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ALTITUDE CALIBRATION OF AN F100, S/N P680063, TURBOFAN ENGINE

by Thomas J. Biesiadny, Douglas Lee, and Jose R. Rodriguez

Lewis Research Center

SUMMARY

An airflow and thrust calibration of an F100 engine, S/N P680063, was conducted at the NASA Lewis Research Center in coordination with a flight test program at the NASA Dryden Flight Research Center to study the airframe and propulsion-system integration characteristics of turbofan-powered high-performance aircraft. The tests were conducted with and without augmentation for a variety of simulated flight conditions with emphasis on the transonic regime.

The resulting corrected airflow data generalized into one curve with corrected fan speed, and corrected gross thrust increased as simulated flight Mach number increased for nonaugmented power. No pronounced trends in augmented thrust ratio with either flight Mach number or altitude were evident for augmented power. Overall agreement between measured data and computed results from a Pratt & Whitney Aircraft in-flight thrust deck was good, with an approximately 1/2 percent difference for corrected airflow and an approximately $-1\frac{1}{2}$ percent difference for gross thrust. The results of an uncertainty analysis are presented for both parameters at each simulated flight condition.

INTRODUCTION

An airflow and thrust calibration of an F100 engine, S/N P680063, was undertaken at the NASA Lewis Research Center as part of a program to study the airframe and propulsion-system integration characteristics of turbofan-powered high-performance aircraft. The calibration, conducted in an altitude test chamber, was done in support of a flight test program at the NASA Dryden Flight Research Center. A second engine, S/N P680059, was also part of this program, and the results of tests with that engine are presented in reference 1.

Attempts to compare wind tunnel and flight data have been plagued by the inability of the wind tunnel simulations to accurately account for propulsion system drag (inlet, boattail, etc.) on the high-performance aircraft used in the flight tests (ref. 2). Engine performance data during flight are normally computed from only a few simple measurements by means of a computer deck supplied by the engine manufacturer. The data obtained at Lewis were compared with values for an average engine predicted by a Pratt & Whitney in-flight thrust deck (ref. 3) to determine to what extent the differences could be applied to the flight test engine.

The Lewis tests (all steady state) were conducted with and without augmentation over a range of simulated flight conditions representing those scheduled for the flight test. The range of conditions was flight Mach numbers from 0.8 to 2.0, altitudes from 4020 to 15 240 meters (13 200 to 50 000 ft), and nonstandard-day as well as standard-day inlet temperatures. The most important performance variables evaluated were corrected airflow and gross thrust. The effect of a simple, but typical, inlet distortion was also evaluated. This distortion was more severe than that used with the F100, S/N P680059 (ref. 1).

Test results for all conditions are presented in terms of corrected airflow and corrected gross thrust as functions of corrected fan speed for nonaugmented power and an augmented thrust ratio as a function of fuel-air ratio for augmented power. Comparisons of measured and predicted data are presented along with the results of an uncertainty analysis for both corrected airflow and gross thrust.

APPARATUS

Engine

The F100 engine, S/N P680063, used in this investigation was classified as an F100 ($2\frac{7}{8}$) engine, which was essentially an F100 (2) configuration with an improved stability fan module and supervisory-control logic changes. The control logic changes were made to maintain fan surge margin with engine deterioration and to provide burner pressure bias on the nozzle area setting. The basic F100 engine, shown schematically in figure 1 along with instrument stations, is a 111-kilonewton (25 000-lbf) thrust class Pratt & Whitney engine. It is a low-bypass, high-compression-ratio, twin-spool turbofan with a mixed-flow augmentor. A more complete description of the engine and its various models can be found in references 4 to 6.

A unified fuel control handled the primary and some secondary controlling functions, and the engine electronic control (EEC) provided fine trim for the engine. One of the signals to the EEC, flight Mach number, was input at the facility by a Mach number simulator. This signal replaced the aircraft Mach number signal, allowing the desired input to be dialed into the control. In supersonic flight, inlet-engine stability is protected by the EEC based on this Mach number signal. The control accomplished this by maintaining a minimum total fan airflow for supersonic flight operation.

