Avro Aircraft Limited

INTER-DEPARTMENTAL MEMORANDUM

Ref: 8355/20/J

Date: July 23, 1957
To: Mr. J.D. Hodge-Technical Flight Test Co-Ordinator
From: J.H. Lucas-Chief of Performance Evaluation
Subject: INSTRUMENTATION REQUIREMENTS FOR PERFORMANCE TESTING OF ARROW II AIRCRAFT WITH COMMENTS ON INSTRUMENTATION FOR ARROW I AIRCRAFT

Attached herewith, please find report 70/Perf/1 on preliminary testing and instrumentation requirements for performance testing of the Arrow II aircraft. It is assumed that two aircraft (numbers 6 & 8) will be available for detailed performance testing and the included list of tests and list of instrumentation should apply. A third aircraft (number 7) may be available for preliminary "airworthiness" tests, which would then be delivered to Orenda Engines Limited for evaluation of the engines. Certain instrumentation should be available in that aircraft (as indicated), in addition to Orenda Engines Limited requirements, in order to obtain general performance data during the preliminary airworthiness tests and possibly during Orenda Engines Limited testing.

It is also assumed that Arrow I aircraft (number 1) will not be devoted specifically to performance testing, although general performance data may be obtainable with the existing instrumentation during the course of other tests. Some compromise between available instrumentation and required performance data will be necessary. A final list of available instrumentation should therefore be made in order to assess the extent of any performance programme.

Appendix I to report 70/Perf/1 summarizes a thrust measurement technique and applicable formula for performance testing of Arrow II aircraft. Appendix II will be issued to cover engine or thrust measurement data obtainable from Arrow I instrumentation.

RW/df

John Lucas

J Lucas

cc Messrs:
J.A. Chamberlin
F. Brame
W.D. Raymond
Central Files - without encl.

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

AIRCRAFT: ARROW 1 & II

REPORT No 70/Perf/1

FILE No

NO OF SHEETS: 18

TITLE:

PRELIMINARY TESTING AND INSTRUMENTATION REQUIREMENTS FOR PERFORMANCE TESTING

PREPARED BY R. Waechter DATE July 1957

CHECKED BY DATE

SUPERVISED BY DATE

APPROVED BY [signature] DATE July/57

ISSUE No	REVISION No	REVISED BY	APPROVED BY	DATE	REMARKS

FORM 1316A

PERFORMANCE

PRELIMINARY TESTING INSTRUMENTATION REQUIREMENTS FOR PERFORMANCE TESTING

Performance tests required on Arrow II aircraft are summarized below, together with a discussion of techniques. It should be realized that test technique is being further investigated and is therefore given in general terms and is subject to change and refinement.

a) Aircraft Drag and Fuel Consumption Data: Previously (on C-100 aircraft) this data was obtained from numerous steady level flight measurements. On supersonic aircraft, such as the Arrow, a vast number of steady level speed points would be required to cover the complete speed and altitude range. Also, it is anticipated that excessive time would be required to stabilize in steady supersonic level flight. Hence other methods are to be employed with only several check points in the steady level flight condition. It is recommended that continuous lift and drag measurements be made by accelerometer and/or energy method. For the accelerometer method, use is made of the following formulae:-

1) $\Delta\theta = g/V\left(\left[n - \cos(\alpha + \theta)\right] \cos\alpha + \left[{}^{A}L - \sin(\alpha + \theta)\right] \sin\alpha \right) \Delta t$

2) $V = U_o + \left(\left[{}^{A}L - \sin(\alpha + \theta)\right] \cos\alpha - \left[n - \cos(\alpha + \theta)\right] \sin\alpha \right) \Delta t$

3) $H_p = H_o + V \sin\theta \times \left({}^{T}s/T\right) \Delta t$

4) $C_D = \left(\dfrac{F_N - W\,{}^{A}L}{qs}\right) \cos\alpha + \dfrac{nW}{qs} \sin\alpha$

5) $C_L = \dfrac{nW}{qs} \times \cos\alpha - \left(\dfrac{F_N - W\,{}^{A}L}{qs}\right) \sin\alpha$

(4 and 5) ref NACA TN3821 App. A

Equations 1, 2 and 3 are applicable for manoeuvres in a single vertical plane only (i.e. straight pull ups, push overs and loops) where n = normal acceleration as measured normal to A/C datum.

A_L = longitudinal acceleration as measured parallel to A/C datum.

α = measured angle of attack corrected for A/C pitching, bending, and upwash.

θ = angle of flight path relative to horizontal.

