AVRO AIRCRAFT LIMITED

INTER-DEPARTMENTAL MEMORANDUM

Ref: 3272/04/J
Date: October, 1957
To: Mr.J.D.Hodge - Technical Flight Test Co-ordinator
From: J.H.Lucas - Chief of Performance Evaluation
Subject: ENGINE THRUST FROM ARROW 1 FLIGHT TESTS

Attached herewith, please find Appendix II to report 70/PERF/1 on Measurement of Engine Thrust from Arrow 1 Flight Tests. This appendix deals with adequacy of available instrumentation, applicable formulae and test bed calibration results.

J.H.Lucas
Chief of Performance Evaluation

c.c Messrs: J.A.Chamberlin
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Central Files - without encl.

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AIRCRAFT: ARROW I	ENGINE PERFORMANCE	REPORT NO. 70/PERF/1 APP II	SHEET NO. 1
		PREPARED BY: R. Waechter	DATE: Oct. 1957
		CHECKED BY:	DATE:

MEASUREMENT OF Y J 75 ENGINE THRUST DURING ARROW I FLIGHT TESTS

1) SUMMARY

This appendix to report 70/Perf/1 summarizes a thrust measurement technique to be used during Arrow I flight tests. Use is made of available instrumentation only; therefore adequacy of instrumentation is discussed. Applicable formulae are given, to be used in conjunction with report 71/Perf/2 on Programming for Performance Data from Arrow I Flight Tests.

Results of Pratt and Whitney Aircraft Run-in and Final Acceptance Engine test bed calibration runs are also contained herein from which the following information is extracted for Y J 75 P - 3 Engines No.610034 (port), No.610029 (stbd), No.610026 (spare) and No. 610027 (spare) for Arrow I Aircraft No.1

Military thrust rating = 15,500 lbs. static sea level standard thrust with corresponding H.P. Comp. R.P.M's of 8630, 8690, 8690 & 8715 respectively. Each of these rpms are increased between 10 and 35 rpm to obtain N_2 Data Plate speeds of 98.80, 99.55, 99.72 and 99.72% respectively based on 8750 as 100% rpm. (see note 1).

Average L.P.Comp. R.P.M. at military thrust = 0.768 N_2 for existing engines

Average "Aerodynamic " nozzle area = 510 (± 2) sq.inches (A/B off)
928, 921, 947 and 917 sq. inches (A/B on).

These aerodynamic nozzle areas should be used in conjunction with the gross thrust coefficient curves of report 71/Perf/2 Charts III and IV to determine effective nozzle area.

NOTE:- 1) Pratt and Whitney Aircraft have stated that 100% N_2 R.P.M. = 8732 and 100% N_1 R.P.M. = 6774 and therefore the above percent data plate speeds should be revised accordingly. i.e. increase percentage by 0.20%

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AIRCRAFT:		REPORT NO. 70/PERF/1 APP. II	
		SHEET NO 2	
		PREPARED BY	DATE
ARROW 1	ENGINE PERFORMANCE	R. Waechter	Oct./57
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2) ADEQUACY OF INSTRUMENTATION FOR THRUST MEASUREMENT

A list of available instrumentation which can be used for thrust measurement during Arrow 1 flight tests is given under report 71/PERF/2 (sheets 2 & 3). Complete engine instrumentation is not available and therefore certain estimations, based on manufacturer's figures for basic engine performance (Charts 3 & 4 Report 71/PERF/2) and on Avro Nobel Test results and a NACA method for ejector performance (Charts 6 & 7 Report 71/PERF/2) must be used to complete the calculations. In addition to using the above manufacturer's data for basic engine performance, their data must also be used to derive final nozzle total pressure P_{10}, and total temperature T_{10}, from measured quantities at the engine inlet and turbine outlet, for use in obtaining installed gross thrust (with ejector) values.

Primary engine gross thrust determination depends on turbine outlet total pressure, P_7, measurement which is adequately measured by manifolded rake measurements. Primary engine momentum drag depends on compressor inlet P_2 (total pressure) and p_2 (static pressure) measurements; the former being adequately measured by 2 or 3 representative probes (after rake calibration while the latter is measured by a single wall static vent mounted on the inboard side on the centre line of the inlet duct approx. 6 feet upstream of the L.P. compressor face. This static vent pressure measurement is primarily used for structural loading investigations and can be ueed to represent static pressure at the engine face for momentum drag consideration since relative areas should be the same (after allowing for aft diffuaing and nose bullet). However, approx. 3% total air weight flow is extracted to the overboard cooler just upstream of the compressor face and therefore the static pressure measurement should be increased approx. 1% to be more representative of the pressure at the compressor face. (See table 1)

Determination of installed gross thrust (with ejector) requires the measurement of air weight flow and total temperature at the by-pass exit. For this, measurements of p_B, T_B and P_B are obtained by various means. i.e. Static pressure, p_B, is measured near the by-pass exit relative to ambient static pressure, p_1. Absolute p_B can therefore be readily obtained by adding p_1 to $(p_B - p_1)$. Total by-pass temperature, T_B, is measured upstream of p_B and assumed equal to T_B near exit. Total pressure, P_B, is measured further upstream (approx. mid jet pipe) but is actually a static presaure measurement. Since the by pass area is large at thia station a relatively low mean Mach number of .31 will be assumed such that measured mid jet pipe $p_B \approx P_B/1.07$. Measuremente of P_B & p_B are made by single probes while T_B is measured on two probee, one above and one below the jet pipe. Disadvantages of single or dual probe measurements compared to rake measurements for average readings are well known and therefore care must be exercised in interpreting the results. (see table 2) By-pass measurements are only made on the port engine and identicel readings will be assumed for the starboard side. Programing for performance data from Arrow 1 flight teste according to report 71/PERF/2 is therefore valid only for symetrical power conditions.

