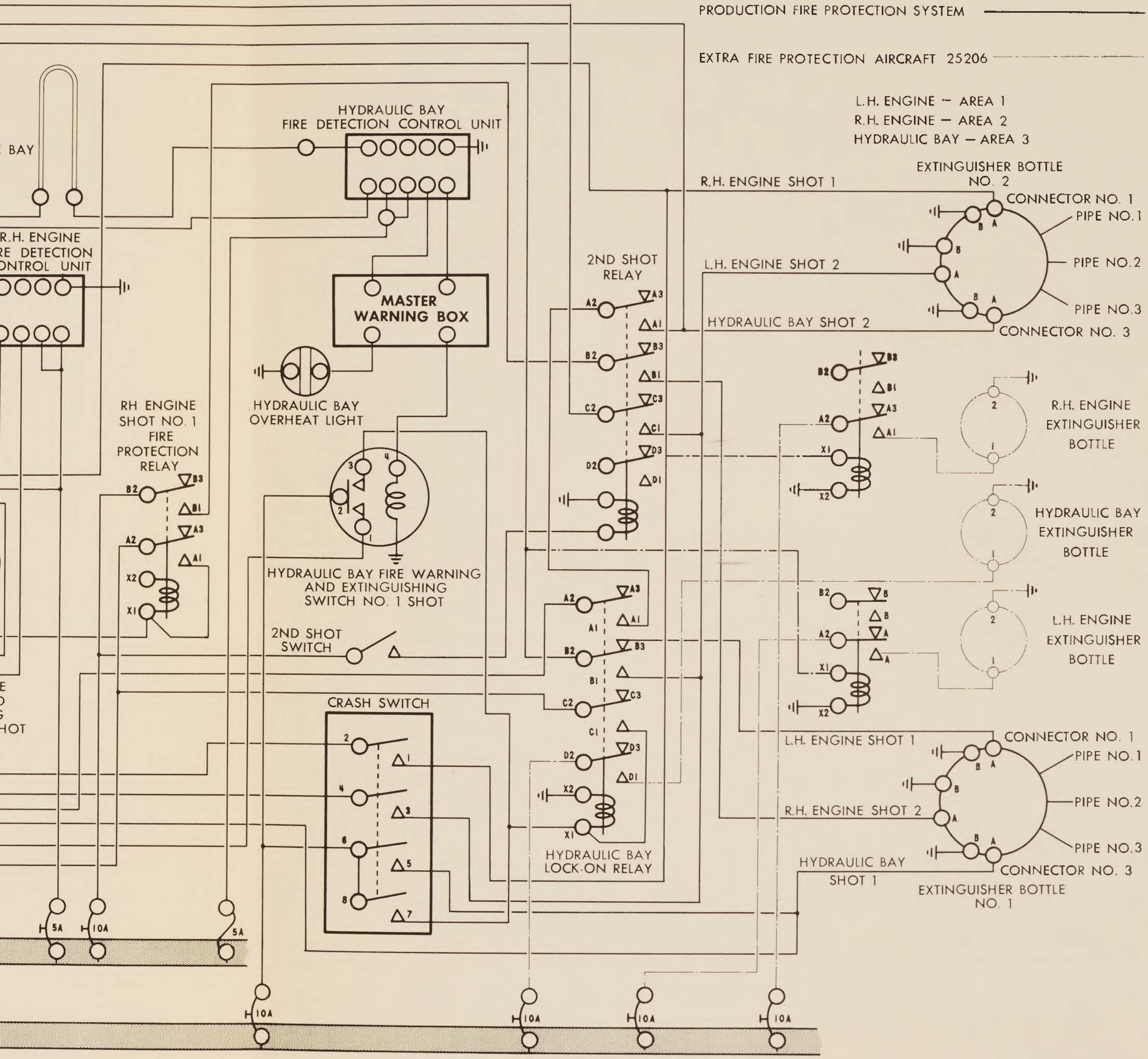


FIG. 25 FIRE PROTECTION SYSTEM WIRING DIAGRA



G. 25 FIRE PROTECTION SYSTEM WIRING DIAGRAM — ARROW 2

13.0 FUEL SYSTEM

13.1 FUEL BOOSTER PUMP

As noted in the previous Quarterly Technical Report the fuel vaporization occurring at pump inlet, contributes to the low delivery rates being obtained with the Pesco pump. This vaporization is due to the internal shape of the pump housing. The vapor formation blocks some of the pump fuel passages, thus limiting the fuel intake area and reducing pumping capacity. Fuel vaporization in the pump inlet can be eliminated by redesigning the pump housing. However, the cost and time involved are considered prohibitive and consequently new pumps will be selected for operational ARROW 2 aircraft.

Operation of the pump bypass has been hindered by improper operation of the pump check valves. Investigation revealed that the cracking pressures are apparently set too low. This permits air from the inoperative pump to enter the engine fuel feed line with subsequent loss of fuel feed pressure. The problem is being rectified by readjusting the check valve cracking pressures.

13.2 FUEL TRANSFER SYSTEM

13.2.1 FUSELAGE FUEL TRANSFER - ARROW 1

The fuel transfer line from each fuselage tank contains a transfer pump in series with a fuel pressure regulator. A combined bypass and refuelling shut-off valve is arranged in parallel with the pump and regulator as shown in Figure 26.

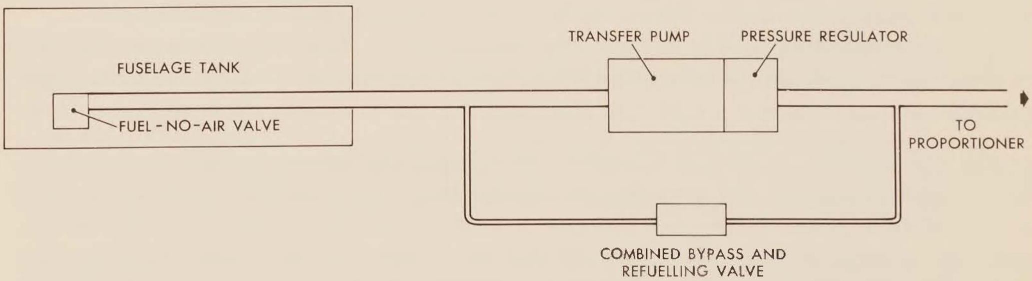


FIG. 26 SIMPLE SCHEMATIC—FUSELAGE FUEL TRANSFER—ARROW 1

Control of the bypass valve for the normal or refuelling mode, and for emergency modes where either the pump or regulator fail, is obtained by hydraulic servo. A bypass over-ride valve provides the necessary control of servo fuel pressures.

13.2.1.1 Transfer Pump

Loss of prime in the transfer pump has been experienced during simulated mission tests on the fuel test rig. The conditon occurs during nose down attitudes when the fuel level in the tank is low. Occurrence of the condition was relatively infrequent and resulted in no adverse centre of gravity effects.

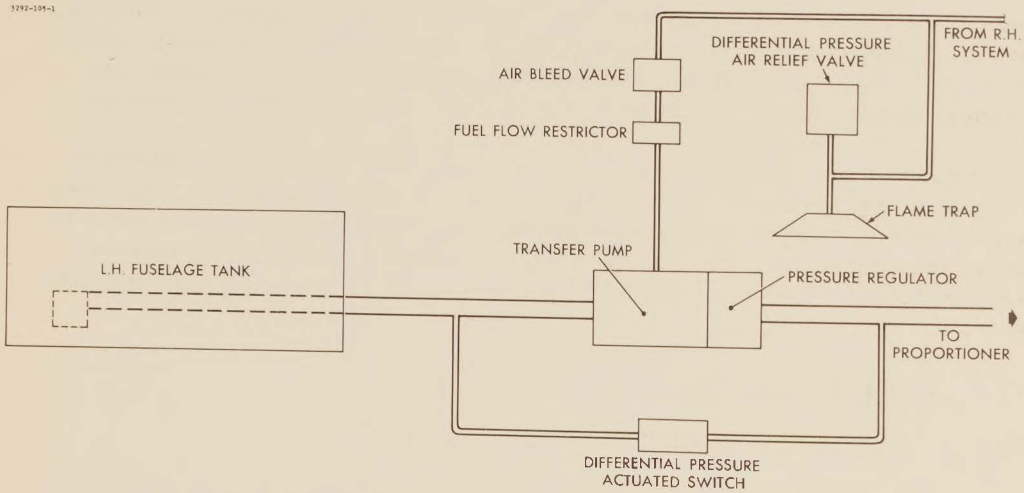


FIG. 27 SIMPLE SCHEMATIC—TRANSFER PUMP AIR BLEED—ARROW 1

Pump repriming will be achieved by introducing a transfer pump air bleed valve as shown in Figure 27. Air is bled directly from the pump through a fuel flow restrictor and a bleed valve, and is discharged to atmosphere through the existing fuselage tank venting system. Loss of prime is signalled by the differential pressure-actuated switch closing the cockpit warning light circuit.

At the same time, a circuit is made energizing the bleed solenoid valve to open. The bleed air, at a pressure higher than atmospheric ambient is thus conveyed to a point on the low pressure side of the air pressure relief valve, then discharged through the vent flame trap. When pump prime has been restored, the pressure-actuated switch breaks the circuit to close the bleed solenoid valve. Should a more serious pump failure occur, the bleed line restrictor will limit fuel lost overboard to a tolerable figure.

13.2.1.2 Pump Bypass and Pressure Regulator

Improper functioning of the bypass valve, its override control valve and the pressure regulator was reported in the previous Quarterly Technical Report. Since the vendor has been unable to provide an immediate solution to the problem, AVRO is taking delivery of the outstanding units and will modify them to meet the necessary requirements.

13.2.2 FUEL FLOW PROPORTIONER AND MANIFOLD - ARROW 1

Performance of the fuel flow proportioner in the test rig and on aircraft 25201 has been much better than anticipated. It appears now that a system using the proportioner will give more satisfactory results than originally anticipated for the ARROW 1 flight test program. Additional fuel flow proportioners have been ordered, and with proportioners available for all ARROW 1 aircraft, fuel flow minifolds will no longer be required. Endurance testing of the fuel flow proportioners will be resumed in an effort to extend the useful life of the unit between overhauls to beyond the present 25 hours.

13.3 PRESSURIZATION AND VENTING SYSTEM

With all tributary tanks pressurized, collector tanks must be vented to effect fuel transfer. Using 19 psi air pressure in the tributory tanks, adequate fuel transfer at low altitudes can be achieved only be evacuation of the collector tanks. Evacuation is obtained by forced ejection of collector tank air.

13.3.1 AIR EJECTOR

Development and testing of AVRO-designed air ejectors has been in progress for several months. The objective of the program is to obtain a unit which will meet the low pressure air flows defined by the design line shown on the pressure vs flow diagram in Figure 28. Ejector designs providing capabilities shown by lines (1) and (2) have been built and tested, and the performance indicated by line (3) seems to be attainable.

The air ejector operates on the same principle as the steam-jet air ejector, except that a high pressure air jet is used instead of a steam jet to create suction. The high pressure air is admitted through a nozzle into the suction chamber Figure 29 where a low pressure air is entrained and discharged with the high pressure air to atmosphere. Performance of the unit is dependent on the design of the high pressure air nozzle and the discharge tube.

The ejectors now available for installation in ARROW 1 aircraft are performing as shown by test line (1) in Figure 28. Suction attainable with this ejector is low at the higher low pressure air flows. Modification of this ejector has improved performance at high tank pressures but resulted in poorer performance in the low pressure range. This is represented by test line (2) in Figure 28. It is expected that performance corresponding to test line (3) can be achieved by increasing the high pressure air flow in the modified ejector. Such a unit would be satisfactory for ARROW 2 installation.

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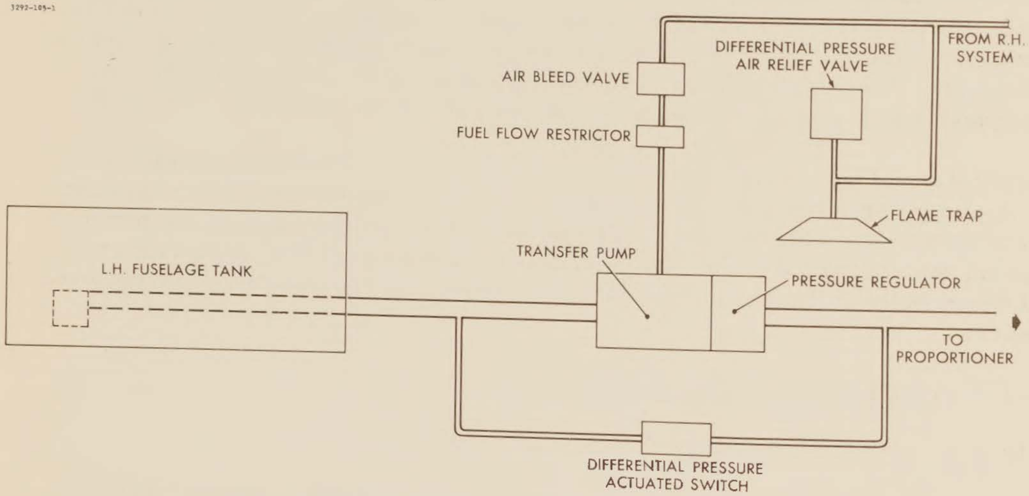

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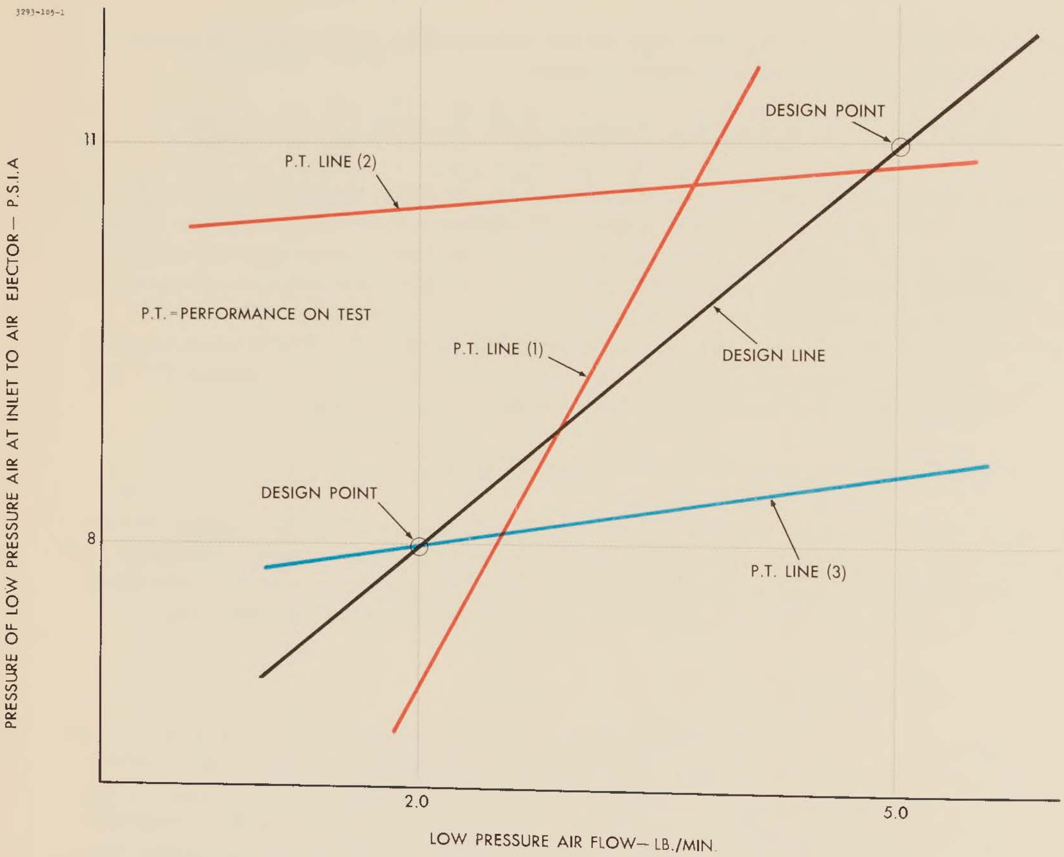

FIG. 28 AIR EJECTOR PERFORMANCE — PRESSURE VS. FLOW FOR LOW PRESSURE AIR

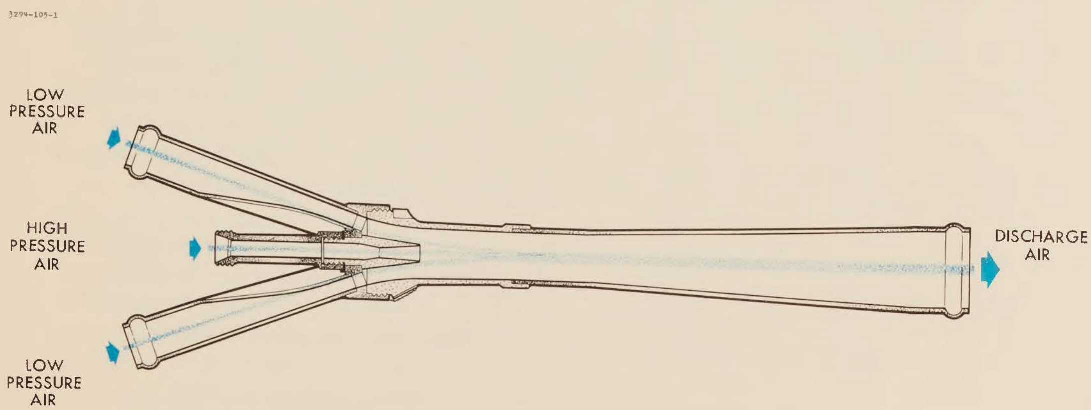

FIG. 29 AIR EJECTOR CROSS SECTION

13.3.2 COLLECTOR TANK AIR RELEASE - ARROW 2

Modification of the collector tank venting system has been necessary to accommodate the increased air release requirements which occur when a tributary tank fuel-no-air valve fails to open. Flow handling capability of the original system was inadequate for this condition because the piping friction head, as determined by pipe sizes and number of pipe bends, restricted air flow. During such an emergency, inadequate release of collector tank air increases collector tank air pressure, eliminating the pressure differential available for fuel transfer. It is considered essential that sufficient transfer capability be retained to allow use of the fuel in the affected system during the return phase of a mission. All pipes to the air ejector Figure 30 have been increased to 1.0 inch diameter.

The aft outboard air release valve has been repositioned inboard of rib No. 2. This makes it effective over a wider range of aircraft attitudes, maintaining the fuel level in the collector tank above the level at which the cockpit warning light is illuminated. The air release valves, as defined by the specification, are capable of adequately handling the larger air flows occurring dueing the condition noted.

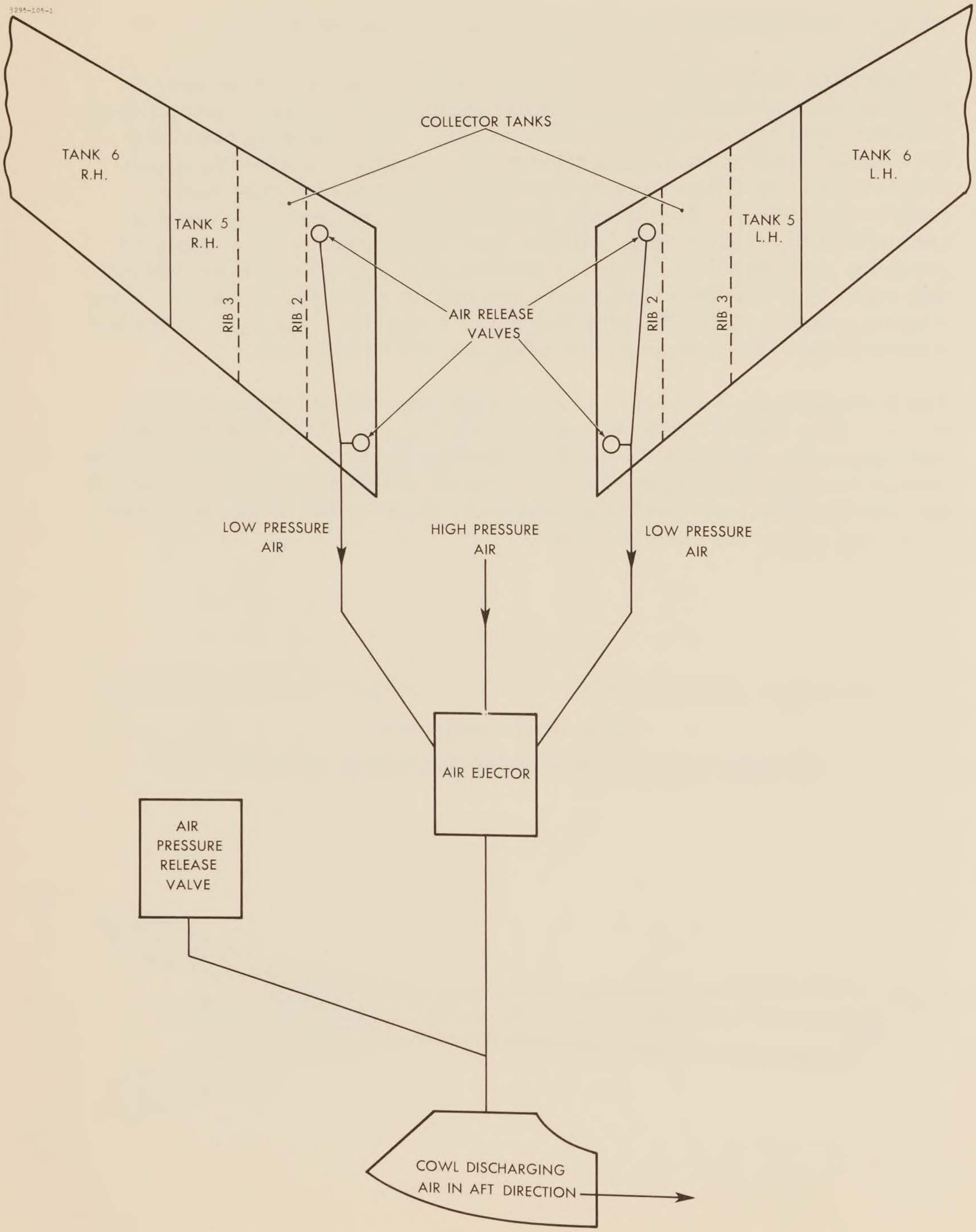

FIG. 30 SIMPLE SCHEMATIC – COLLECTOR TANK AIR RELEASE – ARROW 2

14.0 HYDRAULIC SYSTEMS

14.1 POWER SYSTEM DEVELOPMENT

The modification of the ARROW 2 hydraulic system, to allow the ASTRA I radar antenna to be driven from the utility hydraulics power system, has been further studied. Consideration of system complexity, weight, redesign and testing time, and vulnerability to failure has prompted AVRO to request the RCAF to reconsider this requirement.