Uo = initial T.A.S. for each increment, originally measured under steady state conditions.

Ho = initial pressure attitude for each increment, originally measured under steady state conditions.

$V \sin\theta \times {}^{Ts}/T$ = increment in pressure altitude in which ${}^{Ts}/T$ would need to be neglected for convenience.

In equation 1 successive approximations for θ are necessary to determine $\Delta\theta$. However, one approximation for θ (or using initial θ) for each increment will suffice providing Δt is small.

For the energy method, speed and altitude are measured directly but account must be taken for A.S.I. pressure lag. Lift is determined using the same formula #5 where "n" must be measured and A_L determined from speed measurements or $\sin\alpha$ assumed equal to zero. Drag is determined from the formula : 6) $C_D = \frac{F_N \cos\alpha}{q S} - \frac{W}{q S}\left(\frac{\frac{dp}{dt}}{\rho V} + \frac{\frac{dV}{dt}}{g}\right)$

The accelerometer method is preferred (ref: NACA TN 3821) for which an accurate accelerometer design is given in the referenced report; but the energy method does not require additional instrumentation and therefore can be used as a standby or complimentary method. Both methods require accurate thrust measurement.

Considering now flight technique, it is assumed that gentle or gradual menoeuvres will be carried out and a complete time history of lift, drag and Mach number will be obtainable. One of these variables must be held constant for each run and used as a parameter for plotting the remaining two. Holding lift constant entails carrying out accelerated and decelerated level flight measurements; while holding drag constant entails holding constant (high) power in steady level flight and then gradually tightening into a turn at constant altitude. To hold Mach number constant requires either a climb test or a test initiated in steady level flight at low power and then a gradual increase in power and normal acceleration. A combination of methods is desireable for cross checking, but a primary method should be chosen which will cover as much as possible of the speed and altitude range of the aircraft. It should be noted that if power is steady or changes are gradual it may be possible to obtain fuel consumption data as well, otherwise special tests will need to be carried out.

b) Position Error Data: Position error by the Aneroid (or tower fly past) method is deemed the most convenient and accurate method. It is limited to low altitude and to the maximum permissible speed of the aircraft at or near sea level which for the Arrow II is 700 knots or M = 1.06. At or near this Mach number the position error should fall to approximately zero due to passing of a shock wave over the pitot static tube. Hence it may or may not be necessary to test to higher Mach number. Usually it is sufficiently accurate to "expand" sea level position error data to high altitude but it may be necessary to check the "expansion" at the extreme altitudes of the Arrow II aircraft. If position error data must be established beyond M = 1.06 and at high altitude, (or at varying lift coefficients), the availability of accurate radar tracking facilities should be investigated. Other position error methods exist such as the pacing method (requiring a second aircraft which is accurately calibrated) the speed course method (requiring calibrated vertical ground cameras) and the trailing bomb method (requiring a trailing bomb which trails beyond the sphere of influence of the aircraft). The trailing bomb methods is usually used to determine position error data at or near the stall; but since the Arrow aircraft does not normally operate very near the stall, this method may not be necessary.

c) <u>Climb and Descent Tests</u>: Climb and descent data will be required to check the overal fuel, time and distance to and from altitude; and also to find an optimum technique for best fuel, best time, and best fuel for a given time, to an altitude and speed. Partial climbs will be required to determine the optimum technique to obtain maximum altitude from a "zoom and push over".

d) <u>Take-off and Landing Tests</u>: Take-off and landing data will be obtained from ground measurements for which a long range ground camera with telephoto lens is required and which should cater for the lengthy ground runs (up to at least 6000 feet of the Arrow aircraft).

Instrumentation Requirements are as follows:-

1.0 Pilot's Cockpit Instrumentation

1.1 A.S.I.
1.2 Altimeter
1.3 Machmeter
1.4 Fuel remaining gauges
1.5 Instrumentation recorder controls for various sampling frequencies (1 and 20 per second)
1.6 Signal switch to code instrumentation records. This switch should also operate a "radio link" signal when necessary (for take-off and landing tests) which will instantaniously code ground camera and A/C instrumentation records.

