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Donald Michel Rudnitski

A DETAILED PROCEDURE FOR MEASURING
TURBOJET ENGINE PERFORMANCE IN AN
ENCLOSED SEA-LEVEL TEST FACILITY

By

© D.M. Rudnitski

A thesis submitted to the Faculty of Graduate Studies in partial fulfillment of the requirements for the degree of Master of Engineering.

Faculty of Engineering
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19 August, 1983

The undersigned recommend to the Faculty of Graduate Studies and Research acceptance of the thesis, "A Detailed Procedure for Measuring Turbojet Engine Performance in an Enclosed Sea-Level Test Facility" by Don M. Rudnitski, in partial fulfillment of the requirements for the Degree of Master of Engineering.

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ABSTRACT

The Engine Laboratory of the National Research Council of Canada has recently participated in a NATO sponsored Uniform Engine Test Program. In this international program, two Pratt & Whitney J57P19W turbojets are being tested in ground-level test beds, indoor and outdoor, and altitude facilities with the objective of rationalizing any performance differences that may appear.

Procedures in the laboratory had to be developed to assess engine airflow, fuel flow, and gross thrust. Airflow was calculated based on total pressure and temperature measurements in an annular section behind the entry bellmouth, and inner and outer wall static pressures. Fuel flow was measured with two turbine meters placed in series, and corrected for viscosity and temperature effects. Gross thrust was deduced using the momentum method and scale force measurements.

A rigorous error analysis was performed with the result that airflow, thrust, and specific fuel consumption could be measured with uncertainties of $\pm 0.77\%$, $\pm 0.31\%$, and $\pm 0.52\%$ respectively.

Data comparisons based on rotor speeds, high or low compressor, yielded poor comparisons with two altitude facilities. However, using non-dimensional thrust coefficients as a function of nozzle pressure ratio, the differences were reduced to 0.2%.

ACKNOWLEDGEMENTS

An international program of this type has never been carried out in the past, as it requires the cooperation and dedicated commitment of many organizations and individuals. It is to the personal credit of Dr. Jim Mitchell of the Arnold Engineering Development Centre that has ensured the success of the Uniform Engine Testing Program thus far, as the logistics of dealing with several countries can be quite overwhelming.

I would like to thank Mr. Earl Dudgeon, Director of Mechanical Engineering, who during his term as Section Head of the Engine Laboratory, strongly encouraged National Research Council participation. He felt we were as capable as anyone in testing gas turbine engines, and in areas where we were weak, we would quickly learn. He was right. In the laboratory a project of this magnitude requires the continuing support of many individuals, it cannot be handled alone. Mr. Howard Fowler, as current Head of the Engine Laboratory, recognized the international importance of this program, and continued to provide the necessary encouragement and resources. There are many people in the Laboratory and Division to whom I am indebted - they number no less than 30 - thus to avoid missing anyone, I won't list their names.

A very special thanks goes to Dr. Bernard MacIsaac of GasTOPS. He provided the motivation and encouragement - forceful at times - when it was most needed, and assisted, along with his staff, in data interpretation.

Dr. Herb Seravanamuttoo of Carleton University is a patient man. He has waited a number of years for this thesis. Thank you Herb for several times giving me "one more chance".

The Department of National Defence through CRAD, the AGARD Panel, and DGAEM have provided financial and physical support for this endeavour. Without it, NRCC could not have

participated in the program. We are now in a better position to respond to DND's needs based on our new found knowledge.

Mlle Nicole Brisson has done a commendable job in trying to untangle several reports in her word processor to find all the necessary parts of this thesis. Hers is a most frustrating job, and being the final link in a strong chain, the deadlines always appear to fall on her shoulders. Thanks Nicole for a job well done.

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NOMENCLATURE

Names

AEDC - Arnold Engineering Development Centre
AGARD - Advisory Group for Aerospace Research and Development
ATF - Altitude test facility
GLTB - Ground level test bed
GTP - General test plan
NASA LeRC - National Aeronautic and Space Administration Lewis Research Centre
NATO - North Atlantic Treaty Organization
NRCC - National Research Council of Canada
PEP - Propulsion and Energetics Panel
SOAP - Spectrographic Oil Analysis Procedure
UEIP - Uniform Engine Testing Program

General

A - Area
Cp - Specific heat at constant pressure
CDB - Nozzle airflow coefficient
CG8 - Nozzle gross thrust coefficient
D - Drag
Ec - Compressor efficiency
EPR - Engine pressure ratio
ETR - Engine temperature ratio
F - Thrust
FG - Gross thrust
FN - Net thrust
H - Enthalpy
K - Turbine flowmeter calibration constant
LHV - Lower heating value
M - Mach number
MW - Molecular weight
P_t - Total pressure
PSW - Test cell wall static pressure
R - Gas constant

NOMENCLATURE (CONT'D)

SLS	- Sea level static
SFC	- Specific fuel consumption
T	- Total temperature
TM	- Metal temperature
V	- Velocity
W or WA	- Airflow
Wfe or WF	- Fuel flow
γ	- Ratio of specific heats
$\bar{p}$	- Referred pressure P/P_{ref} .
η	- Efficiency
$\bar{T}$	- Referred temperature T/T_{ref} .
ν	- Kinematic viscosity
ρ	- Density
α	- Linear expansion coefficient

Subscript - General

AV	- Average
comb	- Combustion
e	- Exit
g	- Gross
i	- Inlet
m	- Measured
n	- Net
o	- Free stream
∞	- Ambient

Twin-Spool Turbojet

BV	- Bleed valve
EGT	- Exhaust gas temperature
HPC	- High pressure compressor
LPC	- Low pressure compressor
NH	- High pressure spool speed
NL	- Low pressure spool speed

1.0 INTRODUCTION AND OBJECTIVES

The Propulsion and Energetics Panel (PEP), Working Group 15, of the Advisory Group for Aerospace Research and Development (AGARD) is sponsoring a Uniform Engine Testing Program (UETP). In this program, two Pratt and Whitney J57-19W non-afterburning turbojet engines are being tested under identical conditions, in a variety of altitude and ground level facilities, as a means of correlating these facilities.

The objectives of the UETP, as stated in the General Test Plan, are the following:

- To provide a basis for upgrading the standards of turbine engine testing within NATO countries by comparing test procedures, instrumentation techniques, and data reduction methods, thereby increasing confidence in performance data obtained from various engine test facilities.
- To compare the performance of an engine measured in ground-level and in altitude test facilities at the same non-dimensional conditions and to investigate the reasons for any observed differences.

To achieve these objectives, each participant is to use the facility test procedures, instrumentation arrangements, and analysis methods that are consistent with their normal practice to define three basic engine performance parameters: airflow, gross thrust, and fuel flow. Certain instrumentation has been provided to travel with the engine. This reference instrumentation is being used to set test conditions, and monitor engine health and engine performance degradation.

NASA Lewis Research Center was responsible for the initiation of the UETP, the initial program management, and the procurement of some hardware and instrumentation for use in the remaining phases of the UETP.

The Engine Laboratory of the National Research Council of Canada was invited to participate in the program, with the intent of becoming an independent, internationally recognized gas turbine test facility. As all previous jet engine testing at the laboratory was of a comparative nature, there were no established procedures to measure absolute engine performance to a relatively high degree of accuracy. A detailed study of test cell philosophy was required with particular attention being paid to the influence of cell size and secondary airflow in the determination of gross thrust.

The purpose of this thesis is to detail the efforts required by NRCC to participate in the UETP. It describes the test installation, instrumentation system, calibration techniques, data reduction equations, and actual test results. As a measure of success, the test results are compared with those obtained at two altitude facilities in the United States, operating under simulated sea-level conditions.

2.0 BACKGROUND

Engine manufacturers, airframe suppliers, and the final customers are all interested in knowing the true performance of turbojet or turbofan engines. As engines are sold on the basis of guaranteed thrust and specific fuel consumption, it is crucial to accurately determine these two quantities with confidence, especially since the performance margin is usually only 3 to 4% above guarantee. Engine calibration facilities, equipped with high quality data acquisition systems, are used to test engines under closely controlled conditions. In addition to thrust and fuel flow, airflow and rotor speeds are also measured, thus enabling calibrations of thrust to be obtained as a function of other parameters.

Two basic classes of engine calibration facilities exist, the Ground Level Test Bed (GLTB), and the Altitude Test Facility (ATF). The most common is the GLTB in which the engine operates under the prevailing sea level static conditions.

Intake and exhaust pressures are nominally the same. Altitude Test Facilities are provided with extensive air compressor equipment in order to enable engine inlet and nozzle exhaust pressures to be independently controlled, and inlet air to be conditioned with respect to pressure and temperature. Thus the facility can simulate the inlet and exhaust conditions of an engine over a range of altitude and aircraft flight Mach numbers.

The primary measurements of most concern in the test facilities are gross thrust and airflow. Thrust is usually measured by mounting the engine from a framework which, itself, is suspended from a fixed structure by means of flexures to enable the engine in its framework to move freely in the axial direction. The limit of axial movement is restrained and the resulting force measured, usually by a strain gauge load cell. The air intake to the engine is usually a bellmouth or a venturi designed as an airmeter. This inlet is usually mounted on the thrust bed, so that under static conditions the force on the framework, measured by the load cell, is very close to the gross thrust of the engine (Figure 1, obtained from Ref. 1).

Figure 1(a) schematically represents an open air test bed. If testing is conducted in conditions of zero wind then the measured force is the gross thrust. As such ideal conditions are rare, testing may need to be done with a wind blowing. If there is a head wind, and is not greater than 5 m/s (16.5 ft/sec), the resulting free stream momentum correction to measured thrust will then be in the order of 1 percent.

Enclosure of test cells results in further thrust corrections. In the simplest type of enclosure, Figure 1(b), engine air is constrained to approach the inlet venturi with a small but consistent velocity, which again leads to an approach momentum force deficit on the bellmouth. The most usual form of GLTd (Figure 1(c)) has an ejector tube or detuner that collects and silences the exhaust and provides cell scavenging. Entrained secondary air is drawn over the engine from the test bed intake. This increased approach momentum has to be properly credited to

the engine as it is the prime mover responsible for accelerating the air from rest to the prevailing cell velocity. Drag forces are exerted on the engine external carcass and thrust stand, and the static pressure around the engine is modified to a small degree. Further corrections thus have to be made to the measured force in order to obtain gross thrust.

The load cell measuring system must also be calibrated in-frame by applying axial loads to the complete engine/frame assembly at the engine centreline by means of weights on a master load cell system.

The corrections to the measured load to obtain gross thrust are derived by one of two ways, a) by a careful cross calibration with an open air test facility, and b) by computation using air velocities and pressures measured in the test cell. It should be noted that these corrections are particular to the engine type and test cell arrangement. The magnitude of the corrections can vary from zero up to 5 percent.

An essential feature of Altitude Test Facilities (Figure 1(d)) in the context of thrust measurement, is the physical separation of the inlet from the engine. The plant facilities supply air at the required temperature and pressure via a flow measuring system. The engine exhausts into the free space of the chamber, the chamber itself being exhausted to the required ambient pressure level. In deriving the gross thrust, allowances have to be made for external pressure forces, slip joint leakage, carcass drag resulting from ventilation flows, and inlet momentum.

As stated earlier, one of the objectives was to compare data analysis techniques amongst the participating facilities. The methodology selected was to test a pair of engines in various types of facilities, ground level enclosed and outdoor, and altitude, and then compare the data. The tests were performed in sequence, with the first facility retesting the engines at the end of the program to ensure that no engine

deterioration had occurred. The participating countries include the United States, Canada, United Kingdom, France, and Turkey.

3.0 TEST INSTALLATION

3.1 Test Cell

The test cell designated for the UETP was cell No. 5 of the Engine Laboratory, NRCC, at Ottawa, Canada. A sectional elevation of this sea-level-static test cell is shown in Figure 2. Ambient air enters through three banks of 9.5 m (31 ft.) wide, 4.6 m (15 ft.) high zigzag silencing splitters. Wire mesh on the entrance and exit faces of this group of silencing baffles prevents debris from entering the test cell. The planar dimension of the cell reduces obliquely from 9.5 m (31 ft.) to 4.6 m (15 ft.) in the inlet plenum, leading into the constant 4.6 x 4.6 m (15 x 15 ft.) area of the test section downstream from the plenum. The test cell inlet can be closed with folding doors at Station A.

Engine efflux and entrained secondary air flow are ducted from the cell through an exhaust collector tube to a vertical silencer that discharges to the atmosphere, approximately 6 m (20 ft.) above the cell roof.

The intake section, test cell, and exhaust stack are constructed of heavy reinforced concrete. These concrete walls and a 100 mm (4 in.) thick sand-filled acoustic barrier, spanning the full cross section at the rear of the cell, around the exhaust collector tube at Station B, minimize noise propagation to the outside.

The control, observation, and data acquisition room is located on the starboard side of the cell, on a mezzanine floor, about 1.5 m (5 ft.) above the test cell floor level. In addition to a window at the operator's left side, a remote-control closed-circuit television camera permits visual monitoring of the engine.

The engine installation is schematically shown in Figure 3. The engine bed is suspended by four 300 mm (12 in.) long flat plate flexures from the thrust frame which is anchored to the floor of the test cell. A single dual bridge load cell, mounted in compression, measures thrust at the base of the test bed. A centre-pull apparatus incorporating a reference load cell was used to perform an in-frame calibration.

For controlling secondary airflow, inserts of different diameters can be placed in the exhaust collector. (A radial spray water cooling system, not used for this program, prevents overheating of the exhaust silencer when engines are run in afterburn).

The engine entry bellmouth directed the air into a cylindrical air measuring section, before it entered the compressor. All parts were hard-mounted to the thrust stand so that forces generated by the bellmouth were transmitted to the thrust bed. An antipersonnel screen was anchored to the test cell floor, in front of the bellmouth.

3.2 Test Article

3.2.1 Engine Description

The two test engines were U.S. Air Force supplied Pratt and Whitney J57-P-19W turbojets (S/N 607594 and 615037). The J57-P-19W engine (Figure 4) is an axial-flow two-spool non-augmented turbojet with a fixed-area exhaust nozzle. Rated sea-level-static thrust is 46.71 kN (10,500 lb) at military power. The engine inlet diameter is 932 mm (36.7 in), and the maximum engine dry weight is approximately 1882 kg (4150 lb).

The engine has a sixteen-stage two-spool compressor with an overall pressure ratio of 11.3:1, at a rated airflow of 74.83 kg/s (165 lbm/sec) at sea-level-static, military power conditions. The nine-stage low-pressure compressor (LPC) is connected by a through-shaft to a two-stage low-pressure turbine.

The seven-stage high-pressure compressor (HPC) is connected by a hollow shaft to a single-stage high-pressure turbine. The only variable geometry feature is an intercompressor bleed which discharges air overboard during starting and low power operation.

The combustor section consists of an annular diffuser and a cannular combustor unit with eight flame tubes. Each flame tube contains six dual-orifice fuel spray nozzles. Ignition is accomplished by spark igniters located in flame tubes 4 and 5. Combustion spreads to the other flame tubes by means of cross-over tubes welded to the forward section of each chamber. Power for ignition is provided by a 7-amp, 24-vdc external source.

Engine fuel is metered by a hydromechanical fuel control as a function of power lever position, high-pressure compressor rotor speed, compressor inlet temperature, and compressor discharge pressure. High pressure rotor speed, at a fixed power lever position, is biased by compressor inlet temperature. Burner pressure is limited to a maximum of approximately 1,380 kPa (200 psia).

The test engines were modified to incorporate (1) a cylindrical tailpipe and converging fixed geometry exhaust nozzle assembly (Section 3.2.5), (2) a "reference" Uniform Engine Testing Program Instrumentation Package, and (3) a water/oil cooler to replace the aircraft oil cooler. The test program did not require horsepower extraction, water-injection, customer air bleed, or anti-ice air bleed.