14.2 NOSE LANDING GEAR DOOR OPERATION - ARROW 1

The ARROW 1 nose gear door is now automatically closed, following extension of the landing gear, to improve directional stability during approach and landing. Since the inception of the door reclosing scheme, the following changes have been introduced:

The nose gear scissors switch no longer opens the nose gear door at touch-down. The electrical circuit has been revised to incorporate a ground servicing switch, labelled OPEN - OFF - CLOSE. This is selected by the ground crew, after the aircraft has landed, to open the nose gear door. The same switch can be used to retract the door before take-off, should the pilot wish to take-off with the door closed. To avoid hazard to ground personnel, the door, although selected up by the ground servicing switch, will not close until after the engines are started.

14.2.1 SEQUENCE OF OPERATION

The operating sequence for the nose gear door will now be as follows:

1. On selection of gear DOWN, the nose gear door opens and the gear is extended by the operation of a nose-door-actuated hydraulic 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, thus closing the door.

3. If the aircraft overshoots without landing, a landing gear UP selection signals the nose door selector valve, allowing hydraulic pressure to open the door. When the door is fully open, a "door fully open" limit switch is tripped, allowing the gear retraction system to operate in the normal manner.

4. After landing and when the aircraft has come to rest, the nose door may be opened by means of the ground servicing switch, located in the nose gear bay. This should be done while hydraulic pressure is still

available, to avoid unexpected door operation if pressure is applied after the selection has been made. The switch will be labelled thus:

"To open door: Utility hydraulic pressure must be available before switch is selected momentarily to OPEN".

5. When it is desired to take-off with the door open, the ground servicing switch is left in the OFF position and the landing gear UP selection retracts the gear in the normal manner.

6. For take-off with the door closed, the ground servicing switch is selected by ground crew to CLOSE. The switch will be labelled thus:

"To close door before take-off: Pre-select switch to CLOSE. Door will close at start-up".

When the aircraft is airborne, the landing gear UP selection reopens the door. When the door is fully open, the limit switch is tripped, allowing the gear to retract in the normal manner.

7. Power is disconnected when all doors are closed, but is re-instated if a door is not fully locked in the closed position. The nose landing gear indicator shows UP, only when both the gear and the door are locked up.

8. Emergency extension of the landing gear is still achieved as described in the previous Quarterly Technical Report. The hydraulic system remains unaltered by the above changes to the electrical sequencing.

14.3 NOSE GEAR DOOR OPERATION - ARROW 2

Should flight trials prove the door-closing requirement, it may be possible to introduce a simpler, more reliable system for ARROW 2, than the electrically sequenced system used in ARROW 1. An investigation has been conducted to devise a mechanically-sequenced nose gear hydraulic circuit. The requirements of the system are:

(a) It should close the nose door immediately after extension of the landing gear.

(b) It should be possible to open the door after the aircraft has landed.

(c) It should be possible to reverse a gear selection before it is complete.

(d) For safety reasons it should not be possible for the door to close on the ground after a door open selection has been made.

Equipment to be deleted from the present hydraulic system would comprise the nose door selector valve and emergency bypass valve. New equipment required for mechanical sequencing consists of a mechanical door selector valve, a ground actuated door selector valve, an additional (door actuated) sequence valve, and a bypass sequence valve (refer to Figure 31).

14.3.1 SEQUENCE OF OPERATION

The actuation of the proposed system is arranged to achieve the same sequence of operations as in the ARROW 1, except that the nose gear door cannot be closed on the ground after it has been opened. The sequence of operations is as follows: (Refer to hydraulic circuit, Figure 31).

1. During flight, the mechanical selector valve and the ground operated selector valve (figures 32 and 33 respectively) are both in position 1.

2. On selection of gear DOWN, the nose gear door opens and the gear is extended when the nose gear actuates the sequence valves marked * in Figure 31.

3. When fully extended, the gear actuates the mechanical selector valve to position 2, and the door closes.

4. After landing, the ground operated selector valve is moved by hand to position 2. This allows the door to open.

5. After take-off, the landing gear UP selection is made. As the nose gear retracts, the following actions occur in turn:

   (a) The bypass sequence valve is actuated to maintain the pressure supply to the nose gear jack, if the door actuates the two sequence valves (*) before the gear is locked up.

   (b) The mechanical selector valve takes up position (3) (neutral).

   (c) The ground actuated selector valve reverts to position (1).

   (d) When the gear is fully retracted, the mechanical selector valve takes up position (1), and the door closes.

6. When the landing gear is selected down, the mechanical selector valve takes up position (3) when the gear is falling, and the ground actuated valve remains in position (1). If the selection is reversed before extension is complete the gear will retract as described in 5 above.

7. With the gear extended and the nose door reclosed, the mechanical and ground actuated selector valves are in positions (2) and (1) respectively.

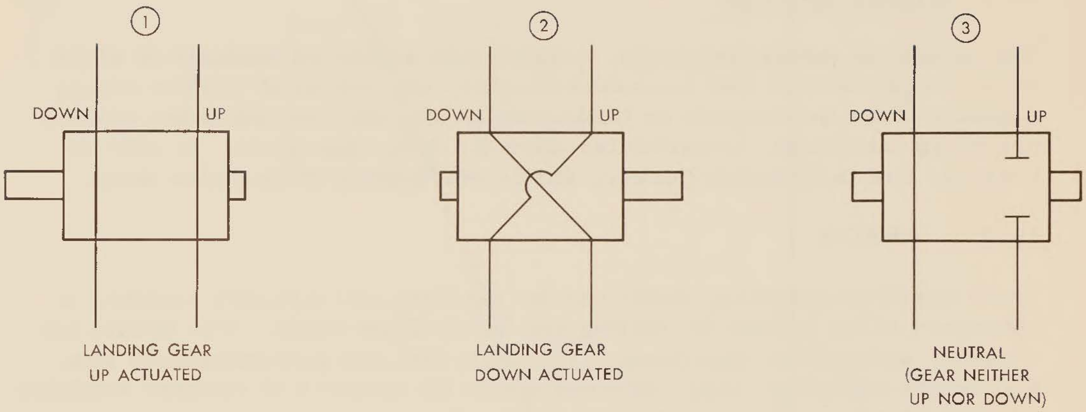

FIG. 32 MECHANICAL SELECTOR VALVE—NOSE LANDING GEAR DOOR

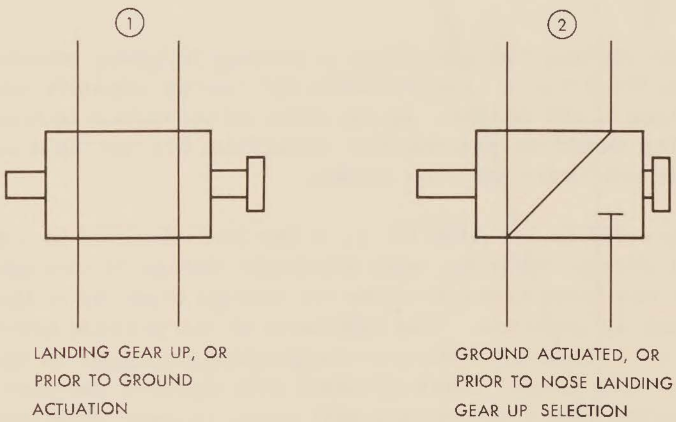

FIG. 33 GROUND OPERATED SELECTOR VALVE—NOSE LANDING GEAR DOOR

Should an UP selection now be made, the door will open, actuating the two sequence valves (*) and allowing the nose gear to retract as in 6 above.

14.4 WHEEL BRAKES

The results of initial taxi trials, coupled with a detailed analysis of wheel brake requirements and increased weights, has indicated that the energy capacity of the brakes may be inadequate to stop the aircraft under certain operating conditions, in particular when the drag chute fails. In addition, brake or tire failures may occur, due to overheating of the brake units.

14.4.1 ARROW 1

Tests based on operating conditions for the first aircraft have resulted in clearance of the brakes for taxiing and initial flight trials. The brakes are capable, under these conditions, of a single full energy (emergency) stop without the use of the drag chute and within the confines of runways available for flight testing. The brakes have to be changed, however, after each full energy stop.

The curve of brake lining temperature vs energy input for the existing brake (Figure 34) illustrates the overheating problem for the energy levels expected.

Temperatures in excess of approximately 700°F at the brake linings are detrimental to the equipment. Temperatures in excess of 300°F at the tire bead seat, due to radiation from the brakes can cause tire failures after landing.

Analysis of the landing run and range of landing weights, speeds and attitudes has shown that the present requirements for energy capacity cannot be met with the existing brake design. In the event of parachute failure a, single emergency stop would exceed the safe maximum energy input of 11×10^6 ft lb per brake unit and "burn out" the brakes.

As an interim solution for ARROW 1, it has been decided to trade brake life for increased energy capacity, with minimum change in configuration. Goodyear Aircraft has been asked to raise the energy capacity of the existing brakes as much as possible. The thickness of the present brake discs will be increased by 0.25 inch, with a corresponding decrease in the thickness of the linings. The larger mass of metal will improve the heat sink but at the same time the life of the linings will be decreased, thus reducing the number of stops available from each set of linings. Expected performance of the modified units will be established by tests as follows:

(a) 25 stops at a normal energy input of 7.5×10^6 ft lb per brake unit.

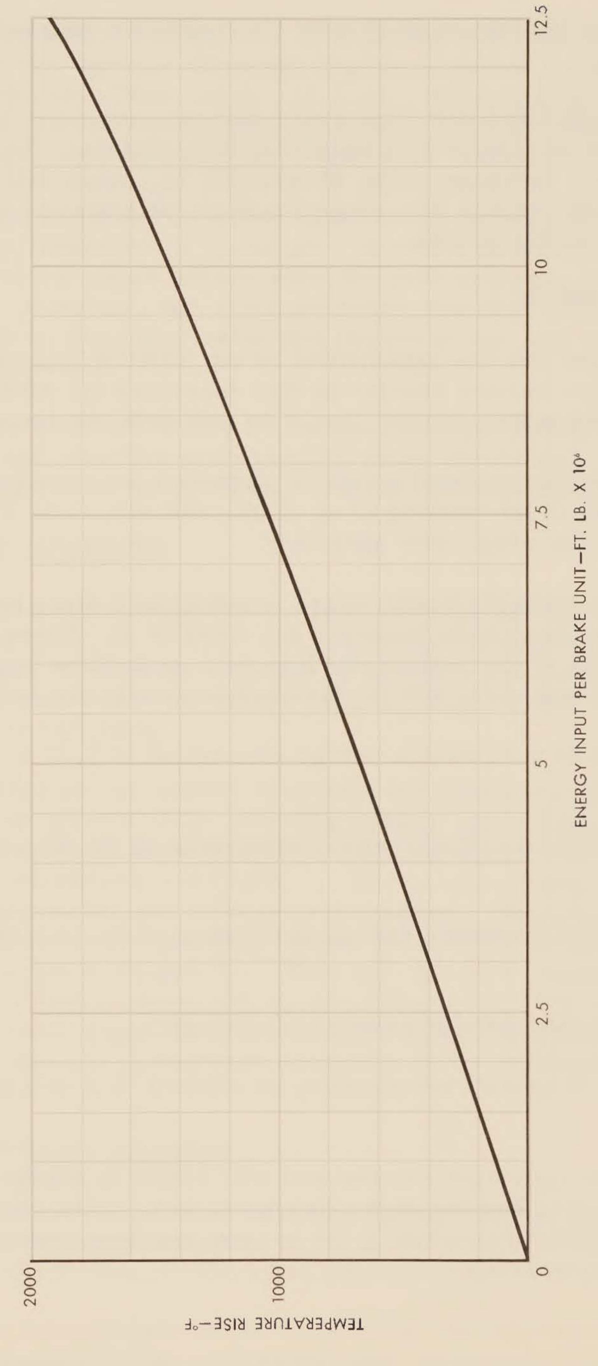

FIG. 34 TEMPERATURE RISE AT WHEEL BRAKE LINING VS. ENERGY INPUT PER BRAKE UNIT

(b) 2 stops at a high energy input of 12×10^6 ft lb per brake unit.

(c) A series of tests to determine the maximum energy the brakes can absorb.

A problem associated with high brake unit temperatures is tire failure after taxiing, due to excessive tire bead seat temperature. To overcome this problem, the inner surfaces of the wheels will be coated with a heat reflecting paint. This will reduce the temperature of the wheel by protecting it from heat radiated by the brakes.

14.4.2 ARROW 2

An investigation into the capabilities of the ARROW 2 brakes has revealed that the position is very similar to that described for ARROW 1. The energy capacity requirements cannot be met with the existing brake units.

Maximum safe energy input would be exceeded for a single emergency stop without the use of the braking drag chute, and only 12 to 15 stops could be achieved with the drag chute operating.

Multi-disc tri-metallic brake units, in which both discs and linings are used as a heat sink, have been proposed for ARROW 2. These will bring the energy capacity of the brakes into line with up-to-date landing speeds and weights. The new units will be designed to satisfy the following requirements:

(a) 45 stops at a minimum energy absorption of 9.25×10^6 ft lb per brake unit.

(b) 5 stops at a minimum energy absorption of 12.75×10^6 ft lb per brake unit.

(c) 1 stop at a minimum energy absorption of 22.25×10^6 ft lb per brake unit.

(d) A maximum brake pressure of 1,500 psi.

If the scheme is proven satisfactory on ARROW 2 it will be retrofitted to ARROW 1.

Adoption of the redesigned brake unit will result in greater energy capacity and therefore less heating of the tire bead seat. This will maintain bead seat temperatures below 300°F for normal and high energy stops, thereby preventing tire failures.

Another suggestion being investigated is the replacement of the existing type VII tire with the new type N low profile tire, which apparently has better high speed characteristics.

14.5 SPEED BRAKES

It is possible that the speed brakes may be deployed when the missiles are fired. In this case the Sparrow 2D would impose a load of 9,700 lb. on the fully extended speed brake (Ref. para. 17.3.2). This missile blast load, added to the normal air load would produce a hinge moment considerably in excess of the normal value required for "blow back" retraction (Ref. Figs.35 and 36). Although the blast load would be of short duration, it would nevertheless produce a pressure surge in the speed brake jack up to 7,000 psi in excess of the design maximum of 5,000 psi. A relief valve would normally allow "blow back" of the speed brakes when the jack pressure exceeds a nominal 5,000 psi. However, the rapid pressure rise due to missile blast could not be handled by the relief valve and failure of the jack or the supporting structure could result.

To obviate the above problem it may be necessary to partially retract the speed brakes when the missiles are extended ready for firing. The structural and system aspects of the problems are currently under investigation.

14.6 NOSEWHEEL STEERING

The present mechanically-operated nosewheel steering system was found to be unsatisfactory on the first aircraft and has been disconnected. High friction in the system resulted in difficulty in steering the aircraft, and produced an undesirable effect on rudder centring and rudder control, with the nose landing gear extended.

It is proposed to redesign the system to reduce the operating forces and to eliminate the adverse effects of the steering linkage upon rudder operation. Efforts will be concentrated on developing a suitable system for ARROW 2, which can later be retrofitted to ARROW 1. Electrical control will be used in place of the present cable and linkage arrangement. Steering control will be non-linear, to obtain corresponding rudder pedal and nosewheel deflections of the order shown in Figure 37. This will prevent oversteering at high speeds (i.e. landing or take-off) by providing a 2:1 ratio of rudder pedal angle to nosewheel angle, for pedal angles up to 7 degrees. Various types of electrical steering system are presently being evaluated to determine their suitability.

14.7 ANTI-SPIN WHEEL BRAKING

The main wheels are prevented from spinning during landing gear retraction by automatic application of the brakes. This is achieved by means of an electrical signal to the brake control valve solenoid when the landing gear up selection is made.