	2.0 Recording Instrumentation	Range	Required Accuracy	Remarks
* 2.1	Record number	-	-	-
2.2	Sampling frequency (at pilot's descretion)	1 & 20 per sec.	-	-
* 2.3	Time scale	-	± 0.1 sec	-
* 2.4	Coding signal	-	"	-
2.5	Parachute selection sig.	-	"	-
* 2.6	A.S.I. diff. pressure (3 off)	0-720,1440 & 2880 psf	± 75% of max	-
2.7	Sensitive A.S.I. diff. pressure	0-144 psf	"	for special tests
* 2.8	Ambient static press. (3 off)	0-288,720 & 2160 psf	"	-
2.9	Sensitive static pressure	1800-2300 psf	± 7 psf	for aneroid P.E.
* 2.10	Ambient air temp.	-65 + 350°F	± .5% of max.	1 sec. response
2.11	Longitudinal acc.	± 1 g	± 1% of max	
2.12	Normal acc.	-3 + 8 g	±.5% of max	-
* 2.13	Elevator angle	-30+20 deg	± .3 deg.	-
2.14	Dive brake angle	0-65 deg.	± 2 deg.	-
2.15	Angle of attack	-10+20 deg.	± .1 deg.	-
2.16	Pitching velocity	-30+30 deg/sec	± .5 sec.	-
2.17	L.P. Comp. r.p.m.	0-110% of max	±.5% of max	port & strb'd
2.18	H.P. Comp. r.p.m.	0-110% of max	±.5% of max	port & strb'd
2.19	Engine fuel flow	600-25000 pph	±.5% of max	port & strb'd
2.20	A/B fuel flow	0-40,000 pph	±.5% of max	port & strb'd
2.21	Total fuel gone	0-3000 gals	±.5% of max	port & strb'd

2.0 Recording Instrumentation	Required Range	Required Accuracy	Remarks
2.22 Inlet duct (near engine) (port & stbd)			
a) dynamic pressure (P_1-p_1)	0-1440 psf	±.75% of max	1 or 2 rakes across the diameter with $10P_1$, $10p_1$ & $6T_1$ probes each. The probes should represent equal annular areas & be manifolded together
	0-720 psf	±.75% of max	
b) static pressure (p_1)	0-5000 psf	±.75% of max	
	0-2160 psf	±.75% of max	
	0-288 psf	±.75% of max	
c) Total Temperature (T_1)	-65 +350°F	±.5% of max.	
2.23 Engine inlet (@ comp. face) (port & Stbd)			
a) dynamic pressure (P_1-p_1)	0-1440 psf	±.75% of max	3 or 4 radial rakes with $5P_1$ $5p_1$ & $3T_1$ each. Each probe should represent equal areas and be manifolded together
	0-720 Psf	±.75% of max	
b) static pressure (p_1)	0-5000 psf	±.75% of max	
	0-2160 psf	±.75% of max	
	0-288 psf	±.75% of max	
2.24 Secondary airflow (@ primary nozzle exit plane)(port & stbd)			
a) dynamic pressure	0-720 psf	±1% of max	average readings from 2 or 3 rakes with 3 probes each
b) static pressure	0-5000 psf	±1% of max	
	0-2160 psf	±1% of max	
c) Total temperature	-65 +550°F	±1% of max	
2.25 Extractor wall statics (inside face, aft of primary nozzle exit, Port & stbd)	0-3000 psf	±1% of max	Approx. 3 longitudinal rows of 8 to 10 vents each.
	0-720 psf	±1% of max	
2.26 Primary nozzle diameter	28" - 38"	±.1"	Average of 3 locations
2.27 By-pass door position	open or closed	-	1 signal switch
2.28 Blow in door position	open or closed	-	1 or 2 signal switches
2.29 Turbine outlet (port & stbd)			
a) total pressure (P_6)	0-8500 psf	±.5% of max	Average readings from at least 1 rake of 5 probes each
	0-3500 psf	±.5% of max	
b) total temperature (T_6)	0-800°C	±.5% of max	
2.30 Swinging Probe (at extractor exit, port & stbd)			
a) dynamic pressure (P_E-p_E)	0-6000 psf	±.5% of max	Swing probes with 4 second immersion @ 5 rpm
	0-2880 psf	±.5% of max	
b) static pressure (p_E)	0-3000 psf	±.5% of max	
	0-720 psf	±.5% of max	
c) Probe position across extractor	60"	±0.1"	

AVRO AIRCRAFT LIMITED
MALTON - ONTARIO
TECHNICAL DEPARTMENT

AIRCRAFT: ARROW	PERFORMANCE	REPORT NO 70/Perf/1	SHEET NO 7
		PREPARED BY: R. Waechter	DATE: July 23/57
		CHECKED BY:	DATE:

3.0 Ground Equipment	Range	Required Accuracy	Remarks
3.1 Engine test bed facilities for thrust measurement	0-26,000 lbs	± .5% of max	
3.2 Aircraft thrust stand facilitites for thrust measurement	0-50,000 lbs	± .5% of max	to be invest-igated
3.3 Aircraft weighting scale	0-70,000 lbs	± .5% of max	
3.4 Radar tracking (P.E.) facilities			
a) height measurement	20,000-80,000'	± .2% of max	to be invest-igated
b) speed measurement	200-1300 knots	± .2% of max	
3.5 Cine camera for T.O and landing meas.			
a) azimuth scale	± 60°	± .1% of max	@ 16 frames per sec.
b) time scale	3 min	± .1% of max	
3.6 Camera for aneroid P.E. tests			
a) height measurement	0-500'	± 1% of max	
3.7 Fuel S.G. hydrometer	.7 to .9	± .002	
3.8 Refuelling gauge	0-3000 gals	± 10 gals	
3.9 Anemometer	0-50 mph	± 1 mph	
3.10 Barometer	28-31" Hg	± .05"	
3.11 Thermometer	± 40°C	± .5°C	

FORM 1319 A

4.0 Additional Equipment

4.1 An air or water cooled jet pipe total head pressure rake installed near the final nozzle of the primary jet, for engine test bed measurements. As many probes as conveniently possible should be incoporated on the rake, spaced to represent equal anular areas and then manifolded to record an average pressure.

Note 1: Items denoted (*) should also be incorporated on Arrow II aircraft ho. 7, for preliminary "airworthiness" tests prior to delivery to Orenda Engine Limited. It is assumed that engine instrumentation (such as RPM, fuel flow and air flow) will be installed according to Orenda Engines Limited requirements.

2: Items 2.22 to 2.30 inclusive constitute a preliminary list of instrumentation requirements for thrust measurement on the Arrow II aircraft (numbers 6 and 8). From the measurements it should be possible to determine total (exit) gross thrust as well as primary and secondary contributions. Momentum drag can also be determined, from engine inlet measurements, including and excluding that through the by-pass. Measurement of airflows to overboard cooler and from compressor airbleed are deemed unnecessary with the existing instrumentation. Investigations are being carried out to determine whether this instrumentation is practical and whether alternative methods would be more practical. (see Appendix I on

Technique and Formula for Use in Thrust Measurement).

3: All pitot measurements in or around the engine should be measured relative to static (i.e. dynamic pressure) in order to decrease the required range and thereby increase the accuracy. Where corresponding static pressure is not available (turbine outlet), total head pressure may be measured on an absolute pressure transducer.

4: A.S.I. data or ambient pitot-static measurements for the test system should be obtained from the nose boom "pitot and primary static" system with the differential and static pressure transducers mounted as near to the source as possible. No other test instrumentation volumes should be incorporated.

5: It is assumed that Arrow I aircraft (number 1) will not be devoted specifically to performance testing, although general performance data may be obtainable with the existing instrumentation during the course of other tests. In general, all items, 2.1 to 2.29 inclusive, will be available in various stages of initial flight testing of that aircraft, with the probable exceptions of:-

a) sensitive dynamic and static pressures (items 2.7 & 2.9)

b) secondary air flow measurements at final nozzle exit plane and extractor wall statics (items 2.24 & 2.25)

c) final nozzle diameter and by-pass and blow-in-door positions (items 2.26 - 2.28)

If possible, these should be measured simultaneously with remaining instrumentation such that preliminary performance data may be obtained when necessary.

Primary thrust data can be obtained from the turbine outlet total pressure measurements, providing a calibration of this is made on an engine test bed. The calibration determines effective final nozzle area and total pressure loss from the turbine outlet to the final nozzle. Determination of total pressure loss requires the measurement of total pressure near the final nozzle by means of air cooled or water cooled probes, and although this is highly desireable, it may be omitted and effective final nozzle area based on turbine outlet total pressure.

Also, during Arrow I testing, A.S.I. differential and ambient static pressure measurements may be required in duplicate, but measured from independent sources. i.e. Basic data will be measured from the "nose boom pitot and primary static system while reference data will be measured from the lower fin pitot static (see P/Systems/18).