3.2.2 Compressor Bleeds

The test engine utilizes two compressor bleed valves (left and right). Each valve is scheduled independently as a function of both LPC speed and compressor inlet temperature. Because of overlapping schedules, high inlet temperature operation of the engine with the bleed valves in this configuration limits the bleeds-closed speed range to approximately 250 rpm HPC. This limited speed range could have prevented use of the

test procedure recommended by the PEP Working Group 15 (nine steady-state power levels between bleeds-closed and military power). To expand the useful speed range, the right hand bleed port was capped off and the left-hand bleed port was enlarged to an acceptable geometry ("Fighter" engine version vs. "Bomber" engine), as described in the General Test Plan (Ref. 2). The increased rotor speed range (approximately 800 rpm HPC) allowed use of the recommended test procedure. In addition, anti-icing and customer bleed ports were capped to ensure that no leaks were present.

3.2.3 Lubricating Oil System

Engine lubrication is provided by a pump which provides a flow of high pressure oil to the main bearings and accessories. A scavenge system drains this oil and pumps it through an externally mounted oil cooler before returning it to the storage tank. A breather system incorporates a pressurizing valve so that a common pressure is maintained between the individual bearing compartments and the oil tank.

The external oil cooler was part of the equipment shipped with the engine. It used water as a coolant and maintained oil temperature at $366 \pm 6K$ ($200^{\circ}F$).

3.2.4 Engine Mount System

The engine mount framework (Figure 5) was supplied by NASA and was bolted directly to the NRCC engine test stand. This frame provided load points at four locations on the engine case, two points on the intermediate casing and two points on the turbine casing. The NRCC test stand included instrumentation patch panels and was used to mount the oil cooler which was part of the engine package supplied by NASA.

Although not really attributable to the engine mount system, the air starter system is a test facility device which contributes to parasitic losses in much the same way as the test

stand. This was mounted on the engine casing throughout all tests and has been treated as part of the engine mount system for purposes of computing losses.

3.2.5 Engine Exhaust Nozzle

Because the tailcone on the standard J57-P-19W extends through the nozzle exit plane (Figure 4C), it was difficult to accurately determine nozzle entry conditions. The Bill-of-Materials nozzle was replaced by a cylindrical tailpipe and a reference convergent nozzle, both fabricated from sheet metal (Figure 6). The cylindrical tailpipe provided a simpler means of establishing nozzle inlet conditions, however it required a calibration test to size the nozzle so that engine performance could be restored to approximately the nominal value. This work was performed by NASA LeRC and although two engines were involved in the program, only one tailpipe-nozzle assembly was used.

3.2.6 Inlet Hardware

The engine inlet nose bullet, which is an aircraft part rather than an engine part, was fabricated by NASA from existing U.S. Air Force designs and modified to mount the inlet instrumentation rakes. An instrumentation spool piece, also provided by NASA, was attached to the engine inlet and used to monitor entry conditions and measure air inflow.

The NRCC airflow measuring station required the removal of the NASA nose cone, insertion of a parallel annular section in front of the spool piece, and replacement of the conical nose with one of a parabolic design. An elliptical fiberglass bellmouth, anchored to the thrust stand, completed the flow entry section. Figure 7 details the differences between the two configurations.

3.2.7 Fuel

Jet A or A-1 fuel rather than Jet B (JP4), the most commonly used fuel for this engine, was used for the UET Program, necessitating a pre-test engine re-trim of the two engines at NASA. The fuel was filtered to 10 microns through water coalescing filters and supplied at the appropriate pressure to the engine fuel pump.

A sample of fuel was taken at the beginning and end of each test period and analyzed for viscosity, specific gravity, and lower heating value. The results are listed in Table 1.

4.0 INSTRUMENTATION

4.1 General

Instrumentation was made up of two components, the primary or facility instrumentation, and the reference or engine-dedicated instrumentation. The primary instrumentation was required to determine engine inlet airflow, fuel flow, and thrust. The reference instrumentation was used to determine environmental test conditions, to monitor engine health and safety, and to record any engine deterioration between tests.

Engine station designations are numbered as shown in Figure 8; the numbering system agrees with the SAE-755A recommendations (Ref. 3) although it differs from that traditionally used for this engine.

4.1.1 Engine and Reference instrumentation

The reference instrumentation which measured pressures, temperatures, speeds, and fuel flow were provided by NASA and were affixed to the engine or support stand to form an integral part of the test package. The engine entry (Station 2, Figure 8) was instrumented with pitot probes, wall statics, and total temperature and boundary layer rakes. This station was

used to measure inlet environmental conditions as well as to determine circumferential and radial pressure and temperature profiles. The boundary layer rakes were required to assess boundary layer thickness as it was expected to vary with each installation and affect average inlet pressure. Airflow calculations for this plane were compared with those for the facility (NRCC) airmeter. Turbulence characteristics were measured by placing a high response (0 - 500 Hz) Kulite pressure transducer, flush mounted to the inlet wall in the plane of the wall static taps.

Pressures and temperatures were measured at the high pressure compressor exit (Station 3), turbine exit (Station 5), and at the exhaust nozzle inlet (Station 7). Overall compressor performance was calculated from Station 3 information, and overall pumping capacity from Station 7 measurements. Ambient pressure was defined as that measured with the exhaust nozzle trailing edge statics (Station 0.5). The equations used are detailed in Ref. 2.

Each engine was provided with a fuel flow measuring system consisting of two turbine flowmeters in series, a thermocouple, and associated piping. The system was calibrated and sealed by NASA, thereby ensuring that fuel flow comparisons could be made at all facilities.

Sixty-tooth reluctance-type speed sensors were installed on each rotor gearbox to provide a common means of monitoring rotor speeds. Other instrumentation for ensuring safe engine operation monitored engine vibration, oil and fuel pressure and temperature, and exhaust gas temperature. A list of the reference instrumentation is found in Table A-2 of Appendix A.

4.1.2 Facility Instrumentation

All participating test facilities were to use their established methods for measuring engine airflow, fuel flow, and

gross thrust. The NRCC method of determining inlet total airflow, described in Appendix B, was based on the integration of flow per unit area assigned to each pitot probe, including the boundary layer. This method assumes that the static pressure is constant across the annulus at the airflow station (Station 1). Accordingly, only outer and inner wall static pressures in the boundary layer and a number of pitot pressures and total temperatures in the free stream were measured. A flow per unit area was determined for each pitot probe using the total and static pressure measurements, an average temperature measurement obtained from probes at Station 2, and the physical properties of air. The total airflow was then determined by means of a numerical integration of the flows per unit areas. The entry bellmouth was sufficiently far upstream to assume that no curvature effects were felt at Station 1.

Fuel flow was measured with two turbine flow meters placed in series, thereby providing confidence in the data. Uniform flow to each meter was assured by utilizing long radius bends followed by 15 diameters of straight tubing upstream and downstream of the meters. Type 'I' thermocouples were used to measure fuel temperature for calculating viscosity and specific gravity. A table look-up routine determined the K factor over the appropriate range of frequency and viscosity. Appendix C describes in detail the calculation of fuel flow.

Engine gross thrust was the most difficult parameter to deduce. While it is relatively straight-forward to measure scale force on the thrust bed, there are a number of correction factors that must be applied when an engine is operated in an enclosed ground level test bed. The cell correction factors are a function of cell configuration, airflow demand of the engine, and the positioning of the engine in the cell. The augmenter size and its position relative to the engine affects the local static pressure at the nozzle exit and the total amount of secondary airflow through the cell (Ref. 10). This additional secondary air has the effect of increasing the approach momentum of the air entering the engine. Additionally, the entrained air

is drawn over the engine giving rise to drag forces on the engine carcass and stand.

The momentum balance for an engine in an enclosed test cell with secondary airflow is given below:

$$F_m + W_i V_{\infty 1} + D_{bm} + D_c + D_{ts} + D_{bt} = WeVe + (P_e - P_{\infty 2}) A_e$$

Where

- F_m = measured thrust from load cell
- F_n = net thrust
- F_g = gross thrust
- F_{ram} = ram drag
- W_i = engine inlet mass flow
- We = mass efflux from the engine
- $V_{\infty 1}$ = free stream velocity upstream of engine inlet
- Ve = nozzle exit velocity
- V_o = equivalent flight velocity
- $P_{\infty 1}$ = cell forward static pressure
- $P_{\infty 2}$ = nozzle base static pressure = P_{amb}
- P_e = nozzle exit static pressure
- A_e = nozzle exit area
- D_{bm} = form drag of bellmouth intake
- D_c = skin friction drag on engine casing
- D_{ts} = form drag on thrust stand
- D_{bt} = bouyancy (boatail) drag on exhaust nozzle.

Considering that

$$(P_e - P_{\infty 2})A_e + W_e V_e = (P_e - P_{\infty 1})A_e + (P_{\infty 1} - P_{\infty 2})A_e + W_e V_e$$

and gross thrust, F_g , is defined as

$$F_g = (P_e - P_{\infty 2})A_e + W_e V_e - (P_{\infty 1} - P_{\infty 2})A_e$$

then gross thrust

$$F_g = F_m + W_1 V_{\infty 1} - (P_{\infty 1} - P_{\infty 2})A_e + D_{bm} + D_c + D_{ts} + D_{bt}$$

and net thrust,

$$F_n = F_g - F_{ram}$$

where $F_{ram} = W_1 V_o$.

V_o is the equivalent flight velocity based on the difference between inlet total pressure, P_2 , or $P_{\infty 1}$ if no significant losses occur due to screens at the engine inlet, and the base nozzle pressure, $P_{\infty 2}$. In effect, the engine appears to experience a forward flight velocity (Ref. 4).

A detailed treatment of the individual terms follows:

1) Inlet Momentum, $W_1 V_{\infty 1}$

Inlet velocity, $V_{\infty 1}$ is calculated from total cell airflow at a particular accounting plane, ∞_1 . The common method employs an energy balance across the cell, provided exhaust stack temperature is measured.

$$W_{cell} = \frac{1000 \cdot W_{fe} (LHV \cdot \eta_{comb} + H_{fuel} - H_{stack})}{H_{stack} - H_{inlet}}$$

$$\text{and } V_{\infty 1} = \frac{W_{cell}}{\rho_{\infty 1} \cdot A_{cell}}$$

$$\rho_{\infty 1} \cdot A_{cell}$$

where W_{cell} = cell mass airflow

W_{fe} = engine mass fuel flow

LHV = fuel lower heating value

η_{comb} = combustion efficiency

H_{fuel} = fuel enthalpy

H_{stack} = exhaust stack enthalpy

H_{inlet} = inlet air enthalpy

$\rho_{\infty 1}$ = air density at plane ∞_1

A_{cell} = cell cross-sectional area

For No. 5 cell, the momentum defect is in the order of 1% F_g .

2) Tailpipe Pressure Correction, $(P_{\infty 1} - P_{\infty 2})A_e$

With the engine tailpipe in close proximity to the exhaust collector, the secondary air must accelerate to pass through the annular gap. This accelerated flow causes the static pressure at the nozzle exit, $P_{\infty 2}$, to be depressed relative to the free stream static, $P_{\infty 1}$. The nozzle then exhausts into a lower pressure area and generates more thrust. Therefore, to correct back to inlet conditions, the area x pressure difference term must be subtracted from the gross thrust. Typical values are in the order of 0.3% gross thrust.

3) Bellmouth Form Drag, Dbm

The thrust contribution of the bellmouth, which is usually attached to the thrust stand, is directly credited to the engine and included as part of the scale force. High secondary airflow could give rise to horizontal pressure gradients along the engine, and in particular behind the bellmouth. If the pressure behind the bellmouth (P_b) differs from $P_{\infty 1}$, then an additional correction term $(P_{\infty 1} - P_b) A_{proj}$ must be applied over the projected annular area of the bellmouth. By keeping the secondary airflow low, P_b will be the same as $P_{\infty 1}$, thus requiring no correction. This was the case for No. 5 cell.

In the NRCC treatment of the bouancy terms in the test cell, the bellmouth drag, Dbm , and the boatail drag, Dbt , are treated separately. Normally a surface integral with respect

to pressure along the exterior surface, omitting the inlet and exhaust planes, would encompass all the terms, but as $P \approx P_1$, is virtually constant along the skin except in the vicinity of the nozzle, these two particular areas were given special attention.

4) Skin Friction Drag, D_c

Scrubbing of the engine carcass by the induced secondary airflow results in a skin friction drag that must be credited to the engine. However, the values are usually negligible and can therefore be ignored (typically 0.003% F_g).

5) Thrust Stand Form Drag, D_{ts}

Secondary airflow also impinges on the frontal area of the thrust stand and creates an additional drag. An estimate of the magnitude can be made by using a flat plate approximation for a drag coefficient. Even though the computed values are small, they are included in the thrust accounting (0.04% F_g).

6) Bouyancy (Boat tail) Drag on the Exhaust Nozzle, D_{bt}

Due to the close proximity of the nozzle exit to the collector inlet, the secondary airflow accelerates past the exhaust nozzle into the mixing tube, and sets up a static pressure gradient along the external surface of the nozzle. The generated net force can be calculated by integrating the surface pressures along the nozzle and resolving the horizontal component. By carefully selecting the insert size and proper spacing of the exhaust nozzle to the collector, this correction can be kept to 0.1% of scale force thrust.

7) Thrust Force on Load Cell, F_m

Scale force thrust is measured with a dual-bridge strain gauge load cell placed in compression between the thrust bed and thrust frame. To account for stand stiffness, spring rate, and hysteresis, the load cell is calibrated "in-frame" uti-

lizing a centre-pull apparatus and a reference load cell. A description of this calibration process is outlined in Appendix D.

The test cell instrumentation consisted of transducers, steady-state readout instruments, and a data acquisition system. The engine operator required, as a minimum, a continuous display of spool speeds (NH and NL), exhaust gas temperature (EGT), oil pressure and temperature, fuel pressure, and vibration levels. These parameters and others such as fuel flow, environmental conditions, oil scavenge pressure, stack temperature, and cell cooling air velocity were displayed on the panel using digital indicators, bourdon gauges, and voltmeters.

Data required to determine engine performance were gathered by a minicomputer-controlled data acquisition system. At the beginning and end of the test program, rigorous calibrations were carried out on all transducers. Load cells were calibrated on a dead-weight tester, then installed in the thrust stand for several in-frame centre pull calibrations. A transferable pressure standard formed the basis for pressure transducer calibration, and a voltage source verified data system accuracy for thermocouples and load cells. At the beginning and end of each test sequence, an offset and a gain reading were recorded and compared against known values, or against the steady-state visual displays such as digital temperature indicators. Throughout the test sequence, a hand written log of all the panel instruments was kept; this practice provided a cross check of a number of key parameters such as fuel flow and thrust.

A layout of the system data flow (Figure 9) traces the experimental and calibration data path. Briefly, the scanivalve system, located in a constant temperature area, directed individual pressures to externally mounted capacitive pressure transducers. Where the number of signals exceeded the capacity for a single scanivalve, they were staged through another scanivalve to enable a single transducer of a given range to be used. While this method delayed the data collection process, it reduced the overall error that comes from using multiple transducers. A

Texas Instruments quartz capsule provided a reference pressure on a selected port against which the transducers could be verified at run time. Thermocouples were changed from type "K" or "J" material to copper wire on a Kaye Uniform Temperature Reference (UTR) plate located in the test cell and connected to the data system. The temperature of the UTR with reference to a Kaye electronic ice-point was analytically added to obtain the true temperature. Frequency inputs, checked by a signal generator, from fuel meters and speed sensors were isolated with differential amplifiers and directly read as a frequency by the data system. The 60 tooth speed pick-offs were replaced with 120 tooth pick-offs to increase speed resolution.