With the present system there is a risk of brakes-on landings should a fault

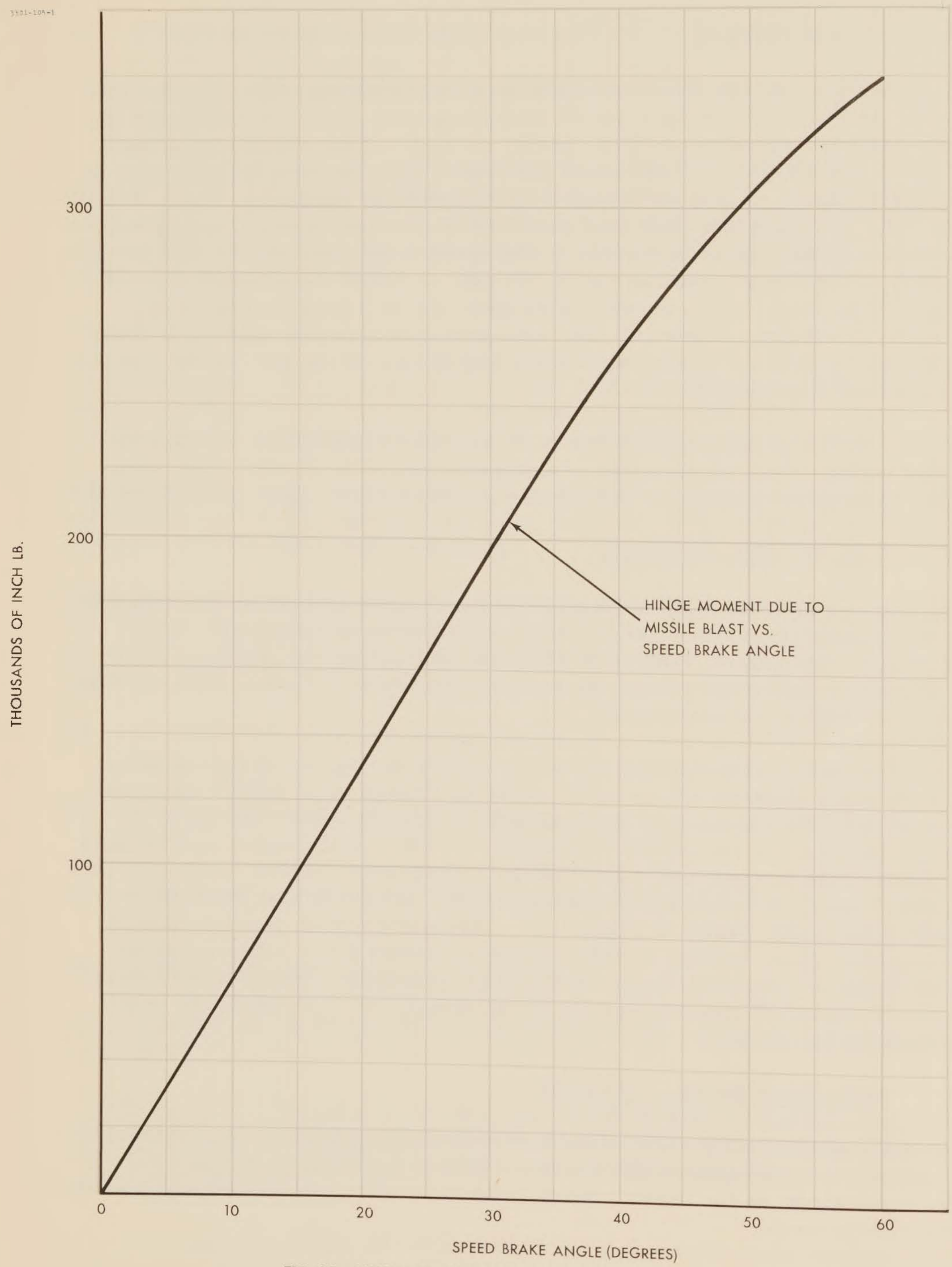

FIG. 35 MISSILE BLAST EFFECT ON ONE SPEED BRAKE

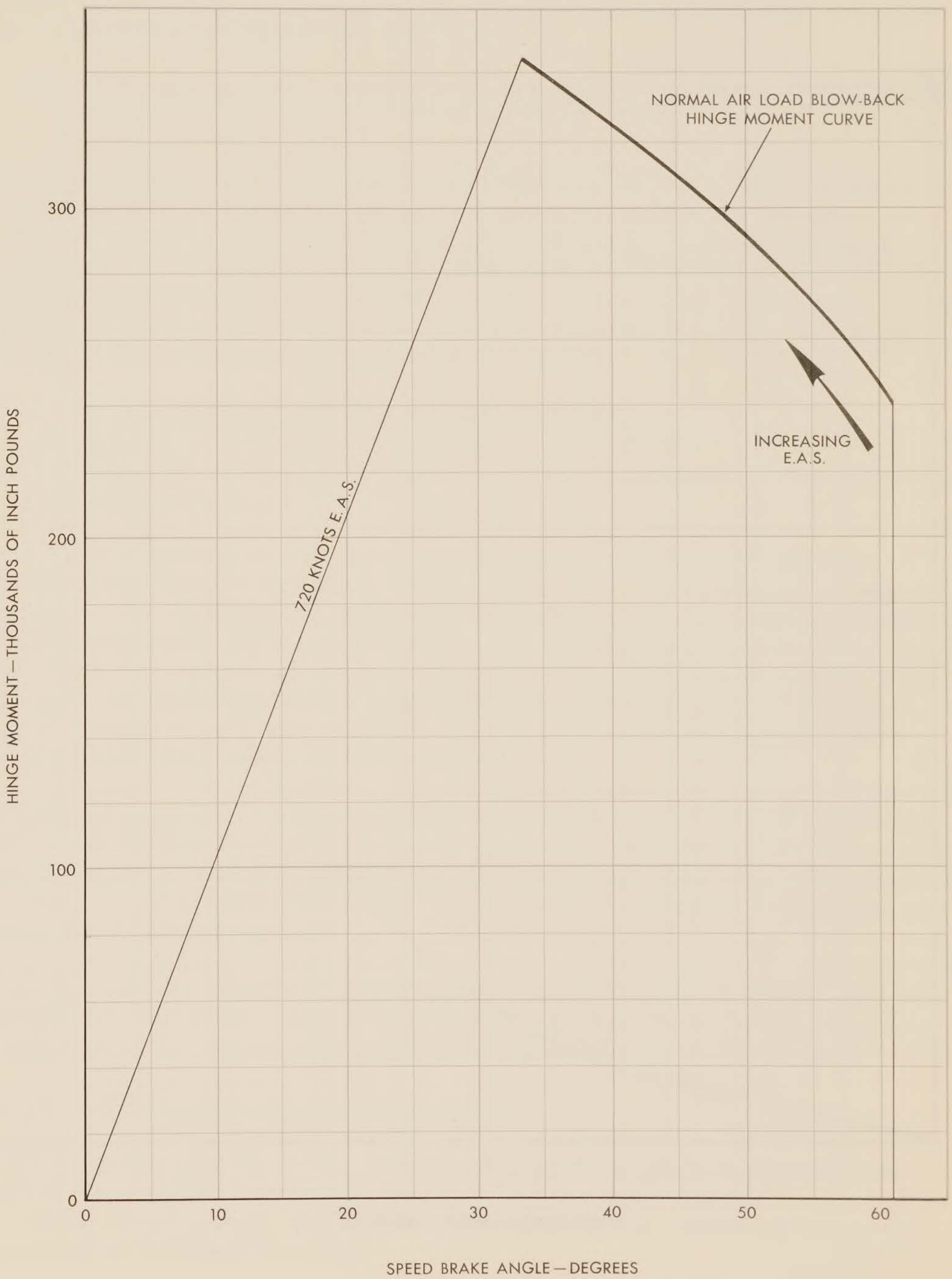

FIG. 36 NORMAL AIR LOAD BLOW-BACK HINGE MOMENT ENVELOPE

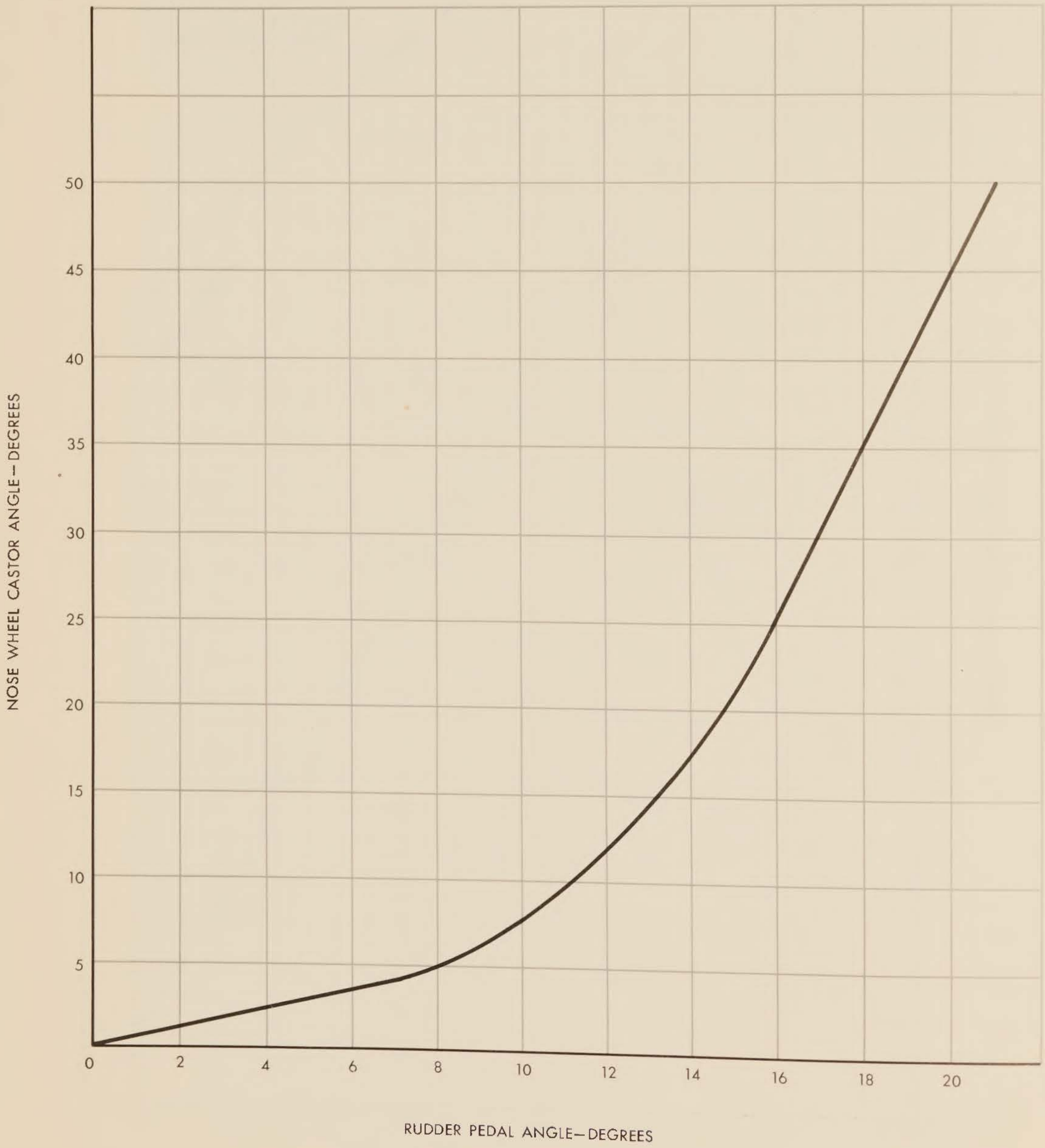

FIG. 37 NOSE WHEEL STEERING CONTROL CHARACTERISTICS

occur in the control valve solenoid. To eliminate this possibility, the electrical valve actuation is being replaced with hydraulic actuation; pressure being applied when GEAR UP is selected.

14.8 FLYING CONTROL SYSTEM-INPUT BOOSTERS (Refer Quarterly Report No. 2)

The hydraulic control circuit for the elevator and aileron input boosters in the flying control system is shown in Figure 38. Hydraulic pressure is supplied to each booster from both the A and B flying controls hydraulic systems simultaneously. In the event of a failure in either hydraulic system, the hydraulic fluid in the failed system is circulated around the booster, through a check valve between the pressure and return lines. The booster installation is further discussed in para. 15.1.1.

14.9 DEVELOPMENT OF FLARELESS FITTINGS

All leakage, burst and impulse tests have been completed for the initial flightworthiness program.

The flexural vibration tests are still incomplete due to difficulty in evolving satisfactory tooling to size the tube end. This tooling is required to apply the sleeve shrink-fit method of assembly to give results comparable to the NRC method. AVRO tooling has so far produced tube ends with a fatigue life of 2×10^6 cycles, as compared to 2×10^7 cycles produced by the NRC method. Although the fatigue life achieved with AVRO tooling is a great improvement on previous results, it is still not considered sufficient to justify proceeding with production tooling.

A batch of unhardened sleeves has been obtained from Weatherhead Canada Ltd., and attempts are being made to pregroove the tube, preset the sleeve into the groove, and then expand the sleeve and tube together. This method is based on successful NRC tests, during which unhardened MS sleeves and tubes were swaged together.

A series of tests have been completed to assess the life of swaged-tail tube assemblies using the constant displacement method of flexure (as standard in the U.S.A.). The results are now being evaluated.

14.10 HIGH PRESSURE STEEL TUBING

A new specification (M-7-14) has been compiled to replace specification M-7-6 for ARROW 2 hydraulic tubing. The new specification requires that all tubing be seamless, and a close tolerance internal diameter has been specified to facilitate forming.

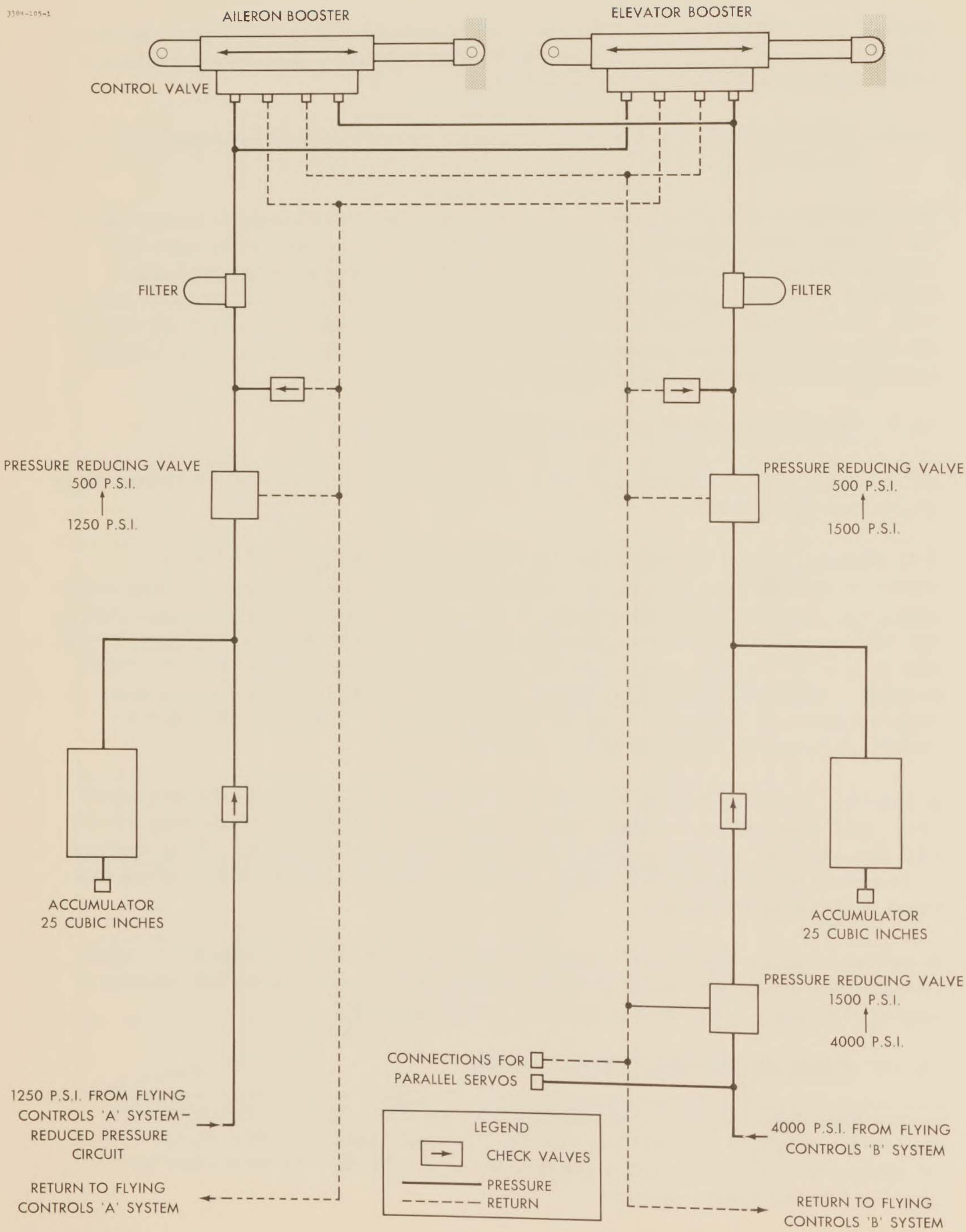

FIG. 38 HYDRAULIC CIRCUIT—FLYING CONTROLS SYSTEM INPUT BOOSTERS

15.0 FLYING CONTROLS AND DAMPER SYSTEM

15.1 FLYING CONTROL SYSTEM

With the completion of qualification tests and the correction of faults applicable to aircraft 25201, the first ARROW was cleared for the first flights within a flight envelope of 40,000 feet and Mach 1.4 or 450 knots EAS, whichever was lower.

Continued tests on the B-1 rig uncovered several faults in the system. Slackening of rudder calbes,qc actuator system unreliability, control valve instability problems and high frictional forces in the feel and trim units were among the faults which required correction. Also, with the incorporation of boosters in the elevator and aileron control systems, several additional system changes were necessary. These included modifications to the front tension regulator quadrants and the relocation of the feel and trim units and parallel servos, at the front quadrant areas.

15.1.1 CONTROL BOOSTERS

As stated in the previous issue of the ARROW Quarterly Technical Report, the unacceptable breakout forces in the elevator and aileron systems were eliminated by the installation of boosters. These are located at the front quadrant area, together with the relocated feel and trim units and parallel servos.

When the system was checked for components functioning, it was found that the booster control valve exhibited unstable characteristics. The arrangement of the booster jack and booster valve levers is such that a follow up process occurs when the control valve lever moves to initiate movement of the booster jack. The free motion of the control valve ram and operating linkage, produced the instability.

This fault was partly corrected by installing a rate spring, between the two booster levers to provide a centring action. In addition, a damper has been fitted on the control valve linkage which further reduced the problem to acceptable limits. The rate spring and damper were then combined to form a damper/feel unit and installed on the elevator and aileron booster jack linkages as shown in Figure 39. The boosters will be installed on all ARROW 1 aircraft, although the installation is complicated on aircraft 25204 and 25205 by the addition of ASTRA I.

15.1.2 CONTROL STICK GRIP

The stick force transducer in the Humphrey grip was proven to be unflightworthy (Ref. 15.2.1). The grip was installed on aircraft 25201, but the transducer was inoperative during flight. Subsequent ARROW 1 aircraft will

employ Humphrey grips with isolated transducer circuits, if the flightworthy Humphrey transducer type grips are not available.

15.1.3 FEEL AND TRIM UNITS

The elevator and aileron feel and trim units produced high frictional forces. These have been reduced to an acceptable level by the removal of felt dust shields at the ram end of each unit.

The booster installation (Ref. para. 15.1.1) necessitated the relocation of the feel and trim units at the front quadrants. In addition, the elevator feel and trim unit was shortened. This was necessary to fit it into the available space and to reduce excessive feel and trim unit deflection incurred at resonant frequencies, which could result in failure of the unit.

An electrical emergency release has been incorporated with the elevator feel and trim unit installation. The electrical disconnect allows the pilot to disengage the unit should it jam or "runway". This is required for the ARROW 1 since a large portion of flying will be done in the emergency mode, and trim motor runaway could be disastrous.

15.1.4 CONTROL VALVES

As mentioned in the previous issue of the ARROW Quarterly Technical Report, instability problems existed on the control valves. Modifications were made to the valves to correct the flow force characteristics but they were still considered unacceptable. An investigation was therefore conducted to determine if suitable damping could be applied to the valve.

Two types of dampers were tested; a Humphrey viscous damper and an AVRO orifice damper. The AVRO damper proved to be the most acceptable. Damping was linear and no fall-off in linearity was observed during 30,000 cycles of test operation. When installed on the flying controls test rig actuator control valves, the damper was sufficient to cure the oscillation of the aileron and elevator control valves and subsequently the rudder control valve.

AVRO dampers have been fitted to all five control valves on aircraft 25201, although the damper installation is different on the rudder, elevator and aileron control valves. In each case however the damper acts directly on the valve spool linkage.

15.1.5 RUDDER CABLE TENSIONER

During tests on the rudder control system, it was found that when maximum force was applied to the rudder pedal, the off-tension side of the control cable run showed about 2.5 inches of slack. This would result in about ten inches of slack per six foot of cable run. In order to eliminate this condition, a rudder cable tensioner has been installed in each cable run.

The rudder cable tensioner consists of a spring enclosed in a tube, with a plunger acting to contract the spring. Suitable cable connections are provided at each end of the unit on the tube and plunger. Tension applied in the cable run contracts the spring through three inches of travel unitl the stop on the plunger contacts the end of the tube.

The three inch range of the unit provides ample length for the removal of slack, and the spring bottoms at a force of approximately 32 lb, providing sufficient tension in the off-tension side of the cable run.

Sponginess is eliminated in the on-tension side of the cable run, since the stop is reached at approximately 50 lb tension and the maximum cable tension is approximately 225 lb. Rudder cable tensioners have been installed on aircraft 25201.

15.1.6 RUDDER TENSION REGULATOR QUADRANT

The rudder cable tension regulator quadrant was required to maintain 50 lb tension. However, when the quadrant had been set, and high forces applied to one cable run, the quadrant would slip to the extent of slackening the off-tension side of the cable run. The rudder tension regulator quadrant on aircraft 25201 has now been modified by the addition of stops which prevents the quadrant from slipping beyond the desired tension setting.

15.1.7 Qc ACTUATOR SYSTEM

Qualification testing has begun on the AiResearch qc actuator system. However, repeated electrical failures of the magnetic amplifier indicated a definite lack of reliability. Failure of the unit was believed to have resulted from poor insulation of the excitation circuit components, leading to heat dissipation problems within the unit.

In order to ensure the availability of a flightworthy system for aircraft 25201, it was decided that AVRO would develop a system which could be used as an alternative to the AiResearch system, should the faults not be corrected in time for the first flight. The simulation laboratories therefore developed a system based on a CF-100 vacuum tube type amplifier, and an additional chassis which included a phase compensation network, a power supply to feed the amplifier, and a system gain control. Two sets were assembled, passed qualification tests, and one set was installed on aircraft 25201.

During vibration tests, the qc actuator, which until this time had functioned satisfactorily, became inoperative due to malfunctioning of a limit switch. This could result in the actuator motor burning out at either extreme of its travel, thus rendering both the normal system and the emergency override inoperative. The limit switches triggering mechanism was modified after which the actuator successfully underwent vibration tests. The qc actuator system was then made fully operative in time for the first flight. Some overheating of the actuator motor has since been noted but the cause of this has not yet been determined.

For increased flight safety on ARROW 1 aircraft, the emergency override is incorporated and is independent of the normal qc actuator system. An override switch on the left-hand side of the pilot's main instrument panel is used to disengage the normal qc actuator system and select the rudder feel mechanism. A three-position switch then enables the pilot to select the amount of hinge moment limitation or feel required for the particular airspeed. A qc actuator position indicator, calibrated in knots EAS, is provided to show the position of the actuator. Consideration is being given to the incorporation of the pilot override system on the ARROW 2.

As a result of the design problems encountered by AiResearch, and the standard of reliability required of the qc actuator system, the present AiResearch system will not be used on the ARROW 2.

15.2 DAMPER SYSTEM - AIRCRAFT 25201

For the first flights of aircraft 25201, the emergency damping system was operative, but, the normal damping system was operative in the yaw axis only. This was necessary since a flightworthy stick force transducer and g limiter were not available. The g limiter, which operates in conjunction with the normal pitch damper will employ transistorized servo accelerometers (ref. para. 15.2.3). The first g limiter is expected to be delivered in June 1958.

The AFCS switch in the control stick grip has been wired as a damper POWER-OFF switch on aircraft 25201 for flight test purposes. This safety feature will also be incorporated on aircraft 25202 and 25203.

Another safety feature was incorporated which enables the pilot to select landing gear down mode while the landing gear is up, and landing gear up mode while the landing gear is down.

During taxi trials a divergent oscillation of the rudder was experienced. The cause was traced to a structural torsional mode which excited the normal and emergency lateral accelerometers located in the armament bay roof. The accelerometer mounts were found to resonate at 8.5 cps, the approximate frequency of the structural torsional mode, and at the nominal flight settings, the accelerometer gain was high enough to cause the unstable condition. An attempt to make the mounts more rigid had a relatively small effect in reducing the oscillation. The condition was relieved for the first flight by changing the forward loop gain and the servo loop gain. However, a filter has now been incorporated.

15.2.1 STICK FORCE TRANSDUCER

Initial tests conducted on the stick force transducer revealed that the output voltage was non-linear with the applied stick force and that the AFCS override device, while not required for aircraft 25201, was inconsistent in operation. The tendency of the AFCS override device to remain disengaged after application of loads was considered unacceptable.

Analog computing has produced a rearrangement of system elements which eliminate stick buzz due to the parallel servo. Work is continuing in an attempt to establish a satisfactory force-displacement curve to produce a good break-out feel, and also to eliminate a "knock" believed to be due to the parallel servo first overshoot. The Bell and Humphrey stick force transducers are being investigated. A stick buzz is also produced by the differential servo. This has not been checked thoroughly as yet, but it is thought that the booster installations will eliminate this condition. This will be checked out on the flying controls test rig.

15.2.2 RUDDER MONITOR

The rudder monitor is used to protect the aircraft against malfunction of the normal damping system by switching over to the emergency damping system, should transverse acceleration or sideslip approach structural integrity limits. The switching mechanism operates on all three axes and is shown in Figure 40.

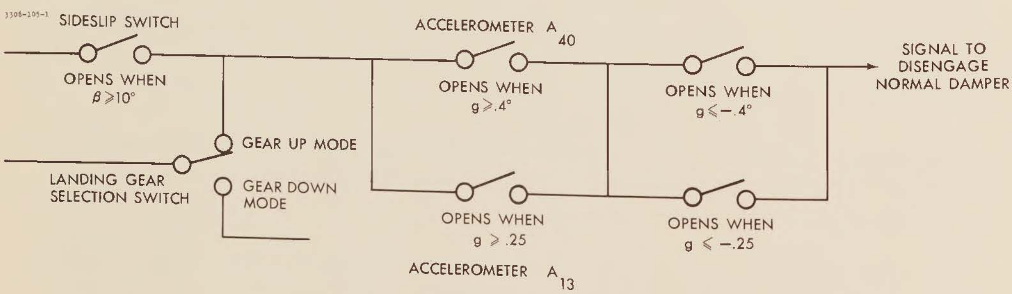

FIG. 40 RUDDER MONITOR CIRCUIT

The circuit includes a sideslip switch (beta vane in the nose boom) and two transverse accelerometers which are located approximately 13 feet and 40 feet, respectively, forward of the aircraft centre of gravity. The sideslip switch is set to open and interrupt the normal damping system supply if the sideslip angle exceeds ten degrees. The landing gear mode selection switch allows selection to landing gear down mode on low speed configuration to permit the pilot to produce intentional sideslip.