APPENDIX I

TECHNIQUE & FORMULAE FOR USE IN TURBO-JET THRUST MEASUREMENT

1) Swinging Boom

Total (primary and extractor) exit gross thrust should be determined from the swinging boom which will measure total pressure P_E and static pressure $\not{P}_E$ on a traverse across the diameter of the exit. The traverse will not be precisely across the diameter of the exit due to the arc described by the boom but by measuring boom position it should be possible to plot boom measurements versus representative anular areas to cover the complete ejector area. Gross thrust is then calculated as follows:-

$$X_{GE} = m_E V_E + (\not{P}_E - \not{P}) A_E = A_E V_E^2 \rho_E + (\not{P}_E - \not{P}) A_E$$

$$\text{Now } V_E = M_E a_E \quad \text{and} \quad M_E = \sqrt{\left[\left(\frac{P_E}{\not{P}_E}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]\frac{2}{\gamma-1}}$$

$$\& \; a = \sqrt{\frac{\gamma \not{P}}{\rho}}$$

$$\therefore \boxed{X_{GE} = A_E \not{P}_E \frac{2\gamma}{\gamma-1}\left[\left(\frac{P_E}{\not{P}_E}\right)^{\frac{\gamma-1}{\gamma}} - 1\right] + (\not{P}_E - \not{P}) A_E} \text{ --------(1)}$$

Total temperature for mass flow and momentum drag considerations, need not be measured on the swinging boom. It would enable momentum drag at the duct inlet to be determined if overboard cooler and airbleed flows were also measured, but duct losses should be differentiated or considered independently and mass flow can be more conveniently and accurately measured at the engine inlet.

2) Primary Jet

Measurement of gross thrust in the primary jet is required in order to differentiate the total swinging boom measurements into primary and secondary contributions. For this measurement, use of a final nozzle total pressure probe (or rake) would give the most direct indication. Due to the high jet pipe temperatures, with afterburner on, the probe would need to be air cooled; but even when cooled a probe at this location has a maximum life of approximately 8 hours with 1 hour afterburner operation. Hence, use of turbine outlet total head pressure measurements is recommended, which should be calibrated, for pressure loss, against an air or water cooled final nozzle total head pressure rake on an engine test bed.

2) <u>Primary Jet</u> (Continued)

These ground measurements would also determine effective final nozzle area. It will then be assumed that the pressure loss is independent of altitude; but if the life of the final nozzle probe permits (after ground calib ration) the pressure loss should be checked in flight at maximum possible altitude.

Hence if turbine outlet total pressure, P6, is measured the pressure loss, ΔP, is subt racted to give final nozzle P7; and then use is made of equation (1) to calculate gross thrust. Some simplifications in the formula are possible since below choking $p_7 = p$ and above choking $P_7 = p_7\left(1+\frac{\gamma-1}{2}M_7^2\right)^{\frac{\gamma}{\gamma-1}} = p_7\left(\frac{\gamma+1}{2}\right)^{\frac{\gamma}{\gamma-1}}$ where $M_7 = 1$

Therefore, below choking $\boxed{\frac{X_G}{A_7 p} = \frac{2\gamma}{\gamma-1}\left[\left(\frac{P_7}{p}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]}$ --------(2)

and, above choking $\boxed{\frac{X_G}{A_7 p} = (\gamma+1)\left[\frac{2}{\gamma+1}\right]^{\frac{\gamma}{\gamma-1}}\left(\frac{P_7}{p}\right) - 1}$

Hence determination of $\frac{X_G}{}$ requires measurements of p, A7 and P7 where p is obtained from ambient A.S.I. data.

A7 is obtained directly from final nozzle measurements and corrected to effective A7 using ground test results and P7 is obtained from P6 measurements less ΔP, where P6 need only be measured on 1 averaging pitot rake.

Again total temperature for mass flow and momentum drag considerations need not be measured in the jet pipe since this will be measured at the engine inlet. However, turbine outlet total temperature should be measured to indicate engine operating conditions.

If a complete temperature survey by rake measurements were made near the final nozzle of the jet pipe, this would eliminate the need for measuring A7. Gross thrust could be determined from the formula;

$$X_G = C_V m_7 \sqrt{2g\frac{\gamma}{\gamma-1} R T_7 \left[1-\left(\frac{p}{P_7}\right)^{\frac{\gamma-1}{\gamma}}\right]} \quad \text{-----(3)}$$

However, measurement of A7 is deemed more convenient (or less difficult) than measurement of T7 which must be shielded from radiation and which requires an air cooled structure.

3) Secondary Thrust

Secondary thrust measurement is desirable to ensure that swinging boom thrust data is equal to the summation of primary and secondary contributions. Measurement of total and static pressures by rake survey in the plane of the primary nozzle exit is required together with a longitudinal survey of extractor wall statics aft of this section. Secondary thrust in the plane of the primary nozzle exit is calculated by means of equation (1).

i.e. $$X_{G_S} = A_S \bar{P}_S \frac{2\gamma}{\gamma-1}\left[\left(\frac{P_S}{\bar{P}_S}\right)^{\frac{\gamma-1}{\gamma}} - 1\right] + (\bar{P}_S - \bar{P})\, A_S$$

where A_S = secondary air flow area at the station of measurement and $\bar{P}$ = ambient static pressure.