Facility

The conventional direct-connect engine installation is shown in figure 1 installed in the altitude test chamber. The engine was hung from a mounting structure, which was attached to a thrust bed. The thrust bed, in turn, was suspended by four flexure rods attached to the chamber supports and was free to move except as restrained by a dual load-cell system, which allowed the thrust bed to be preloaded and was used to measure thrust.

The chamber included a forward bulkhead, which separated the inlet plenum (5.5 m (18 ft) diam) from the test chamber (7.3 m (24 ft) diam). Air of the desired temperature and pressure flowed from the plenum through the bellmouth to the inlet duct. A labyrinth seal was used to isolate the inlet ducting from the bellmouth and bulkhead. The inlet ducting, in turn, was mated to the engine through an inflatable flex joint which served to minimize the loading on the engine front flange.

Engine exhaust gases were captured by a collector that extended through the rear bulkhead, thereby minimizing the possibility of exhaust gas recirculation in the test chamber.

Distortion Screen

An engine-inlet total-pressure distortion was produced by the screen pictured in figure 3 and located 0.73 meter (2.39 ft) from the engine inlet flange. The screen, (an F100-PW-100 Engine Table IIB Distortion Screen, part No. RA518-21AA 1-7F), had been used to simulate the engine inlet profile at maximum power for a flight Mach number of 0.9 at an altitude of 9140 meters (30 000 ft). The distortion pattern produced by this screen at maximum power for one test condition using the technique described in reference 7 is shown in figure 4. The distortion factor for this condition was 26 percent, based on the difference between the maximum and minimum total pressures divided by the average total pressure.

Instrumentation

Only flight-qualified steady-state instrumentation, including the inlet rake described in reference 8, was mounted in the engine. Further, the amount of instrumentation was minimized to include only those measurements considered necessary for setting test conditions, measuring airflow and gross thrust, and monitoring engine health. The locations of the majority of the instruments are shown schematically in figure 1. (See the appendix for a list of symbols and their definitions.)

The steady-state pressures, including the pressures at the engine inlet, were recorded on 12 scanivalves. Tests with engine S/N P680059 (ref. 1) were conducted using individual absolute transducers which were part of the inlet rake. However, the scanivalves were more reliable and thus were used for the tests reported herein. In two locations, the fan inlet and the compressor inlet, high-response transducers were used as stall indicators. Chromel-alumel thermocouples referenced to a 339 K (610° R) oven were used throughout the installation to measure temperatures.

The equations used to calculate airflow and gross thrust and an uncertainty analysis for airflow and gross thrust are presented in reference 1.

TEST PROCEDURES

Engine Conditions

Engine inlet pressure and temperature and exhaust pressure were determined from the flight Mach numbers, altitudes, and inlet recovery factors specified by Dryden. Inlet pressure was established using an average total pressure at the engine inlet. This was true with uniform inlet flow as well as with inlet distortion. The inlet temperature was an average of the thermcouple measurements in the inlet plenum, it being assumed that there was no heat lost between the plenum and engine inlet. The simulated altitude conditions were determined in the test cell using the static pressures on the exterior surface of the nozzle. In particular, the ring of static-pressure taps farthest from the nozzle exit plane were used to set test conditions. However, the differences among all of the nozzle static pressures were insignificant. A complete list of the simulated flight conditions can be found in table I.

A few of the tests, namely, conditions 2 and 4 in table I, were conducted with the EEC from engine S/N P680059, while functional tests were being performed on the EEC from engine S/N P680063. Therefore, the inlet distortion testing, condition 4, was only comparable with the uniform inlet testing, condition 2, when this EEC was installed. After the EEC from engine S/N P680063 had been reinstalled and the engine retrimmed, testing was conducted over the entire range of uniform inlet flow conditions specified in table I.