Switching to the emergency mode will occur only when both accelerometers register accelerations larger or equal to values shown in Figure 40. The required combinations of accelerations to cause disengagement is only possible in the case of rudder runaway.

Analytical work was conducted on the rudder monitor for the flight envelope of aircraft 25201 for the first flights. The results of this work showed that better protection could be obtained by disconnecting accelerometer A_{13} from the circuit. For the first flights of 25201, accelerometer A_{13} has therefore been eliminated from the rudder monitor circuitry.

15.2.3 G LIMITER

The function of the g limiter was described in paragraph 4.0 of the previous Quarterly Technical Report. This system employed two spring mass balanced accelerometers.

Honeywell have now proposed a new g limiter which operates in exactly the same manner as the former system, but employs transistorized servo accelerometers. These accelerometers are installed in two sets of two, each set being packaged separately. This type of accelerometer was proposed as a means of reducing the system error which at best was ± .6g when employing matched spring accelerometers. The system error is reduced to ± .3g when the servo accelerometers are used.

16.0 OXYGEN SYSTEM

16.1 OXYGEN SYSTEM ARROW 1

The ARROW 1 aircraft is equipped with a shock-mounted ARO oxygen converter, which has a gaseous flow of 40 litres per minute. An emergency supply cylinder is mounted under each seat, with high pressure (1800 psi) piping to a charging valve, and a pressure reducing valve.

The low pressure pipe from the liquid oxygen converter passes through the bulkhead, adjacent to the cockpit entrance step. This has been considered acceptable for ARROW 1 aircraft, in view of structural alterations required to reroute the pipe. On ARROW 2 aircraft, the pipe has been rerouted to clear the entrance step.

16.1.1 OXYGEN CONTENTS INDICATOR

The master oxygen contents indicator on ARROW 1 aircraft is located in the front cockpit, with a repeat indicator in the rear cockpit. Each indicator is marked 0 to 5 litres of liquid oxygen. In the event of electrical power failure, a warning flag, marked OFF, appears on each indicator dial.

16.2 OXYGEN SYSTEM ARROW 2

ARROW 2 aircraft will be fitted with a Bendix oxygen converter. This unit has a gaseous flow of 60 litres per minute.

Improvements have been made in the emergency system design. The supply cylinder pressure gauge charging and starting valves, and the pressure reducing valve have been incorporated into one unit, together with the system actuating handle. The emergency system is now completely self contained, eliminating the interconnecting high pressure pipes. A complete unit is to be installed on the back of each seat.

16.2.1 OXYGEN CONTENTS INDICATOR

On ARROW 2 aircraft, the oxygen contents indicators are marked in percentage of liquid oxygen available, and the master indicator will be equipped with a test switch to check that the indicator is functioning. This is in addition to the OFF flags provided on each indicator.

17.0 ARMAMENT SYSTEM

17.1 GENERAL

The ARROW 2 operational aircraft is equipped with four Sparrow 2D air-to-air guided missiles.

17.2 ARROW 1

ARROW 1 aircraft, with the exception of aircraft 25203, do not incorporate missile carrying facilities. Aircraft 25203 can carry four modified Sparrow 2D missiles. These missiles will not contain any electronic guidance or war-head and are known as simulated aerodynamic vehicles (SAV). Their use on this aircraft is to check the missile lowering and retraction gear, jettison, firing of missiles and initial launch projectory. Photographic coverage of missile launching and jettison, will be obtained by cameras mounted on both the launch and chase aircraft.

At a meeting with AVRO on 7 February 1958, Canadair Aircraft Ltd. suggested that they would like to fire Control Test Vehicles (CTV) early in 1960. The only ARROW aircraft that will be available at that time, is aircraft 25203 which would be used for weapon pack development during that period. AVRO will consider their request when planning the activities of this aircraft. The necessary altitude of missile operation, could be obtained by launching from a CF-100 in a nose-up attitude. No final decisions have yet been made on this subject.

17.3 ARROW 2

The production version of the ARROW 2 will be equipped with an ASTRA I electronic system, and will carry four Sparrow 2D air-to-air guided missiles. However, early ARROW 2 aircraft (25260 - 25208) will be equipped with a partial ASTRA I system only and will not carry missiles. The weapon pack bay will be used to carry instrumentation. Aircraft 25209 will be a missile-carrying aircraft equipped with the developmental ASTRA I system.

The following items are currently under review:

1. Elasticity effect on antenna line of sight
2. Installation of Infra-Red system - (see para. 7.0)
3. Dynamic study of the missile launcher
4. Missile firing with extended air brakes
5. Firing circuits

6. Interference between missile and rail

17.3.1 RIGIDITY EFFECT ON ANTENNA LINE OF SIGHT

The aircraft antenna and missile antenna lines of sight are parallel within acceptable limits on a static check. Due to structural deflection however, these limits may become unacceptable during flight.

The magnitude of the deflections and their effect on the lines of sight is now being investigated. Flight tests will also be made on aircraft 25203 to measure the deflections between the aircraft and missile antenna sight lines.

17.3.2 FIRING CIRCUITS

General agreement has been reached between the RCAF, AVRO and RCA, regarding the firing circuits for ARROW 2 aircraft. These agreements were recorded in paragraph 16.1.4 of the previous Quarterly Technical Report. RCA is now preparing a revised statement of the circuitry so that complete agreement may be established.

17.3.3 INTERFERENCE BETWEEN MISSILE AND RAIL

Preliminary investigations indicate that when all adverse tolerances and deflections are taken into account, it is possible for a foul to exist between the missile body and launcher rails. The magnitude of the interference has not yet been established, although the nominal clearance is 0.26 inches. Due to the nature of the deflections involved, it is not possible to immediately arrive at a nominal clearance to prevent fouling. A design scheme is being prepared to restrict those deflections that tend to reduce the clearance between the missile and missile launcher. This will be achieved by fitting a pair of small diameter rollers on the launcher so that under static conditions they are located just above the missile body, at missile station 47.9. The function of these rollers is to prevent the missile attachment buttons from jamming in the launcher rails during launching. The rollers will be automatically retracted by the action of the missile button on passing under the rollers.

Side deflection of the launcher and missile will tend to prevent an extended missile from being retracted back into its stowed location within the weapon pack. To prevent fouling of the missile and sealing edges, during retraction, a scheme is being prepared involving the use of plates to guide the missile to its correct stowed position.

17.4 MISSILES

A meeting was held between AVRO and Canadair to establish missile responsibilities. AVRO is responsible for determining and recording the environment in which the Sparrow 2D missiles must exist and function, up to the time the missile wings are unlocked. In addition, AVRO is to eliminate the environmental effects encountered by the missiles but only where this is feasible within the limitations imposed by the existing ARROW airframe configuration. Items within the missile which are not compatible with the environment of the fully developed ARROW armament system, must be rectified by the missile contractor.

Missile environment testing, planned on aircraft 25203, will be AVRO's responsibility. Instrumented missile test vehicle will be supplied by Canadair, who will be responsible for the instrumentation within vehicles. Environmental test vehicles will also be prepared by AVRO. This vehicle will be instrumented to AVRO's requirements and is AVRO's responsibility. The aircraft instrumentation will be arranged so that all telemetry channels in each missile may be read separately. At the same time certain key channels in each missile can be read simultaneously to facilitate instantaneous comparison of results from the missiles.

17.4.1 MISSILE PROTECTION

Missile skin temperatures will be approximately 250°F after being subjected to some flight conditions. The present missile configuration is not designed to withstand this temperature. In an effort to reduce the effects of this environmental condition, AVRO is investigating the possibility of fitting a jettisonable protective covering over each missile.

In addition to providing protection from a temperature rise, the missile covering will protect the missile nose.

17.4.2 UMBILICAL PLUG

AVRO are still awaiting confirmation from the RCAF that a suitable socket will be fitted to the missiles. The new socket should accommodate the plug being designed by AVRO to suit the missile launchers installed on the aircraft.

17.4.3 MISSILE WING ACTUATION TIME

Negotiations between the RCAF and AVRO are underway on the method of unlocking the missile wing controls after initial launching. In reply to AVRO's suggestion of a time delay mechanism, the RCAF has suggested that the unlocking should be controlled by a double integrating accelerometer. This would ensure missile arming at a fixed distance from the launcher, instead of a fixed time after firing as suggested by AVRO. Final agreement has not yet been reached.

17.4.4 WEAPON/INSTRUMENTATION PACK

A special pack is being designed to accommodate a combined weapon and instrumentation installation. The basic weapon installation will be the same as that designed for ARROW 2 aircraft with the exception that only two missiles will be carried, both of which will be located to one side of the aircraft centre line. The other half of the pack will be used to house the instrumentation. This pack will be interchangeable with the standard weapon pack and may be installed on any of the development aircraft.

A study of vibration and noise during missile firing is to be undertaken, and the necessary measurements will be made during the ground firing trials.

17.5 ALTERNATIVE WEAPONS

At the RCAF's request AVRO is conducting a study on the feasibility of installing GENIE rockets.

18.0 ESCAPE SYSTEM

A study of aircraft escape systems used in both the United Kingdom and United States, forms the basis of a development study intended to make the present ARROW escape system compatible with the aircraft flight capabilities.

18.1 STATIC EJECTION

Static ejection tests from both the forward and rear cockpits have been conducted at AVRO using the Martin-Baker C-5 seat. The tests indicated that the legs of the dummy occupants made contact with the instrument panels during ejection. It was felt that the use of a non-rigid type survival pack might contribute to the occupant slumping down in the seat during ejection, thus exposing him to the possibility of both spinal and leg injury.

The tests were discussed with Martin-Baker, and it was agreed that they would conduct ejection tests with dummy and live occupants in order to study leg and body reactions. These tests were conducted at the RAE Farnborough, England, and the results indicate that occupants can escape without injury when rigid type survival seat packs are used. Martin-Baker do not think that leg restraints are necessary during ejection, as even with the legs initially outstretched, the inertia force during ejection, pulls the feet back to the seat.

The RCAF Institute of Aviation Medicine is developing a rigid type survival pack. Martin-Baker tests showed that the chances of the occupant slumping down when a rigid seat pack is used are greatly reduced, thus decreasing the possibility of injury to the lower legs.

18.2 ARM AND HEAD RESTRAINTS

Martin-Baker has agreed to the RCAF request for arm and head restraint, but information on their progress is not yet available.

18.3 DROGUE CHUTE

Martin-Baker is currently redesigning the 22 inch diameter drogue chute used on the Mk 5-C seat in order to reduce the shock load due to chute opening. For test purposes, two seats have been equipped with 24 inch diameter drogue chutes and anti-squid lines. The g stop, controlling the opening of the crewman's parachute and his separation from the seat, has been reduced from 6g to 4g. The tests on these seats are to be conducted by Martin-Baker.

18.4 DUPLICATE SEAT EJECTION CARTRIDGES

Martin-Baker has agreed to investigate the RCAF request for a duplicate cartridge in the seat ejection system. Information on the progress of this investigation is not yet available.

18.5 SLED TESTING OF THE ESCAPE SYSTEM

Initial contractual authority has been received from DDP to proceed with the sled testing of the ARROW escape system. Proposals for conducting the tests have been received from the following subcontractors: Coleman Engineering, Hunter-Bristol Corporation, and Cook Research Laboratories. These proposals have been studied by AVRO and contractual negotiations are being conducted with Coleman Engineering. AVRO has proposed that the testing should be conducted at Hurricane Mesa, and preliminary negotiations are underway for the use of this track.

19.0 DRAG-CHUTE

19.1 DRAG CHUTE ARROW 1

Pre-flight tests were conducted on the drag chute brake operation. Several modifications were required on aircraft 25201 and the system was thoroughly checked during high speed taxi trials before the first flight.

An investigation will be conducted on Dacron drag chute risers employing glass sleeving. The suitability of Dacron will be established through high temperature tests, and the effectiveness of glass sleeving will be evaluated during flight testing.

19.2 DRAG CHUTE - ARROW 2

An investigation is being conducted to determine the maximum strength of the parachute consistent with the ARROW 2 structural strength. This may require an increase in parachute box volume.

20.0 STRESS ANALYSIS

20.1 THIRD STRUCTURAL MATRIX

The setting up of the third structural matrix is now in progress. Since this is a more comprehensive study than those previously conducted, it will provide an accurate check on the results of the static test on the complete aircraft. The analysis will provide a method for determining the effects of thermal stresses.

The so-called lumping method, employed on previous matrix analyses, is convenient for use with the IBM 704 computer. However, it omitted several valuable factors in the strain energy formula. A less elaborate grid was employed than that now used, in that several ribs and spars were combined for simplicity.

The analysis now underway will include the posted box inner wing, and an accurate representation of the centre fuselage frames. Further refinements of the third structural matrix include a more exact presentation of ribs 10 and 12, thus establishing a more accurate picture of the inner wing to outer wing transport joint. Also, the complete wing area is now represented, and this will enable the introduction of control surface effects.

20.2 STRESSING OF HYDRAULIC PIPING

The present work involves stress clearance of the ARROW 1 hydraulic piping. ARROW 2 piping will be considered later.

The system for determining hydraulic piping fatigue life was devised some time ago. The method accounts for pressure fluctuations and relative movements of the piping ends, and was based on actual tests to determine the physical characteristics of piping. A certain amount of fatigue trouble had been experienced on the original SAE 4130 pipes with brazed end fittings. These were subsequently replaced by stainless steel pipes with flareless fittings, thus overcoming the major difficulty.

20.3 THERMAL STRESSING

20.3.1 INVESTIGATION OF CREEP PROPERTIES OF 75-S-T6

The introduction of high skin temperatures through aerodynamic heating, demands that a thorough investigation be conducted to determine the effects of creep on the aircraft structure. A method has therefore been established for determining the effects of creep at highly stressed structural locations.

This work, produced in report 7-0500-37, Investigation of Creep Properties of 75-S-T6, considered creep generally and derived a creep law which related creep strain to temperature, stress and time. The derived law was then used to analyze typical structure cross-sections under bending and end loads, although the relationships have not as yet been verified by test results. A 'T' section with a bending moment applied was analyzed, as were a non-symmetrical section and 'T' section using both end load and bending moment. A numerical analysis of a 'T' section with an applied bending moment was then performed. These analyses did not account for the change in position of the neutral axis. However, this is of little consequence, and modifications to account for this would not impose much difficulty. The change of applied bending moment due to changes of curvature could also be incorporated in the analysis. Creep data for 75-S-T6 at 250°F was not available and this had to be extrapolated from information supplied by Alcoa.

This work served to form a basis for future work which will deal with creep buckling.

20.3.2 TEMPERATURE DISTRIBUTION THROUGH TYPICAL STRUCTURAL SPECIMENS

Two basic joint configurations were tested to determine the transient heating effects across various joint conditions under various heating rates. While serving mainly as a check against NACA data, the tests also gave some indication of the effects of using sealing compounds and primers between joint surfaces.

21.0 MOCK UP

21.1 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. The effects of these changes are being determined by AVRO, RCA, Martin-Baker Limited, and the RCAF.

21.2 STATUS OF MOCK-UP CHANGE REQUESTS

The current status of the 252 change requests are listed below:

Subject	Code	Items Not Evaluated	Change Req'st	Status: Under Initial Investi-gation	Under-going Correc-tive Action	Com-pleted	Demonstra-tion Req'd
Cockpit	A	1	62	20	19	23	5
Structure	B	2	51	10	13	28	2*
Engine Installation	C	#	17	-	1	16	#
Electrical	D	-	22	4	2	16	-
Air Cond'g	E	-	7	3	-	4	-
Low Press. Pneumatics	F	-	1	-	-	1	-
Fire Exting. System	G	-	3	-	3	-	-
De-Icing	H	-	2	-	-	2	-
Fuel System	I	-	11	2	1	8	1
Hydraulics	K	-	15	3	5	7	-
Oxygen System	L	2	5	-	4	1	2
Instruments	M	-	7	2	1	4	-
ASTRA 1	N	1	34	11	9	14	1*
Armament	O	1	15	4	5	6	1
TOTAL		7	252	59	63	130	12

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

Three items previously listed as not evaluated were approved, after demonstration, on 17 February 1958. At the same time, one other item in the same category was re-demonstrated and approved.

21.3 FUTURE DEMONSTRATIONS

The seven items not evaluated during the mock-up conference, will be demonstrated at future dates.

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	A. 24	Aircraft
Console lights	A. 8 (cat. 1)	Mock-up
Cockpit lights	A. 12 (cat. 1)	"
Map lights	A. 37 (cat. 1)	"
Intensity of light	A. 38 (cat. 1)	"
Facilities for bladder tank removal	B. 11 Pt. 2	Letter
Emergency lowering of electronic bay access door	B. 25	Letter
Facilities to remove fuel booster pump gearbox	I. 1 (cat. 1)	Mock-up
Liquid oxygen converter	L. 3	"
Seat oxygen equipment	L. 5	Aircraft
Antenna multiplexing	N. 18	RCA to clear by letter
Missile umbilical plug and rail	0. 7	Test rig

Items in the above list with their category marked in brackets e.g. (cat. 1) were changes as a result of the mock-up conference, and will be redemonstrated.

Although on actual aircraft seat was shown during the mock-up conference, it was not demonstrated. It is now proposed to demonstrate the latest type seats (Martin-Baker Mk C-5), with the seat oxygen equipment.

22.0 COMPONENT DESIGN

22.1 WING DESIGN - ARROW 1 and 2

Aileron deflection tests have been conducted in the flying controls test rig. When the aileron was operated under the full wing deflection condition, failure occurred along the aileron trailing edge. As a result of this failure, the ailerons on aircraft 25201 have been split. This involved cutting a 3/4 in. slot through each aileron from the trailing edge to just aft of the spar at rib 4A, and packing the open slot with sponge rubber.

This unexpected failure, while having an immediate affect on aircraft 25201, is under investigation to determine possible discrepancies in the test which led to the increased bending moments. The effects on future ARROW aircraft will depend upon the results of this investigation. An investigation is also being conducted to determine further requirements for splitting the other control surfaces.

The main landing gear door uplocks and surrounding structure will be redesigned for all ARROW I aircraft because of increased aerodynamic loads on the top surface. In addition, the redesign of the main landing gear door, will be investigated for the ARROW 2 to reduce manufacturing problems. A material similar to "lockfoam" will be considered for the door components.

A report on the interchangeability of the trailing edge control boxes is currently being prepared.

22.2 WING STRESS

An investigation will begin shortly on the inner wing to outer wing joint. This will serve mainly to check the expected fatigue life of the structure in this area.

22.3 RADAR NOSE DESIGN

22.3.1 RADAR NOSE DESIGN - ARROW 2

Ballast will be required in the ARROW 2 radar nose when ASTRA I is not fitted. This will consist of 600 lbs. of lead blocks mounted on a vertical shear panel which will be designed to pick up at the existing shock mount positions. An additional 300 lb ballast will be mounted on a slide rail between stations 67.5 and 96.24. Design work on this item has been completed.

22.3.2 RADAR NOSE DESIGN - ARROW 1

Present work on the radar nose concerns the provisioning for the ASTRA I electronic system in aircraft 25204 and 25205. Structural provisions for the system involve mainly the replacement of the ARROW 1 radome and radar nose structure with the corresponding ARROW 2 components.

22.4 RADAR NOSE STRESS

Present work is concerned with the radome. A strength test proposal has been received from the radome manufacturer and this is now being analyzed.

22.5 FRONT FUSELAGE DESIGN

Venting holes have been provided in the top panel of the nose wheel well on aircraft 25204 and 25205. This prevents the pressure build up which formerly existed in the nose wheel well.

22.5.1 FRONT FUSELAGE DESIGN - ARROW 2

In the interest of saving space and weight, a new transparent material is being investigated for the windshield and canopy. This new material is Sierracin 880 with sierracote demisting medium The material used at present is glass with thermapane.

An investigation is underway for the possible use of "Infaration" de-icing equipment on the ramp and intakes of the ARROW 2. This equipment, which is approximately .040 in. thick, is applied to the basic structure and fitted into the existing de-icing boot rebate. It is then covered with an .040 inch aluminum sheet.

The cockpit ducting and silencers of the ARROW 2 air conditioning system will be redesigned to reduce the cockpit noise level. This change will be effected on aircraft 25206 and subsequent aircraft

22.6 FRONT FUSELAGE STRESS

22.6.1 FRONT FUSELAGE STRESS - ARROW 1

It was found that several bleed holes in the top and bottom air intake ramp fillet were blanked off by the interior structure. An investigation was instituted to determine if the existing bleed hole pattern could be improved, while maintaining a satisfactory area and flow distribution for all flight speeds. An improvement has been accomplished by rearrangement of the holes.