Rake measurement of P_s and $\bar{P}_s$ at the primary nozzle exit plane is made difficult by the variable area nozzle and by the associated change in position of the reference plane. However, it should be possible to choose a compromise position between the final nozzle closed and fully open positions.

Additional thrust of the extractor must be measured by placing wall statics along the convergent and "12 degree" divergent nozzle aft of the primary exit plane. These wall statics may be manifolded to obtain an average pressure. Then, additional gross thrust $= \frac{\pi}{4}(\bar{P}_W - \bar{P})(D_e^2 - D_p^2)$

where $\bar{P}_W$ = average wall static pressure

$\bar{P}$ = ambient static pressure

De = extractor exit diameter (= 49" for Arrow II aircraft)
(= 45" for Arrow I aircraft)

D_p = extractor internal diameter at primary nozzle exit plane

Measurement of mass flow is also required in the secondary air passage and this will be most conveniently obtained by measuring total temperature at the primary nozzle exit plane where total and static pressures are already measured. This mass flow is required for momentum drag consideration when the blow in doors are open and the by-pass door closed (i.e. for flight at low Mach number and at low altitude.)

3) Secondary Thrust (Continued)

Mass flow is calculated as follows-

$m = \rho A V = \rho A\, M a$ where A = area

$$M = \text{Mach number} = \sqrt{\left[\left(\frac{P}{p}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]\frac{2}{\gamma-1}} = \sqrt{\left[\frac{1-(p/P)^{\frac{\gamma-1}{\gamma}}}{(p/P)^{\frac{\gamma-1}{\gamma}}}\right]\frac{2}{\gamma-1}}$$

a = Local speed of sound $= \sqrt{\gamma R T}$

ρ = Density = p/RT

T = Static temperature = $T_T \Big/ \left(\frac{p}{P}\right)^{\frac{1-\gamma}{\gamma}}$

whence

$$\boxed{m = \frac{pA}{\sqrt{RT_T}\left(\frac{p}{P}\right)^{\frac{\gamma-1}{\gamma}}}\sqrt{\frac{2\gamma}{\gamma-1}\left[1-\left(\frac{p}{P}\right)^{\frac{\gamma-1}{\gamma}}\right]}} \quad \text{------------------(4)}$$

4) <u>Inlet Momentum Drag</u>

Momentum drag must be measured at the inlet to the engine installation in order to derive net thrust from the installed engine, i.e. $X_N = X_G - M V$ where X_G is derived as praviouly described and $M V$ = mass flow x velocity (at the engina inlet) = momentum drag. $M V$ is therefore determined by making total and static pressure measurements at the inlet and using an equation similar to (1) i.e. $M V$ =

$$A\, p \left(\frac{2\gamma}{\gamma-1}\right)\left[\left(\frac{P}{p}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]$$

where A = geometric area of the inlet and P and p are obtained from rake measurements across the inlet. Rake pressure measurements should be made on probes spaced on equal annular areas and then manifolded to obtain an average pressure.

It is raquired that measurements be made at the engine inlet, slightly upstream of the L.P. compressor face, and also completely across the inlet duct near the engine inlet in order to separate momentum to engine proper from momentum of the complete installation including by-pass flow. Net thrust can then determined for the engina only and for the installed engine. It should be noted that momentum drag of the installed engine will include the momentum drags of the flow to the overboard cooler and of the compressor airblead to air conditioning and pressurization; whila momentum drag of the engine proper includes momentum drag of the compressor air-bleed flow.

4) Inlet Momentum Drag (Continued)

Mass flow measurement across the inlet duct is also required in order to derive momentum drag of the duct itself. i.e. Momentum drag of the duct can be determined by subtracting airflow momentum at the intake of the engine installation from the free stream momentum of the intake mass flow. Hence , total temperature must be measured in conjunction with the pressure measurements taken at the intake to the complete engine installation only. Mass flow is then determined using equation (4).

In conclusion it is stated that mass flow measurements can be more conveniently and accurately measured at the engine intake than at the exit due to the lower extremes of pressure and temperature.

Further, by measuring mass flow at the intake, leakage losses of airflow through the engine are properly accounted for ; although total mass flow at the exit is probably similar or only slightly less due to fuel addition off-setting leakage and airbleeds.