FORM 1319A

Geometric areas must be measured at the reference stations (by-pass exit and engine face inlet) and then corrected to an effective area using an estimated coefficient for use in appropriate air weight flow formulae. Air weight flows will be obtained as an output from IBM 704 computations as well as final net thrust. If these air weight flows do not reasonably agree with estimates (due to inadequate instrumentation) the area coefficients or pressure measurement coefficients can be adjusted such that a good measure of thrust is obtainable which should be consistent for all flights using the same instrumentation. A preliminary estimate of area coefficients is .97 and .93 for engine face and by-pass exit respectively. Probable tolerance on these area coefficients is approximately .025.

In conclusion, it is stated that a reasonable or consistent measure of thrust is expected from available instrumentation. Some deterioration in thrust measurement accuracy is expected at the lower engine powers at high altitude (low ambient static pressure) due to inadequate instrumentation. Use of two or three range pressure measurement instrumentation at the engine face and at the turbine outlet would improve overall accuracy.

TABLE 1 EFFECT OF INSTRUMENTATION ACCURACY ON DETERMINATION OF AIR WEIGHT FLOW AND MOMENTUM DRAG AT ENGINE FACE

Note: 1) $W_E = \rho_E \, A_2 \, V_2 = \frac{p_2}{Rt_2} \times A_2 \times M_2 \times \sqrt{\gamma g \, Rt_2} =$

$$= \sqrt{\frac{\gamma g}{Rt_2}} \times p_2 \, A_2 \, M_2$$

where "static t_2" = "total T_2" $\left(\frac{p_2}{P_2}\right)^{\frac{\gamma-1}{\gamma}} = T_1 \left(\frac{p_2}{P_2}\right)^{.286}$

2) $m_E \, V = \frac{W_E \times V}{g}$ where V = free stream velocity

3) $M_2^2 = \left[\left(\frac{P_2}{p_2}\right)^{\frac{\gamma-1}{\gamma}} - 1\right] \times \frac{2}{\gamma-1} = 5.0 \left[\left(\frac{P_2}{p_2}\right)^{.286} - 1\right]$

For 1% error in pressure measurement at engine face

M_2 = .5 tolerance = ± .015 = ± 3.0%

M_2 = .4 tolerance = ± .019 = ± 4.75%

M_2 = .3 tolerance = ± .025 = ± 8.3%

Hence if P_2 measurement is in error by ± 1% then t_2 will be error ± .143% and W_E and $m_E V$ will be in error ± 3.14% at M_2 = .5

± 4.89% at M_2 = .4

± 8.44% at M_2 = .3

If p_2 measurement is in error by 1% the effect on W_E and $m_E \times V$ will be similar to that for P_2 but with the absolute value of the error reduced by 1%.

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REPORT NO.	70/PERF/1 APP II	
SHEET NO.	5	
AIRCRAFT: ARROW 1	ENGINE PERFORMANCE	
PREPARED BY	R. Waechter	DATE: Oct. 1957
CHECKED BY		DATE

TABLE II EFFECT OF INSTRUMENTATION ACCURACY ON DETERMINATION OF AIR WEIGHT FLOW AT BY-PASS EXIT

NOTE:- 1) $W_B = \rho_B \; A_B \; V_B = \frac{P_B}{Rt_B} \times A_B \times M_B \times \sqrt{\gamma g \, Rt_B}$

$$= \sqrt{\frac{\gamma g}{Rt_B}} \; P_B \; A_B \; M_B$$

2) $M_B^2 = \left[\left(\frac{P_B}{p_B} \right)^{\frac{\gamma - 1}{\gamma}} - 1 \right] \frac{2}{\gamma - 1} = 5.0 \left[\left(\frac{P_B}{p_B} \right)^{.286} - 1 \right]$

For 1% error in pressure measurement

$M = 1.0$ tolerance $= \pm .009 = \pm 0.9\%$

$M = 0.8$ tolerance $= \pm .010 = \pm 1.25\%$

$M = 0.6$ tolerance $= \pm .013 = \pm 2.2\%$

Hence if P_B measurement is in error by $\pm$ 1%

Then W_B will be in error $\pm$ 1.04% at $M_B = 1.0$

$\pm$ 1.39% at $M_B = 0.8$

$\pm$ 2.34% at $M_B = 0.6$

If p_B measurement is in error by 1% the effect on W_B will be similar to that for P_B but with the absolute value of the error reduced by 1%.

Any error in T_B measurement causes an error in W_B which is inversely proportional to $\sqrt{T_B}$.