A cockpit limit and proof pressure test has been completed on the ARROW 1

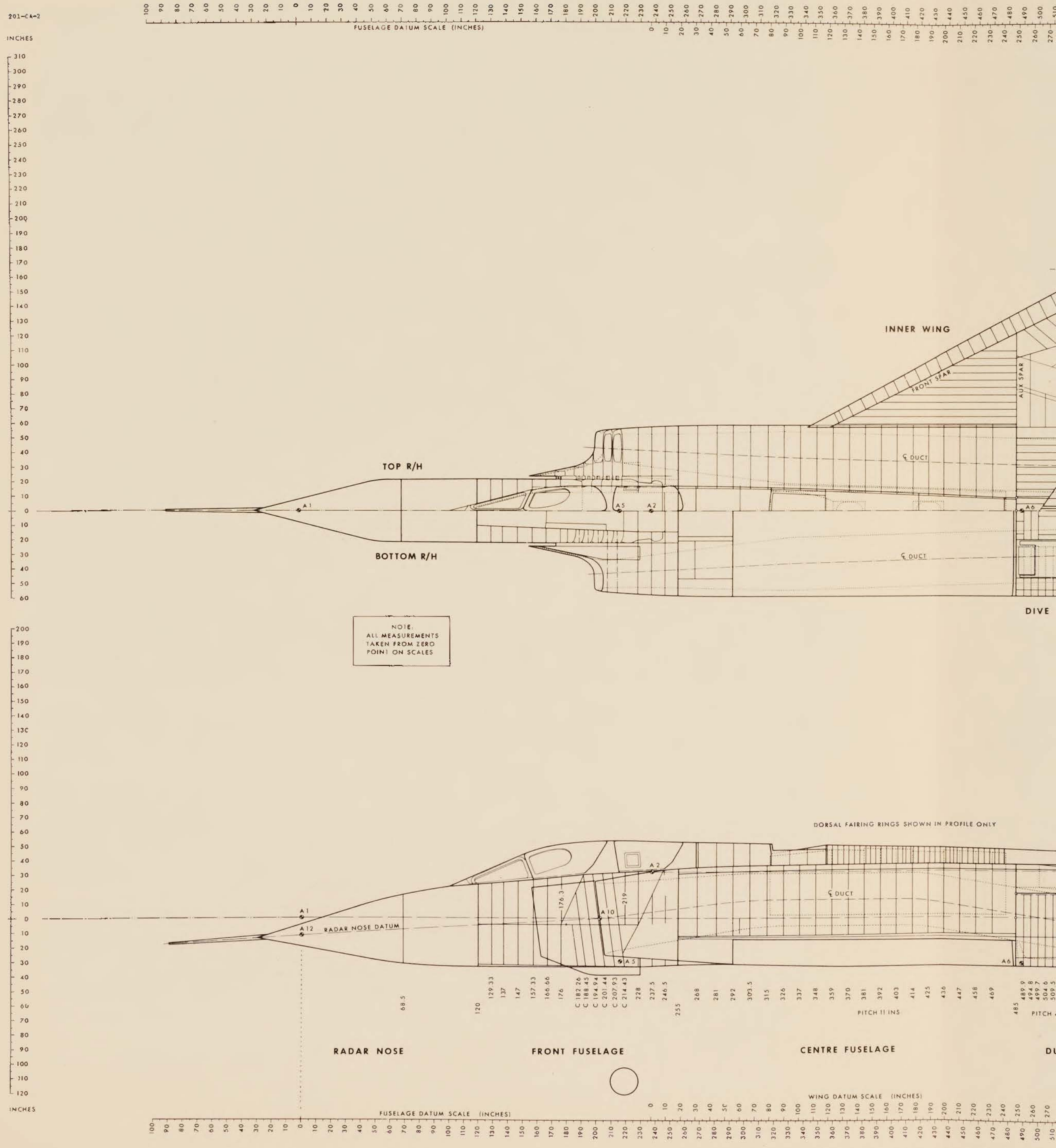

FIG. 41 ARROW STATION AND DATUM LIN

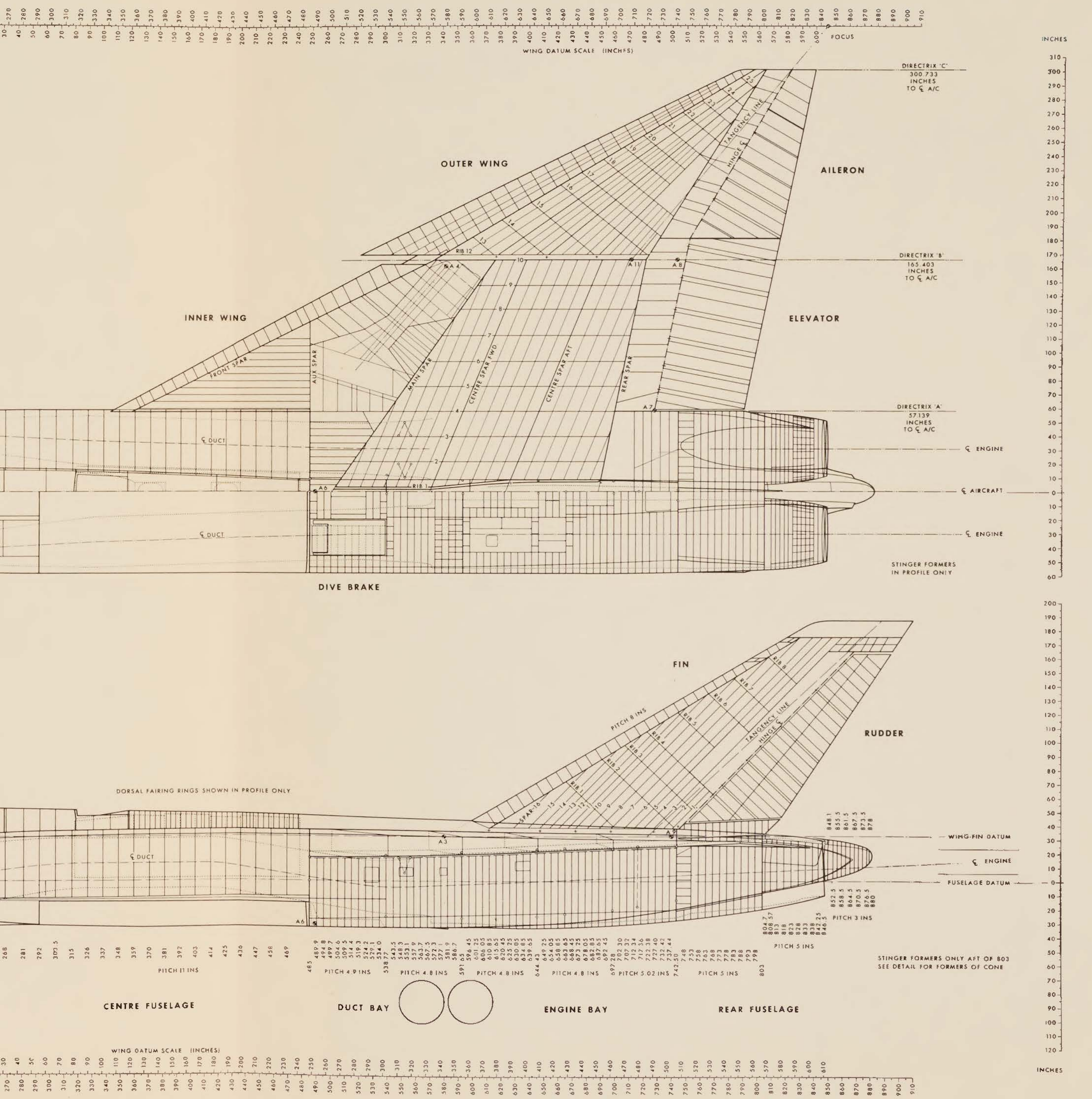

FIG. 41 ARROW STATION AND DATUM LINES

static test aircraft. The tests were conducted at 4 psig, 5.75 psig and finally 7.66 psig, the pressure being reduced to zero after each test. Deflections were measured and strains recorded by a Kelk automatic strain recorder. Observations and inspections during and after the tests indicated that no structural deformation occurred. The tests showed an acceptable leak rate (17 cfm at 2 psig) and that the cockpit structure would satisfactorily withstand the proof pressure of 7.66 psig.

22.7 CENTRE FUSELAGE DESIGN

22.7.1 CENTRE FUSELAGE DESIGN - ARROW 1 and 2

A design investigation on a plastic heat exchanger outlet duct has been completed. Several bonding materials were considered for the Fibreglass ducts; silicone resin, Narmco and phenolic resin (NA-91LD). The phenolic resin was considered to be the most suitable with regard to temperature requirements. The Metallurgical Department is developing the Fibreglass duct fabrication process employing the phenolic resin.

22.8 CENTRE FUSELAGE STRESS

22.8.1 CENTRE FUSELAGE STRESS - ARROW 1

Pressure and suction tests have been completed on the air conditioning system transition duct (heat exchanger) to fan. The duct successfully withstood the limit pressure (15.3 psig) and limit suction (-5.7 psig). A later attempt was made to pressurize the duct to 30.4 psig (twice limit pressure). Failure occurred at 30 psig due to buckling of the stiffeners.

The preliminary load analysis is being completed for the weapon/instrument pack.

The magnesium castings for the missile extension gear have been tested, and in all cases, except one, the casting classification has been raised from 1C to 1B. The test on the remaining casting (upper drag link) will be completed shortly.

Heat exchanger outlet duct stressing has shown that a weight saving of 4 lb. can be expected with the plastic duct compared to the ARROW 2 stainless steel duct. The decision to use the plastic duct will be made when the duct has been manufactured and tested. Manufacture on both the metal and plastic ducts is now proceeding.

22.9 DUCT BAY DESIGN

22.9.1 DUCT BAY DESIGN - ARROW 1 and ARROW 2

All major design work has been completed on the duct bay.

22.10 DUCT BAY STRESS

22.10.1 DUCT BAY STRESS - ARROW 2

An analysis was previously conducted to check the strength of the ARROW 2 speed brake operation at M = 2. A further investigation is now underway to check the integrity of the speed brakes and fuselage structure when missiles are fired with the speed brake extended. The partial retraction of the speed brakes by electro-mechanical means, upon initiation of the missile firing signal, is being considered.

22.11 ENGINE BAY DESIGN

All major work has been completed on the engine bay.

22.12 ENGINE BAY STRESS

Engine bay stressing is continuing and some minor investigations are underway.

22.13 REAR FUSELAGE DESIGN

22.13.1 REAR FUSELAGE DESIGN - ARROW 1

During drag chute jettisoning, the drag chute clevis has been damaging the fibreglass fairing on the aft ramp of the drag chute stowage box. The redesign of this ramp fairing is being investigated. The fairing must either be be made capable of withstanding the clevis impact or be removable so that the damaged fairing may readily be replaced.

22.13.2 REAR FUSELAGE DESIGN - ARROW 2

A jettisonable nozzle, to be inserted in the ejector nozzle, has been proposed for the ARROW 2 (Ref. para. 3.3). It would consist of an inner cylinder with sixteen longitudinal ribs spaced around the periphery with two circumferential rings at the load transfer points. The outer skin would be stretch-formed in two pieces.

An investigation has been completed on the proposed enlargement of the drag chute box. This is in conjunction with the investigations of a larger and improved drag chute (ref. para. 19.2).

22.14 REAR FUSELAGE STRESS

22.14.1 REAR FUSELAGE STRESS - ARROW 2

The addition of the jettisonable nozzle mentioned in para. 22.13.2 may require strengthening of the rear fuselage which would result in a slight increase in weight. Stress work will begin when the loads have been established.

22.15 FIN AND RUDDER DESIGN - ARROW 1 and 2

On 25209 and subsequent aircraft, the fin leading edge will be modified to permit the installation of an IR seeker, with its associated piping and wiring. The preliminary schemes for this installation have been completed and a trial installation will be made on aircraft 25202.

22.16 FIN AND RUDDER STRESS - ARROW 1

Combined fin and rudder deflection tests have been successfully completed. The test involved the application of 100% limit load. This produced a fin tip deflection of approximately 18 inches, at which time full rudder movement was applied. The rudder top hinge attachment failed under this condition. This was considered satisfactory, however, since in actual flight conditions, maximum fin deflection would not be associated with maximum rudder deflection. The test showed that the rudder did not fail when deflected to a degree consistent with maximum fin deflection.

Loads approaching the structural limits on the fin do not occur during normal flight. In the event of failure of the normal damper mode yaw axis however, it was suspected that a hard-over rudder deflection could occur which would produce significant loads before the emergency damping system could come into operation. This condition has been investigated and found to be safe.

22.17 LANDING GEAR - ARROW 1 and 2

An investigation is being conducted into the possibility of incorporating a self aligning bearing at the aircraft attachment fitting of the nose landing gear jack. The present landing gear jack attachment makes assembly difficult, and could induce bending in the jack piston rod, due to possible offset between the aircraft attachment and the landing gear attachment fitting.

22.18 LANDING GEAR STRESS - ARROW 1 and 2

Fatigue tests are continuing on the main landing gear. Fatigue failure occurred on the back stay and a redesign is in work.

22.19 LONG RANGE TANK DESIGN

Production drawings have been started on the metal version of the long range tank.

22.20 LONG RANGE TANK STRESS

Schemes for the long range tank stress are currently being processed.

PART 5

RELIABILITY, MAINTENANCE AND SUPPORT

23.0 MAINTENANCE AND RELIABILITY

23.1 MAINTENANCE ENGINEERING

During the past quarter, the efforts of the Maintenance Engineering group have been mainly concentrated on the Personnel Requirements Data (PRD) study, preflight test coverage, the issuing of a ground equipment list to support 37 aircraft, inspection schedules and maintenance instructions. A revised list of the ground support equipment required for the 37 aircraft program has been prepared and will be issued in April.

23.2 PERSONNEL REQUIREMENTS DATA STUDY

AVRO has now received authority to proceed with the Personnel Requirements Data (PRD) study. Work is continuing on the revised concept and the completion date has been rescheduled to April 1959. In February, AVRO personnel working on this study were supplemented by four members of the RCAF maintenance appraisal team. Several meetings have been held between AVRO, RCA and Orenda representatives to discuss the scope of work involved in the PRD study as it affects associated contractors.

23.3 PRE-FLIGHT TEST COVERAGE

A twenty-four hour pre-flight coverage of aircraft 25201 continued. Maintenance personnel have made recommendations to the Product Design department for improving the maintenance and accessibility aspects of the aircraft and its associated ground support equipment.

23.4 INSPECTION SCHEDULES

A revised inspection schedule, consisting of pre-flight, post-flight, between flight and primary inspections has been issued by the Maintenance Engineering group. An Appendix 'A', listing the special inspection requirements for aircraft 25201, is also included with the inspection schedule. A revised periodic inspection schedule is currently in work.

23.5 INCIDENT INVESTIGATIONS

One incident report has been prepared as a result of investigations conducted by the Maintenance Engineering group. Incident Report No. 1 covers investigations of the damage incurred when a foreign body entered the right-hand engine during ground running.

23.6 RELIABILITY ENGINEERING

The total number of equipment items requiring qualification for the ARROW 1 increased from 927 to 965 in the past quarter. This was mainly due to the addition of a booster system in the flying controls hydraulic system. Of the 965 items, 578 have been fully qualified; 58 items being qualified during the past quarter. All items required for aircraft 25201 have been approved for the initial flight program. Emphasis is now on improving the approval status for those items which have not yet received full qualification.

A total of 924 ARROW 2 items require qualification. Of this total, 131 items are applicable only to the ARROW 2 and 793 items are common to both the ARROW 1 and 2. A total of 293 items have already received qualification, leaving a balance of 631 items to be qualified.

In January 1958 the Equipment Design Department issued Report No. 71/ REL 00/1-2 (Qualification Status Airborne Equipment). Amendments No. 1 and No. 2 to this report were issued in March.

23.7 RELIABILITY ANALYSIS

The Service Analysis covering the activity on aircraft 25201 have used the Reliability Analysis Group's Defect Report forms to record equipment reliability problems in detail. These reports have been transferred to IBM punched cards, which can be sorted, printed or counted at high speed to facilitate the analysis of defect occurrence. The Reliability Engineering Group has investigated the more immediate reliability problems, and has called for design or manufacturing action where necessary. The results of these investigations are included in the IBM records.

In order to provide the essential background information, without which the significance of reported defects cannot be properly measured, a further reporting and recording system has been initiated during the quarter. This is the Utilization Data project which aims at recording the different types of operating hours being accumulated by the various aircraft systems on the ground and in the air. It is intended to record the utilization of about 500 types of serialized component per aircraft, taking account of their possible removal and replacement. The recording in this case will be on magnetic tape, by means of AVRO'S IBM 704 computer.

24.0 GROUND SUPPORT EQUIPMENT

24.1 GENERAL

Although the essential ground support equipment for the first flight of the ARROW has been delivered to the Test Department, six items remained outstanding to complete the equipment for the ARROW 1, Engineering work has been completed on three of these items: the fuselage maintenance stand, engine exhaust cover and the portable boarding ladder. The J75 P5 afterburner sling is still in abeyance pending information from Pratt & Whitney. A tool for opening the drag chute door, and a J75 nose bullet extractor, were added to the equipment list, and have been completed.

Drawings for the following items of ARROW 2 ground support equipment have been completed during this quarter:

- Air conditioning outlet cover
- Engine intake cover
- Radome cover and sling
- Rudder and rudder control box sling
- Elevator sling and armament pack test stand

The following items of ARROW 2 equipment are in work:

- Missile trailer/hoist,
- Iroquois engine change stand
- Iroquois engine maintenance trailer,
- Iroquois engine change crane
- Iroquois radar maintenance stand

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

- Engine starting truck
- Power and air conditioning truck
- Armament harmonization stand
- Weapon pack test console
- Engine exhausts cover
- Auxiliary external fuel tank trailer
- Fuel tank test intercooler unit

Aircraft component slings for:

- Aileron control box
- Ailerons
- 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
Main landing gear tie-down

The requirement for rigging boards has been deleted.

24.2 MOBILE GROUND POWER UNITS

Report No. 72/GEQ/1, (ARROW 2 Design Study of Mobile Ground Power Units) issued in December 1957, reviews in detail, both air and vapour cycle refrigeration systems. These systems are combined with an electrical power generator, which might be considered for supporting the ARROW 2 electronic system during field maintenance or forward base readiness. The performance estimate of this equipment was based on the environmental conditions specified by the RCAF The RCAF reviewed the report recommendations and indicated that it preferred the air cycle equipment, because of its compact size and light weight.

The RCAF has since relaxed the environmental conditions to bring them within the scope of the air cycle equipment, and deleted the requirement for cockpit air conditioning from the mobile ground equipment. AVRO has been requested to review all the air cycle systems previously considered with respect to the revised conditions. Addendum 1 to Report No. 72/GEQ/1 (ARROW 2 Design Study of Mobile Ground Power Units) is currently being prepared to meet these requirements and will be issued shortly.

25.0 AIR BASE FACILITIES

25.1 ARROW DEVELOPMENT AND DEMONSTRATION PROGRAM - COLD LAKE

In January 1958, Report No. 72/GEQ/4 (Preliminary Report on the Base Facilities Required at RCAF Cold Lake for the ARROW Development and Demonstration Program) was issued to comply with the RCAF's request for preliminary information on the air base requirements to support the ARROW Weapon System. The information contained in the report is preliminary and subject to confirmation.

25.2 PROPOSED ALTERATIONS TO RCAF READINESS HANGARS

In compliance with an RCAF request, Report No. 72/GEQ/6 (Comments on Proposed Alterations to RCAF Readiness Hangars) was issued in February 1958. The report included comments on cooling air supply, engine starting air duct, quick disconnect lanyards, maintenance control room construction and the specification for cooling air units.

25.3 ARROW DEVELOPMENT PROGRAM - FACILITIES AT MALTON

In February 1958, Report No. 72/GEQ/5 (Proposed Special Equipment for ARROW 2 Development in AVRO Hangars), was issued. The report outlines the requirements for the support of ASTRA-equipped ARROW aircraft in the AVRO flight test hangars. The ground servicing hangar facilities for the development of the aircraft and its ASTRA electronic system, requires the introduction of complex equipment.

The study is based on a hypothetical hangar bay containing four aircraft and illustrates the facilities required for supporting ASTRA equipped aircraft. The layout provides for optimum utilization of space and equipment.

To service four aircraft, it is recommended that cooling air be supplied from a fixed installation, consisting of two electrically driven vapour compression refrigeration systems. Each unit would supply two aircraft, and should be capable of supplying cooling air at 260 lb/min at 55°F. The cooling air supply would flow through a 14 inch diameter duct beneath the hangar floor. From the duct, the air is distributed through a manifold connected to eight 45-foot lengths of flexible hose, thus completing the circuit to the aircraft.

It is recommended that the electrical power be supplied to the aircraft from mobile motor generators. The expected maximum load requirement with all aircraft services operating is 31.5 KVA. However, it is recommended that a generator with a rated output of 37 to 40 KVA be specified to allow for aircraft development.

In addition to the aircraft power supply, electrical power outlets will be required at each aircraft position for 550V, 60 cycle, 3 phase and 110V, 60 cycle single phase, alternating current.

25.4 1st LINE MAINTENANCE AND TURNAROUND FACILITIES

The RCAF requested that a report be prepared on the ARROW 2 first line maintenance and turnaround facilities. This report, No. 72/GEQ/3 which will be issued in April, was incorrectly referred to as Report No. 72/GOPS/2 in the previous quarterly report.

The report outlines the requirements for equipment, procedures, personnel and facilities for turnaround and first line maintenance. The report also reviews the ground servicing equipment for 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, between-flight inspections and first flight maintenance would be performed in this hangar. It is recommended that the turnaround hangar be located near the readiness facility to isolate high noise level activities from other areas of the base.

The study indicates that the airframe inspection may be completed in approximately 1 hour 5 minutes, and a primary inspection of the ASTRA I sub-systems may occupy 2 hours 40 minutes. 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 turnaround operations within 15 minutes in a prepared turnaround hangar. These figures are preliminary and are to be confirmed by the Personnel Requirements Data Study. Estimates are given of power requirements for turnaround and first line maintenance. Provision of emergency power supplies is essential.

The advantages of hydrant refuelling are marginal with respect to pumping time, but this method will be necessary to supply cool fuel, if flights above Mach 1.5 are contemplated. Tests are required to establish the fuel temperatures attainable in the fuel tanks while the aircraft are parked in the open under tropical conditions, and while parked in hangars in the summer. Tests are also necessary to establish refuelling tanker and hydrant system delivery temperatures under similar circumstances, to determine the necessity of fuel cooling.