As mentioned in the APPARATUS section, the engine control maintained a minimum total fan airflow for supersonic flight operation. However, for the calibration tests it was considered necessary to investigate part-power operation while simulating supersonic flight. This was accomplished by dialing a subsonic Mach number into the Mach number simulator, which resulted in the removal of the airflow lockout function and allowed part-power scheduling of the engine power lever. Test conditions were maintained at simulated supersonic flight pressures and temperatures.

Nonstandard Day and Inlet Distortion

The effects of nonstandard-day test conditions were evaluated at a ram pressure ratio (i.e., engine inlet pressure/exhaust pressure) of 1.66, which for uniform inlet flow corresponds to a flight Mach number of 0.9, and an altitude of 7380 meters (24 200 ft). Also included was the evaluation of the effects of inlet flow distortion at this condition. This flight Mach number and altitude were chosen because instrument accuracies were judged best at these values due to the relatively high pressure levels, and the distortion screen was designed to simulate the inlet conditions near this test condition. High instrument accuracy was required because small differences were expected.

Calculation Program

Engine performance data during flight are normally computed from only a few simple measurements by means of a computer deck supplied by the engine manufacturer. The method, in which such a deck was used to calculate gross thrust and corrected airflow as a comparison with measured values, is presented herein.

Gross thrust. - The Pratt & Whitney In-Flight Thrust Computing Deck (ref. 3) was used to compute gross thrust. The only program modification was the substitution of measured nozzle area ratios. All inputs, including correct airflow, were obtained from measured data.

Corrected airflow. - The corrected airflow data in the Pratt & Whitney In-Flight Thrust Deck was curve fit by Dryden and formed a basis of comparison for the measured corrected airflow data. The baseline data are defined as those data which excluded corrections for such factors as Reynolds number effects, guide-vane angle variation, inlet distortion, and a term referred to (ref. 3) as an "Airflow Correction from deck to engine." The curve fit required only measured corrected fan speed and engine pressure ratio as input to determine corrected airflow.

An exception to this method of determining baseline corrected airflow was made at a corrected fan speed of 8500 rpm. At this corrected fan speed, previous data (ref. 1) had shown larger than normal differences between predicted and measured corrected airflow. As a consequence, the curve fit at 8500 rpm was redone by Lewis.

RESULTS AND DISCUSSION

Airflow and thrust calibration tests were conducted with and without augmentation for a variety of simulated flight conditions with emphasis on the transonic regime. Results from these tests are presented in figures 5 to 10 in terms of corrected airflow and corrected gross thrust as functions of corrected fan speed for nonaugmented power and an augmented thrust ratio as a function of fuel-air ratio for augmented power. Comparisons of these measured results with calculated data are displayed in figures 11 and 12, and the results of an uncertainty analysis for both measured corrected airflow and gross thrust are shown in table II. The design corrected airflow used to normalize the data was 98.4 kilograms per second (217 lbm/sec), and the nominal corrected gross thrust was arbitrarily chosen as 111 kilonewtons (25 000 lbf).

Standard Day

Corrected airflow data as a function of corrected fan speed over a range of flight Mach numbers and altitudes are presented in figure 5. A range of corrected fan speeds from part power, approximately 5000 rpm, to intermediate power (the maximum nonaugmented power setting) were investigated with all data collapsing into a single curve with little scatter. Data at Reynolds number indexes of 0.29 and 0.34 showed no apparent shift from the other test conditions, which were above a Reynolds number index of 0.5.

The presentation of corrected gross thrust as a function of corrected fan speed (fig. 6) for a range of Mach numbers, altitudes, and Reynolds number indexes contrasts with the airflow curve of figure 5 in that each resulted in a distinct set of data. In addition, corrected gross thrust increased with increasing flight Mach number for a given corrected fan speed. The influence of Mach number can further be seen in the data at a constant Mach number of 0.9, two different altitudes of 7380 and 13 720 meters (24 000 and 45 000 ft), and three inlet temperatures of 278, 295, and 252 K (501°, 531°, and 453° R) produce the same curve. In contrast, two different Mach numbers, 1.4 and 2.0, at the same altitude, 15 240 meters (50 000 ft), produce two distinct curves.