3) THRUST MEASUREMENT TECHNIQUE

The following symbols are used in the derivation of ensuing formulae:-

capital T = total temperature °K
capital P = total pressure
small p = static pressure
F = installed engine thrust (with extractor)
X = basic or primary engine thrust

FORM 1319A

subcript G = gross
" N = net
" 1 = station 1, free stream
" 2 = " 2, engine face
" 7 = " 7, turbine outlet
" 10 = " 10, final nozzle (primary)
" B = " 10, in by-pass

<u>Primary engine gross thrust</u> is derived from turbine discharge total pressure measurement as follows:-

$$X_G = \left[1.255 \frac{P_7}{p} - 1\right] C_g \times A_{10}\, p \text{ for } \frac{P_7}{p} \geqslant 1.852 \qquad (1)$$

$$X_G = 8.06 \left[\left(\frac{P_7}{p}\right)^{.248} - 1\right] C_g \times A_{10}\, p \text{ for } \frac{P_7}{p} < 1.852 \quad (2)$$

for $\gamma = 1.33$ and p = ambient static pressure

Effect of $\gamma = 1.33$ down to approx. 1.30 for afterburner operation has negligible effect on the above formulae. (See also P.W.A. Curve No. Inst. 16624).

Nominal A_{10} = 3.68 sq.ft. = 530 sq.inches (cold and relaxed) A/B off
= 6.398 sq.ft. = 921 sq.inches (est. operating) A/B on

However A_{10} values, based on P.W.A. engine test bed calibration runs on engines for A/C No. 1 = 510 ($\pm$ 2) sq.inches A/B off
= 928 ($^{+19}_{-11}$) sq.inches A/B on

C_g = gross thrust coefficient from P.W.A. curve no inst. 27214 and 18151.

To cover engine idling cases at low altitude curve 27214 for A/B off must be extended downward to $P_7/p \approx 1$ where C_g will then be approximately .80 based on data from log book of engines 610026 and 610027 (See charts III and IV of report 71/PERF/2).

<u>To determine primary net thrust</u>, momentum drag to the engine proper must be subtracted from X_G to obtain X_N. Total and static pressure are measured at or near the engine face and therefore momentum drag is determined from the formulae:-

$$m_E V = A_2\, p_2 \left(\frac{2}{\gamma-1}\right) \sqrt{\frac{\left[\left(\frac{P_2}{p_2}\right)^{\frac{\gamma-1}{\gamma}} - 1\right] \times \left[1 - \left(\frac{p}{P}\right)^{\frac{\gamma-1}{\gamma}}\right]}{\left(P_2/P_2\right)^{\frac{\gamma-1}{\gamma}}}} = \frac{W_E \times M \times 65.8 \sqrt{T_o}}{g}$$

where A_2 = geometric area at engine face corrected by an area coefficient

The next stage in thrust measurement consists of determining net thrust of the complete engine installation including intake duct and ejector.

Total gross thrust, F_G, is determined from the ratios $\frac{F_G}{X_{i_G}}$ for A/B off & $\frac{F_G}{X_G}$ for A/B on, where X_G is the primary gross thrust as previously determined and X_{i_G} is an ideal primary gross thrust.

Ideal primary gross thrust assumes isentropic expansion to ambient pressure behind the final nozzle and is determined as follows

$$X_{i_G} = m_{10} \sqrt{\frac{2 \gamma R_g T_{10}}{\gamma - 1} \left[1 - \left(\frac{p}{P_{10}} \right)^{\frac{\gamma - 1}{\gamma}} \right]}$$

Now for the momentum gross thrust $(mV)_{10}$ in the final nozzle, ambient p in the above formula is substituted by final nozzle p_{10}, and for choked flow $\left(\frac{P_{10}}{p} \geq 1.852 \right)$, $M_{10} = 1$ and $p_{10} = \left(\frac{2}{\gamma + 1} \right)^{\frac{\gamma}{\gamma - 1}} \times P_{10}$

Then
$$\frac{X_{i_G}}{(mV)_{10}} = \sqrt{\frac{1 - \left(p/P_{10} \right)^{\frac{\gamma - 1}{\gamma}}}{\frac{\gamma - 1}{\gamma + 1}}} = 2.66 \sqrt{1 - \left(\frac{p}{P_{10}} \right)^{.248}}$$

for $\gamma = 1.33$

Now equation (1) can be rearranged such that

$$\frac{X_G}{(mV)_{10}} = \frac{1.255 \frac{P_{10}}{p} - 1}{.714 \frac{P_{10}}{p}}$$

Hence
$$\frac{X_{i_G}}{X_G} = \frac{1.90 \frac{P_{10}}{p} \sqrt{1 - \left(\frac{p}{P_{10}} \right)^{.248}}}{1.255 \frac{P_{10}}{p} - 1} \qquad (3)$$

See Chart V Report 71/Perf/2

This equation is used for A/B off only, for which it can be assumed that $P_{10} = .99\ P_7$ (where P_7 is measured).

The ratios $\frac{F_G}{X_{i_G}}$ and $\frac{F_G}{X_G}$ are obtained from experimental data

i.e. for A/B off, a NACA solution is used (Chart VI of 71/Perf/2)

and for A/B on, Avro Nobel Test results are used (Chart VII of 71/Perf/2)

To determine $\frac{F_G}{X_{i_G}}$ and $\frac{F_g}{X_G}$ values of $\frac{P_{10}}{p}$ and μ must be known.