It should be noted that net thrust derived herein is the net thrust of the engine only, including and excluding the effect of secondary airflow, but not including intake momentum and duct losses. Intake momentum and duct losses will be considered separately but eventually will be subtracted from engine net thrust to give total installed net thrust.

Intake spillage drag, to be determined by suitable flight test (decelerated levels at varying power), being a function of mass flow will also be considered as a reduction in installed net thrust.

The effect of γ and the use of a mean value in the various equations for thrust measurement will be investigated, where γ varies from approximately 1.4 for air, 1.33 for air + fuel in the primary jet without afterburner and 1.28 for air + fuel in the primary jet with full afterburner.

DISCUSSIONS ON TECHNIQUE & FORMULAE FOR USE IN TURBO-JET THRUST MEASUREMENT

1. In conjunction with the Appendix on thrust measurement technique, visit #1 was made to Wright Aeronautical Development Centre (W.A.D.C.) in Dayton, Ohio on July 16, 1957, to discuss its application. The visit was made by the author, in company with Mr. D. R. Woolley of Flight Test Engineering. At W.A.D.C. discussions were held with the following personnel:-

Mr. R.E. Conover) Major Graf)	Flight Test Instrumentation Dept. (Area C)
Mr. J. King) Mr. P. Schumaker)	Flight Research & Test Dept. (Area C)
Mr. E.C.Simpson Mr. O'Brian	Propulsion Laboratory (Area B) %Aircraft Thrust Stand

Some useful comments were obtained, but in general, W.A.D.C. had no direct experience in making the detailed measurements proposed herein. A swinging boom at the jet exit (installed by North American Aviation Co.) on a F-100 aircraft was inspected and was seen to a practical installation. Total and static pressures and temperature were all measured on one probe and the operating mechanism was installed in place of the tail skid on the underside of the fuselage. Some trouble was experienced when the probe and boom locked in the centre of the jet during a take-off and burned off. To date, no results have been obtained. W.A.D.C. had instelled an air cooled rake for total pressure measurements in the primary jet of an F-86D aircraft but due to other problems and priorities they were not able to do any testing. In connection with engine intake measurements it was stated that rakes or probes at this location were "risky" if and when structural failure occurs (usually in fatique). However, the probes could be mounted on existing struts or on reinforced rakes firmly attached at both ends.

Other problems discussed were as follow:-

(a) Primery jet swirl: This swirl depends on intake temperature and R.P.M. and may be as high as 50°. Total head tubes mounted in the primary jet should therefore be designed to accommodate this swirl.

(b) Nozzle stabilization time: Usually with variable area nozzles, after a chenge in conditions, the nozzle may take approximately 3 minutes to stabilize with a corresponding overshoot and 3 minute stabilization of thrust.

Other problems (Continued)

(c) Measurement of final nozzle area: It was agreed that measurement of either final nozzle T_7 or A_7 was necessary and that measurement of A_7 is probably less difficult, although problems would be encountered to measure it accurately. It was therefore suggested to measure T_7 and A_7 and average the results. i.e. Gross thrust in the primary jet can be determined by measuring total pressure and area or by measuring total pressure and temperature using equations (2) or (3) of the Appendix respectively. The validity of equation (3) for the relatively high jet pipe pressure ratios of the Arrow aircraft is questionable since report NACA RM E 55 Do5a states that it may be used for pressure ratios up to approximately 4.0 only. However, verifications will be made, since WADC stated that the equation should be satisfactory for pressure ratios to at least 12.0.

(d) Wall statics: WADC have had experience and obtained satisfactory results with wall static measurements at engine inlets and foresee no practical difficulties in making these measurements at the extractor outlet to determine additional thrust of the extractor.

(e) Thrust stand measurements: It was tentatively stated that the thrust stand at W.A.D.C. could accommodate the Arrow aircraft with a single engine thrust up to 25,000 lbs. or a two engine thrust up to 50,000 lbs. A further check on wheel base requirements should be made. An accuracy in thrust measurement of 0.1 per cent of full scale reading was quoted although some doubt was cast on this accuracy under actual test conditions.