Cell static pressure was read out by a Digiquartz absolute pressure transducer located on the cell wall (PSW2) in the plane of the bellmouth. All differential transducers were referenced to this location thereby eliminating the need to correct for test cell depression. Four wall statics were used, one at the upstream location, Stn. 00, one in the bellmouth plane, one in the exhaust plane, and one mid-way along the engine. A mercury in glass barometer was used as a check on the absolute transducer when the engine was inoperative.

All the signals from the various transducers were recorded by a dedicated data system comprising a DEC PDP 11/23 minicomputer and a DEC Compact System Controller. In general, all low level signals, except the thermocouples, were filtered with 10 Hz filters, amplified to a nominal 10 volts full scale, multiplexed, and digitized with a 12-bit analog to digital converter.

Fuel flow and thermocouple calibrations were handled by look-up tables, while linear curve fits were used for pressure transducer calibrations. After conversion to engineering units, the data were stored on floppy diskettes for subsequent off-line analysis.

The list of transducers and reference instrumentation is given in Table 3 and in Table A-1 and A-2 of Appendix A. Hand readings, listed in Table A-3, were logged for each data point.

5.0 TEST DESCRIPTION

5.1 Engine Operating Procedures

The two test engines were operated in accordance with the operating procedures outlined in sections 8.1 to 8.3 and Appendix III of the General Test Plan (Ref. 2). The procedures included: engine depreservation, fuel/oil leak check, engine starting and shutdown, engine and fuel control system functional check, and engine performance testing. The engine operating limits observed were those listed in Appendix E.

5.2 Test Conditions

As NRCC has no inlet temperature control, tests were run at the available extremes of ambient temperature and pressure. This approach allowed NRCC to evaluate its correction and standardization methods for observed data.

Operation of the engine in an indoor test cell results in some secondary airflow in the cell, due to the engine exhausting into the collector and acting as an ejector. Although some airflow is necessary to avoid recirculation and to cool the engine, high secondary airflow can cause pressure gradients in the cell, a large approach momentum, and parasitic drag forces on the engine and test stand, all of which require calculated corrections. Test cell velocity was kept within 5 to 6 m/s (16 to 20 ft./sec.) by sizing the exhaust collector to 0.84 m (33 in.) diameter, and locating it 0.46 m (18 in.) behind the engine exhaust plane. By keeping air velocities low, computed corrections are made small enough that any errors become insignificant.

Engine performance is significantly affected by high humidity in the intake air. While air properties (C_p , γ , MW, R)

can be corrected for reasonably low levels of absolute humidity, high relative humidity may result in condensation in the engine inlet. The phase change from vapour to liquid in the accelerating air inflow stream results in a temperature rise in the air stream, which is difficult to handle in air flow calculations. Subsequent evaporation in the fan stream reverses the process. Furthermore, wetness of fan blade surfaces may affect fan efficiency. Visible moisture in the air could also enter the pressure probes introducing biases in the pressure measurements. Thus, in order to avoid condensation in the air inlet stream, the following atmospheric limitations were observed:

maximum relative humidity: 75%

maximum absolute humidity: 14 g water /kg air (0.014 lbm water/lbm air or 98 grains).

For sea-level testing, the engine inlet total pressure was defined as the numerical average of the 20 total pressure measurements at Station 2.0. The boundary layer rakes were not used in the determination of P2. Engine inlet total temperature was defined as the numerical average of the 10 total temperatures at Station 2.0. A recovery factor of 1.0 was used for the temperature probes, as they were constructed of small diameter, exposed loop thermocouple material in ~~deeply~~ recessed radiation shields (Ref. 5).

Two regions of engine operation were specified:

- 1) From engine power settings from "bleeds just closed" speed to military power, and
- 2) From engine power settings from "bleeds just open" speed to idle power.

As the NRCC facility has no environmental control, the engine power settings had to be established for each test temperature. For the high power region, values of high rotor speed (NH) were established for bleed valve closed (BVC) and MIL

power (MIL power is defined as 90° power lever angle, and is set up at engine trim time to deliver a specified level of thrust). The test range was then divided into 8 equal increments resulting in 9 values of NH. Similarly for the low power region, the speed range between idle and BVC was divided up to obtain 9 values of NH. Table 2 outlines the test points and sequence, for both ranges.

Specifically, the test sequence was as follows:

- 1) The engine was lit as defined in Section 5.1 and allowed to stabilize at idle. After recording NH, the engine was slowly accelerated to the point of bleed valve closed, and NH again recorded. This defined the speed band for the low power region. A mechanical flapper, connected to a limit switch and indicator light on the operator's console, was used to monitor bleed valve status.
- 2) The engine was then accelerated to MIL power and allowed to stabilize for 5 minutes. NH at this setting and the BVC NH defined the high power region.
- 3) Hand readings of key engine parameters and environmental conditions were taken concurrently with the data scan. Two data scans were taken at each power setting.
- 4) Nine engine power settings were determined as per Table 2. The sequence of power settings is shown graphically below.

Note: Test point 3 was chosen 30 rpm above BVC rpm to ensure stable engine operation.

- 5) Step 3 was repeated at each power setting 2 to 9 (725 rpm) allowing a 2 minute engine stabilization time before the initial data scan.

5.3 Data Acquisition and Reduction

At the beginning and end of each test sequence (9 power settings), a data scan verified the transducer zero and gain values, and established a load cell tare. If the post test load cell tare value differed from the initial, the final value was used. A centre-line pull calibrated the thrust stand at the beginning and end of the test series for each engine, (Appendix D).

For these steady state tests, each parameter input signal was sampled at a constant rate of approximately 100 Hertz, over a 5 to 10 second period. Filtering of the input signals removed frequencies above 5 Hertz. The digitized samples were then arithmetically averaged to yield a single value. Two, back to back, scans of all parameters were done at each throttle setting to demonstrate repeatability, and to detect intermittent signals. Each scan took approximately 5 minutes, during which time hand recordings were logged from the visual display instruments as listed in Table A3 of Appendix A.

Parameter mean values were automatically converted from digital counts to engineering units; pre-stored calibrations, which had been checked prior to the run, were used to perform the conversion before display on a printer. A visual comparison of the computer-acquired data to the hand recordings was made prior to storage on a magnetic disk.

In addition to recording values for each individual sensor, expressed in SI engineering units, the General Test Plan (GTP) Section 9.0 specified many additional calculated parameters

such as airflow, pressure, and temperature ratio. A software performance package incorporating the equations was developed to calculate the desired parameters off-line. The calculation of airflow and fuel flow are described in detail in Appendices B and C respectively.

All data were copied to magnetic tape, according to a NASA LeRC-supplied format, for transfer to NASA for plotting and distribution to other test centres. Bad data, determined by inspection, were identified on the tape and removed from averaging routines or further calculations.

The details of the data reduction equations used for this program are those provided in the General Test Plan (Ref. 2, section 9.0). The thermodynamic properties routines (γ , C_p), given in Ref. 6, were modified using a weighted average technique for absolute humidity (Ref. 7). Scale force thrust was corrected by the dry/moist air ratio.

5.4 Measurement Uncertainty

Estimates of measurement uncertainty are necessary to establish the quality of data before any meaningful comparisons can be made with other test facilities. Measurement uncertainty is the maximum error which might reasonably be expected between a measurement and the true value defined by the National Standard.

Measurement error has two components: a fixed or bias error, and a random or precision error. The bias error is the constant or systematic error and is determined by comparison with a true value. Through a calibration process, correction factors are generated that can eliminate or reduce large biases. Precision error, which is the variation between repeated measurements, can often be reduced by taking several repeated observations and averaging.

Overall measurement uncertainty combines the elemental bias and precision error and expresses it as a single number for

a reasonable limit of error. The procedure outlined in Reference 8 was followed to generate values of bias, precision, and uncertainty for each measurement. Calibration, data acquisition, and data reduction error sources were combined in a root-sum-square method to yield overall values for the errors; these are shown in Table 4. Since some error estimates are based on full scale values, and others as percentages of reading, two operating points were chosen, one at a high power setting (8950 rpm NH), and another at a low power setting (7150 rpm NH). The uncertainty estimates were then stated in engineering units and as a percentage of reading for both power settings.

As basic measurements were used to calculate engine performance parameters, i.e. pressure ratio or corrected speed, uncertainty estimates were produced for the parameters of interest. These are listed in Table 5.

6.0 TEST RESULTS

6.1 Engine Operational Data

Engine SN607594, being the first engine tested, accumulated a total of 39 running hours at NRCC, of which 5 hours were spent at MIL power. The vibration levels, SOAP analysis, and exhaust gas temperature spread were all within prescribed limits. Seven T7 thermocouples were unserviceable, while 6 others were intermittent during portions of both engine tests.

Engine SN615037 would have accumulated only 21 hours, however, during preservation, a fuel control gasket failed. The fuel control was replaced, and the engine retested, bringing the total number of running hours to 28.5, of which 3 were at MIL power. Vibration levels, though higher on this engine, were still well within acceptable limits. EGT spread was near the limit of 80°K, but rather than a hot streak, it was a cool spot that caused this spread. This cool spot could be caused by a malfunctioning fuel nozzle in one of the combustor cans, but as this temperature profile was also noted in previous testing at

two other facilities, any attempt to remedy the problem would invalidate any meaningful comparisons. Appendix F summarizes pertinent mechanical data for both engines.

6.2 Airmeter Calibration

As described in Appendix B, the NRCC facility airmeter required the insertion of rakes at Station 1, and replacement of the NASA conical nose with an extension piece and a parabolic nose. A flow coefficient valid over the applicable range of inlet Mach numbers was derived by forming a ratio of the integrated airflow to the ideal airflow based on outer wall static pressure measurements at Station 1. All subsequent testing was done without Station 1 rakes; however, data were gathered with both nose bullet configurations to quantify any performance differences. Over the range from 0.10 to 0.35 inlet Mach Number, the Station 1 flow coefficient remained relatively constant at 0.9908. This coefficient was used on all test runs with the NRCC nose cone to determine facility airflow (WA1) at Station 1. Comparing integrated Station 1 airflow (WA1) to Station 2 airflow (WA2) resulted in an average ratio of $WA1/WA2 = 0.9804$. Thus to deduce WA1 with the NASA nose installed, WA2 was multiplied by 0.9804 for the complete range of inlet Mach numbers.

Airflow measurements, although quasi-steady state in an overall sense, are still subject to a certain level of turbulence in the engine inlet. Measurements taken at the Station 2 air metering station with a high response static pressure transducer showed that the root mean-square pressure variation was less than 0.2% of the average total pressure at military power. The predominant frequencies were in the 100 to 200 Hertz range.

6.3 Performance Plots

In an engine correlation program, some basis of comparison has to be established. If, as is usually the case, there are differences in data, it is first necessary to ascertain whether there has been a real shift in engine performance. A

shift in performance may be caused by either engine deterioration or by external cell effects. Cell effects determine the environment in which the engine operates, and include items such as inlet air turbulence, temperature and pressure distribution, and boundary layer growth within the bellmouth. Secondly, it must be established whether the differences are real, or simply within the data scatter estimated by the measurement uncertainty. It would be unrealistic to explain a 0.5% performance difference if the data were gathered with a 1.0% error band.

The main engine performance parameters of interest were engine airflow (WAI), net thrust (FN), and fuel flow (WF) or specific fuel consumption (SFC), normalized to sea-level-static (SLS) conditions. For comparative purposes, it was agreed to plot these parameters against high pressure compressor rotor speed, corrected to standard day SLS conditions ($NH/\sqrt{\theta}$), as that was the engine parameter used in setting test conditions. As a technique for data comparison, the data for each facility were fitted using a least-squares quadratic equation; then using a nominal value of 8900 rpm $NH/\sqrt{\theta}$ (91.75%), the appropriate value of the dependant variable was calculated and recorded. Data comparisons with NASA and AEDC are expressed as percentage differences normalized to the NRCC data, ie: $\% \text{ DIFF} = ((\text{NASA Value} - \text{NRCC Value}) / \text{NRCC Value}) \times 100$. The differences only, and not the actual values, are tabulated in Table 6.

In the following discussion, plots for only engine SN 615037 will be presented, as it was this engine that was previously run under simulated SLS conditions in two altitude facilities. On each graph, results from two NRCC test runs are plotted, run 22 with the NRCC nose bullet, and run 24 with the NASA nose bullet. Run 24 will be the basis of comparison, however differences between the two runs will be discussed. Both tests were conducted reasonably close to standard day temperature conditions, between 289 and 293 K. The test range extended from the point where the bleed valve was just closed to military power.

On each graph, data points obtained from the NASA Lewis Research Centre and the Arnold Engineering Development Centre (AEDC) were plotted and identified with appropriate symbols. NASA data (test point NASA (3)), were generated with an inlet temperature (T2AV) of 288 K, and an inlet pressure (P2AV) of 82.7 kPa. AEDC data were available for T2AV of 288 K and P2AV of 82.7 and 101.3 kPa. They are identified as AEDC (3) and AEDC (11) respectively. The data with P2AV of 82.7 kPa were corrected to standard day conditions.

For security reasons, actual values along the vertical axis are not shown, but the relative magnitude of the scale is indicated by a 1% band near the "target value" of 8900 rpm $NH/\sqrt{\theta}$ (91.75%).

The discussion of data is grouped into three sections; (1) engine health and repeatability, (2) engine performance, and (3) exhaust nozzle coefficients.

6.3.1 Engine Health and Repeatability

For a twin-spool fixed nozzle turbojet with no variable geometry, a good indicator of engine health is the ratio of spool speeds as a function of either spool speed. Other indicators are compressor performance, and overall engine pumping capacity. A comparison of speed ratio (Figure 10) indicates that between AEDC and NRCC there is little change (0.1%), but between NASA and NRCC there is a shift in the order of 0.5%. To ascertain whether this shift is due to Reynolds number effects or compressor degradation, it is necessary to look at the other compressor parameters. Inspecting plots of pressure and temperature ratios ($P3/P2$ and $T3/T2$) vs. high pressure compressor speed ($NH/\sqrt{\theta}$), and the compressor pumping characteristics, $P3/P2$ vs. $T3/T2$ (Figures 11, 12, and 13), close agreement exists between NRCC (run 24) and AEDC (run 11) data, but a distinct difference can be noted between NRCC and NASA data. The differences (tabulated in Table 6) are easily seen on the $P3/P2$ vs. $T3/T2$ plot (indicative of compressor efficiency) which shows good agreement

with AEDC (11), 0.2%, but a -3.0% difference with NASA (3). Plotting the same data as compressor efficiency (E_c) on a more expanded scale (Figure 14), good agreement is found between NRCC and AEDC, -0.2%, but a -1.8% difference exists with NASA. Since NASA ran the engine at a P2AV of 82.7 kPa and corrected the results to 101.3 kPa, the observed differences show that this particular compressor may be sensitive to Reynolds number. This contention is supported by the AEDC (3) data, generated at the same Reynolds number as NASA (3), in which the same E_c was obtained (Figure 14). This shift, however, appears to be rather large for this small change in Reynolds number. The difference in E_c between NRCC and NASA cannot be attributed to compressor degradation, as E_c would not be expected to increase with time; the AEDC (11) and the NRCC data were generated after NASA (3) and AEDC (3).

At first glance, an effect of the NRCC nose cone (run 22) on compressor delivery pressure appears to exist. However, not noticed during the test, the P3 transducer behaved erratically, as evidenced by the E_c and P3/P2 plots (Figures 14 and 11). Hence, all parameters that contain P3 data in run 22 are considered unreliable.