$\frac{P_{10}}{p}$ must be derived from measured $\frac{P_7}{p}$, which for A/B off

$= .99\ \frac{P_7}{p}$ (as above) while for A/B on, $\frac{P_{10}}{p}$ is derived as follows:-

According to J T 4A Engine Installation Handbook

$P_{10} = 0.1297\ X_G + 0.8\ p$ for A/B on, and if $A_{10} = 6.398$ sq.ft.,

then $\frac{X_G}{.965 A_{10}\ p} = 1.250\ \frac{P_{10}}{p} - 1$

It is therefore evident that a nozzle coefficient, based on $\frac{P_{10}}{p}$, of .965 was used.

Now C_g (Chart IV Report 71/Perf/2) is based on $\frac{P_7}{p}$; and the difference between C_g and .965 is assumed to account for the pressure loss from P_7 to P_{10}

$$\therefore \frac{X_G}{C_g\ A_{10}\ p} + 1 = 1.250\ \frac{P_7}{p} \qquad \text{(based on } P_7\text{)}$$

and $$\frac{X_G}{.965\ A_{10}\ p} + 1 = 1.250\ \frac{P_{10}}{p} \qquad \text{(based on } P_{10}\text{)}$$

Solving these two equations

$$\boxed{\frac{P_{10}}{p} = \frac{C_g}{.965} \times \frac{P_7}{p} - \frac{C_g}{1.206} + 0.8} \quad \text{---(4)}$$

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MALTON - ONTARIO
TECHNICAL DEPARTMENT

REPORT NO 70/PERF/1 App. II	SHEET NO 9	
AIRCRAFT: ARROW 1	ENGINE PERFORMANCE	
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For evaluation of μ which equals $\frac{W_B}{W_{10}}\sqrt{\frac{T_B}{T_{10}}}$ the following assumptions are made:

a). W_B (lbs/sec) and T_B (^{o}K) are air weight flow and total temperature as measured in the by-pass near the final nozzle.

$$W_B \neq \frac{p_B A_B \sqrt{g}}{\sqrt{RT_B}\left(\frac{P_B}{p_B}\right)^{\frac{\gamma-1}{\gamma}}}\sqrt{\frac{2\gamma}{\gamma-1}\left[1-\left(\frac{p_B}{P_B}\right)^{\frac{\gamma-1}{\gamma}}\right]} = \frac{p_B A_B \sqrt{g}}{\sqrt{RT_B}\left(\frac{P_B}{p_B}\right)^{.286}}\sqrt{7\left[1-\left(\frac{p_B}{P_B}\right)^{.286}\right]} \quad (5)$$

for $\gamma = 1.4$, R =96.02

b) W_{10} = air weight flow through the primary nozzle, which for the A/B off case is assumed = engine inlet W_E and for the A/B on case is assumed = $W_E + \frac{Q_{A/B}\text{ (A/B fuel flow, lbs/hr)}}{3600}$ i.e. engine fuel flow (approx 2.5% of W_E) is assumed to offset airbleed & leakage

$$W_E = \frac{p_2 A_2 \sqrt{g}}{\sqrt{RT_2}\left(\frac{P_2}{p_2}\right)^{.286}}\sqrt{7\left[1-\left(\frac{p_2}{P_2}\right)^{.286}\right]} \quad \text{for } \gamma = 1.4 \text{ , R = 96.02} \qquad (6)$$

b) $T_{10} = T_7$ (^{o}K) for A/B off and

$$\sqrt{T_{10}\ (^{o}K)} = \left[\frac{.745 \times X_G}{W_E + \frac{Q_{A/B}}{3600}}\right] \times \left[\frac{.0009\left(\frac{X_G}{p}\right)+0.8}{.00219\left(\frac{X_G}{p}\right)-.02}\right] \quad \text{for A/B on (p in psi)} \qquad (7)$$

Formula (7) is obtained by combining two equations from J.T.4A Engine Installation Handbook and letting $W_{10} = W_E + \frac{Q_{A/B}}{3600}$

i.e. Combine $T_{10} = \left[\frac{X_G/W_{10}}{2.435 - 1.897\frac{p}{P_{10}}}\right]^2$ for T_{10} in ^{o}R, with P_{10} = $0.1297\, X_G + 0.8\, p$ for p in psf.

FORM 1319A

To derive installed net thrust (excluding external pressure drag around ejector), momentum drag and spillage drag must be subtracted from installed gross thrust.

i.e. $F_N = F_G - m_V - D_S$

Momentum drag, m_V, is determined by mass flow measurements at the engine face, to which is added the mass flow as measured in the by-pass near the exit and then multiplied by airplane velocity.

$$\text{i.e. } m_V = \frac{(W_E + W_B) \; M \times 65.8\sqrt{T_0}}{32.2}$$

where W_E is obtained from equation (6).
W_B is obtained from equation (5).
M = true Mach number
T_0 = true ambient air temperature °K

It should be noted that

1. W_B is airflow that enters through the by-pass doors and/or through the blow in doors. It is assumed that air entering the blow in doors (at low Mach number, low altitude) is extracted from the free stream at free stream velocity.

2. Airflow through the inlet duct to the overboard oil cooler is neglected since this airflow is less than 3% of total air flow, and approximately 50% of the momentum drag of the oil cooler air flow, is regained as thrust at the overboard cooler exit. The effect therefore is less than 1½% on total momentum drag and less than .75% on net thrust.