The corrected gross thrust data for augmented power (fig. 7) are plotted as a ratio of the corrected gross thrust at selected augmentor power lever angles to the corrected gross thrust at intermediate power for the same test condition. Except for minimum and maximum augmentation, these data were recorded at a power lever angle corresponding to the midpoint of each augmentor segment. The augmentor fuel-air ratio is the ratio of the augmentor fuel flow to the unburnt air - that is, the air associated with the oxygen consumed in the primary combustor was subtracted from the total airflow. This is a method used previously in reference 9.

In general, the data scatter, even with the sensitive augmented thrust ratio, was small enough that each set of data could be separated, with the exception of the two sets of data at 0.9 Mach and 7380 meters (24 200 ft), to form individual curves. However, because of the many variables such as flight Mach number, altitude, inlet temperature, nozzle area, and the wide range of values covered, no general trends are evident.

Nonstandard Day and Inlet Distortion

Inlet temperature variation. - An inlet temperature variation from 278 to 295 K (501° to 531° R) was accomplished at a flight Mach number of 0.9 and 7380 meter (24 200 ft). Corrected airflow (fig. 5) and corrected gross thrust (fig. 6) are plotted against corrected fan speed for nonaugmented power, and augmented thrust ratio is plotted against fuel-air ratio in figure 7. As previously mentioned in the discussion of these figures, little or no differences were evident on the three plots with inlet temperature variation.

Inlet flow distortion. - The corrected airflow and corrected gross thrust data for the nonuniform inlet condition are plotted in figures 8 to 10 in a fashion similar to the previously presented data. The screen distortion, which resulted in an approximately 26 percent distortion factor at maximum power, was not enough to cause a difference in corrected airflow and corrected gross thrust between uniform and the noninform inlet conditions within the accuracy of the data.

Data Comparisons and Accuracies

A primary objective of the calibration tests, besides providing data for the flight program, was to compare measured results with calculated data - in this case those from an in-flight thrust computing program (ref. 3). It follows that the accuracy of the measured data is also of prime importance. The results of the comparison between measured and calculated data (figs. 11 and 12) and an uncertainty analysis of the measured data (table II) are presented herein.

Data comparisons. - In figures 11 and 12 corrected airflow and gross thrust (as measured) are compared with the results from the calculation program. The deviation of corrected airflow (fig. 11) is the difference between measured and calculated values divided by the calculated value and is plotted against measured corrected airflow.

For the range of flight Mach numbers, altitudes, and inlet temperatures presented in figure 11(a), all were with uniform inlet flow. Figure 11(b) compares data for uniform and nonuniform inlet flow - both of which were recorded with the EEC from engine S/N P680059 installed. In figure 11(a) the mean of the difference between the measured and calculated data was approximately 1/2 percent with majority of the data within a band of ±1 percent about this mean. There are notable exceptions, however, particularly between 65 and 70 percent of design correct airflow where the data for each test condition are between 0 and -2 percent. This appeared to be a problem area with the calculation program. The data between 45 and 55 percent also are of interest because differences are 2 percent or greater. Measurement uncertainties, which will be presented in the discussion of table II, are the most likely cause of these discrepancies where airflows are low.

In figure 11(b) the mean through the data was near 1/4 percent with the majority of the data included in a band of approximately ±1 percent. If the data below 50 percent design corrected airflow were ignored, the mean was close to 0 percent with a band of ±3/4 percent enclosing the data. The resulting differences between the two sets of data were from 1/2 to 3/4 percent with the exception of 70 percent flow.

In summary, for a variety of conditions the vast majority of the measured and calculated corrected airflow data were in good agreement.