In summary, based on the change in efficiency (E_c) as a function of inlet pressure (P2AV), the compressor of engine SN615Q37 appears to be sensitive to Reynolds number.

In considering overall engine health, it is common to look at exhaust nozzle inlet conditions, Station 7, referenced to compressor entry, Station 2. Plotting P7/P2 (Engine Pressure Ratio, EPR) and T7/T2 (Engine Temperature Ratio, ETR) against $NH/\sqrt{\theta}$ points to a shift in engine operation between NRCC and both AEDC and NASA. To remove speed dependence from the data, a measure of engine pumping capacity can be obtained by plotting the engine pressure ratio against the temperature ratio (Figure 17). If the engine is thought of as a pump wherein a given pressure rise (P7/P2) requires a specified temperature rise (T7/T2), this plot can be interpreted as a measure of overall engine

efficiency. If the measurements were true, the data would indicate that less energy ($T7/T2$) was required to generate a given pressure rise ($P7/P2$) at NRCC than at either NASA or AEDC (-1.5 and -1.2%, respectively). These differences do, however, fall within the measurement uncertainty. Not included in the error assessment is the definition of $P2AV$. Inlet pressure takes no account of the boundary layer, which is different for each installation. Consequently, even if numerically the same, true inlet pressure may vary.

In addition to pressure errors, even greater errors exist in establishing a nozzle entry temperature ($T7AV$). Each facility had various numbers of unserviceable and intermittent $T7$ probes. Since there was no consistent method of synthesizing replacement values in a highly non-uniform temperature field, the calculated $T7AV$ could be quite different for the same gas temperature profile. A better measure of energy input is fuel flow, an inherently more accurate signal. Plotting EPR against fuel flow (Figure 18) shows a different trend than the EPR - ETR plot. Now more energy, rather than less, was required for a given EPR. Consequently, specific fuel consumption (SFC) based on NRCC data is expected to be higher; the SFC vs. FN plot will be shown in the performance section. EPR differences between NRC data and NASA data is -0.1%, and AEDC data 0.8% for a given fuel flow.

6.3.2 Engine Performance

The three performance parameters of interest were airflow, fuel flow, and thrust. As for engine health, the agreed upon basis for comparison was high pressure compressor speed ($NH/\sqrt{\theta}$). Facility airflow ($WA1\sqrt{\theta}/\delta$), facility fuel flow ($WF/(\delta\sqrt{\theta})$), and net thrust (FN/δ) were plotted against $NH/\sqrt{\theta}$ (Figures 19, 20, and 21). Generally speaking, the engine behaves "smaller" at NRCC than at NASA and AEDC. Airflow, fuel flow, and thrust are lower than NASA by 2.5%, 3.4%, and 3.2% respectively, and against AEDC by 1.4%, 1.6%, and 3.2%. It is evident that there are inconsistencies in either the NASA or AEDC data, as both thrust and fuel flow should follow airflow; the actual rela-

tionship will be shown later. Even though AEDC and NASA differ on airflow and fuel flow, they both manage to arrive at the same value of thrust.

The low pressure compressor determines the swallowing capacity, or airflow, of the engine. Since thrust and fuel flow are functions of airflow, the three performance parameters were plotted against low rotor speed ($NL/\sqrt{\theta}$). Comparing NRCC values of airflow, fuel flow, and thrust with NASA, the differences became smaller, 1.7%, 1.8%, and 1.5% respectively (Figures 22, 23, and 24). Against AEDC data, the differences are only slightly less, 1.3%, 1.4%, and 3.0% for airflow, fuel flow, and thrust. These trends are not unexpected, as differences were observed in NL/NH between NRCC and NASA, but only a small difference between NRCC and AEDC. Again the inconsistency in the NASA-AEDC thrust data is evident.

Using measured airflow as the base rather than $NL/\sqrt{\theta}$, the thrust and fuel flow difference between NRCC and NASA is -2.3% and -2.0% and between NRCC and AEDC 0.2% and -1.4% (Figures 25 and 26). If the uncertainty band of 0.77% on airflow is applied, the data could be said to agree.

Another way of presenting fuel flow and thrust is to divide fuel flow by thrust to obtain specific fuel consumption (SFC). SFC was plotted against $NH/\sqrt{\theta}$ (Figure 27), or what is generally more common, against FN/δ (Figure 28). The NASA and NRCC data agree remarkably well, +0.2% and +0.3%, using $NH/\sqrt{\theta}$ and FN/δ as a base, whereas AEDC show a lower SFC, -1.7% and -1.6% respectively. If the NRCC data were taken as correct, the AEDC data indicates that their value of net thrust is too high. To remove Reynold's number effects, NASA (3) and AEDC (3) data were directly compared, as the test points were generated at the same inlet temperature and pressure. Figure 28 shows that for the same Reynold's number, AEDC and NASA disagree by 1.4% in SFC.

In Figures 27 and 28, the NRCC data for both nose cones don't collapse on a single curve (runs 22 and 24). Data

scatter in run 22 is more pronounced than in run 24, suggesting that the engine may not have been "heat soaked" for a sufficient period of time before data gathering was initiated. Keeping in mind the error band of $\pm 0.52\%$ on SFC, $\pm 0.23\%$ on $NH/\sqrt{\theta}$, and engine repeatability, the results from these two runs could be considered the same.

6.3.3 Exhaust Nozzle Coefficients

Another method of data comparison, that can also provide relative checks on airflow and thrust, is to examine exhaust nozzle performance. Based on temperature and pressure measurements at the nozzle entry plane, Station 8, and knowing the nozzle area, A_8 , ideal values of airflow and gross thrust were calculated. Facility measured values of airflow (WA_1) and thrust (FG) were then divided by the corresponding Station 8 ideal values to yield flow (CD_8) and thrust (CG_8) coefficients.

$$CD_8 = WA_1/WA_{I8}$$

$$CG_8 = FG/FG_{I8}$$

CD_8 = flow coefficient

CG_8 = thrust coefficient

WA_1 = facility airflow, Station 1

FG = measured gross thrust

WA_{I8} = ideal airflow, Station 8 = $f(\gamma, R, P_{7AV}, T_{7AV}, A_8, P_{amb})$
(fuel flow is factored out)

FG_{I8} = ideal nozzle gross thrust = $f(\gamma, P_{7AV}, P_{amb}, A_8)$

A_8 = nozzle area corrected for thermal expansion

The flow and thrust coefficients are plotted as functions of nozzle pressure ratio, P_{7AV}/P_{amb} (Figures 29 and 30). For CD_8 , a larger difference exists between NRCC and NASA (2.4%) than between NRCC and AEDC (0.8%). Reasons for the differences are attributed both to the uncertainty of T_7 measurements, and in the facility airflow (WA_1).

The thrust coefficients (CG8) are more tightly grouped owing in large part to the smaller uncertainty in pressure measurements (P7AV). The differences between NRCC and NASA, and NRCC and AEDC are 0.6% and 0.2% respectively.

Two other well known non-dimensional thrust groups (Ref. 1) are used for evaluating thrust. They are the so-called " $W\sqrt{T}$ " and "AP" options. The " $W\sqrt{T}$ " method divides gross thrust, FG, by engine inlet airflow, WA1, and nozzle entry temperature, T7AV. The second option divides gross thrust by nozzle area, A8, and either nozzle entry pressure, P7AV, or the local ambient pressure, Pamb.

$$\frac{FG}{WA1 \cdot \sqrt{T7AV}} \quad \text{and} \quad \frac{FG}{A8 \cdot P7AV}$$

The "AP" option, also known as the FAP function, contains the smallest measurement uncertainty, especially for a fixed nozzle engine. The " $W\sqrt{T}$ " option, because of the uncertainty in T7AV, tends to be less accurate, unless it is calculated from fuel flow measurements. The results bear this out, as $FG/(WA1\sqrt{T7})$ vs. $P7/Pamb$ (Figure 31) shows differences of -1.6% and -0.4% between NRCC and NASA, and NRCC and AEDC. The $FG/(A8 \cdot P7)$ vs. $P7/Pamb$ plot (Figure 32) shows near perfect agreement with both NASA and AEDC (-0.2% and -0.1%). This latter method is the one preferred by General Electric to correlate their F404 facilities. Remembering that one of the objectives was to compare facility measurement methods, the FAP function shows the least sensitivity to engine performance changes in the gas generator.

7.0 CONCLUSIONS

1) Detailed methods have been presented to assess turbojet engine performance in an enclosed ground-level test bed.

2) Engine airflow in the annular inlet was calculated based on the integration of elemental airflow for each ring sector centered around individual pitot or boundary layer probes.

Static pressure was measured on the inner and outer wall of the annulus and if there was a gradient, it was assumed to vary linearly across the passage. Local temperature was measured with stagnation type thermocouples.

3) Fuel flow was measured with two turbine flowmeters placed in series. Individual values for fuel flow were calculated then averaged to yield a single result. Corrections were made for viscosity, temperature, and the non-linear characteristics of each meter.

4) A rigorous momentum accounting method was used to deduce the gross thrust of a turbojet engine operating in an enclosed facility. The scale force accounted for approximately 99% of the thrust, inlet momentum for around 1%, and several other minor positive and negative terms that cancelled themselves out. The relative magnitude of these terms will be maintained if the approach velocity is kept down below 7 m/s, and secondary airflow is equal to or less than the engine airflow. The ratio of exhaust nozzle to collector diameter is the primary means of controlling secondary airflow. The distance between the tailpipe and the collector strongly influences the static pressure depression in the plane of the exhaust nozzle.

5) A rigorous error analysis determined that airflow, gross thrust, and specific fuel consumption could be measured to accuracies of $\pm 0.77\%$, $\pm 0.31\%$, and $\pm 0.52\%$ respectively.

6) Due to apparent changes in engine performance from facility to facility, engine performance as a function of spool speed, HP or LP, is not a good method to compare data.

7) Facility correlation, especially thrust, is best achieved using the method employing non-dimensional exhaust nozzle performance plotted as a function of nozzle pressure ratio. The inherently most accurate nozzle thrust coefficient is the so-called "FAP" function, which yielded agreement better than 0.2% amongst three facilities.

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- 10) A Simulation of a Jet Engine Test Cell, NRCC LTR-ENG-110, Capt. R.R. Hastings, June 1983.

TABLE 1 FUEL ANALYSES

FUEL TYPE: JET A-1

ENGINE S/N	RUN NO.	SAMPLE NO.	SAMPLE DATE	HEAT OF COMBUST. (LHV) KJ/Kg	SULPHUR CONTENT % BY MASS	API GRAVITY 60%	S.G. 60/60°F	KINEMATIC VISCOSITY CST			
								-17.8°C	0°C	25°C	40°C
607594	1-12										
607594	13-19	FLO - 82190	3 Aug. 82	43212	0.009	46.4	0.7954	2.87	1.96	1.29	1.05
615037	20-25	FLO - 82203	18 Aug. 82	43196	0.04	45.7	0.7985	3.06	2.06	-	1.09
615037	26-28	FLO - 82229	8 Sep. 82	43275	0.003	44.6	0.8035	3.39	2.25	-	-

TABLE 2

SEA-LEVEL FACILITIES-BLEED CLOSED SETTINGSTEST POINTS - HIGH RANGE

SPEED SETTING	HIGH COMPRESSOR SPEED CALCULATION	HIGH COMPRESSOR SPEED, NH RPM
N1	$N_{MIL} \quad (1)$	
N2	$\frac{(N1+N3)}{2}$	
N3	$N_{BVC} + 30$	
N4	$\frac{(N2+N3)}{2}$	
N5	$\frac{(N1+N2)}{2}$	
N6	$\frac{(N2+N4)}{2}$	
N7	$\frac{(N3+N4)}{2}$	
N8	$\frac{(N2+N5)}{2}$	
N9	$\frac{(N1+N5)}{2}$	
N_{BVC}		

- (1) Military power speed, defined by power lever angle position during the initial trim to a specified thrust level, varies as a function of inlet temperature, thus the actual speed must be determined at run time.

TABLE 2 (Cont'd)

SEA-LEVEL FACILITIES-BLEED OPEN SETTINGSTEST POINTS - LOW RANGE

TEST POINT	SPEED CALCULATIONS	HP SPEED (N ₂) RPM
N10	NBVC-30	
N11	$\frac{N10+N12}{2}$	
N12	NIDLE	
N13	$\frac{(N11+N12)}{2}$	
N14	$\frac{(N10+N11)}{2}$	
N15	$\frac{(N11+N13)}{2}$	
N16	$\frac{(N12+N13)}{2}$	
N17	$\frac{(N11+N14)}{2}$	
N18	$\frac{(N10+N14)}{2}$	
NBVC:		

TABLE 3A: PRESSURE MEASUREMENT SYSTEM DESCRIPTION

Parameter Designation	Type Measuring Device	Type Recording Device	Applicable Calibration Standard Laboratory
Station 1.0 (NRC METERING)			
Total Pressures Except Radial B.L. Probes A, B, M, N	Capacitance type Transducer 0 to 10 Torr (0 to 1.3 kPa)	Automatic Multiple Pressure Scanning Onto Sequential Sampling Analog Voltage To Digital Converter With Magnetic Storage	Betz Micromanometer N/A
Static Pressures And Radial B.L. Total Pressure Probes A, B, M, N.	Capacitance Type Transducer 0 to 100" H ₂ O (0 to 25 kPa)	"	Texas Instruments Pressure Calibrator NBS-P-1602-3118
Station 2.0 (ENGINE INLET)			
Total Pressures and Bullet B.L. Probes D, E; Bellmouth B.L. Probes A to E	Capacitance Type Transducer 0 to 10 Torr (0 to 1.3 kPa)	"	Betz Micromanometer N/A
Static Pressures and Bullet B.L. Probes A, B, C; Bellmouth B.L. Probes F to H	Capacitance Type Transducer 0 to 100" H ₂ O (0 to 25 kPa)	"	Texas Instruments Pressure Calibrator NBS-P-1602-3118
Station 1.3, 1.4, 3.0			
Total and Static Pressures	Capacitance Type Transducer 0 to 200 psig (0 to 1380 kPa)	"	Texas Instruments Pressure Calibrator NBS-P-1602-3118

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TABLE 3A: PRESSURE MEASUREMENT SYSTEM DESCRIPTION (Cont'd)

Parameter Designation	Type Measuring Device	Type Recording Device	Applicable Calibration Laboratory	Standard National
<u>Station 5.0, 7.0</u>				
Total and Static Pressures	Capacitance Type Transducer 0 to 100" Hg (0 to 340 kPa)	Automatic Multiple Pressure Scanning Onto Sequential Sampling Analog Voltage Digital Converter With Magnetic Storage	Texas Instruments Pressure Calibrator	NBS-P-1602-3118
<u>Station 0.4</u>				
Static Pressure Probes 08,17,26,35	Capacitance Type Transducer 0 to 10 Torr (0 to 1.3 kPa)	"	Betz Micromanometer	N/A
<u>Station 0.5</u>				
Static Pressure Probes 08,17,26,35	Capacitance Type Transducer 0 to 10 Torr (0 to 1.3 kPa)	"	Betz Micromanometer	N/A
<u>Engine Carcass</u>				
Static Pressures Probes Near Stations 1.0,1.4.2.0,3.0	Capacitance Type Transducer 0 to 10 Torr (0 to 1.3 kPa)	"	Betz Micromanometer	N/A

TABLE 3B: TEMPERATURE MEASUREMENT SYSTEM DESCRIPTION

Parameter Designation	Type Measuring Device	Type Recording Device	Applicable Calibration Standard
Facility Fuel Inlet And Cell Temperature	Copper-Constantan Thermocouple	Automatic Scanning Onto Sequential Sampling Millivolt To Digital Converter with Magnetic Storage	Fluke DC Voltage Calibrator And Millivolt Substitution Into Temperature vs Millivolt Table. NBS Report 213385 NBS Circular 561 and NBS Report 9712
All Other Static and Total Temperatures	Chromel-Alumel Thermocouple	"	ASTM Standard E230-72 Based on IPIS-68