Intake spillage drag is not directly measureable, except by suitable flight tests such as accelerated levels at various engine settings, and hence wind tunnel data will be used. This data may change as the tests progress. Intake spillage drag data is presented as curves of $\frac{\partial C_{D_S}}{\partial\left(\frac{m_i}{m_o}\right)}$ and $\frac{P_{th}}{P_1}$

versus M (see chart $\overline{\text{VIII}}$ & $\overline{\text{IX}}$ Report 71/PERF/2 Issue 2)

where C_{D_S} = spillage drag coefficient $= \dfrac{D_S}{\frac{\gamma}{2} pM^2 S} = \dfrac{\partial C_{D_S}}{\partial\left(\frac{m_i}{m_o}\right)} \left(\dfrac{m_i}{m_o} - \dfrac{m_i^*}{m_i}\right)$

S = area = 40.98 sq.ft.
p = ambient static pressure
M = Mach number
m_i = actual inlet air weight flow
= air flow to engine, W_E, as measured
+ air flow in by-pass, W_B, as measured
+ air flow to overboard cooler, W_C, (assume mean value = 5 lb/sec.)
m_i^* = inlet air weight flow corresponding to choked conditions at the throat
m_o = nominal or capture air weight flow

Adding W_B and W_C to W_E to obtain m_i is not valid when the by-pass doors are closed. This only occurs at low altitude, low Mach number and therefore is unimportant for spillage drag considerations.

Now effective inlet throat area = .97 x 5.6 = 5.43 sq.ft.
and capture area = 10.18 sq.ft.

Hence:-

$$m_o = \rho AV = \frac{p}{RT_o} \times 10.18 \times 65.8\, M\sqrt{T_o} = 6.97 \frac{pM}{\sqrt{T_o}} \quad \text{for } p \text{ in psf}$$

and T_o (amb. static temp.) in ^oK

$$m_i^* = (\rho AV)_{throat} = \left(\frac{p}{Rt}\right)_{th.} \times 5.43 \; 65.8 \left(M\sqrt{t}\right)_{th.} \quad \text{where } t = \text{static temp.}$$

$$= 1.962 \frac{p_{t_h}}{\sqrt{t_{t_h}}} \quad \text{for } M_{th} = 1 \text{ ; } P_{t_h} = 1.894\, p_{t_h} \text{ \& } R = 96.02$$

$$= 1.962 \frac{P_1}{\sqrt{t_{t_h}}} \left(\frac{P_{t_h}}{P_1}\right)$$

Now $T_o \; (1 + .2M^2) = t_{t_h} \; (1 + .2M^2)_{t_h} = 1.2\, t_{t_h}$

$$\text{Hence } \frac{m_i^*}{m_o} = \frac{1.962 \; P_1}{6.97 \; pM} \left(\frac{P_{t_h}}{P_1}\right) \times \sqrt{\frac{1.2\, T_o}{(1 + .2M^2)T_o}} = .308 \frac{(1 + .2M^2)^3}{M} \left(\frac{P_{t_h}}{P_1}\right)$$

Therefore, spillage drag can be evaluated knowing W_E, W_B, p, M, T_o and using charts VIII & IX of Report 71/PERF/2 Issue 2.

Installed net thrust can then be compared to primary net thrust as determined previously, in order to verify installation effects. The latter thrust is also useful for comparison to engine brochure curves (after correcting for ambient temperature to standard, intake pressure recovery power extraction and air bleed) and should be more reliable since it will have been based on P.W.A. approved thrust coefficients and turbine outlet and engine face total pressure measurements but with the exception of unapproved static pressure measurement at the engine face.

AVRO AIRCRAFT LIMITED MALTON ONTARIO TECHNICAL DEPARTMENT		REPORT NO 70/PERF/1 APP.II SHEET NO 12	
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AIRCRAFT ARROW 1 A/C No. 1	ENGINE PERFORMANCE	R. Waechter	Oct. 1957
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CORRECTED ENGINE PERFORMANCE (ARROW 1 A/C No.1)

(P.W.A. RUN IN AND FINAL ACCEPTANCE ENGINE TEST BED CALIBRATIONS)

Engine:- 610034 (port)

Power	F_N	N_2	N_1	T.S.F.C.	T_{T_7}	P_{T_7}/P_{T_2}	N_2 Data Plate Speed ([illegible]IAS 60°) R.P.M.
Military	15,500	8630	6630	.792	1057	2.50	8645 at 98.80%
Normal	13,780	8380	6250	.759	947	2.33	
75% Norm.	10,300	7940	5550	.726	732	1.99	

Engine:- 610029 (stbd.)