The comparison of measured and calculated gross thrust is presented in figure 12(a) for a range of conditions with uniform inlet flow. The mean of the plotted data was approximately $-1\frac{1}{2}$ percent, and the deviation about this mean was $\pm 1\frac{1}{4}$ percent. This deviation, which was larger than that encountered for tests with engine S/N P680059 (ref. 1), was apparently due to a number of reasons. The data at a flight Mach number of 2.0 obviously contributed to the scatter since much of these were between 0 and 2 percent. Another factor was the data at a flight Mach number of 0.9 and an altitude of 13 720 meters (45 000 ft). There was some difficulty in maintaining the inlet temperature of 252 K (453° R), which resulted in more frequent than normal adjustments in the valves controlling the inlet total pressure. As a consequence, test condition stability was affected. The data below 12 percent gross thrust reflected the proportionately greater influence of gross thrust uncertainties at low thrust than at high thrust levels.

Data with and without uniform inlet flow at one flight Mach number - altitude condition with the EEC from engine S/N P680059 are presented in figure 12(b). No distortion factors were entered into the calculation program. The resulting differences between the two sets of data were less than 1 percent. Otherwise, the mean of all the data was $-1\frac{3}{4}$ percent with a deviation about the mean of ±1 percent. Again, as expected the greatest discrepancies occurred at the lowest gross thrust measurements.

As with the corrected airflow data, the agreement between measured and calculated gross thrust data was good except where noted.

Measured data uncertainty. - The results of an uncertainty analysis for the measured data at selected test conditions are listed in table II for corrected airflow and gross thrust. These uncertainties were calculated using the instrument inaccuracies and the uncertainty equations presented in reference 1. At intermediate power the corrected airflow and gross thrust uncertainties for the majority of the data were less than 1 percent. Since instrument errors were considered fixed, the measurement uncertainties increased as corrected airflow and gross thrust decreased for minimum power setting, as shown in table II. At the maximum power setting for each condition, gross thrust uncertainties were ±0.8 percent or less. The decrease in uncertainties from those at intermediate power was anticipated, because the calculated parameters (e.g., inlet momentum) were fixed while only the load-cell measurements, inherently more accurate, changed.

SUMMARY OF RESULTS

An airflow and thrust calibration was conducted with an F100 engine, S/N P680063, for a variety of simulated flight conditions. The principal results of this calibration were

1. All corrected airflow data generalized into one curve against corrected fan speed with no apparent shift in the correlation for data at a Reynolds number index less than 0.5.

2. Corrected gross thrust increased with increasing flight Mach number for nonaugmented power, but no pronounced trends in augmented thrust ratio with either flight Mach number or altitude were evident for augmented power.

3. Corrected gross thrust correlated with corrected fan speed for nonaugmented power and with fuel-air ratio for augmented power at two different inlet temperatures at the same flight Mach number.

4. Overall agreement between measured and calculated data using a Pratt & Whitney computer deck was good, with approximately 1/2 percent difference for corrected airflow and approximately $-1\frac{1}{2}$ percent for gross thrust. The measured data uncertainties for the majority of the data were less than ±1 percent for corrected airflow and gross thrust at intermediate power and ±0.8 percent or less for gross thrust at the maximum attainable power for each test condition.

5. No significant changes were evident in corrected airflow and corrected gross thrust with an inlet distortion factor of approximately 26 percent.

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APPENDIX - SYMBOLS

F_G	gross thrust, N
f/a	fuel to air ratio
M	Mach number
N	rotor speed, rpm
P	pressure, Pa
RNI	Reynolds number index, $\delta/(\mu/\mu_{SLS})\sqrt{\theta}$
T	temperature, K
W	mass flow rate, kg/sec
δ	ratio of total pressure to standard sea-level static pressure
θ	ratio of total temperature to standard-sea level static temperature
μ	absolute viscosity, kg/(m · sec)