TABLE 3C: MISCELLANEOUS PARAMETER MEASUREMENT SYSTEM DESCRIPTION

Parameter Designation	Type Measuring Device	Type Recording Device	Applicable Calibration Standard
Scale Force	Dual Bridge Shear Web-Type Load Cell	Automatic Scanning Onto Sequential Millivolt to Digital Converter with Magnetic Storage	Dual Bridge Shear Web Type Working Standard Calibrated Against A Deadweight Standard Force Machine Canadian National Std Held By NRCC, Div. of Physics
Fuel Flow	Two Turbine-Type Flowmeters in Series	Automatic Scanning Onto Sequential Frequency to Digital Converter with Magnetic Storage	Microtrak Calibrator at Flow Technology Inc. NBS42634
Shaft Rotational Speeds	Toothed Wheels on Engine Gear-boxes	"	Digital Equipment M405, 1 MHz Clock
Air Flow	Total, Static and Boundary Layer Pressure rakes	Automatic Multiple Pressure Scanning onto Sequential Sampling Volt to Digital Converter with Magnetic Storage	Applicable Calibrations Pressure, Temperature and Area Measurements

TABLE 4 : ESTIMATED MEASUREMENT UNCERTAINTIES
All uncertainty estimates are based on thirty or more degrees of freedom

PARAMETER	NOMINAL VALUE OR RANGE	PRECISION INDEX (±)		BIAS LIMIT (±)		UNCERTAINTY (±)	
		VALUE	PERCENT	VALUE	PERCENT	VALUE	PERCENT
Scale force	5857 N 32525 N	12.40	0.21	26.74	0.46	51.5	0.88
		28.32	0.087	39.39	0.12	96.0	0.29
Pressure (0 to 1380 kPa)	345 kPa 927 kPa	0.252	0.073	0.495	0.14	0.999	0.29
		0.509	0.055	1.12	0.12	2.14	0.23
Pressure (0 to 340 kPa)	114 kPa 189 kPa	0.074	0.065	0.150	0.13	0.298	0.26
		0.105	0.056	0.227	0.12	0.437	0.23
Pressure (0 to 25 kPa)	1.5 kPa 6.2 kPa	0.013	0.89	0.015	1.0	0.042	2.78
		0.013	0.22	0.017	0.27	0.044	0.71
Pressure (0 to 1300 Pa)	30 Pa 60 Pa	0.45	1.5	0.34	1.1	1.24	4.1
		0.45	0.75	0.34	0.57	1.24	2.1
Temperature (Cu-Con)	290° K	0.085	0.829	0.48	0.17	0.65	0.22
Temperature (CR-AL)	290° K 640° K	0.085	0.029	1.14	0.39	1.3	0.45
		0.15	0.023	2.48	0.39	2.8	0.43
Fuel Flow	187 gm/sec 690 gm/sec	0.226	0.12	0.099	0.53	0.55	0.29
		0.824	0.12	0.36	0.052	2.01	0.29
Rotational Speed	4160 RPM (NL) 8085 RPM (NH)	0.245	0.0059	0.42	0.010	0.91	0.022
		0.470	0.0058	0.81	0.010	1.75	0.022

TABLE 5 : ESTIMATED PERFORMANCE PARAMETER UNCERTAINTIES

All uncertainty estimates are based on thirty or more degrees of freedom

PARAMETER		PRECISION INDEX(±)		BIAS LIMIT(±)		UNCERTAINTY(±)	
DESIGNATION	MEASUREMENT RANGE OR NOMINAL VALUE	PERCENT	MEASUREMENT	PERCENT	MEASUREMENT	PERCENT	MEASUREMENT
P2AV	99 kPa	0.010	0.010	0.015	0.015	0.035	0.035
PAMB (P05AV)	99 kPa	0.010	0.010	0.015	0.015	0.035	0.035
PS2AV	98.5 kPa	0.017	0.017	0.021	0.021	0.055	0.055
P3AV	345 kPa 927 kPa	0.073 0.055	0.25 0.51	0.14 0.12	0.50 1.12	0.29 0.23	1.00 2.14
P5AV, P7AV	115 kPa 190 kPa	0.061 0.055	0.070 0.105	0.13 0.12	0.15 0.23	0.25 0.23	0.29 0.44
P2/PAMB	1.0	0.014		0.021		0.050	0.0005
PS2AV/P2AV	1.01 1.06	0.020 0.021		0.026 0.027		0.066 0.069	0.00066 0.00073
P3/P2	3.5 9.3	0.074 0.056		0.14 0.12		0.29 0.23	0.010 0.021
P7/P2	1.1 1.9	0.062 0.056		0.13 0.12		0.25 0.23	0.0028 0.0044
P7/PAMB	1.1 1.9	0.062 0.057		0.13 0.12		0.25 0.24	0.0028 0.0046
FG	5857 N 32525 N	0.21 0.087	12.4 28.3	0.46 0.12	26.7 39.4	0.88 0.30	51.5 96.0
T2AV	290° K	0.029	0.085	0.39	1.14	0.45	1.3
T3AV	500° K	0.030	0.15	0.50	2.50	0.56	2.8

TABLE 5 : ESTIMATED PERFORMANCE PARAMETER UNCERTAINTIES (CONT'D)

PARAMETER		PRECISION INDEX (±)		BIAS LIMIT (±)		UNCERTAINTY (±)	
DESIGNATION	MEASUREMENT RANGE OR NOMINAL VALUE	PERCENT	MEASUREMENT	PERCENT	MEASUREMENT	PERCENT	MEASUREMENT
T7AV	650° K	0.023	0.15	0.38	2.5	0.43	2.8
T3/T2	1.5 2.0	0.035 0.039		0.47 0.58		0.54 0.66	0.008 0.010
T7/T2	1.9 2.5	0.040 0.036		0.59 0.52		0.67 0.59	0.012 0.015
WFE	187 gm/sec 690 gm/sec	0.086 0.084	0.16 0.58	0.069 0.044	0.13 0.30	0.24 0.21	0.45 1.45
WF/δ/θ (WFR)	189 gm/sec 704 gm/sec	0.1 0.1	0.19 0.70	0.21 0.20	0.4 1.4	0.39 0.39	0.74 2.6
NL/√θ (NLR)	4160 RPM	0.016	0.67	0.20	8.2	0.23	9.6
NH/√θ (NHR)	2045 RPM	0.016	1.3	0.20	16	0.23	19
NL/NH	0.41 0.60	0.011 0.0070		0.018 0.012		0.039 0.026	0.00016 0.00016
WA	30 kg/s 63 kg/s	0.70 0.17	0.21 0.11	0.94 0.27	0.28 0.17	2.36 0.61	0.71 0.38
*CD2	0.96 0.96	1.0 0.25		1.3 0.38		3.4 0.87	0.032 0.008
M2	0.1 0.25	0.72 0.15		0.94 0.20		2.4 0.50	0.0024 0.0012

TABLE 5 : ESTIMATED PERFORMANCE PARAMETER UNCERTAINTIES (CONT'D)
 Test conditions: P2 = 101.325 kPa; T2 = 288° K; Ram ratio = 1.0; Target NH = 8900 RPM.
 All uncertainty estimates are based on thirty or more degrees of freedom

PARAMETER		PRECISION INDEX(±)		BIAS LIMIT(±)		UNCERTAINTY(±)	
DESIGNATION	MEASUREMENT RANGE OR NOMINAL VALUE	PERCENT	MEASUREMENT	PERCENT	MEASUREMENT	PERCENT	MEASUREMENT
Airflow (WAIR)	30 kg/s 63 kg/s	0.73 0.22	0.22 0.14	0.95 0.33	0.28 0.21	2.41 0.77	0.72 0.48
Net Thrust (FNR)	6 kN 33 kN	0.23 0.090	0.014 0.030	0.47 0.13	0.028 0.043	0.93 0.31	0.056 0.10
Specific Fuel consumption (SFCR)	34 gm/kN.s 22 gm/kN.s	0.27 0.14	0.091 0.031	0.55 0.24	0.19 0.053	1.10 0.52	0.37 0.11
FG A8.P7	212 713	0.22 0.11		0.47 0.17		0.91 0.39	1.90 2.80
FG WALVT7	8.3 kN 19.1 kg/K	0.73 0.20		1.06 0.34		2.50 0.73	0.21 0.14

TABLE 6 - TARGET VALUE COMPARISONS - 8900 RPM NH

ENGINE S/N	RUN NO.	NOSE CONE	CIT - K
615037	22	NRCC	291-293
615037	24	NASA	289-292

PARAMETER		(A) NASA - % DIFF.	(B) AEDC - % DIFF.
NL/NH	vs. $NH/\sqrt{\theta}$	0.46	0.10
P3/P2	vs. $NH/\sqrt{\theta}$	0.09	0.03
T3/T2	vs. $NH/\sqrt{\theta}$	0.97	-0.06
P3/P2	vs. T3/T2	-2.92	0.24
Ec	vs. $NH/\sqrt{\theta}$	-1.84	-0.20
P7/P2	vs. $NH/\sqrt{\theta}$	1.69	1.66
T7/T2	vs. $NH/\sqrt{\theta}$	1.83	1.61
P7/P2	vs. T7/T2	-1.47	-1.18
P7/P2	vs. $WF/(\delta\sqrt{\theta})$	-0.06	0.83
$WA1\sqrt{\theta}/\delta$	vs. $NH/\sqrt{\theta}$	2.49	1.42
$WF/(\delta\sqrt{\theta})$	vs. $NH/\sqrt{\theta}$	3.38	1.58
FN/ δ	vs. $NH/\sqrt{\theta}$	3.17	3.22
$WA1\sqrt{\theta}/\delta$	vs. $NL/\sqrt{\theta}$	1.73	1.30
$WF/(\delta\sqrt{\theta})$	vs. $NL/\sqrt{\theta}$	1.75	-1.39
FN/ δ	vs. $NL/\sqrt{\theta}$	1.50	3.01
FN/ δ	vs. $WA1\sqrt{\theta}/\delta$	-2.28	0.16
$WF/(\delta\sqrt{\theta})$	vs. $WA1\sqrt{\theta}/\delta$	-2.00	-1.35
SFC/ $\sqrt{\theta}$	vs. $NH/\sqrt{\theta}$	0.17	-1.71
SFC/ $\sqrt{\theta}$	vs. FN/ δ	0.25	-1.59
CD8	vs. P7/Pamb	2.41	0.80
CG8	vs. P7/Pamb	0.56	0.21
FG/($WA1\sqrt{T7}$)	vs. P7/Pamb	-1.57	-0.40
FG/(A8 * P7)	vs. P7/Pamb	-0.15	-0.10

Note: (A) For data comparison, NASA values were obtained from test point No. 3 and AEDC test point No. 11. All data were corrected to standard day conditions and compared at the target value of 8900 rpm $NH/\sqrt{\theta}$

(B) The differences were normalized by using the NRCC values as the base i.e. $\text{Diff} = \frac{\text{NASA value} - \text{NRCC value}}{\text{NRCC value}}$

(C) Run 24 with the NASA nose cone was used as the NRCC value.

FIGURE 1 ENGINE TEST CELL ARRANGEMENTS

FIGURE 2 NUMBER 5 TEST CELL - ENGINE LABORATORY - NRCC

FIGURE 3 SCHEMATIC OF J57 TURBOJET ENGINE ON TEST STAND

1. Compressor section
2. Combustion chamber, turbine, and exhaust section
3. Accessory section

FIGURE 4A PRATT AND WHITNEY J57-19-W TURBOJET ENGINE

FIGURE 4B EXPLODED VIEW OF J57-19-W TURBOJET ENGINE

- A. EPR PROBE (INLET PRESSURE)
- B. LOW PRESSURE COMPRESSOR
- C. OIL SUPPLY TANK
- D. HIGH PRESSURE COMPRESSOR
- E. BURNER CANS
- F. FIRST STAGE TURBINE
- G. SECOND AND THIRD STAGE TURBINE
- H. NOZZLE
- I. TAIL CONE
- J. EPR PROBE (EXHAUST PRESSURE)
- K. ACCESSORY DRIVE ELBOW
- L. ACCESSORY DRIVE HOUSING
- M. MODIFIED TAILPIPE AND NOZZLE ASSEMBLY

FIGURE 4c J57-19-W ENGINE SCHEMATIC

FIGURE 5 ENGINE INSTALLATION IN NASA-SUPPLIED TEST STAND

FIGURE 6 BOM NOZZLE AND TAILPIPE WITH CONVERGENT NOZZLE

FIGURE 7 AIR-INFLOW MEASUREMENT CONFIGURATION

PSW3

PSW4

PSW1

PSW2

STATION NO.

DESCRIPTION

0.0	UPSTREAM OF BELLMOUTH
1.0	NRCC AIRFLOW STATION
2.0	ENGINE OR LPC INLET
1.3	LPC BLEED ANNULUS
1.4	LPC BLEED PORT
3.0	COMBUSTOR INLET
3.1	COMBUSTOR DIFFUSER EXIT
5.0	LPT EXIT
7.0	EXHAUST NOZZLE INLET
0.4, 0.5, 0.8	EXHAUST NOZZLE (EXTERNAL)
PSW1, PSW4	WALL STATIC TAPS

FIGURE 8 INSTRUMENTATION STATION LOCATIONS

FIGURE 8 (CONT'D) AERODYNAMIC PRESSURE INSTRUMENTATION AT STATION 1

FIGURE 8 (CONT'D) INSTRUMENTATION AT STATION 2.0

FIGURE 8 (CONT'D) INSTRUMENTATION AT STATIONS 1.3, 1.4, AND 3.0

FIGURE 8 (CONT'D) INSTRUMENTATION AT STATIONS 3.1 AND 5.0

FIGURE 8 (CONT'D) INSTRUMENTATION AT STATION 7.0

FIGURE 8 (CONT'D) INSTRUMENTATION AT STATIONS 0.4, 0.5, AND 0.8

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JST-19-W TURBOJET
S/N 615037

$\frac{N_L}{N_H}$ VS $\frac{N_H}{V\theta}$

LEGEND

- + NRCC (22)
- NRCC (24)
- NASA (3)
- AEDC (3)
- △ AEDC (1) SL

1% OF READING

NRCC UNCERTAINTY ESTIMATES

$\frac{N_H}{V\theta} : \pm 0.23\%$

$\frac{N_L}{N_H} : \pm 0.03\%$

84	85	86	87	88	89	90	91	92	93	94	95	96	97	98	%	$\frac{N_H}{V\theta}$
VS CORRECTED HIGH SPOOL SPEED																

FIG. 10

15/7/83

J57-19-W TURBOJET

S/N 615037

$\frac{P_3}{P_2}$ VS $\frac{N_H}{\sqrt{\theta}}$

LEGEND

- + NRCC (22)
- NRCC (24)
- NASA (3)
- AEDC (5)
- △ AEDC (11) SL

1% OF READING

NRCC UNCERTAINTY ESTIMATES

$$\frac{N_H}{\sqrt{\theta}}: \pm 0.23\%$$

$$\frac{P_3}{P_2}: \pm 0.23\%$$

84	85	86	87	88	89	90	91	92	93	94	95	96	97	98	%	$\frac{N_H}{\sqrt{\theta}}$
COMPRESSION PRESSURE RATIO VS CORRECTED HIGH SPEED																