Power	F_N	N_2	N_1	T.S.F.C.	T_{T_7}	P_{T_7}/P_{T_2}	N_2 Data Plate Speed R.P.M.
Military	15,500	8690	6640	.797	1073	2.51	8710 at 99.55%
Normal	13,780	8450	6280	.757	960	2.34	
75% Norm.	10,300	8020	5600	.728	837	1.93	

Engine:- 610026 (spare)

Power	F_N	N_2	N_1	T.S.F.C.	T_{T_7}	P_{T_7}/P_{T_2}	N_2 Data Plate Speed R.P.M.
Military	15,500	8690	6670	.783	1068	2.49	8725 at 99.72%
Normal	13,730	8445	6310	.760	950	2.33	
75% Norm.	10,300	7955	5580	.733	790	2.01	

Engine:- 610027 (spare)

Power	F_N	N_2	N_1	T.S.F.C.	T_{T_7}	P_{T_7}/P_{T_2}	N_2 Data Plate Speed R.P.M.
Military	15,500	8715	6715	.806	1065	2.51	8725 at 99.72%
Normal	13,780	8450	6335	.775	965	2.35	
75% Norm.	10,300	7915	5560	.737	815	2.02	

Note:- **Percentage** (N_2) R.P.M.'s based on 8750 as 100%

Average Military N_1 = .768 N_2

FORM 1319A

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AIRCRAFT: ARROW 1 A/C 1	ENGINE PERFORMANCE	REPORT NO. 70/PERF/1 APP. II SHEET NO. 13	
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AFTERBURNER PERFORMANCE

Y J 75 P - 3 ENGINES FOR ARROW 1 A/C No. 1

(P.W.A. RUN IN AND FINAL ACCEPTANCE ENGINE TEST BED CALIBRATIONS)

Engine:- 610034(port)

Power	Corr. F_N	Obs. N_2	T.S.F.C.	BIAS.T.S.F.C.	T_{T_2} (°F)
Mil. Non A/B	16,840	8600	.327	-	32
Afterburner	25,520	8610	1.933	1.945	32

Engine:- 610029 (stbd)

Power	Corr. F_N	Obs. N_2	T.S.F.C.	BIAS.T.S.F.C.	T_{T_2} (°F)
Mil. Non A/B	16,630	8670	.320	-	33
Afterburner	25,220	8665	1.92	1.93	32

Engine:- 610026 (spare)

Power	Corr. F_N	Obs. N_2	T.S.F.C.	BIAS.T.S.F.C.	T_{T_2} (°F)
Mil. Non A/B	17,030	8690	.324	-	32
Afterburner	26,000	8685	1.90	1.915	31

Engine:- 610027(spare)

Power	Corr. F_N	Obs. N_2	T.S.F.C.	BIAS.T.S.F.C.	T_{T_2} (°F)
Mil. Non A/B	16,675	8700			40
Afterburner	25,150	8695	1.94	1.95	40

AIRCRAFT		PREPARED BY	DATE
ARROW 1 A/C No. 1	ENGINE PERFORMANCE	R. Waechter	Oct. 1957
		CHECKED BY	DATE

DETERMINATION OF " AERODYNAMIC" NOZZLE AREA

Y J75 P - 3 ENGINES FOR ARROW 1 A/C No. 1

P.W.A. RUN IN AND FINAL ACCEPTANCE TEST BED CALIBRATIONS

Assumptions

1) $$\frac{X_G}{C_g\ A_{10}\ p} = 1.255 \frac{P_7}{p} - 1$$

or $$A_{10} = \frac{X_G/(C_g \times p)}{1.255 \frac{P_7}{p} - 1}$$

2) $$\frac{P_7}{p} \equiv \frac{P_7}{P_2} \quad \text{and} \quad \frac{X_G}{p} \equiv \frac{F_N}{14.7\ \delta_{T_2}}$$

3) At same N_2, $\frac{P_7}{P_2}$ A/B on = P_7/P_2 A/B off

4) C_g from 71/PERF/2 Chart III A/B off
Chart IV A/B on

5) P.W.A. Final Acceptance Test Bed Calibrations have been corrected to standard sea level conditions

Note:-

1) Nominal A_{10} for Y J75 P - 1, 3 and 11 engines

= 530 sq.in. (A/B off)
= 921 sq.in. (A/B on)