Subscripts:

a	air
amb	ambient
aug	augmentor
calc	calculated value from Pratt & Whitney computer deck (ref. 3)
design	design point
F	fan
int	intermediate power setting
meas	measured value
nom	nominal value
SLS	standard sea level static conditions
t	total
0	station 0, or free stream
1	station 1, airflow measuring station
2	station 2, engine inlet
2.5	station 2.5, fan exit
4.5	station 4.5, fan-turbine inlet

6	station 6, fan-turbine exit
6.5	station 6.5, augmentor liner
6.7	station 6.7, augmentor liner
6.9	station 6.9, augmentor liner

REFERENCES

1. Biesiadny, T.; Lee, D.; and Rodriquez, J.: Airflow and Thrust Calibration of an F100 Engine, S/N P680059, at Selected Flight Conditions. NASA TP-1069, 1978.
2. Bellman, D. R.; Burcham, F. W.; and Taillon, N. V.: Techniques for the Evaluation of Air-Breathing Propulsion Systems in Full-Scale Flight. NASA TM X-68305, 1972.
3. F100 (3) In-Flight Thrust Calibration Deck. CCD 1088-2.0, Pratt & Whitney Aircraft, 1975.
4. Sams, H.: F15 Propulsion System Design and Development. AIAA Paper 75-1042, Aug. 1975.
5. Staley, E. I.: F15 Propulsion Flight Testing Experience. AIAA Paper 75-1052, Aug. 1975.
6. Taylor, John W. R., ed.: Jane's All the World's Aircraft, 1975-1976. Franklin Watts, Inc., 1975.
7. Dicus, John H.: Fortran Program to Generate Engine Inlet Flow Contour Maps and Distortion Parameters. NASA TM X-2967, 1974.
8. Farr, A. P.: Evaluation of F15 Inlet Dynamic Distortion. AIAA Paper 73-784, Aug. 1973.
9. McAulay, John E.; and Abdelwahab, Mahmood: Experimental Evaluation of a TF30-P-3 Turbofan Engine in an Altitude Facility: Afterburner Performance and Engine-Afterburner Operating Limits. NASA TN D-6839, 1972.