25.5 CURRENT STUDIES

The following studies are currently being prepared by the ground support engineering group:

3108-105-2

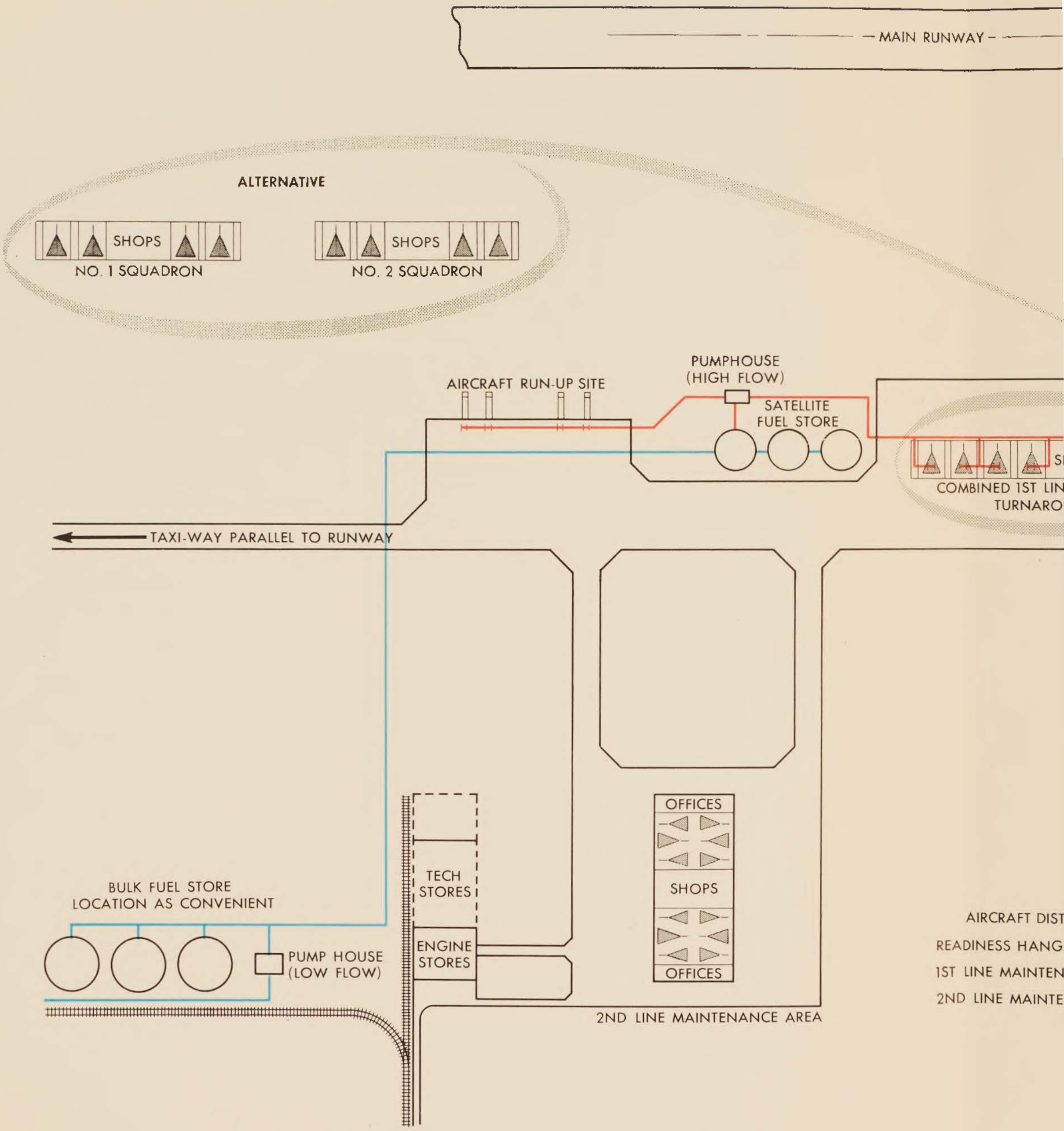

FIG. 42 PROPOSED AIR BASE LAYOUT FOR 2 S
(TURNAROUND AND 1ST LINE MAINTENANCE C

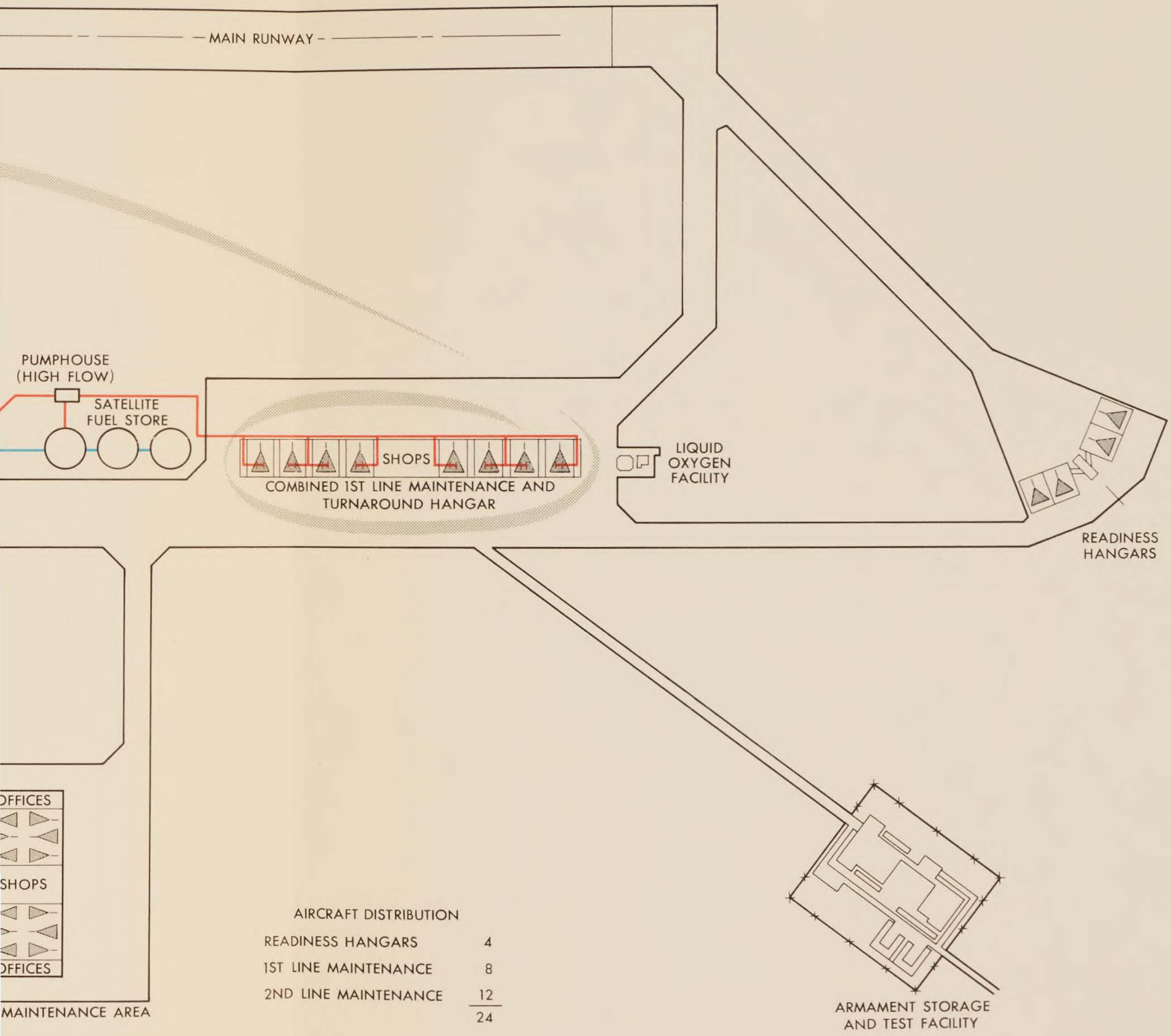

FIG. 42 PROPOSED AIR BASE LAYOUT FOR 2 SQUADRONS (TURNAROUND AND 1ST LINE MAINTENANCE COMBINED)

(a) Iroquois engine run-up facility.

(b) An evaluation of the Honeywell proposal for a fuel distribution test set, the results of which will form the basis for the preparation of a specification.

(c) A design study on a mobile water replenishment unit.

(d) A study of second line maintenance facilities.

(e) Power air condition unit.

(f) Starter unit.

(g) Evaluate damper servo test rig.

(h) Revision of armament storage and test facility.

(j) Weapon pack test console (less electronics).

26.0 WEAPON SYSTEMS TRAINER

26.1 AIRCRAFT SYSTEMS TRAINER

A technical proposal for the ARROW 2 Aircraft Systems Trainer (AST) was submitted to the RCAF in January 1958 for consideration and approval.

26.2 GROUND EQUIPMENT TRAINER

At the RCAF's request, AVRO are preparing a technical proposal for a Ground Equipment Trainer (GET). This proposal will be issued shortly.

The purpose of the study is to propose to the RCAF, suitable facilities for training technicians who will be engaged in the servicing and maintenance of ARROW 2 aircraft. The method of presentation for the ground equipment trainer is similar to that used for the Aircraft Systems Trainer.

Proposed methods of instruction include dynamic trainers, illuminated schematic panels, projection slides, and actual items of equipment. A comprehensive series of projection slides cover the majority of the ground equipment training program. This method of presentation when used in conjunction with prepared lectures and engineering orders, illustrates the testing procedure to be followed. The following test panels are covered by means of projection slides:

- Air conditioning test panel
- Air conditioning and a-c generator test panel
- Windshield and canopy de-icing test panel
- Fuel quantity indicator test panel
- Electrical power test panel
- Engine change equipment and towbar

Certain items are considered unconventional and their operation requires a high level of instruction. In these cases, it is proposed to provide either actual ground servicing equipment as used in the field, or an illuminated schematic panel. Equipment in this category include, the hydraulic test stand and panel, the air/nitrogen compressor, and the Sparrow missile trailer/hoist.

The engine starting system trainer is presented as an illuminated schematic diagram. This is designed to illustrate the engine starting sequence and familiarize personnel with the control and operation of the ground starting unit. Although it is considered unnecessary to provide actual equipment, the operation of the unit is sufficiently complex to warrant a more practical training device then projection slides.

A table of electrical power requirements is included in the proposal.

27.0 STRUCTURAL GROUND TEST PROGRAM

27.1 INTRODUCTION

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

27.2 STATIC TESTING OF THE COMPLETE AIRCRAFT

Limit and proof pressure tests of the static test aircraft cockpit were satisfactorily completed by mid-January. The leak rates were high, but were within the required limits.

Cockpit pressure tests were followed by ejection system tests from the cockpits of the static test aircraft. A report on these tests, is contained in para. 18.1 (Escape System).

Installation of the complete static test aircraft in the test rig began on 5 February 1958, although preparations had been in progress since completion of the seat ejection tests in January. The installation of external strain gauges is continuing. Tension patches are being installed, using Bostik cement as the adhesive. Tests have demonstrated that Bostik satisfactorily resists the action of the Pella oil, which is used as the pressurizing medium in the wing tanks during static testing. Erection of the test rig upper structure is almost complete. The testing of strain gauge circuits is proceeding. Additional test rig and instrumentation drawings for the landing gear springback case and the rolling pullout case tests have been completed, and drawings for subsequent tests are in work. Landing gear spring-back tests will be conducted when the static test aircraft has been installed in the rig.

Tests are being conducted to investigate the effects of time and environment on strain gauges. These tests are of particular importance to the strain gauges designated for long term testing, and so far the results have been satisfactory.

27.3 TESTING OF MINOR COMPONENTS

Structural strength testing of minor components has continued. All component testing vital to the safety of the first flight was completed by the end of February. However, structural strength testing of a wide range of components under loading conditions for the entire flight envelope is continuing.

27.4 RUDDER STIFFNESS AND LIMIT LOAD TEST

The results of the limited rudder strength and stiffness test program have been analyzed and issued to the Technical Design department. The remainder of the program is held in abeyance pending completion of functional testing which requires the use of the same test rig.

27.5 COMPLETE AIRCRAFT VIBRATION TEST

As noted in the previous Quarterly Technical Report, the fin characteristics encountered during aircraft vibration tests did not agree with previous theoretical analyses. The tests were repeated in January with satisfactory results and are now considered complete.

27.6 COMBINED LOADING AND TRANSIENT HEATING OF WING BOX

The wing box specimen, required for the combined loading and transient heating tests has been completed. The installation of strain gauges and associated wiring is now in progress. Test rig drawings have been completed and the rig is being manufactured.

Investigation to determine the most suitable fluid for use as a pressurizing medium during these tests have shown that Monsanto 'Aroclor' 1221 approximates the thermal transfer characteristics of JP-4 fuel. Stress Department approval is now being sought to use Aroclor for these tests.

27.7 TEMPERATURE DISTRIBUTION THROUGH TYPICAL STRUCTURAL SECTIONS

The test program on typical joint specimens has been completed and the results are now being evaluated by the Technical Design department. The installation of strain gauges on a small box specimen representing the forward fuselage side skins and ducts, is proceeding.

Testing will resume during April, after the erection of partitions to exclude drafts from the test area.

27.8 FATIGUE TESTS OF MODEL FUEL TANK NO. 4

Subsequent to the unsatisfactory tests reported in the previous Quarterly Technical Report, the tank model was repaired and modified. These modifications involved the strengthening of skins and stringers. Tests were conducted on the modified model, and the lower stringers failed after a total of 1700 cycles of load and pressure. This was still unsatisfactory and further repairs and modifications were made to the specimen.

Subsequent testing resulted in failures of the rib trusses and upper stringers after approximately 2300 cycles. This is considered satisfactory for these parts, and repairs are now being made to allow the spars and skins to be fatigue tested up to the required 3000 cycles of load.

27.9 ENGINE DOOR STRENGTH TEST

The door specimen used in previous engine door strength tests was repaired and steel side plates provided for the door latches. Subsequent tests on the latches proved satisfactory and no further testing is anticipated.

27.10 PANEL RESPONSE TO SOUND PRESSURE AND FREQUENCY

Further tests have been conducted on representative panels from critical areas of the aircraft structure. Magnesium panels were developed as a result of previous testing, and have proved satisfactory. Tests on further magnesium side skin panels and a stainless steel stringer panel will continue on a low priority basis, using a variety of test conditions.

27.11 FATIGUE TEST OF ELEVATOR LINKS

As reported in the previous issue of the Quarterly Technical Report, the use of incorrect adaptor fittings resulted in failure of the elevator link test specimens. The correct fittings were despatched to Krouse Testing Machines Inc., at Columbus, Ohio and the tests resumed. Failure of the first specimen occurred after 993,000 cycles. This is considered satisfactory and tests are continuing on further specimens, to confirm the results obtained.

27.12 DEVELOPMENT OF HIGH TEMPERATURE STRUCTURAL TEST TECHNIQUES

As part of a development program on high temperature structural testing techniques, the Metallurgical Laboratory has tested, at room temperature, a range of sheet metal specimens to which strain gauges has been cemented. The specimens were subjected to various adhesive curing cycles, and the optimum curing procedure established for strain gauge adhesives in conjunction with certain materials. Some preliminary tests conducted on strain gauges cemented to magnesium sheet have shown that, at elevated temperatures, the temperature compensating characteristics of the gauges were unsatisfactory. Further work is being done to improve the gauge characteristics and cementing techniques.

27.13 BEARING RETENTION DEVELOPMENT

Development testing has continued on methods of retaining bearings in housings. Minor modifications to the roll swaging tool have delayed testing of the last ten specimens, which will complete work on these tests. Testing will resume shortly.

27.14 STRENGTH TESTS OF LANDING GEAR FRONT PIVOT BEARING

Manufacture of the test rig for landing gear front pivot bearing strength tests was completed by mid-January and tests of the bearing to limit load were started immediately. Loading was repeated several times, so that any progressive wear in the bearing could be observed. Testing was completed by late February, and results were satisfactory. The amount of play which developed in the bearing was not excessive, and remained within the design limits.

27.15 MODEL TEST OF WORM AND GEAR ENGINE MOUNT

Initial functioning tests on the worm and gear engine mount resulted in damage to the parts and jamming of the gears. The mechanism was redesigned and further tests proved satisfactory. The specimen was reallocated to a fatigue test program.

27.16 ENGINE LIFTING MECHANISM FATIGUE TEST

Engine lifting mechanism fatigue tests were conducted at Krouse Testing Machines Inc. The specimen failed after 3715 cycles of limit load had been achieved. This satisfied the design requirements. Further specimens are to be tested under various load applications.

27.17 PRESSURE TESTS OF TRANSITION DUCT (HEAT EXCHANGER TO TURBINE)

Initial tests on the transition duct revealed minor leaks in the test rig. These were resealed, and the specimen successfully withstood 150% of ultimate design pressure, which was considered satisfactory.

27.18 PRESSURE TEST ON RAM AND FAN EXHAUST DUCT

The ram and fan exhaust duct was subjected to pressure testing with satisfactory results. This duct has since been blanked off into smaller sections and is now being pressure tested over critical regions.

27.19 AILERON JACK FATIGUE TEST

During preparations for fatigue tests of the aileron jack, some reworking of the jack attachment block was necessary. This has been completed and tests are now proceeding.

27.20 ELEVATOR JACK EXTENSION FATIGUE TEST

Fatigue tests on the elevator jack extension have been conducted at Krouse Testing Machines Inc. The first specimen withsto d over 2. 2 million applications of ± 17875 lb loading before failure. The second specimen withstood more than 1.5 million cycles of ± 21450 loading, and the third specimen withstood 66000 cycles of application of ± 57200 lb. loading. This was considered satisfactory, and further testing at AVRO has been cancelled.

27.21 BASIC INVESTIGATIONS INTO ACOUSTICAL FATIGUE OF STRUCTURES

Tests have been conducted to investigate the acoustical fatigue of structures. Six simple panel specimens have been tested so far, and the results are being analyzed.

27.22 ADDITIONAL TESTS ON INNER WING POSTED BOX

A specimen representative of a critical area of the inner wing box has been manufactured, and strain gauges have now been installed in the specimen. The test rig is now being prepared to receive the specimen and torsion and bending tests will commence shortly, with the box under internal pressure.

27.23 TRANSIENT HEATING TEST OF FORMER 469 (ARROW 2)

Design of the specimen former and associated testing required for transient heating tests of former 469, is proceeding.

27.24 PRESSURE TEST OF ARROW 2 ARTICULATED DUCT

Design of the test rig required for pressure tests of the ARROW 2 articulated duct is proceeding.

28.0 SYSTEM GROUND TEST PROGRAM

28.1 FUEL SYSTEM

The ARROW fuel system test program has been revised and will now be conducted in approximately the following order:

1. Changeover to low pressure system
2. ARROW 1 system tests with low pressure equipment.
3. Changeover to ARROW 2 fuel management system.
4. ARROW 2 fuel management system tests.

28.1.1 TEST PROGRESS

The high pressure system test program has been completed. Tests to determine the air release requirements for tank No. 5 were completed, and showed that they are quite small. Investigation into the loss of prime of a fuel transfer pump is discussed in para. 13.2.1.1.

The changeover of the test rig to low pressure operation was completed in February. Relief valve tests were conducted and the results are now under examination. Refuelling tests on tank No. 5 were completed and the results are being analyzed.

It was found that the air pressure regulating valves installed both on the rig and on the aircraft, were regulating 1 psi too high, after the maximum allowable adjustment had been made. The spring from one valve was ground to reduce its length, and further tests confirmed that the valve would operate at the pressure required. The fuel pressure regulator valves from the aircraft were similarly modified, and springs are now being manufactured for use in the valves on the test rig.

Tests to develop a suitable air ejector for the fuel collector tank are in abeyance pending a closer definition of requirements. Airworthiness tests on a Purolator fuel strainer have been performed but the results have not been satisfactory. Examination of the filter revealed damage to the element, and the strainer has been returned to the manufacturer for repairs.

Tests have been conducted to assess the qualities of "O" rings used in fuel pipe flexible couplings. After the initial soaking in fuel, the sample manufactured by Pacific Moulded Products became swollen. Tests of this sample in a Wiggins coupling gave satisfactory results. However, on dismantling the coupling after testing, some damage was revealed. Testing of "O" rings and couplings continues.

28.1.2 FUEL SYSTEM PRE-FLIGHT TESTS

As malfunction of the fuel bypass override control valve had occurred on the fuel system test rig, it was considered advisable to conduct further tests of the aircraft fuel bypass override system. These tests disclosed that the fuel bypass override system on the left-hand side of the aircraft was not operating correctly, due to a defective bypass valve. The valve was repaired and further tests were satisfactory.

28.2 FLYING CONTROLS SYSTEM

The main flying controls system test program consists of an evaluation of the complete flying control system from the cockpit controls to the control surfaces. Testing is further broken down into individual tests on hydraulic and mechanical components.

The program for the flying controls system testing has been revised and is now as follows:

1. Complete development of the input system for the manual mode.
2. Operation of controls with deflected hinge lines.
3. Development of the stick force mode system.
4. Impedance tests of the output system.
5. Strength and stiffness tests of the output system.
6. Frequency response tests with loaded surfaces.
7. Simulated flight tests with loaded surfaces.
8. Duty cycle testing.

28.2.1 TEST PROGRESS

28.2.1.1 Complete Mechanical System Test Rig

In order to overcome excessive friction and damping forces in the flying control system, it was found necessary to incorporate hydraulic boosters into the system. These were located in the cockpit and operated in the elevator and aileron control circuits. AVRO-designed dampers were installed on all actuators and a centring spring and rudder cable tensioner units were installed. Frequency response tests were then conducted to assess the system prior to flight simulation tests Results were satisfactory and the flying controls system was considered suitable for the flight simulation tests to be performed.

Flying control system flight simulation tests were conducted without the damper in operation. Tests were performed with two engine, single engine, single system and single hydraulic pump operation. The damper simulator was then connected into the test rig, and flight simulation tests were conducted with the yaw damper operative. Records from the flight simulation test programs are still being analyzed, but the pilots were satisfied that control of the aircraft could be safely maintained.

Production type Waterlift boosters, complete with dampers, were installed in the test rig, replacing the prototype Waterlift boosters. Booster evaluation tests were completed on the aileron controls, and the new booster was considered satisfactory.

Tests were conducted to investigate slipping in the aileron cable tension regulator quadrant, and to determine the extent of cable loading induced by sudden operation of the aileron control. Initial tests produced failure of a rig part simulating the control column base casting. Testing was delayed while the part affected was repaired. The amount of slip observed during tests on the cable tension regulator quadrants was excessive, and the regulators were modified. Further tests are proceeding.