2) A_{10} values are calculated at actual test points

FORM 1319A

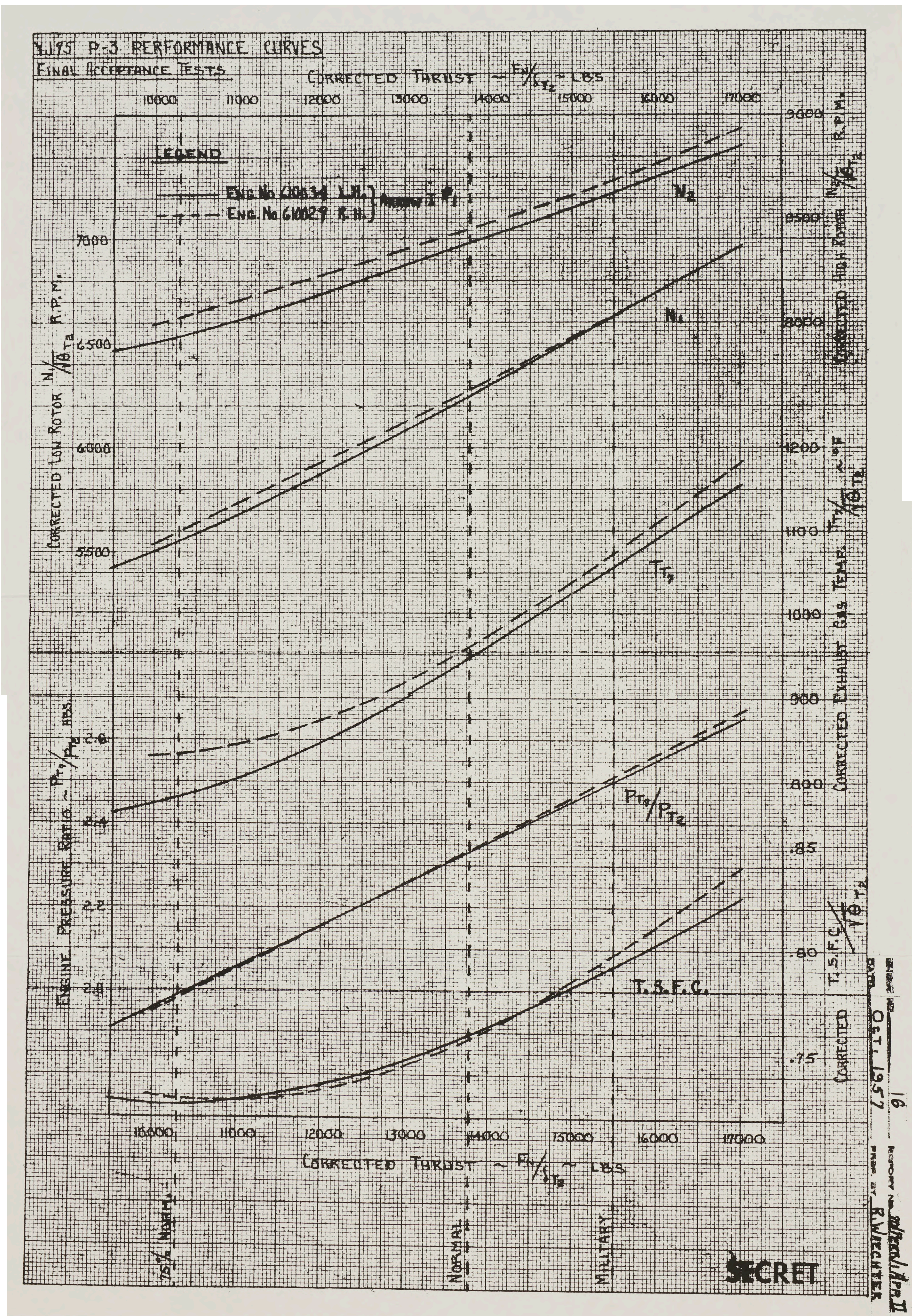
AVRO AIRCRAFT LIMITED
MALTON - ONTARIO
TECHNICAL DEPARTMENT

REPORT No	70/PERF/1 APP. II
SHEET No	15
PREPARED BY	DATE
R. Waechter	Oct. 1957
CHECKED BY	DATE

AIRCRAFT: ARROW 1, A/C No. 1

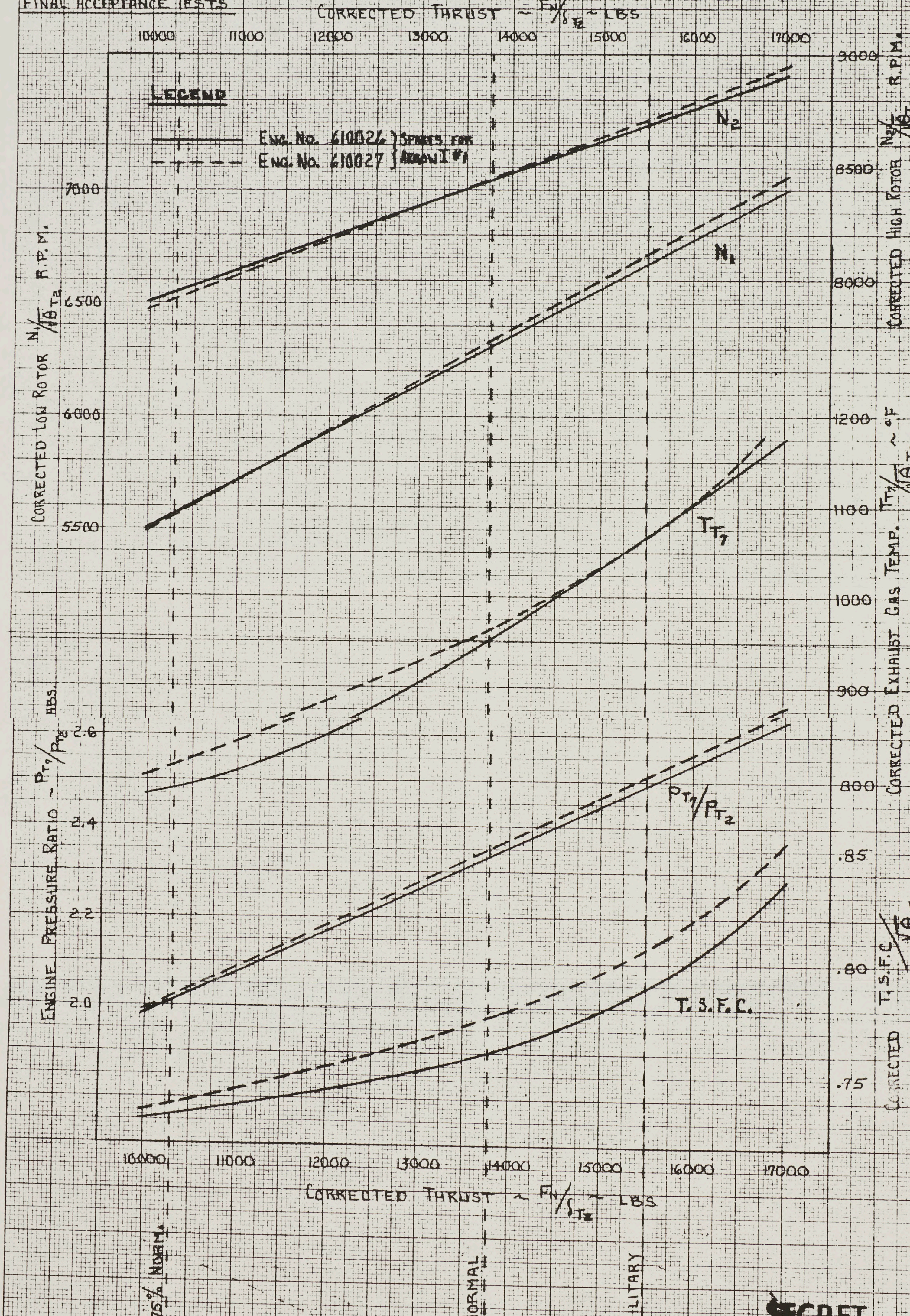
ENGINE PERFORMANCE

Engine	$\frac{F_N}{\delta T_2}$	$= \frac{14.7 X_G}{P_2}$	$\frac{P_7}{P_2}$	$\frac{X_G}{C_g A_{10} P_2}$	C_g	$\frac{X_G}{A_{10} P_2}$	$14.7 A_{10}$	A_{10} □"	A_{10} (aver.)
610026 (spare)	9,860		1.950	1.445	.910	1.317	7490	509	
	12,150		2.131	1.737	.937	1.628	74[illegible]0	508	
	12,180		2.150	1.700	.935	1.591	7650	521	
	14,400		2.400	2.010	.959	1.929	7460	507	512 □"
	15,200		2.470	2.100	.963	2.022	7510	511	
	16,570		2.600	2.260	.973	2.201	7520	512	
	17,120		2.640	2.310	.976	2.258	7580	516	
610027 (spare)	10,390		1.988	1.495	.903	1.351	7690	523	
	12,420		2.220	1.785	.940	1.678	7400	503	
	12,610		2.230	1.800	.942	1.697	7430	505	
	14,310		2.430	2.050	.960	1.970	7520	512	509 □"
	15,210		2.480	2.110	.966	2.040	7450	507	
	16,630		2.630	2.300	.974	2.241	7420	505	
	16,790		2.639	2.310	.975	2.255	7445	507	
610029 (stbd)	10,500		2.016	1.530	.913	1.398	7500	510	
	11,730		2.120	1.660	.930	1.544	7590	516	
	12,670		2.235	1.805	.943	1.702	7430	506	
	14,800		2.441	2.063	.962	1.987	7440	506	508 □"