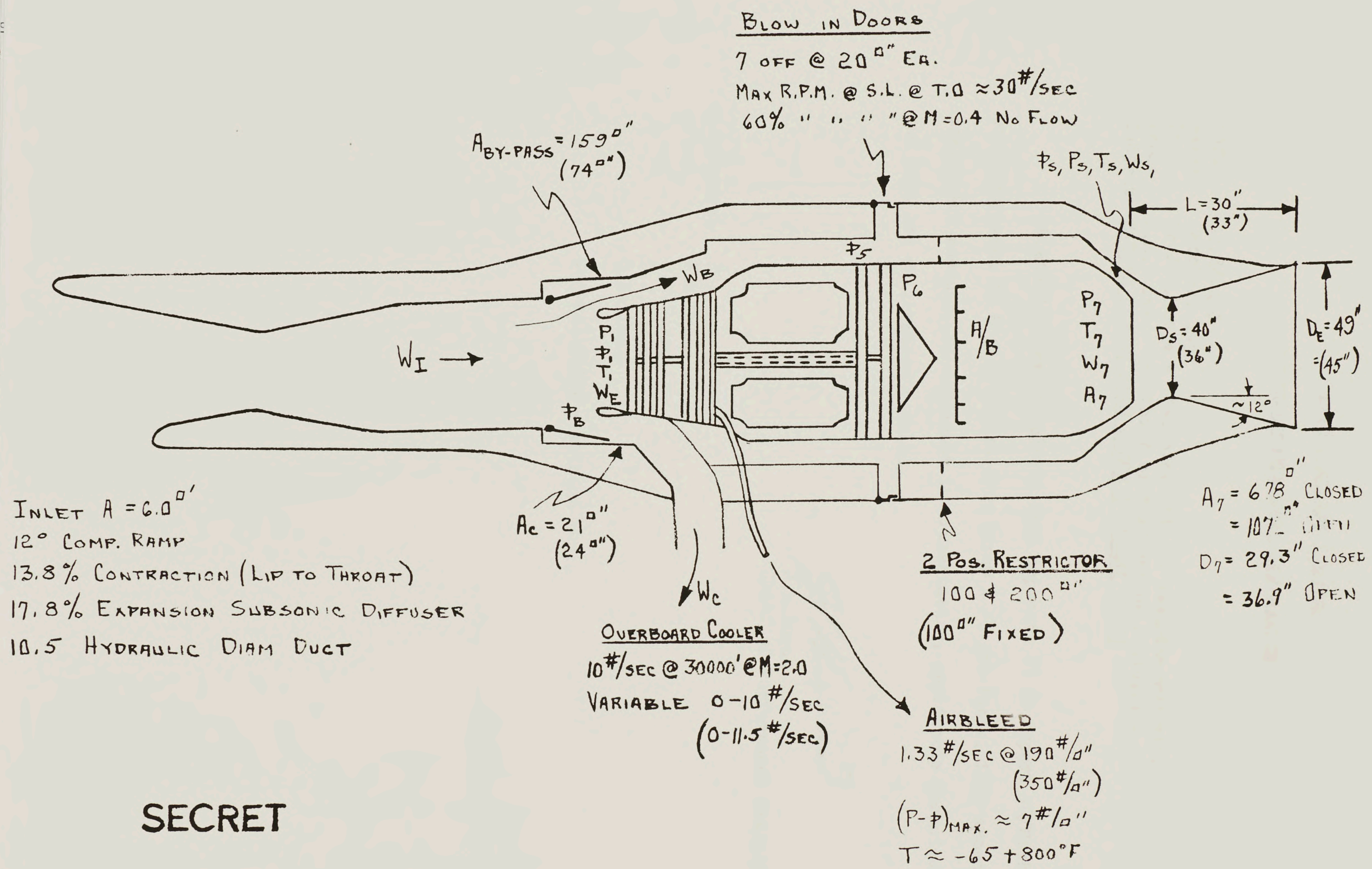
(f) Other techniques for thrust measurement: Strain gauging of the engine mounts for thrust measurement is a simple and straightforward method but is highly inaccurate due to vibrations and large fluctuations. The method would not include thrust or drag contributions of a cooling air extractor.

Primarily, W.A.D.C. were interested only in total net thrust measurement at the exit and suggested that prime contractors such as North American Aviation Co. would be more interested in separating engine performance from aircraft performance and would therefore have more advice on detailed measurements.

(f) Continued.......

Copies of the appendix on Technique and Formulae for use in Turbo-jet Thrust Measurement were left with Mr. J. King and Mr. Simpson for further study and comments to be forwarded to Avro by mail.

The problem therefore remains to design suitable instrumentation. Further discussions with experienced personnel would be useful.



SECRET

R. WAECHTER JULY/57

NOTE: ALL DIMENSIONS & AREAS ARE GIVEN FOR PS-13 INSTALLATION EXCEPT WHERE SHOWN IN BRACKETS FOR J-75 INSTALLATION