FIG. 11

J57-19-W TURBOJET

S/N 615037

$\frac{T_3}{T_2}$ vs $\frac{N_4}{\sqrt{\theta}}$

LEGEND

- + NRCC (22)
- NRCC (24)
- NASA (3)
- AEDC (3)
- △ AEDC (11) SL

10% OF READING

NRCC UNCERTAINTY ESTIMATES

$$\frac{N_4}{\sqrt{\theta}} : \pm 0.23\%$$

$$\frac{T_3}{T_2} : \pm 0.66\%$$

FIG. 12 COMPRESSOR TEMPERATURE RATIO VS CORRECTED HIGH SPOOL SPEED $\frac{N_4}{\sqrt{\theta}}$

J57-19-W TURBOJET
S/N 615037

E_c vs $\frac{N_H}{\sqrt{\theta}}$

LEGEND

- + NRCC (22)
- NRCC (24)
- NASA (3)
- AEDC (3)
- △ AEDC (11) SL

1% OF READING

NECC UNCERTAINTY ESTIMATES

$\frac{N_H}{\sqrt{\theta}} : \pm 0.23\%$

84 85 86 87 88 89 90 91 92 93 94 95 96 97 98 99 % $\frac{N_H}{\sqrt{\theta}}$

FIG. 14 COMPRESSOR EFFICIENCY VS CORRECTED HIGH SPOOL SPEED

15/7/83

$\frac{P_1}{P_2}$

J57-19-W TURBOJET
S/N 615037

$\frac{P_1}{P_2}$ vs $\frac{N_H}{V_0}$

- LEGEND
- + NRCC (22)
 - NRCC (24)
 - NASA (3)
 - AEDC (3)
 - △ AEDC (1) SL

1% OF READING

NRCC UNCERTAINTY ESTIMATES

$\frac{N_H}{V_0} : \pm 0.23\%$

$\frac{P_1}{P_2} : \pm 0.23\%$

15/7/83

FIG. 15 ENGINE PRESSURE RATIO VS CORRECTED HIGH SPOOL SPEED $\frac{N_H}{V_0}$

J57-19-W TURBOJET

S/N 615037

$\frac{T_7}{T_2}$ vs $\frac{N_1}{V_0}$

LEGEND

- * NRCC (23)
- o NRCC (24)
- o NASA (3)
- o AEDC (3)
- Δ AEDC (1)

1% OF READING

NRCC UNCERTAINTY ESTIMATES

$\frac{N_1}{V_0}$: ±0.23%

$\frac{T_7}{T_2}$: ±0.59%

15/7/83

FIG. 16. ENGINE TEMPERATURE RATIO vs CORRECTED HIGH SPOOL SPEED $\frac{N_1}{V_0}$

$\frac{P_7}{P_2}$

17/8/83

FIG. 18

15/7/83

J57-19-W TURBOJET

S/N 615037

$\frac{W_{AV}}{\delta}$ VS $\frac{N_H}{\sqrt{\theta}}$

LEGEND

- + NRCC (22)
- NRCC (24)
- NASA (3)
- AEDC (3)
- △ AEDC (1) SL

1% OF READING

0+

0+

0+

0+

0+

0+

0

0+

NRCC UNCERTAINTY ESTIMATES

$$\frac{N_H}{\sqrt{\theta}} : \pm 0.23\%$$

$$\frac{W_{AV}}{\delta} : \pm 0.77\%$$

$\frac{N_H}{\sqrt{\theta}}$

%

98

97

96

95

94

93

92

91

90

89

88

87

86

85

FIG. 19 CORRECTED AIR INFLOW VS CORRECTED HIGH SPOOL SPEED

J57-19-W TURBOJET

S/N 615037

$\frac{W_{FR}}{\delta \sqrt{\theta}}$ VS $\frac{N_H}{\sqrt{\theta}}$

LEGEND

- + NRCC (22)
- NRCC (24)
- NASA (3)
- AEDC (3)
- △ AEDC (1) SL

1% OF READING

NRCC UNCERTAINTY ESTIMATES

$\frac{N_H}{\sqrt{\theta}} : \pm 0.23 \%$

$\frac{W_{FR}}{\delta \sqrt{\theta}} : \pm 0.39 \%$

84 85 86 87 88 89 90 91 92 93 94 95 96 97 98 % $\frac{N_H}{\sqrt{\theta}}$

FIG. 20 CORRECTED FUEL FLOW VS CORRECTED HIGH SPOOL SPEED

14/7/83

J57-19-W TURBOJET

S/N 615037

$\frac{F_N}{\delta}$ VS $\frac{N_H}{V_0}$

LEGEND

- + NRCC (22)
- NRCC (24)
- NASA (3)
- AEDC (3)
- △ AEDC (1) SL

NRCC UNCERTAINTY ESTIMATES

$\frac{N_H}{V_0} : \pm 0.23\%$

$\frac{F_N}{\delta} : \pm 0.31\%$

1% OF READING

FIG. 21 CORRECTED NET THRUST VS CORRECTED HIGH SPOOL SPEED

$\frac{N_H}{V_0}$

%

95 96 97 98

94 93 92 91 90 89 88 87 86 85

84

FIG. 22 CORRECTED AIR INFLOW AT STATION 1 VS CORRECTED LOW SPOOL SPEED

FIG. 24 CORRECTED NET THRUST VS CORRECTED LOW SPOOL SPEED

J57-19-W TURBOJET

S/N 615037

LEGEND

- + NRCC (23)
- NRCC (24)
- NASA (3)
- AEDC (3)
- △ AEDC (11) SL

$$\frac{F_N}{\delta} \text{ VS } \frac{W_A \sqrt{\theta}}{\delta}$$

1% OF READING

NRCC UNCERTAINTY ESTIMATES

$$\frac{F_N}{\delta} : \pm 0.31\%$$

$$\frac{W_A \sqrt{\theta}}{\delta} : \pm 0.77\%$$

50 52 54 56 58 60 62 64 66 68 70 72 74 76 78 80 $\frac{W_A \sqrt{\theta}}{\delta}$

FIG. 25 - CORRECTED NET THRUST VS CORRECTED AIR INFLOW AT STATION 1

14/7/83

J57-19-W TURBOJET

S/N 615037

$\frac{SFC}{\sqrt{\theta}}$ VS $\frac{N_H}{\sqrt{\theta}}$

LEGEND

- + NRCC (22)
- NRCC (24)
- NASA (3)
- AEDC (3)
- △ AEDC (4) SL

1% OF READING

NRCC UNCERTAINTY ESTIMATES

$\frac{N_H}{\sqrt{\theta}} : \pm 0.23\%$

$\frac{SFC}{\sqrt{\theta}} : \pm 0.52\%$

14/7/83

$\frac{N_H}{\sqrt{\theta}}$

84 85 86 87 88 89 90 91 92 93 94 95 96 97 98 99 %

FIG. 27 CORRECTED SPECIFIC FUEL CONSUMPTION VS CORRECTED HIGH SPOOL SPEED

157-19-W TURBOJET
S/N 615037

LEGEND

- + NRCC (22)
- NRCC (24)
- NASA (39)
- AEDC (3)
- △ AEDC (11) SL

$\frac{SFC}{\sqrt{\theta}}$ VS $\frac{F_N}{\delta}$

NRCC UNCERTAINTY ESTIMATES

$$\frac{F_N}{\delta} : \pm 0.31 \%$$

$$\frac{SFC}{\sqrt{\theta}} : \pm 0.52 \%$$

16 18 20 22 24 26 28 30 32 34 36 38 40 42 44 46 kN $\frac{F_N}{\delta}$

FIG. 28 CORRECTED SPECIFIC FUEL CONSUMPTION VS CORRECTED NET THRUST

W/7/83

157-19-W TURBOJET
S/N 615037

- LEGEND
- + NRCC (23)
 - NRCC (24)
 - NASA (3)
 - AEDC (3)
 - △ AEDC (1) SL

CDB VS $\frac{P_7}{P_{AMB}}$

1% OF READING

NOZZLE UNCERTAINTY ESTIMATES

$$\frac{P_7}{P_{AMB}} : \pm 0.24\%$$

4/7/83

FIG. 29 EXHAUST NOZZLE FLOW COEFFICIENT, CDB, VS NOZZLE PRESSURE RATIO, $\frac{P_7}{P_{AMB}}$

2 OF 2

OF / DE

22
OF / DE

83

4/7/83

J57-19-W TURBOJET
S/N 615037

CG8 VS $\frac{P_7}{P_{AMB}}$

- LEGEND
- + NRCC (22)
 - NRCC (24)
 - NASA (3)
 - AEDC (3)
 - △ AEDC (11) SL

10% OF READING

NRCC UNCERTAINTY ESTIMATES

$$\frac{P_7}{P_{AMB}} : \pm 0.24\%$$

FIG. 30 EXHAUST NOZZLE THRUST COEFFICIENT, CG8, VS NOZZLE PRESSURE RATIO $\frac{P_7}{P_{AMB}}$

84

$\frac{F_G}{W_A \sqrt{T_7}}$

J57-19-W TURBOJET

S/N 615037

LEGEND

- + NRCC (22)
- NRCC (24)
- NASA (3)
- AEDC (3)
- △ AEDC (4) SL

1% OF READING

$\frac{F_G}{W_A \sqrt{T_7}}$ VS $\frac{P_7}{P_{AMB}}$

NRCC UNCERTAINTY ESTIMATES

$$\frac{F_G}{W_A \sqrt{T_7}} : \pm 0.73\%$$

$$\frac{P_7}{P_{AMB}} : \pm 0.24\%$$

11/7/83

FIG. 31 CORRECTED SPECIFIC GROSS THRUST FUNCTION VS NOZZLE PRESSURE RATIO

85

FIG. 32 CORRECTED GROSS THRUST VS NOZZLE PRESSURE RATIO

An ideal airflow (WAI2) was found by averaging the pitot pressures, the static pressures, and the temperatures. Pitot pressures and temperatures were averaged arithmetically, since these probes were placed at centroids of equal cross-sectional areas. Inner and outer pressures were weight-averaged, favouring the values from the outer taps.

Hence

$$WAI2 = 1000 \cdot P2AV \cdot MI2 \cdot A2 \left[\left(\gamma/R \right) \left(1/T2AV \right) \left(PS2AV/P2AV \right) + \left(\left(\gamma+1 \right) / \gamma \right) + \left(1/2 \right) \right]$$

where

$$MI2 = \left[\left[2 / \left(\gamma - 1 \right) \right] \left[\left(P2AV / PS2AV \right) \left(\left(\gamma - 1 \right) / \gamma \right) - 1 \right] \right] + \left(1/2 \right)$$

Allowing for thermal expansion:

$$A2 = A2S \left[1 + \lambda (TMI2AV - 294) \right]^2$$

λ = Linear coefficient of expansion for stainless steel
(16.2×10^{-2})

A2S = Geometric reference flow area @ 294°K (0.53992 m^2)

The ideal airflow, WAI2, neglects boundary layers and will, therefore, give an artificially high value. Together with a flow coefficient, obtained by calibration against an accurate measuring device, an actual airflow can be calculated.

A real airflow was established at station 2 in a fashion similar to that of station 1. The NASA approach, however, was to average the total pressure values of all probes located at the same radius, and to apply that average to the whole ring. Static pressures were weighted averages of opposing outer and inner statics, in relation to the rings radial position. Temperatures for the ring flow areas were obtained either from the appropriate probe in the main stream or from the nearest probe in the boundary layer.

TABLE A-1: NRCC PRIMARY AND FACILITY INSTRUMENTATION DESCRIPTION.

<u>Measurement Type</u>	<u>Instrument Type</u>	<u>Range</u>	<u>Calibration</u>
1. Scale Force Working Standard	Interface Dual Bridge Load Cell: Model 1120-BF S/N 21513	0 to 44 kN	30 June 82
2. Pressure Standards	Betz Micromanometer: Type 2500 S/N 10420 Texas Instruments: Model 157-02 S/N 112 Pressure capsule S/N 7952 S/N 7951	0 to 2.5 kPa	August 77
3. Barometric Reference	Paroscientific Digiquartz: Model 2200-AS-002 S/N 13230	0 to 34 kPa 0 to 1380 kPa	January 82 January 82
4. Pressure Transducers	Datametrix Model 590 Barocel - SN 1038831 - SN 1038826 - SN 1038824 - SN 1038825 Kulite Dynamic XTL5-140-50 Special SN 716-1-15 (R4-91)	0 to 1380 kPa	February 82
5. Temperature References	Reference Plate: Kaye UTR-AS SN-NRC/EL1,2 Ice point Reference: Kay K150-8C SN 7655	0 to 1.3 kPa 0 to 25 kPa 0 to 340 kPa 0 to 1380 kPa 0 to 102 kPa 0 to 500 Hz	June 82 " " " " "
6. Time Reference	Oscillator Crystal: Digital Equipment M8033-YA	1 MHz	

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Table A-1 (cont'd): NRCC Primary and Facility Instrumentation Description

<u>Measurement Type</u>	<u>Instrument Type</u>	<u>Range</u>	<u>Calibration</u>
7. Fuel Flow	Turbine Flow Meter: Flow Technology - 1.9 cm S/N 1201004 - 2.5 cm S/N 1601892	10 to 1210 gm/sec 30 to 2900 gm/sec	September 81 "
8. Voltage	DC Voltmeter: Hewlett Packard HP3456A SN 2201A04613 Voltage Source: Fluke 343A SN 0870002 Power Supply: Anatek 0L12-14 SN 114 External Amplification: NEFF 126	0 to 11VDC 6 to 14 VDC	February 82 December 82 May 82
9. Overall Data Acquisition	Pressure Scanning: Scanivalve Model 4808GM Compact System Controller: Digital Equipment CSC11 SN ZC-H0658-00		June 83
10. Control and Computation	Digital computer: Digital Equipment PDP-11/23		

TABLE A-2

REFERENCE INSTRUMENTATION LIST

Station No.	Description	Parameter Name	Number	SI (English)	Instrument-Range SI (English)	Engine Health (1) Parameters
2.0	Static Pressure	PSZ	8	10.34 (1.5)	103.42 KPa (15 psia)	
2.0	High Response Static Pressure	PSZX	2	10.34 (1.5)	"	x
2.0	Total Pressure	PZ	46	10.34 (1.5)	"	x
2.0	Total Temperature	TZ	10	244.3 (-20)	310.9°K (100°F)	x
2.0	Skin Temperature	TMZA	2	"	"	x
1.3	Static Pressure	PSI3	1	34.47 (5.)	344.7 KPa (50 psia)	
1.4	Bleed Discharge Total Pressure	PB	2	34.47 (5.)	344.7 KPa (50 psia)	x
1.4	Bleed Discharge Total Temperature	TB	4	255.4 (0)	699.8°K (800°F)	x
3.0	Total Pressure	P3	6	206.8 (30.)	206.8 KPa (300 psia)	
3.0	Total Temperature	T3	6	255.4 (0.)	699.8°K (800°F)	
3.0	Static Pressure	PSI1	2	206.8 (30.)	206.8 KPa (300 psia)	x
3.1	Total Pressure	PSAV	1	34.47 (5.)	344.7 KPa (50 psia)	x
5.0	Total Temperature	T5	5	255.4 (0.)	1088.7°K (1500°F)	x
7.0	Static Pressure	PS7	4	34.47 (5.)	344.7 KPa (50 psia)	
7.0	Total Pressure	P7	36	34.47 (5.)	344.8 KPa (50 psia)	
7.0	Total Temperature	T7	36	255.4 (0.)	1088.7°K (1500°F)	
7.0	Skin Temperature	TM7	4	255.4 (0.)	1088.7°K (1500°F)	
7.0	Static Pressure	PS	12	10.34 (1.5)	103.42 KPa (15 psia)	x
0.4, 0.5 and 0.8 Oil System	Oil Breather Pressure	PM8	1	-69.95 (-10)	68.95 KPa (10 psia)	x
"	Main Oil Pressure	MOF	1	68.95 (10.)	689.5 KPa (100 psia)	x
"	Main Oil Temperature	MOI	1	255.4 (0.)	422°K (300°F)	x
"	Oil Temperature Into Oil cooler	TOI	1	255.4 (0.)	477.6°K (400°F)	x
"	Oil Temperature Out of Cooler	TOO	1	255.4 (50.)	477.6°K (400°F)	x
Fuel System	Fuel Pump Inlet Pressure	PI	1	68.95 (10.)	389.5 KPa (100 psia)	x
"	Fuel Pump Discharge Pressure	PD	1	344.7 (50.)	344.7 KPa (500 psia)	x
"	Fuel Pump Inlet Temperature	TFI	1	277.6 (40.)	310.9°K (100°F)	x
"	Engine Flowmeter Discharge Temperature	TFE	1	277.6 (40.)	477.6°K (400°F)	x
Engine Misc.	Rotor Speed	NI	2	0. (0.)	1047.2 Hz (10,000 rpm)	x
"	Power Lever-Angle	PLA	1	0.	100 deg	x
"	Inlet Case Vibration	VIC	1	0.	10 m/s	x
"	Diffuser Case Vibration	VDC	1	0.	10 m/s	x
"	Turbine Exhaust Case Vibration	VTC	1	0.	10 m/s	x
"	Accessory Drive Case Vibration	VAD	1	0.	10 m/s	x
"	Turbine Cooling Air Supply Pressure	PTCA	1	206.8 (30.)	206.8 KPa (300 psia)	x
"	Engine Fuel Flow	FF	1	206.8 (30.)	206.8 KPa (300 psia)	x