28.2.2 AILERON CONTROL SYSTEM FUNCTIONAL TESTS

High temperature tests were conducted on production type AVRO-designed dashpot dampers. Control surface oscillations were effectively eliminated at temperatures up to 250°F.

Aileron function tests were then conducted with a bent aileron hinge line.

Buckling of the aileron skin occurred when the hinge line was bent to 40% limit load deflection. It was decided to split the aileron approximately midway along its length before further tests were conducted. Results from subsequent tests up to 50% limit load deflection were considered sufficiently satisfactory to grant limited flight approval for the flying aircraft, whose ailerons were similarly split. This was justified because the test rig stiffness was unrepresentative of aircraft stiffness. It was considered that 50% limit load stresses in the rig mounted aileron were roughly equivalent to 100% stresses in the aircraft mounted surface.

28.2.3 RUDDER CONTROL SYSTEM FUNCTIONAL TESTS

Testing was conducted to establish the breakout forces of the feel units at the hinge moment limiter. With the nosewheel steering disconnected, rudder centring was within ± 5/8°. Further tests with the nosewheel steering connected showed roughness in the operation of the controls. This was considered to be due to wear in the gear and pinion of the nosewheel steering disconnect. The steering disconnect was replaced, and found to develop wear very rapidly, with consequent roughness in the rudder control again evident during tests. The nosewheel steering disconnect is being redesigned and further tests will be conducted when the new unit becomes available.

Rudder operation tests with the hinge beam in a deflected condition resulted in some hinge bracket attachment bolt failures. In addition, severe buckling of the rudder trailing edge occurred and the rudder skin buckled when the rudder was moved through approximately 13°, with the hinge beam bent to 100% limit load deflection.

Repairs to the rudder are now proceeding, and further testing will be conducted when the repairs have been completed.

28.2.4 ELEVATOR CONTROL SYSTEM FUNCTIONAL TESTS

The elevator tests were similar to those performed on the aileron. It was operated with the hinge beam deflected to 95% limit load deflection. The specimen showed no sign of failure, but the test was suspended at this point to reevaluate the rig strength, before proceeding to higher loads. Testing will resume when the strain gauges have been installed on the test rig. This will enable rig loads to be monitored while tests are in progress.

28.2.5 HYDRAULIC CONNECTIONS FOR 4,000 PSI

Testing has continued in an effort to develop hydraulic connections capable of withstanding high temperature flexural fatigue. Results from testing so far completed are now being assessed, and the necessity for further tests is to be established.

28.2.6 WIG-O-FLEX COUPLINGS FOR LOW PRESSURE HYDRAULICS

Tests are nearing completion on a further range of specimens.

28.2.7 RUDDER CONTROL VALVE

A rudder valve was subjected to low temperature evaluation tests. Results were satisfactory and no further tests are anticipated.

28.2.8 FATIGUE TESTS OF AILERON AND RUDDER PRESSURE LINES

Fatigue testing of various types of aileron and rudder pressure line specimens has continued. Failure of some stainless steel specimens has occurred prematurely at 250,000-350,000 cycles, at a temperature of 250°F. Modification of the specimen configuration resulted in failure at approximately 700,000 cycles. It has been found necessary to replace several rig parts, and it is felt that the rig wear rendered the results of the latest tests invalid. Further specimens were ordered and subjected to test on a rig which is now more representative of the aircraft piping installation. After achieving approximately 1,000,000 cycles on the new specimens, the rig tests were suspended in favour of pressure cycling tests on U-shaped pipes, bent to critical radii. These tests are now in progress at the Weatherhead Co. In parallel with this latest program, tests are being made to determine influence coefficients on actual aileron, elevator and rudder pressure and return pipe assemblies.

28.3 AIR CONDITIONING SYSTEM

The remaining test program for the ARROW 1 air conditioning system now covers only the development testing of the temperature control system.

28.3.1 AIR CONDITIONING SYSTEM TESTS

The ARROW 2 air conditioning system tests are now scheduled to start in mid-May. The test program is as follows:

1. Start up and calibration.
2. System trials and adjustments, and equipment evaluation at moderate inlet conditions, utilizing some restricted performance equipment.
3. Incorporation of fully operational equipment and ducting, and recalibrations.
4. Finalized system trials and adjustments, and equipment evaluation over the full range of inlet conditions.

28.3.2 TEST PROGRESS

Tests on a pneumatic turbine outlet temperature control system were conducted. The results obtained were superior to those achieved with an electrical control system. The pneumatic temperature control system, manufactured by AiResearch, was approved for installation in the aircraft. However, further testing will be required before unlimited approval can be given to this system. Tests on an AVRO-designed breadboard-type cockpit temperature control unit gave satisfactory results. A prototype was manufactured based on the breadboard, but subsequent tests showed that it was too sensitive to voltage and frequency fluctuations. The breadboard cockpit temperature controller was redesigned and further testing is in progress.

For the first flights of the aircraft, a cockpit temperature controller, manufactured by Hamilton Standard Division of United Aircraft Corporation was installed and approved for operation under limited conditions.

28. 3. 3 COCKPIT ENVIRONMENT TESTS - ARROW 1 and ARROW 2

Efforts to reduce the noise level in the cockpit have continued. Tests have been conducted on the test rig and metal mock-up using mufflers attached to the outlet ducts. These are expected to reduce the cockpit noise level to an acceptable level.

Preliminary tests of the temperature distribution in the front and rear cockpits of the metal mock-up, disclosed minor faults in the test arrangements. These have been rectified, but testing has been further delayed due to the necessity of relocating the metal mock-up. Tests will be continued during April.

28. 3. 4 LEAK DETECTION SYSTEM TEST

Tests of the single point and Walter Kidde continuous wire leak detection system proved unsatisfactory. Orders have been placed for leak detection equipment manufactured by Fenwall Inc. and tests will begin on 1 May (see para. 10. 5).

28. 4 <u>ELECTRICAL SYSTEM</u>

A test program has been established for the ARROW 2 electrical system, as follows:

1. Individual circuit operation tests.

2. Voltage loss tests of loaded circuits.

3. Simulated flight operation of circuits.

4. Power system tests.

28. 4. 1 TEST PROGRESS

The remaining electrical system pre-flight tests for the ARROW 1 were conducted during pre-flight tests on the utility hydraulics system and the flying controls system. The behaviour of all electrical equipment was satisfactory.

28. 4. 2 ELECTRICAL SYSTEM - ARROW 2

Manufacture of the breadboard and loadbanks is being completed as the necessary equipment is received.

28. 4. 3 ELECTRICAL BREADBOARD TEST OF ARMAMENT SYSTEM

Tests were conducted on a breadboard of the armament electrical system.

This breadboard incorporated a circuit which automatically jettisoned any misfired missile. Tests on this breadboard were cancelled in favour of tests on a second armament electrical system breadboard which allowed a choice between jettisoning and retaining a misfired missile. Initial testing of the second breadboard revealed some circuit faults, and testing will resume when the necessary corrections have been made.

28.5 LANDING GEAR SYSTEM

Separate tests of the nose landing gear and of the left-hand main landing gear without its associated door valve have been completed.

The revised test program is as follows:

1. Right-hand main landing gear and door tests.

2. Tests of full landing gear system with various flight case loadings.

28.5.1 TEST PROGRESS

28.5.1.1 Nose Landing Gear System Tests

From experience gained during pre-flight testing on the aircraft, it was considered necessary to alter the nose wheel door sequence of operations. The nose landing gear test rig hydraulic piping and the electrical switching arrangement were modified to conform to the altered sequence of operation. Functional tests were then conducted, and the system operated satisfactorily.

Operation of the steering disconnect cable confirmed that its behaviour was satisfactory during nose landing gear actuation. Evidence of wear was discovered on the steering disconnect gear and pinion (ref. para. 28.2.3). Further tests performed on the nosewheel steering system showed that it is still possible to engage the nosewheel steering without first centring the rudder control. The selector valve has been removed from the test rig for examination.

In view of the high friction forces evident in the nosewheel steering during rudder centring tests, and malfunction of the steering valve, consideration is being given to redesigning the system.

28.5.2 MAIN LANDING GEAR

As reported in the previous issue of the Quarterly Technical Report, the retraction jack pick-up bracket bushing and the retraction jack head end bearing required repairs before further tests could be conducted. These repairs were completed and retractions of the main landing gear left-hand leg, under critical loading conditions, were completed satisfactorily.

Modifications to add a door and associated actuating gear to the test rig for the left-hand main landing gear have been completed. Assembly of the door whiffletree is almost complete. Construction of a test rig for the right-hand main landing gear is nearing completion. This will enable full system testing to be undertaken under various loadings as soon as the main gear legs presently installed in the static test aircraft are available. However, it is intended that the preliminary door operation tests for various loads will proceed in advance of the leg installation.

28.5.3 UTILITY HYDRAULICS SYSTEM PRE-FLIGHT TESTS

Pre-flight tests on the utility hydraulics system were completed by the end of February. These included tests on the wheel brakes, and tests to demonstrate correct functioning of the landing gear for both normal and emergency operation. The results were considered satisfactory, though it was noted that the landing gear selector did not lock in the emergency extension position.

28.5.4 MISCELLANEOUS SYSTEM TESTS

28.5.4.1 Assessment of Pressure Switches

Assessment tests were conducted on pressure switches from three different manufacturers: Parmatic, Maletron and Manning, Maxwell and Moore. Results showed that the Parmatic and Maletron switches operated beyond the specified pressure limits at all temperatures. The switch manufactured by Manning, Maxwell and Moore operated beyond the specified pressure limits at high temperatures. During endurance testing Meletron switch casing failed. Efforts are continuing to find a switch that will meet the design requirements under all operation conditions.

28.5.5 PRE-FLIGHT TESTS ON LOW PRESSURE PNEUMATICS

Tests on the noseboom pitot static system were completed in January. Canopy seal tests followed, and were completed satisfactorily in February.

28.6 ESCAPE SYSTEM

28.6.1 PROGRAM

The canopy emergency opening and ejection seat tests from the cockpits of the static test aircraft has been completed. Arrangements are being made to proceed with rocket sled testing of the escape system.

28.6.2 TEST PROGRESS

Seat ejection tests were conducted from the cockpits of the static test aircraft prior to its installation in the test rig (ref para. 18. 1) Film records indicated that the dummy's legs struck the instrument panel during ejection It was considered that the behaviour of the dummy seat occupant had not been entirely representative of that of a live occupant and this was confirmed by tests at the Royal Aircraft Establishment, conducted by the Martin-Baker Company.

Tests were conducted to establish the damping constant of the canopy emergency jack and recurperator The results were considered satisfactory.

Arrangements are progressing to test the rear cockpit escape system on the first aircraft.

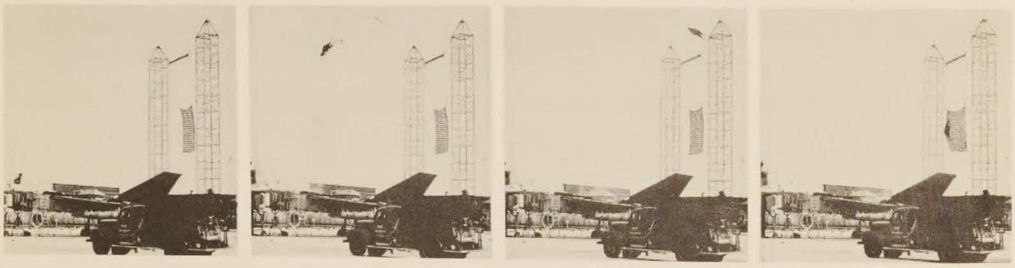

FIG. 43 ESCAPE SYSTEM TESTS

28. 7 SPARROW MISSILE PACKAGE

The test program for the Sparrow missile package has been revised as follows:

1. Launcher development testing.
2. Single missile mechanism tests (production version).
3. Complete package functioning and firing tests.
4. Complete package strength and stiffness tests.

28. 7. 1 TEST PROGRESS

28. 7. 1. 1 Missile Package Door Tests

The missile package door functional tests were conducted at ambient and high temperatures. 500 cycles of operation were successfully achieved at each temperature. Functional tests at low temperatures were interrupted while the cause of door jamming was investigated. The door guide rollers were decreased in diameter and length, to give greater clearances. It was also apparent that the aluminum door drive levers were distorted. Steel levers were substituted, but the door jamming was still evident. Further investigation revealed that the door actuating linkage was misaligned. Tests were suspended at this point, as even with the misalignments and jamming, the door had operated successfully up to 54 cycles at low temperatures.

28.7.1.2 Missile Extension Mechanism Test

The missile extension mechanism test rig is now being assembled. The design of the loading system has been completed, and assembly drawings have been issued. The compensating valves failed to operate satisfactorily and have been returned to the manufacturer for investigation. Missile extension mechanism tests will begin when the compensating valves have been returned.

28.7.1.3 Missile Installation Tests - ARROW 2

The design of a test rig for functional and firing tests on the ARROW 2 missile installation has been completed.

28.7.1.4 Development Testing of Missile Launcher

Initial firing tests of a dummy missile from a rail, representative of the launcher rail on the ARROW 2, resulted in damage to the rail and to the missile. Investigation showed that the test rig was misaligned. This was corrected, a new rail manufactured, and repairs to the missile were effected. Further firing tests were successful and are continuing. To date, over 50 firings have been achieved, but extensive damage has been caused to the anodized surface of the new launcher rail. This problem is under investigation.

The assembly of the umbilical cord and electrical connector for the missile is nearing completion. Tests firings with the cord connected will be undertaken early in April.

28.7.1.5 Shear Pin Test

Shear strength tests have continued on the shear pins used to retain the missile in its launching position. Tests on production type pins have revealed that their strength characteristics are too high. Investigations are now proceeding to design shear pins of a material with more suitable strength characteristics, so that pin failure occurs within the required load limitations.

28.8 PRE-INSTALLATION TESTING OF BOUGHT-OUT EQUIPMENT

Pre-installation functional tests of bought-out parts for the first five aircraft have continued. Tests on thirty-one items of equipment were completed, and nine items are continuing on test.

29.0 FLIGHT TESTING

29.1 PRE-FLIGHT TAXI TESTS

Pre-flight taxi testing continued, to investigate the damper system operating and to assess the reliability of the drag chute at speeds up to 120 kts. The speeds at which the nose of the aircraft lifts under influence of various elevator angles was also determined.

After taxi test No. 15, the pilot reported depletion of the oxygen supply, and separate tests were made to investigate this condition. These tests were inconclusive as the system was found to operate satisfactorily.

During the taxi trials, brake tests were conducted and brake temperatures recorded. The wheel rims were painted with Thermindex paint, which, by its changes in color, indicated the temperature attained by the wheel rims during braking.

The telemetry system was used during the taxi tests, so that the required parameters could be continuously monitored. Experience in the use of the telemetry was gained in preparation for flight testing.

Drag chute tests showed that the chute would stream properly at 100 kts., and that use of the chute effectively assisted the braking. With the drag chute deployed the aircraft braked to a standstill from 100 kts in 3,100 feet without overheating the brakes and wheels. Without the drag chute, a distance of 4,000 feet was required to stop the aircraft from the same speed. In this case, the

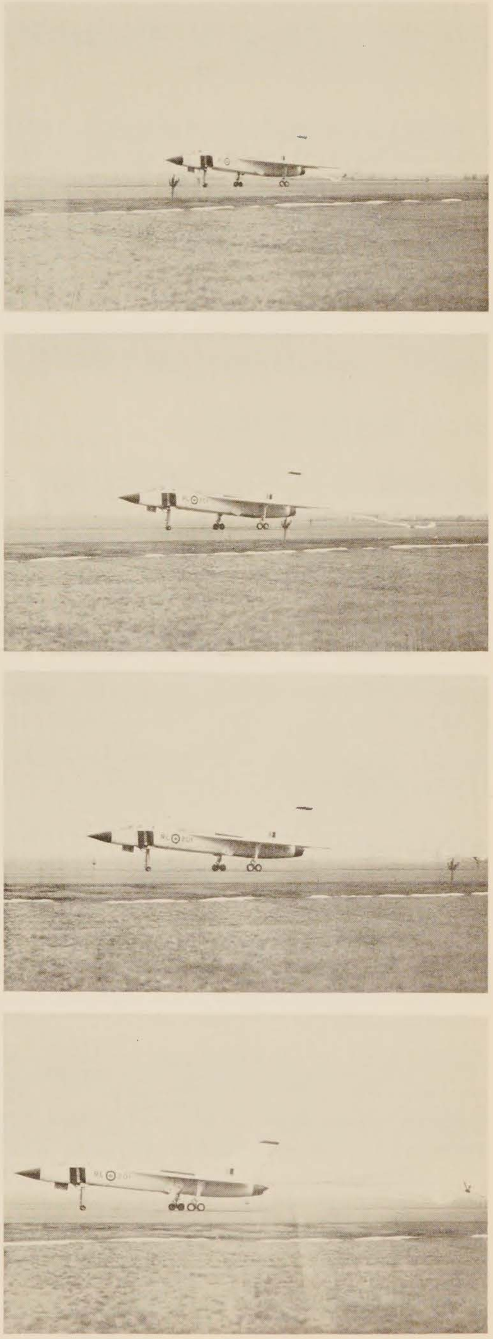

FIG. 44 DRAG CHUTE TESTS

wheels and brakes overheated, and the tires burst on the right-hand main landing gear.

The nose was found to lift at speeds from 115 kts and upwards, depending on the elevator trim.

Damper system tests revealed that the yaw sensing lateral accelerometer was influenced by vibration in the airframe, initiated by fin oscillations. The consequent spurious yaw signals kept the rudder in a condition of oscillation. The sensitivity of the damper system was therefore reduced in an effort to minimize the rudder oscillation. This was successful, but was not considered acceptable for flying at speeds in excess of 350 kts. Subsequent to the first flight, AVRO has been investigating this problem, and a filter circuit has been devised by AVRO which is to undergo tests shortly, when further taxi trials are conducted.

29.2 FIRST FLIGHT

The first flight of the ARROW took place on 25 March 1958. During the 31 minute flight, the aircraft was flown at speeds up to 250 kts, and at altitudes up to 11,000 feet. Take-off was accomplished with the damping system at off, but normal and emergency damping modes were tested during the flight. It must be noted that damping is only operative in the yaw axis of the aircraft at present. After operating the air brakes with the damping system in normal mode, the aircraft was landed with the damping system in normal engagement. During the flight no "up and locked" indications were displayed in the pilot's cockpit for either nose or main landing gear, due to the landing gear actuating system microswitches being incorrectly set.

An air conditioning temperature controller failed in flight, causing only cold air to be supplied to the transformer-rectifier units and other electrical components. The pilot reported that the cockpit temperature was comfortable throughout the flight. No cabin pressurization was required for this flight because of the low altitudes involved. The air conditioning turbine tachometer failed during the flight.

Brake temperatures rose to 210°F during landing, which is considered satisfactory, and the brakes were found to be serviceable for further flights. Damage to the tires was confined to bad tears at the base of the treads. This was considered to have been caused by tire scrubbing when the aircraft turned at the end of its landing run.

During landing, the maximum vertical deflection of the main landing gear was only 1 in. from the static deflection. Landing weight of the aircraft was approximately 56,000 lb.

Vibration of the aircraft was noticed by the pilot just after take-off and was found to be caused by the main landing gear doors buffeting when open. The vibration disappeared when the landing gear was retracted and the doors closed. The drag chute streamed successfully after touchdown, but the pilot chute and main canopy suffered damage when the chute collapsed during taxiing.

The AN/ARC 34 UHF radio used in the aircraft displayed good transmitting and receiving qualities throughout the flight.

The AN/APX-6A (IFF) installation was in operation during the flight. GCI Station Edgar reported that the ARROW responded to Mode 1 interrogations.

Of the 28 parameters which it has been intended to record from the aircraft telemetry system, 20 were successfully monitored.

29.3 SECOND FLIGHT

The second flight took place on I April, and was of 45 minutes duration.

The object of the flight was to extend the pilot's preliminary assessment of the aircraft's handling qualities. Limiting altitude and speed for this flight were specified as 30,000 feet and Mach 0.95.

Due to the failure of the nose landing gear door to retract, no attempt was made to fulfill the test requirements specified in the briefing. Instead, an assessment was made of the aircraft handling within the restrictions (250 knots) imposed by the open door. The efficiency of the cockpit pressurization system was also assessed, up to an altitude of 30,000 feet. The landing gear indicator in the pilot's cockpit again failed to indicate up and locked. This was again caused by maladjustment of the microswitch.

Faults that required post-flight investigation were:

(i) An air conditioning warning indication at 25,000 feet.

(ii) The landing gear indication failure.

(iii) Failure of the UHF transmitter ten minutes after take-off.

(iv) With the damper emergency mode engagement, there was I° of side-slip to the right.

The oxygen system operated satisfactorily during this flight, and oxygen consumption was normal.

Temperature conditions in critical regions around the engine shrouds did not exceed 240°F and this is considered satisfactory.

On servicing the fuel system after flight, a severe fuel leak was found in the line from tank No. 1, at the pressure regulator valve. This leak was due to a damaged "O" ring in the valve connection, caused by misalignment of the line. This was rectified and a new "O" ring fitted.

The air conditioning warning indication noted during flight was considered to be due to cycling of the cooling turbine. This has now been temporarily disconnected in an attempt to improve cockpit air conditioning conditions. The air turbine outlet temperature control system is to be replaced with a system utilizing an AiResearch pneumatic controller.

The drag chute operated normally and sustained no damage during the landing and subsequent taxiing. The tires were again damaged by tire scrubbing on sharp turns during taxiing. The left-hand rear tire has been replaced.

Sanborn recorders were set up to record 15 parameters during the flight. Two channels failed to operate. It was again found that directional control in flight was improved by use of the damper in either normal or emergency mode.

29.4 FLIGHT TEST INSTRUMENTATION

Design and installation of instrumentation for use in the taxi tests and flight tests continued. The telemetry system was in its operation prior to the flight testing.

29.4.1 TELECOMMUNICATION AND NAVIGATION SYSTEM

Servicing of the telecommunication trailer and its equipment has continued in readiness for further antenna testing.