	14,800		2.441	2.063	.962	1.987	7440	506	
	16,650		2.630	2.300	.974	2.241	7420	505	
	16,650		2.630	2.300	.974	2.241	7420	505	
610034 (port)	9,540		1.910	1.395	.905	1.263	7550	513	
	11,830		2.150	1.700	.935	1.590	7450	507	
	11,830		2.150	1.700	.935	1.590	7450	507	
	14,110		2.370	1.975	.956	1.890	7460	508	508 □"
	15,000		2.450	2.075	.963	1.999	7500	510	
	16,510		2.616	2.280	.974	2.221	7430	506	
	16,880		2.639	2.310	.975	2.254	7430	509	
610026	26,000	A/B On	2.635	2.305	.809	1.867	13920	947 }	932 □"
610027	25,150	" "	2.634	2.305	.809	1.867	13480	917 }	
610029	25,220	" "	2.630	2.300	.809	1.862	13540	921 }	924 □"
610034	25,520	" "	2.639	2.310	.809	1.871	13630	928 }	


YJ95 P-3 PERFORMANCE CURVES
FINAL ACCEPTANCE TESTS
CORRECTED THRUST ~ Fn/δT2 ~ LBS
10000
11000
12000
13000
14000
15000
16000
17000
LEGEND
ENG. No. 610634 L.H.
ENG. No. 610029 R.H.
N2
N1
TT7
PT7/PT2
T.S.F.C.
CORRECTED LOW ROTOR N1/√θT2 R.P.M.
7000
6500
6000
5500
ENGINE PRESSURE RATIO ~ PT7/PT2 ABS.
2.6
2.4
2.2
2.0
9000
8500
8000
CORRECTED HIGH ROTOR N2/√θT2 R.P.M.
1200
1100
1000
900
800
CORRECTED EXHAUST GAS TEMP. TT7/θT2 ~ °F
.85
.80
.75
CORRECTED T.S.F.C. /√θT2
75% NORM.
NORMAL
MILITARY
16
OCT. 1957
R. WECHTER
SECRET

YJ75 P-3 PERFORMANCE CURVES

FINAL ACCEPTANCE TESTS



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SHEET NO. 17 REPORT NO. 90/PERF/1 APP. II

DATE OCT. 1957 PREP. BY R. WAECHTER