TABLE A-3

HAND RECORDED MEASUREMENTS

<u>Parameter</u>	<u>Symbol</u>	<u>Units</u>	<u>Readout</u>
Rotor Speed	NL,NH	RPM	Anadex Calculating Counter Model 1601R
Fuel Flow	WFM,WFE	lbm/h, Hz	Anadex Calculating Counter Model 1603R
Screen Temperature	TSCRN	°C	Fluke 2170A Digital Thermometer
Fuel Temperature	TWFM,TWFT	°C	Doric Trendicator Model 405A
EGT	TSAV	°C	Doric Thermocouple Indicator DS-520
Scale Force	FA, FB	Millivolts	Hewlett Packard 3456A Digital Voltmeter
Supply Voltage	VS	volts	Hewlett Packard 3456A Digital Voltmeter
Barometer	Test cell (P500AV)	psia	ParoScientific Digiquartz pressure Computer Series 700
Barometer	control room	psia	ParoScientific Digiquartz pressure Computer Series 700
Test Cell Static Depression	PSW1,PSW2 PSW3,PSW4	psia	Dwyer Mark XX Model 25 Manometer
Vibration	VIC,VDC VTC,VAD	mils	Consolidated Electrodynamics Corp. Vibration Meter Type 1-117
Main Oil Temp.	MOT	°C	Analogic Thermometer Model 2572
Oil Cooler Temp.	TCOOL	°C	Analogic Thermometer Model 2572
Oil Breather Press.	POIL	psig	US Gage 200 psi
Fuel Pressure	PLPF	psig	US Gage 60 psi
Cell Velocity	VCUP,VVANE	volts	Fluke Digital Multimeter 8024A
Dew Point Temp.	DP	°F	General Eastern Dew Point Hygrometer System 1100AP
Ambient Temp.	OAT	°F	General Eastern Dew Point Hygrometer System 1100AP
Stack Temperatures	TSTACK	°C	Analogic Thermometer Model 2572
Temp. Ref. Plate	TREF	°C	Analogic Thermometer Model 2572

APPENDIX B

AIRFLOW CALCULATION

One of the major objectives of the program was the accurate determination of engine air inflow. Since NRCC does not have an airflow measuring device for gas turbines, provision had to be made for deducing airflow by means of pressure and temperature measurements in the air stream, without the use of flow coefficients obtained from calibration against a reference nozzle. To form an airmeter, the engine inlet, as supplied by NASA, was modified by extending the duct and exchanging the nose cone for an NRCC-designed one, together with an extension spool, see Figure B1. This modification created a 35 inch (889 mm) long high-velocity annular intake, with two measuring stations in series. NRCC's probes were placed at station 1, while NASA's probes remained at the original plane, labelled station 2. Throughout the test program, the intake duct extension remained in place, but the NASA and NRCC nose cone and extension were interchanged.

The instrumentation layout of the NASA measuring station 2 is given in Figure 8. The five pitot probes of the four rakes at 0° , 90° , 180° and 270° were placed at centroids of equal-area annuli, which permitted simple averaging of readings. In addition, four boundary layer rakes, two on the outer wall and two on the nose cone extension, were installed. Four wall static taps each were placed on the inner and outer walls of the annular intake duct in line with the leading edge of the pitots. The steady-state instrumentation was completed with two 5-point temperature rakes. Similar to the pitot probes, the thermocouples were located at centroids of equal-area annuli.

NRCC's instrument package at plane 1 was very similar to NASA's at the downstream location, see Figure B2. In order to avoid effects of probe wakes, the NRCC pitot rakes were rotated 20° counter-clockwise from the NASA ones, looking downstream. The inner and outer boundary layer rakes were rotated about 90° and placed in-line, whereas the NASA rakes were offset by 22.5° . The NRCC station had no temperature probes; values from the NASA rakes at station 2 were used in all computations.

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As the additional probes at station 1 shed wakes that reached the compressor, the rakes at station 2 no longer measured the mean inlet total pressure. Additionally, the effective flow area at station 2 would no longer be the same as for the other test facilities. Several data points were gathered to establish a flow coefficient for station 1 using the outer wall statics as the reference measurement. The rakes were then removed along with the centrebody instrumentation struts, thus leaving a calibrated airmeter as long as the NRCC nose section was in place. As the NRCC nose could affect engine performance, data were gathered using both configurations. Station 1 could no longer be used as an airmeter, therefore airflow was deduced by calibrating station 2 against station 1.

Implicit in this airmeter, is the assumption that the static pressure field is constant in the measurement plane. No attempt was made to verify this assumption in plane 2, however, a crude static pressure survey in plane 1 and a numerical analysis of the NRCC flow entry region supported this assumption (Ref. 9).

Airflow Calculation at Station 1.

The annular gap of the cross-section in the plane of station 1 was accurately measured. The annulus was then divided into four equal circular ring sectors, such that each would have one of the four pitot rakes at its axis of symmetry. The ring sectors were further subdivided radially to obtain 17 small circular ring sectors, one for each pitot probe, including the boundary layer. Because of overlap with the boundary probes, the outermost pitot of each rake was not used. Also, equal spacing of boundary layer probes, and overlap between boundary layer and pitot rakes made it impossible to place each probe at the centroid of its respective circular ring sector. Generally, the main probes were at centroids of areas whereas boundary layer probes were located at near radii of their ring sectors.

The outer wall static tap nearest to the pitot rake was assigned to that ring sector. The inner wall taps were not

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used because in subsequent tests they were to become unavailable. For air temperature, the average of all temperatures measured by the rakes at station 2 was used in each calculation.

Airflow at plane 1 was obtained by the sum of flows through each ring sector of the four rake ring sectors, i.e. $4 \times 17 = 68$ areas:

$$WA_1 = \sum_{i=1}^{68} WA_{1S} \cdot (A_1/A_{1S})$$

$$WA_{1S} = \sum_{i=1}^{68} 1000 \cdot P_{1i} \cdot M_{1i} \cdot A_{1i} \left[\left(\frac{\gamma}{R} \right) \left(\frac{1}{T_{2AV}} \right) \left(\frac{P_{S1i}}{P_{1i}} \right) + \left(\frac{\gamma+1}{\gamma} \right) \right] + (1/2)$$

where

$$M_{1i} = \left[\left(\frac{2}{\gamma-1} \right) \left[\left(\frac{P_{1i}}{P_{S1i}} \right) + \left(\frac{\gamma-1}{\gamma} \right) - 1 \right] \right] + (1/2)$$

Allowing for thermal expansion

$$A_1 = A_{1S} [1 + \lambda (T_{M1AV} - 294)] + 2$$

- WA_1 = Engine air mass inflow at Station 1
- WA_{1S} = Engine Air mass inflow through reference area
- A_1 = Geometric flow area corrected for temperature expansion
- A_{1S} = Geometric reference flow area @ 294°K (0.54009M²)
- P_1 = Pitot pressure
- P_{S1} = Wall static pressure
- M_1 = Local Mach No.
- T_{2AV} = Average inlet air temperature
- T_{M1AV} = Average duct metal temperature
- λ = linear coefficient of expansion for aluminum (22.9 x 10⁻⁶)
- R = Gas constant for air 287.05 J/(Kg K)
- γ = Ratio of specific heats f(T_{2AV})

Airflow Calculation at Station 2

Calculations of airflow at station 2 followed the outline given in Reference 1.

An ideal airflow (WAI2) was found by averaging the pitot pressures, the static pressures, and the temperatures. Pitot pressures and temperatures were averaged arithmetically, since these probes were placed at centroids of equal cross-sectional areas. Inner and outer pressures were weight-averaged, favouring the values from the outer taps.

Hence

$$WAI2 = 1000 \cdot P2AV \cdot MI2 \cdot A2 \left[\left(\frac{\gamma}{R} \right) \left(\frac{1}{T2AV} \right) \left(\frac{PS2AV}{P2AV} \right) + \left(\frac{\gamma+1}{\gamma} \right) + \left(\frac{1}{2} \right) \right]$$

where

$$MI2 = \left[\frac{2}{(\gamma-1)} \right] \left[\left(\frac{P2AV}{PS2AV} \right) \left(\frac{\gamma-1}{\gamma} \right) - 1 \right] + \left(\frac{1}{2} \right)$$

Allowing for thermal expansion:

$$A2 = A2S [1 + \lambda (TM1AV - 294)]^2$$

λ = Linear coefficient of expansion for stainless steel
(16.2×10^{-2})

A2S = Geometric reference flow area @ 294°K (0.53992 m^2)

The ideal airflow, WAI2, neglects boundary layers and will, therefore, give an artificially high value. Together with a flow coefficient, obtained by calibration against an accurate measuring device, an actual airflow can be calculated.

A real airflow was established at station 2 in a fashion similar to that of station 1. The NASA approach, however, was to average the total pressure values of all probes located at the same radius, and to apply that average to the whole ring. Static pressures were weighted averages of opposing outer and inner statics, in relation to the rings radial position. Temperatures for the ring flow areas were obtained either from the appropriate probe in the main stream or from the nearest probe in the boundary layer.

Thus, real air flow at station 2 was obtained by
18

$$WA2 = (1/2) \sum_{n=2} WA2QA_n (AP2_j - AP2_k)$$

where

$$j = n-1$$

$$k = n+1$$

$$WA2QA_n = 1000 \cdot P2_n \cdot M2_n [(\gamma/R)(1/T2_n)(PS2_n/P2_n) + ((\gamma+1)/\gamma)]$$

$$M2_n = ([2/(\gamma-1)][(P2_n/PS2_n) + ((\gamma-1)/\gamma) - 1] + (1/2)) + (1/2)$$

$$PS2_n = C_n \cdot PS2AVOA + (1-C_n) PS2AVOB$$

where C_n are tabulated weight factors

Allowing for thermal expansion,

$$AP2_n = AP2Sn [1 + \lambda(TM1AV-294)]^2$$

PS2AVOA = Average static pressure on duct outer wall

PS2AVOB = Average static pressure on duct inner wall

Values of $P2_n$, C_n , $T2_n$, and $AP2Sn$ for each value of n are listed in the following table.

Table Of Pressure And Temperature Relations And Areas

n	P2n	Cn	T2n	AP2Sn
1	-	-	-	0.72712
2	PB02AVOH	0.0134	T2AVOD	0.71756
3	PB02AVOG	0.0334	T2AVOD	0.70336
4	PB02AVOF	0.0536	T2AVOD	0.68924
5	PB02AVOE	0.0736	T2AVOD	0.67533
6	PB02AVOD	0.0937	T2AVOD	0.66149
7	PB02AVQC	0.1137	T2AVOD	0.64787
8	PB02AVOB	0.1473	T2AVOD	0.62538
9	PB02AVOA	0.1874	T2AVOD	0.59897
10	P2AVOD	0.2403	T2AVOD	0.56510
11	P2AVOC	0.4203	T2AVOC	0.45705
12	P2AVOB	0.6232	T2AVOB	0.34903
13	P2AVOA	0.8608	T2AVOA	0.24097
14	PB12AVOE	0.8829	T2AVOA	0.21102
15	TB12AVOD	0.9097	T2AVOA	0.20588
16	PB12AVOC	0.9365	T2AVOA	0.20080
17	PB12AVOB	0.9772	T2AVOA	0.19579
18	PB12AVOA	0.9900	T2AVOA	0.19084
19	-	-	-	0.18720

AIR-INFLOW MEASUREMENT CONFIGURATION
PROVIDED BY N.A.S.A.

FIG. B-1

FIGURE B-2 AERODYNAMIC PRESSURE INSTRUMENTATION AT STATION I

APPENDIX C

FUEL FLOW CALCULATION

Fuel flow at the Engine Laboratory is measured by two or more turbine flowmeters, placed in series, with the necessary length of flow straightening (see list of instrumentation, Appendix A). Such meters are designed to measure volume flow (Ref. C-1), although it is mass flow which is used in engine performance analysis. Consequently, account must be taken of fuel specific gravity and fuel viscosity at the temperature of the fuel in the flowmeter. These data can then be used together with flowmeter calibrations, for fuel mass flow calculations.

1. Specific Gravity Correction

Given a specific gravity reading of a fuel sample at 15.5°C (60°F), normally called the 60/60 specific gravity, Figure C-1 can be used to calculate the specific gravity at the measured fuel temperature.

2. Viscosity Correction

Fuel viscosity is highly temperature dependent, with an increase in temperature causing a decrease in viscosity. While differing values of viscosity will not affect the accuracy of turbine flowmeters in the linear high-flow range, a potential inaccuracy exists in the low flow range. The calculation of the viscosity, from a calibration, will be outlined below, together with a numerical example.

For a given fuel sample, the NRCC Fuels and Lubricants Laboratory provided a five-point viscosity calibration, as a function of temperature. When plotted on a viscosity-temperature chart, obtained from ASTM (D 341-39), the points formed a straight line, see Figure C-2.

The equation for this line is taken from Reference C-2 and presented as follows:

$$\log \log Z = A - B \log T \quad (1)$$

where: Z = defined below

$\log$ = logarithm to base 10

ν = kinematic viscosity, cST (mm^2/S)

T = temperature, $^{\circ}\text{K}$ or $^{\circ}\text{R}$

A and B = constants

Kinematic viscosity can be written as:

$$\nu = [Z-0.7] - \exp(-0.7487 - 3.295 [Z-0.7] + 0.6119[Z-0.7]^2 - 0.3193[Z-0.7]^3) \quad (2)$$

In order to obtain the constants A and B , an expression for Z is given by:

$$Z = \nu + 0.7 + \exp(-1.47 - 1.84\nu - 0.51\nu^2) \quad (3)$$

With two calibration points of ν vs T , the constants A and B can be found by solving equation (1) as two simultaneous equations. Since the equation applies most accurately to viscosities below 2 cST, the two points should be chosen in that range.

The equations were found to give very good results for viscosities up to 4.6 cST (-30°F).

3. Turbine Flowmeter Calibration

Turbine flowmeters were calibrated by the manufacturer on a ballistic flow calibrator, by passing a given volume of fluid through the meter, at a constant rate, and counting the number of flowmeter pulses for a measured time period. Calibrations were carried out with various liquids, with viscosities ranging from 0.82 to 3.3 cST, to allow measurement of fuel flows over a temperature range of approximately -40 to 40°C .