29.4.2 STRAIN GAUGE INSTRUMENTATION IN ARROW AIRCRAFT

29.4.2.1 Aircraft 25201

A complete check of all strain gauges installed in aircraft 25201 revealed that approximately thirty gauges had been damaged by personnel working on the aircraft. These will be replaced following the installation of the instrumentation wiring patch panel. Rewiring of the patch panel, necessitated by the decision to use smaller gauge wiring is in progress and will be completed by April.

The installation of the transducers is approximately 80% complete.

29.4.2.2 Aircraft 25202

Installation of strain gauges in aircraft 25202 is complete, but awaits inspection.

Rewiring of the patch panel is proceeding, and transducer installation is approximately 60% complete.

29.4.2.3 Aircraft 25203

Installation of all available gauges in aircraft 25203 has been completed. Further installation is delayed, pending delivery of additional strain gauges.

Transducer installation is approximately 40% complete.

29.4.3 DATATAPE DEVELOPMENT TESTING FOR THE ARROW

All compound modulation/frequency modulating units manufactured by Consolidated Engineering Corporation for use in conjunction with potentiometer transducers have been modified to increase their effective input impedance. Some component failures have occurred in datatape units which had been subjected to several hours continuous service. This is symptomatic of components overheating, and cooling and ventilating of the datatape equipment is under investigation. Further investigations have been made into the operation of datatape equipment installed in a CF-100 aircraft. The ground station decommutor was found to be operating satisfactorily. Further experimentation with the airborne system confirmed that the signal pulse characteristics were improved by increasing the record current. The contacts in the commutator switches were polished to provide distortion-free output signals. Subsequent ground tests performed on the datatape equipment in the CF-100 gave good results. Flight tests, however, have proved unsatisfactory and records have been unintelligible. Further investigations of the datatape system are necessary and the future test program is under review.

29.4.4 GROUND STATION TELEMENTRY ARRANGEMENT

The telemetry ground station has operated satisfactorily during the many system tests conducted during January and February. During the taxi trials conducted in March, the telemetry ground station was in operation, and monitoring of the instrumentation associated with the telemetry system was satisfactory, with the exception of the intelligence transmitted by the PW/M/ASCOP system. The FM/FM continuous transmission data was subject to periodic fade during flight tests and further investigations of antenna location on the aircraft are progressing.

The commutated signals proved unsatisfactory, and the future use of low level commutated signals is under review.

29.4.5 DATA PROCESSING

The Sanborn recorders in the readout room have been in continuous use during the past month. Their performance has been satisfactory, though damage to pens has resulted from the high frequency, large amplitude noise emanating from the telemetry receiver when no signal is present. Twenty pens were damaged, and safeguarding measures are under investigation.

A new design of millisadic intervalometer with a reduced sampling rate is being developed. Its purpose is to enable the most economical use of the IBM machines used for recording and sorting information gained during flight tests.

30.0 SPECIFICATIONS ISSUED

30.1 MODEL SPECIFICATIONS

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

30.2 AVROCAN SPECIFICATIONS

To date, approximately 420 AVROCAN equipment specifications (series 'E') have been issued for the ARROW. An index of these specifications (AVRO ref. E.I. Gen. 489/193 dated 31 March 1958) has been issued to the RCAF. Additional AVROCAN specifications not included on the index, are listed below.

Specification No.	Description	Issued
M-16-2	Grease, Aircraft -65° to + 400°F Temperature Range	Mar. 1958
P-3-3	Hard Anodizing, Aluminum Alloys	Mar. 1958
P-10-3	Application of Epoxy Base Primer and Enamel	Mar. 1958
P-14-5	Processing of Iron Base Alloy (N-155) Sheets	Mar. 1958
P-14-6	Processing of Chromium, Molybdenum Vanadium Steel for Aircraft Parts	Mar. 1958

30.3 DESIGN CERTIFICATES

71/PROJ/7/1	Design Certificate for Flight Trials of ARROW 1 - Aircraft Serial #25201	Feb. 1958
	Amendment #1	Mar. 1958
	Amendment #2	Mar. 1958
	Amendment #3	Mar. 1958

31.0 REPORTS ISSUED

31.1 PRELIMINARY DESIGN PROPOSALS

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

31.2 WEIGHT AND BALANCE REPORTS

Since weight and balance reports are issued monthly, as required by CAP 479, a further index of these reports will not be included in the Quarterly Technical Report. The following report was issued, but was not previously recorded.

Report No.	Description	Issued
71/WEIGHTS/1	Weight Distribution for CF-105	Feb. 1958

31.3 WIND TUNNEL DATA

Report No.	Description	Issued
71-2/W.TUNN/3	Missile Crossplots - CAL Windtunnel Tests Feb. 1957	Jan. 1958
71/W.TUNN/6	Wind Sensor WT Tests (Data Reduction Feb 1958 - CAL	Mar. 1958
71/W.TUNN/7	Wind Sensor W T. Tests (Operations Log and IBM Data) Feb 1958 - CAL	Mar. 1958
71/W.TUNN/8	Wind Sensor WT Tests (Vane Calibration Log) Feb 1958 - CAL	Mar. 1958
71/W.TUNN/9	Wind Sensor WT Tests (Plots and Correction Data) Feb 1958 - CAL	Mar. 1958

31.4 PERFORMANCE REPORTS

Report No.	Description	Issued
13	Periodic Performance Report (ARROW 2)	Jan. 1958
72/PERF/13	Performance for Employment in SAGE	Feb. 1958
72/PERF/16	Improvement of ARROW 2 Overload Range Mission	Mar. 1958

31.5 STRUCTURAL STRENGTH TESTS

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

31.6 AIRCRAFT GROUND * AND FLIGHT TEST

Report No.	Description	Issued
72/FAR/9.155.3	Instrumentation - Aircraft 25208 (ARROW 2)	Jan. 1958
71/FAR/10	Launching and Jettison Tests on ARROW 1 Photographic Coverage Required for Sparrow Lowering	Jan. 1958
71/FAR/12	Reduction and Presentation of Flight Test Data for Stability Analysis	Feb. 1958
71/FAR/13	Stability Derivatives from Steady State Tests	Feb. 1958

* No Ground Test reports are available

31.7 FUNCTION TYPE TESTS

Each item of equipment 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 at AVRO.

31.8 VENDORS' REPORTS

Vendors' reports on equipment supplied to AVRO, for use on 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 procurred to an AVROCAN specification.

31.9 ASTRA 1 SYSTEM

AVRO has not compiled any formal reports relative to the ASTRA 1 System.

31.10 STRESS ANALYSIS REPORTS

Report No.	Issue	Description	Issued
7.0500/27	1	Rigid Wing Landing Case Loads	Jan. 1958

Report No.	Issue	Description	Issued
7/0500/34	1	ARROW MK 1 & 2 - Aileron Fatigue Loading - Case 27.7 (a)	Jan. 1958
7/0500/36	1	ARROW MK 1 - Jacking to Determine Wing Deflections	Feb. 1958
7/0500/38	2	Sparrow 2 Missile Pack - Stress Office Test Requirement	Mar. 1958
7/0554/38	1	Former & Bottom Longeron Analysis Due to Pressure Loading in Missile	Feb. 1958
7/0554/42	1	Expected Stresses - Test Former Under Load	Jan. 1958
7/0556/20	3	Manufacturing Joint at Station 485	Mar. 1958
7/0558/66	1	Side Skin - Station 485 to 742	Mar. 1958
7/0559/11	1	Rear Fuselage Rudder Dorsal Fairing Stress	Mar. 1958
7/0562/17	1	Outer Trailing Edge	Feb. 1958
7/0562/18	1	Front Spar & Leading Edge - Inner Wing	Feb. 1958
7/0583/1	1	Fin Stringer Post - Test Box	Jan. 1958
7/0584/7	1	Tension Field Analysis of Deflection Rudder	Jan. 1958
7/0591/3	2	Nose Landing Gear Door	Mar. 1958

31.11 SYSTEMS

Report No.	Description	Issued
70/SYSTEM 11/93	Electrical System - Loads and Controls	Jan. 1958
70/SYSTEM 13/84	Assessment of the Simple Stable Platform	Feb. 1958
70/SYSTEM 13/101	ARROW Model Antenna - Pattern Tests	Jan. 1958
70/SYSTEM 13/109	Antenna Work on the ARROW by Sinclair Radio Ltd.	Feb. 1958

Report No.	Description	Issued
70/SYSTEM 13/117	Sinclair Radio Ltd. Work Schedule for March 1958	Feb. 1958
70/SYSTEM 13/131	Progress Report for Sinclair Radio Ltd. from Jan. 16 to Mar. 16, 1958	Mar. 1958
70/SYSTEM 19/85	Drainage of Aircraft Hydraulic System	Jan. 1958
70/SYSTEM 24/43	Development Program for the Crew Escape System in ARROW Aircraft	Feb. 1958
70/SYSTEM 24/89	Theory of Trajectories of Ejected Bodies and Conditions for Seat Requirements to Miss the Aircraft Fin	Feb. 1958
70/SYSTEM 24/90	Study of the Thrust Required to Keep Deceleration of Flying Bodies. Such as Ejected Seats Within Predetermined Limits	Jan. 1958
70/SYSTEM 24/125	Appraisal of the Existing ARROW Escape System	Mar. 1958
71/SYSTEM 13/91	UHF Antenna Coverage - A/C 25201	Jan. 1958
71/SYSTEM 13/97	The AiResearch q_c Actuator System	Jan. 1958
71/SYSTEM 13/99	Temporary Solution to the Problems Encountered on the ARROW Simulator	Jan. 1958
71/SYSTEM 13/100	Simulator Flying Controls	Jan. 1958
71/SYSTEM 13/102	Modification to Scheduling Servo Amplifiers	Jan 1958
71/SYSTEM 13/113	Post-Installation Check Setting-up Procedure of the q_c Actuator	Feb. 1958
71/SYSTEM 13/121	Post-Installation Check of ARROW Antenna for A/C #25206, 25207, 25208 & 25211	Mar. 1958
71/SYSTEM 15/94	Development of the Valve Control Dampers	Jan. 1958

Report No.	Description	Issued
71/SYSTEM 15/106	q_c Actuator System Problems	Mar. 1958
71/SYSTEM 15/108	State of the B-1 Rig at Flight Simulation on 23 Jan 1958	Mar. 1958
71/SYSTEM 15/114	Aileron Support with Hydraulics Off	Feb. 1958
71/SYSTEM 15/115	Pre-flight Flying Control Check on A/C 25201	Feb. 1958
71/SYSTEM 15/116	Rudder Centring	Mar. 1958
71/SYSTEM 15/118	Pre-Installation Tests on AVRO q_c System	Feb. 1958
71/SYSTEM 15/119	Control Cable Friction Test	Feb. 1958
71/SYSTEM 15/120	Report on Nose Wheel Steering Valve Investigation	Mar. 1958
71/SYSTEM 16/57	Booster Pump Adequacy in 19 psi Fuel System	Jan. 1958
71/SYSTEM 16/98	First Aircraft Fuel System Flight Limitations	Jan. 1958
71/SYSTEM 18/92	Low Pressure Pneumatics	Jan. 1958
71/SYSTEM 19/81	Nose Landing Gear Retraction Restrictor	Jan. 1958
71/SYSTEM 19/86	Nose Landing Gear Door - Closing Scheme	Jan. 1958
71/SYSTEM 19/104	ARROW Braking Equipment & Performance	Feb. 1958
71-2/SYSTEM 19/129	Anti-skid Installation	Mar. 1958
71-2/SYSTEM 19/124	Nose Gear Door Jack - Pressure Investigation	Mar. 1958
71/SYSTEM 22/133	Temperature Setting of Turbine Outlet Limiting Regulator	Mar. 1958

Report No.	Description	Issued
71/SYSTEM 25/123	Engine and Accessory Oil System Production	Mar. 1958
71/SYSTEM 26/78	Redesigned Missile System for Aircraft 25203	Jan. 1958
71/SYSTEM 32/96	Flying Control Booster Schematic	Jan. 1958
72/SYSTEM 13/95	Power Switching and Launch Control	Feb. 1958
72/SYSTEM 13/121	Post Installation Check of Antenna System for Aircraft 25206, 25207, 25208 & 25211	Mar. 1958
72/SYSTEM 13/134	Weapon System Evaluation - ARROW	Mar. 1958
72/SYSTEM 15/135	Simplification of Elevator and Aileron Booster Circuits	Mar. 1958
72/SYSTEM 15/138	Damper System Reliability Requirements	Mar. 1958
72/SYSTEM 19/80	Dual Pressure Range Pumps	Jan. 1958
72/SYSTEM 19/122	Reclosable Nose Undercarriage Door	Mar. 1958
72/SYSTEM 19/136	Missile/Instrument Pack Schematic	Mar. 1958
72/SYSTEM 22/49	Piping Schematic Cooling Air - Constant Speed Drive and Accessories Gear Box	Jan. 1958
72/SYSTEM 22/127	Requirements for ARROW 2 Cockpit Temperature Control	Mar. 1958
72/SYSTEM 23/110	Extra Fire Protection System	Feb. 1958
72/SYSTEM 23/128	Fire Extinguishing System, Functional Test Procedure	Mar. 1958
72/SYSTEM 26/105	Captive Flight - Sparrow 2 Missile	Mar. 1958
72/SYSTEM 26/112	ARROW 2 Performance Proposals	Feb. 1958
72/SYSTEM 29/130	Constant Speed Drive - Separate Oil System	Mar. 1958

31.12 EQUIPMENT DESIGN - Airborne & Ground Equipment - Maintenance, Reliability

Report No.	Description	Issued
72/GEQ/2	ARROW 2 Technical Proposal for Aircraft Systems Trainer	Jan. 1958
72/GEQ/5	ARROW 2 Development Program - Study of Electrical and Cooling Air Flow Requirements (Malton Facilities)	Jan. 1958
72/GEQ/6	ARROW 2 - Comments on Proposed Alterations to RCAF Readiness Hangars	Feb. 1958
72/GEQ/7	AVRO Associate Contractor Demonstration of ARROW 2 Aircraft	Mar. 1958
72/GEQ/8	ARROW 2 Design Study on Engine Ground Run-up and Testing	Mar. 1958
71/AIREQ 00/1	ARROW 1 Pressure Checking Hydraulic System	Mar. 1958
71/QIREQ 15/1	ARROW 1 Report on Testing q_c Actuator System	Jan. 1958
71/AIREQ 23/1	ARROW Fire Protection Supply Unit	Feb. 1958
71/AIREQ 25/1	A Proposed Method of Controlling Power Plant Equipment Design Changes, and Accessory Utilization Time Cycles	Feb. 1958
71/AIERQ 25/2	Preliminary Operating Instructions ARROW 1 J75-P3 Power Plant Installation (Aircraft 25201 only)	Feb. 1958
71/AIREQ 25/3	Reprogramming of Engine Testing and Installation Calibration - Checks on P3 and P5 Pratt & Whitney Engines	Mar. 1958
71/AIREQ 25/4	A Requirement for the Maintenance of ARROW 1 Engine Installations and Associated Equipment	Mar. 1958

Report No.	Description	Issued
71/AIREQ 25/5	Special Maintenance Instructions - Daily and Periodic Inspection Pre-flight - Engine Performance Check ARROW 1	Mar. 1958
72/AIREQ 13/6	Cockpit Instruments and Damper Equipment (A/C 25206, 25207 & 25208)	Jan. 1958
72/AIREQ 16/2	Investigation of the Feasibility of Developing an Improved Fuel-No-Air Valve	Mar. 1958
72/AIREQ 95/1	ARROW 2 - Lucas-Rotax Pneumatic Starter, Power Plant	Feb. 1958
71/REL 00/1	ARROW 1 Qualification Status, Airborne Equipment (Amendment 2)	Feb. 1958
71/REL 11/6	Qualification Test History of CS-R-122 Relay (10 amp. Class B Hermetically Sealed) Manufactured by U.S. Relay Co. Ltd.	Mar. 1958
71/REL 11/7	ARROW 1 - Report on Meeting Called by RCAF to Discuss the Alternator and Transformer Rectifier Unit	Mar. 1958
71/REL 11/8	ARROW 1 - Qualification Testing of Lucas-Rotax Ltd. Transformer - Rectifier Units with Zener Diodes	Mar. 1958
71/REL 16/2	12 1/2 Hour Inspection of the Flow Proportioning Units Installed on ARROW 1 (A/C 25201)	Feb. 1958
71/REL 22/2	ARROW 1 - Vibration Test History Investigation at Barbour Colman Co. on AVRO Part No. 7-2252-2 Air Conditioning	Feb. 1958
71/REL 32/1	ARROW 1 - Valve Minifold Delivery - Flying Control Hydraulic System	Feb. 1958
71/MAINT 00/4	Camlock Fastener Summary	Mar. 1958

Report No.	Description	Issued
71/MAINT 13/7	Maintenance Instruction - Electronics Interphone System AN/AIC-10	Jan. 1958
71/MAINT 16/3	Maintenance Instruction - Refuelling and Defuelling - Fuel System	Jan. 1958

31.13 GENERAL TECHNICAL DESIGN

Report No.	Description	Issued
72/INT AERO/8	Windmilling Buzz Boundaries	Mar. 1958
71-2/AERO DATA/6	Revised Rigid Rudder Derivatives in Body Axes	Jan. 1958
72/AERO DATA/8	External Tank Drag	Mar. 1958
72/ELASTICS/3	Review of Noise Problem of ARROW Aircraft	Feb. 1958
GEN/ELASTICS/1	Comparison with Test of Analytically Derived Influence Coefficients, for a Straight Aspect Ratio Model Wing	Feb. 1958
70/STAB/15	ARROW Lateral Dynamic Stability	Jan. 1958
71/STAB/16	Digital Computation and Analysis of ARROW Lateral Response in Emergency Mode	Feb. 1958
71/STAB/17	Digital Computation and Analysis of ARROW Longitudinal Response in Emergency Mode	Feb. 1958
71/STAB/27	Longitudinal Stability of the Elastic Aircraft	Mar. 1958
72/STAB/26	Aerodynamic Effects of Underwing Fuel Tanks	Mar. 1958
70/LOADS/6	Pilots Canopy - Dynamic Loads	Feb. 1958
72/LOADS/10	Preliminary Load Analysis - Infra-Red Seeker	Feb. 1958

Report No.	Description	Issued
71/GEOM/1	Control System Kinematics	Mar. 1958
70/THERMO/7	Engine Bay Cooling	Feb. 1958
70/THERMO/20	A Sample Analytical Solution for Temperature Response in an Insulated Skin	Mar. 1958
70/THERMO/22	Calculation of Two-Dimensional Temperature and Thermal Stress, Distributions in Structures	Feb. 1958
70/THERMO/25	Summary of Investigations of Temperature and Thermal Stress Distributions	Mar. 1958
71/THERMO/4	Overload Air Bleeds	Feb. 1958
71/THERMO/5	Approach and Landing Transight	Feb. 1958
72/THERMO/6	Permament Engine Rail	Feb. 1958
72/THERMO/12	Determination of Heat Transfer Coefficients Over a Wing at Supersonic Speeds with Reference to Unequal Heating Rates of Upper and Lower Surfaces	Feb. 1958
GEN/THERMO/1	Effective Changes in Local Heat Transfer Coefficient on the Total Heat Transfer Coefficient from Boundary Layer to Fuel	Mar. 1958
71/SIMUL/11	Evaluation of the Required Instrument Responses for Flight Simulator	Jan. 1958
71/SIMUL/13	Tests on Simulator Control System	Jan. 1958
71/SIMUL/15	Check on the Yaw Damper Installed in A/C 25201	Feb. 1958
72/TACTICS/1	A Preliminary Study of the Tactical Implications of Lowering Missiles Independently in Pairs	Feb. 1958

Report No.	Description	Issued
72/TACTICS/3	Calculation of the Angular Firing Error from the Computed Miss Distance	Feb. 1958
72/TACTICS/5	Preliminary Study of the Steering Gain	Feb. 1958
72/TACTICS/7	Effect of Trim, Limit Conversion Manoeuvres on the Terminal Phase Capabilities of the ARROW When Operating Against Equal Speed Targets	Mar. 1958
72/TACTICS/8	Notes on Digital Computer Programms for Tactical Evaluation of the ARROW	Mar. 1958

31.14 METALLURGICAL REPORTS

Report No.	Description	Issued
M 4098 Add Il	The evaluation of Epon 828 Resin with Various Reinforcing Materials as a Faying Surface Filler on Machined Aluminum Alloy Surfaces	Jan-Mar. 1958
M 4521 Add 1	To establish a reinjection limit for PR 701 M Channel Sealant	Jan-Mar. 1958
M 4777	Tensile Properties of 2024-T81 Sheet to T86 Stretch Formed Sheet	Jan-Mar. 1958
M 4780	The Effect of Shot Peening on the Fatigue Strength of EN30B and 7075 Alloy	Jan-Mar. 1958
M 4782 & M 4782 Add. I	Examination of Some AM350 Strips Dimpled in Sub-zero Cooled and Tempered Condition	Jan-Mar. 1958
M 4788	Correlation of Ultrasonic "Blip" Sizes with Fatigue Properties of 7075-T6 Aluminum Alloy Plate	Jan-Mar. 1958
M 4790	Examination of Vinson Hydraulic Valve Unit Vinson Part No. A 60240	Jan-Mar. 1958
M 4808	Strength of Welds in Cast ZH62 Magnesium Alloy	Jan-Mar. 1958

Report No.	Description	Issued
M 4825	Tensile Property Survey of Androformed Alclad 2024-T3 Aluminum Sheet Panel After Aging to 2024-T81	Jan-Mar. 1958
M 4827	The Bonding of Ramp No. 1 De-icer Boot, ARROW 1 for Qualification Test	Jan-Mar. 1958
M 4836	Examination of Third Stress Relieved 7079-T65 Sample Hand Forging Supplied by Kaiser Aluminum Company	Jan-Mar. 1958
M 4837	Examination of Hydraulic Compensator on 1st ARROW Aircraft	Jan-Mar. 1958
M 4848	The Evaluation of Braided Glass Sleeving as a Means of Identing Cables in Fuel Tanks	Jan-Mar. 1958
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