The calibration involves the following measurements:

- D = displacement volume (litres or gallons),
- A = displacement time for volume D (seconds),
- B = time period (seconds),

C = flowmeter pulses during time period B,
 S = fluid reference specific gravity (water at 4°C)
 v = fluid kinematic viscosity (cST)

Thus we can write the mass flow as

$$M_f = \frac{D}{A} \left[\frac{\text{US gal}}{\text{s}} \right] 3600 \left[\frac{\text{s}}{\text{hr}} \right] 8.34517 \left[\frac{\text{lbm H}_2\text{O}}{\text{US gal}} \right] S \left[\frac{\text{lbm fuel}}{\text{lbm H}_2\text{O}} \right]$$

$$\text{or } M_f = 30042.61 \frac{SD}{A} \left[\frac{\text{lbm fuel}}{\text{hr}} \right] \text{ or } 3.785 \frac{SD}{A} \left[\frac{\text{kg fuel}}{\text{s}} \right]$$

The calibration of a flowmeter establishes a functional relationship between flowmeter pulse frequency and a calibration factor in pulses per mass flow unit, derived as:

$$K = \frac{C}{D} \left[\frac{\text{pulses}}{\text{US gal}} \right] \frac{1}{S} \left[\frac{\text{lbm H}_2\text{O}}{\text{lbm fuel}} \right] \frac{1}{8.34517} \left[\frac{\text{US gal}}{\text{lbm H}_2\text{O}} \right]$$

$$\text{then } K = 0.119830 \frac{C}{SD} \left[\frac{\text{pulses}}{\text{lbm fuel}} \right]$$

$$\text{or } 0.26418 \frac{C}{SD} \left[\frac{\text{pulses}}{\text{kg fuel}} \right]$$

A plot of K versus turbine flowmeter output frequency (pulses per second) divided by temperature corrected viscosity (see section 2 of this appendix) has been shown to yield a unique curve for a range of viscosities. Figures C3, C4 and C5 present the calibrations for the three facility meters.

The calibration curves shown on the plots were entered into the data reduction software in tabular form. Linear interpolation was used for determining a K value for a given turbine meter frequency and fuel temperature as read by the data acquisition system.

4. Fuel Flow Computation

Given the fuel flowmeter calibration of section 3, based on a reference specific gravity, the actual fuel flow was calculated as follows.

$$M_f = \frac{C}{B} \left[\frac{\text{pulses}}{s} \right] \frac{1}{K} \left[\frac{\text{lbm}}{\text{pulses}} \right] \frac{S}{S_{ref}} \left[\frac{\text{actual spec. gravity}}{\text{ref spec. gravity}} \right]$$

$$= \frac{C}{BK} \frac{S}{S_{ref}} \left[\frac{\text{lbm}}{s} \right] \text{ or } 0.45359 \frac{C}{BK} \frac{S}{S_{ref}} \left[\frac{\text{kg}}{s} \right]$$

References

- C-1. Zimmerman, R. Handbook for Turbine Flowmeter Systems,
Deery, D. Flow Technology, Inc. 4250 East
Broadway Road, Phoenix, Arizona, 85040,
August 1977.
- C-2. ASTM 1981 Annual Book of ASTM Standards,
Part 23, Petroleum Products and
Lubricants (1): D56-D1660, D341.

Fig. C-2 Viscosity vs. Temperature Chart of Jet A-1

FIGURE 1-4 MANUFACTURED-CALIBRATED CALIBRATION, 2" ELEMENT 8/11/1992

FIGURE C-5 MANUFACTURER-SUPPLIED CALIBRATION OF 1-1/2" FLOWMETER S/N 2401321

APPENDIX D

SCALE FORCE CALIBRATION

Scale force thrust was measured with a strain-gauge type double-bridge Interface load cell, placed in compression between the thrust bed and the thrust frame, see Figure D1. The thrust bed was suspended by flat plate flexures inside the thrust bed, which, in turn, was anchored firmly to the floor. Signal output from the load cell was input to the data acquisition system and into a millivolt meter for visual display. All load cells used had been calibrated by the traceable dead-weight tester in the NAE Structures Laboratory.

Stand stiffness, spring rate, and hysteresis had to be accounted for as there were measurable deflections of the thrust bed relative to the ground reference. As the method of pinning the ends of the flat plate flexures caused them to behave like springs, squeezing the thrust bed with a reference load cell in the plane of the measurement cell could conceivably calibrate the overall system. However, because the engine thrust vector was 115 cm (45 in.) above the load cell, a centreline thrust calibration was performed to compensate for any moment effects (Figure D1). While it was of academic interest to determine the magnitude of the stand stiffness, the only matter of importance was the repeatability of the pre and post-test calibrations. A measure of the repeatability is shown in Figure D2, which shows a maximum deviation of approximately 0.045 KN (10 lbf) for both bridges. The difference in loading between the bridges is due to a side load on the load button of the measurement load cell.

To take into account the deviation between centreline force and base force measurements, the readout from the base load cell was correlated directly to the centreline force applied, see Figure D3. Outputs from bridges A and B of the 25 K base load cell were correlated to the corresponding thrust measurement of the single-bridge centreline reference load cell (10 K limit).

Least-squares-fits of data points taken before and after the test runs yielded mild quadratic equations for bridges A and B of the base load together with the centreline load cell used for the calibration. Separate equations were obtained from pre and post-test data, the coefficients of which were then combined by taking arithmetic means. During the tests, the output of both bridges were averaged to provide a single thrust value, F_m .

For the first engine tested, S/N 607594, the equations were:

Bridge A

$$F_A = 0.012047 + 26.2347 \frac{(mV)}{V_s} + 0.02138 \frac{(mV)^2}{V_s}$$

Bridge B

$$F_B = 0.017207 + 26.2620 \frac{(mV)}{V_s} + 0.01420 \frac{(mV)^2}{V_s}$$

For the second engine, S/N 615037, the following equations were derived:

Bridge A

$$F_A = 0.001709 + 26.1801 \frac{(mV)}{V_s} + 0.05456 \frac{(mV)^2}{V_s}$$

Bridge B

$$F_B = 0.002900 + 26.2701 \frac{(mV)}{V_s} - 0.005329 \frac{(mV)^2}{V_s}$$

where

F_A and F_B = thrust forces measured by bridges A and B of 25,000 lbf load cell in kN

mV_s = electrical output from bridges A and B of 25,000 lbf load cell in mV

V_s = electrical supply voltage to the load cell in V

$$F_m = \frac{F_A + F_B}{2}$$

Load Cell Documentation

Base Load Cell: Interface, Model 1120 BF, Serial No.
18734, capacity 25000 lbf.

Centreline Load Cell: Interface, Model 1210 AF, Serial No.
(calibration only) 21513, capacity 10000 lbf.

III-FRAME CALIBRATION OF THRUST
MEASUREMENT SYSTEM

FIGURE D-1

FIGURE 2-2 CENTRELINE PULL LOAD CELL CALIBRATION - ENGINE 607594

FIGURE D-3 THRUST LOAD CELL OUTPUT VS. CENTRELINE FORCE - ENGINE S/N 615037

APPENDIX E
ENGINE OPERATIONAL LIMITS(1)

<u>DESCRIPTION</u>	<u>PARAMETER</u>	<u>LIMITS(2)</u>
Fuel pressure at Inlet to Engine	PFO	34.5 kPag (5 psig) - Minimum 344.7 kPag (50 psig) - Minimum
Main Oil Pump Discharge Pressure	MOP	241.3 kPag (35 psid) - Minimum at Idle 310.3 (45) $\pm$ 34.5 kPag (5) psid - Minimum at Military
Main Oil Breather Pressure	PMB	13.55 kPa (4 in. Hg) - Maximum Steady-State 37.25 kPa (11 in. Hg) - Maximum Transient
Main Oil Temperature	MOT	394.3°K (250°F) - Maximum Steady-State 405.4°K (270°F) - Maximum Transient
Start Fuel Flow Rate	WFE	1259.98 g/sec (1000) lbm/hr - Maximum
Low Rotor Speed	NL	6460 rpm - Maximum
Low Pressure Compressor Bleed Control Limits		See Fig. E-1
High Rotor Speed	NH	9925 rpm - Maximum
High Rotor Speed Idle Trim Limits	NH Idle	Limit See Fig. E-2
Turbine Cooling Air Pressure	PCP	See Fig. E-3
Exhaust Gas Temperature	TSH	723.15°K (834°F) - Maximum Starting 883.15°K (1130°F) - Maximum Steady-State 910.93°K (1180°F) - Maximum Transient
Exhaust Gas Temperature Individual Probe Limit	T5A01	922.04°K (1200°F) - Maximum
Exhaust Gas Temperature Spread (Max-Min)	Spread	83°K (150°F) - Maximum

ENGINE OPERATIONAL LIMITS (1)
(continued)

<u>DESCRIPTION</u>	<u>PARAMETER</u>	<u>LIMITS(2)</u>
Exhaust Gas Temperature Starting	TI7	722.54°K (840°F) - Maximum Steady-State
Oil Temperature Heat Exchanger Operating	TOCO	366.5 (200) ± 5.6°K (10°F)
Inlet Case Vibration	VIC	5.0 Mils Peak-to-Peak - Maximum
Diffuser Case Vibration	VDC	4.5 Mils Peak-to-Peak - Maximum
Turbine Exhaust Case Vibration	VTC	4.5 Mils Peak-to-Peak - Maximum
Accessory Drive Case Vibration	VAD	7.0 Mils Peak-to-Peak - Maximum
Oil Consumption		.526 ml/sec (4 Pints/Hr) - Maximum
Accessory Fuel Drain Line Leakage	P&D Valve	10 cc/Min - Maximum
	Fuel Cont.	20 cc/Min - Maximum
	Fuel Pump	10 cc/Min - Maximum

Spectrometric Oil Analysis Program Sample (SOAP) Limits

Element - Maximum(*)	
Fe	11 PPM
Ag	3 PPM
Al	5 PPM
Cr	3 PPM
Cu	3 PPM
Mg	5 PPM
Ni	3 PPM
Ti	3 PPM

(*) Maximum limits of accumulated wear metal contaminants in parts per million (PPM) concentrations.

Notes: (1) Limits extracted from I.O. 2J-J57-13, Section IX.

(2) Limits are steady-state unless otherwise noted.

NOTE: LIMITS EXTRACTED FROM
T.O. 2J-J57-13, FIG. 9-5

FIG. E-1. COMPRESSOR BLEED SYSTEM PERMISSIBLE VARIATION OF N_1 OPERATING SPEEDS.

FIG. E-2. IDLE HIGH COMPRESSOR TRIM LIMITS.

NOTE: LIMITS EXTRACTED FROM
T.O. 2J-157-13, FIG. 9-11

FIG. E-3. MINIMUM TURBINE COOLING AIR PRESSURE REQUIREMENTS.

APPENDIX F

SUMMARY ENGINE DATA Engine SN 607594

- 1) Total Engine Run Time - 39.07 hrs
Total Engine Time at MIL - 4.98 hrs
Total Engine Time at MAX EGT - 0.0 hrs (610°C)
- 2) MAX EGT Recorded MIL - 576°C
- 3) Trim Speed (Mechanical) N_H - 9350 RPM ± 15 RPM
- 4) Vibration

VIC	0.7 - 0.8 MILS
VDC	0.4 - 0.5 MILS
VTC	0.6 - 0.8 MILS
VAD	0.9 - 1.0 MILS
- 5) Oil Analysis - 6 Samples

$\frac{Ti}{<2}$	$\frac{Cu}{<1}$	$\frac{Fe}{<1}$	$\frac{Pb}{<1}$	$\frac{Ni}{<1}$	$\frac{Ag}{<1}$	$\frac{Cr}{<1}$	$\frac{Al}{<1}$
-----------------	-----------------	-----------------	-----------------	-----------------	-----------------	-----------------	-----------------
- 6) EGT Spread

T5A02-552°C	T5A20-582°C	T5H-572°C
T5A11-605°C	T5A29-550°C	ΔT5H-55°C
- 7) Instrumentation Defects -
 - a. PBL2E25 - Tube broken off
PBL2F05 - Tube plugged
 - b. TB14A21 - Intermittent - NRC used mechanical flapper
TB14B21 - " indicator for B.V. Open/Close
 - c. T7802 - Open junction - unservicable
T7L02 - " "
T7A11 - " "
T7G11 - " "
T7K11 - " "
 - d. T7P02, T7E11, T7I11, T7H20, T7J20, T7I29 were intermittent during portions of the test. Some connectors were replaced.

SUMMARY ENGINE DATA
Engine SN 615037

- (A) Fuel Control 570376L21 SN 4980
(B) Fuel Control 570377L22 SN 8333

	(A)	(B)	TOTAL
1) Engine Run Time (hrs)	20.85	7.63	28.48
Engine Time at MIL (hrs)	1.88	1.22	3.10
Engine Time at MAX EGT (610°C)	0.0	0.0	0.0

2) MAX EGT Recorded at MIL 577°C 574°C

3) Trim Speed -N_H-RPM @ 288°K 9500 9480

4) Vibration at MIL Power	VIC	1.3-1.7	2.0
MILS	VDC	0.7-0.8	0.9
	VTC	1.0-1.2	1.1
	VAD	0.8-1.0	1.1

5) Oil Analysis - 4 Samples

$\frac{Ti}{<2}$	$\frac{Cu}{<1}$	$\frac{Fe}{<1}$	$\frac{Pb}{<1}$	$\frac{Ni}{<1}$	$\frac{Ag}{<1}$	$\frac{Cr}{<1}$	$\frac{Al}{<1}$
-----------------	-----------------	-----------------	-----------------	-----------------	-----------------	-----------------	-----------------

6) EGT Spread
(MIL) T5A02-581°C T5A20-575°C T5H-575°C
T5A11-611°C T5A29-533°C ΔT5H-78°C

7) Instrumentation Defects -

- a. PBL2E25 - Tube broken off
PBL2F05 - Tube plugged
- b. TB14A21 - Intermittent - NRC used mechanical flapper
TB14B21 - " indicator for B.V. Open/Close
- c. T7B02 - Open junction - unservicable
T7L02 - " "
T7A11 - " "
T7G11 - " "
T7K11 - " "
- d. T7P02, T7E11, T7I11, T7H20, T7J20, T7I29 were intermittent during portions of the test. Some connectors were replaced.

ENGINE SN 615037

- 1) After completion of testing, during engine preservation a gasket in the fuel control failed. In the region of failure, a lug on the mounting flange appeared to be cracked, thus precluding replacement of only the gasket. A replacement fuel control was obtained and installed on 13 September 1982. The engine was trimmed to the same nominal N_H mechanical speed, and another power hook obtained. The results with both fuel controls were the same within the data scatter. The fuel scheduling parameter $\frac{W_f}{P_3}$ vs N_2 was plotted for both controls, producing the same results.
- 2) The oil cooler control operated erratically for this engine. When tested off line, the controller opened at 92°C and closed at 88°C , however during the engine test, the oil temperature remained low.
- 3) The inlet support strut at the 6 o'clock position appeared to have a couple of hairline cracks propagating from the surface scratches near the outer wall. The photographs supplied by AEDC show these scratches in the "as received" condition. A close eye should be kept on these.
- 4) During test with the new fuel control, the vibration level at the inlet flange increased from approximately 1.5 to 2.0 mils. The allowable limit is 5 mils, however, and particular attention should be paid to this area.
- 5) The TSH spread was quite close to the allowable limit of 83°K ; we observed values of $78-80^\circ\text{K}$ at mil. Rather than a hot streak, there appears to be a cool spot at the 290° location.

END

10	5	10	9	8	4
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FIN