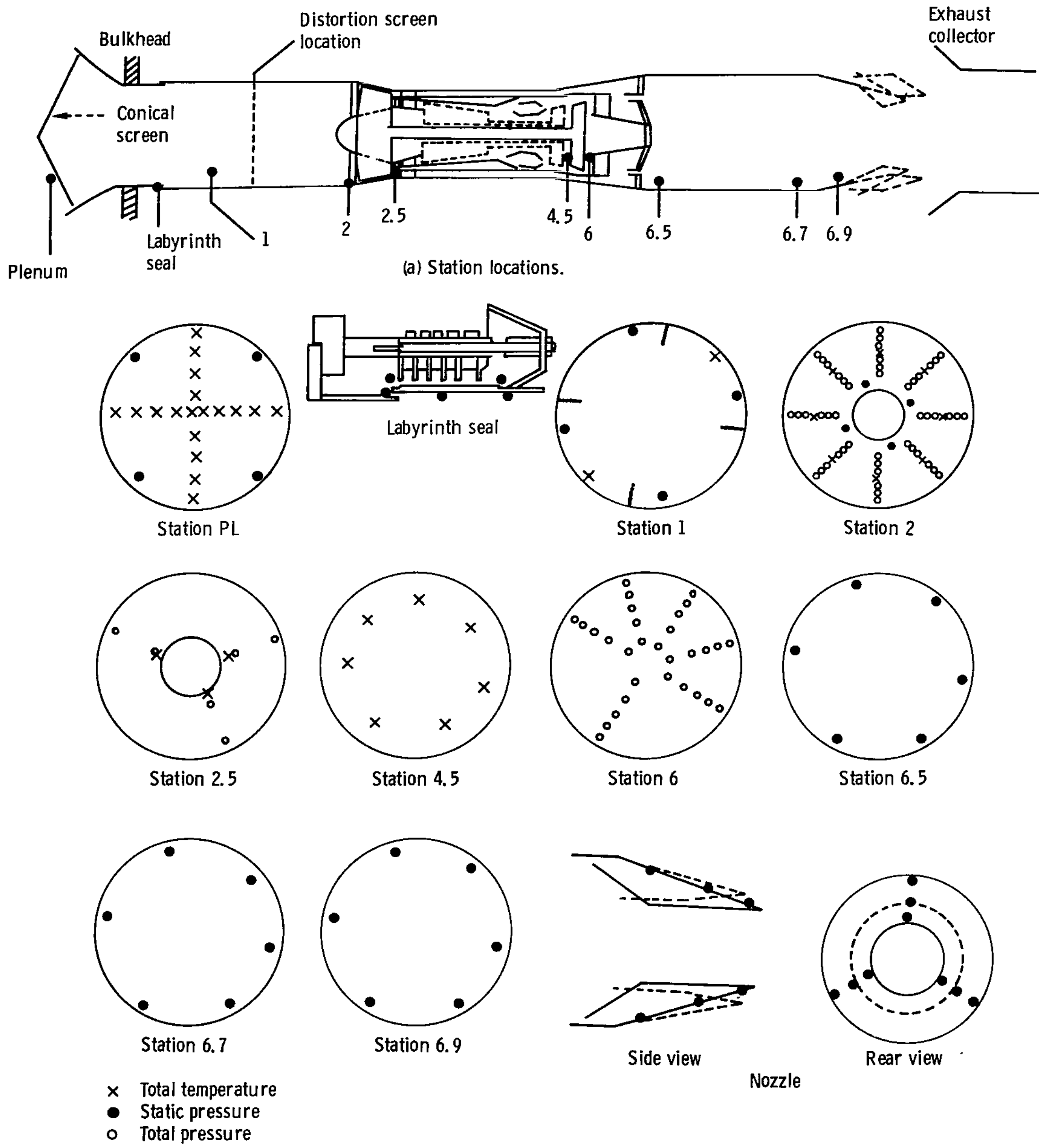
TABLE I. - SIMULATED FLIGHT CONDITIONS

Simulated flight condition number	Mach number, Mo	Altitude		Inlet pressure, P_{t2}		Inlet temperature, T_{t2}		Ambient pressure, P_{amb}		Reynolds number index, RNI	Inlet flow	Day
		m	ft	kPa	psia	K	^oR	kPa	psia			
1	0.8	4 020	13 200	92.7	13.44	296	532	61.4	8.91	0.89	Uniform	Standard
2	.9	7 380	24 200	64.5	9.36	278	501	39.0	5.65	.66	Uniform	Standard
3	.9	7 380	24 200	64.5	9.36	295	531	39.0	5.65	.62	Uniform	Hot
4	(a)	7 380	24 200	64.5	9.36	278	501	39.0	5.65	.66	Nonuniform	Standard
5	.9	13 720	45 000	24.8	3.59	252	453	14.8	2.14	.29	Uniform	↓
6	1.4	15 240	50 000	36.1	5.24	301	542	11.6	1.68	.34	↓	
7	1.6	9 140	30 000	120.0	17.40	339	610	30.1	4.37	.96		
8	2.0	15 240	50 000	85.6	12.42	390	702	11.6	1.68	.58	↓	↓

[a]Ram pressure ratio (based on average station 2 pressure with inlet distortion), 1.66.

TABLE II. - MEASUREMENT UNCERTAINTIES

Simulated flight condition number	Corrected airflow		Gross thrust		
	Power lever angle				
	Min.	Int.	Min.	Int.	Max.
	Uncertainties, ± percent of measurement				
1	0.8	0.6	1.8	0.4	---
2	1.0	.7	2.1	.5	0.3
3	1.0	.7	2.1	.5	.3
4	1.1	.7	2.4	.5	.3
5	1.5	1.4	2.0	1.3	.8
6	1.4	1.0	2.1	.9	.6
7	.7	.3	.8	.5	.2
8	.8	.5	1.0	.5	.3



(b) Individual stations.

Figure 1. - Engine instrumentation.

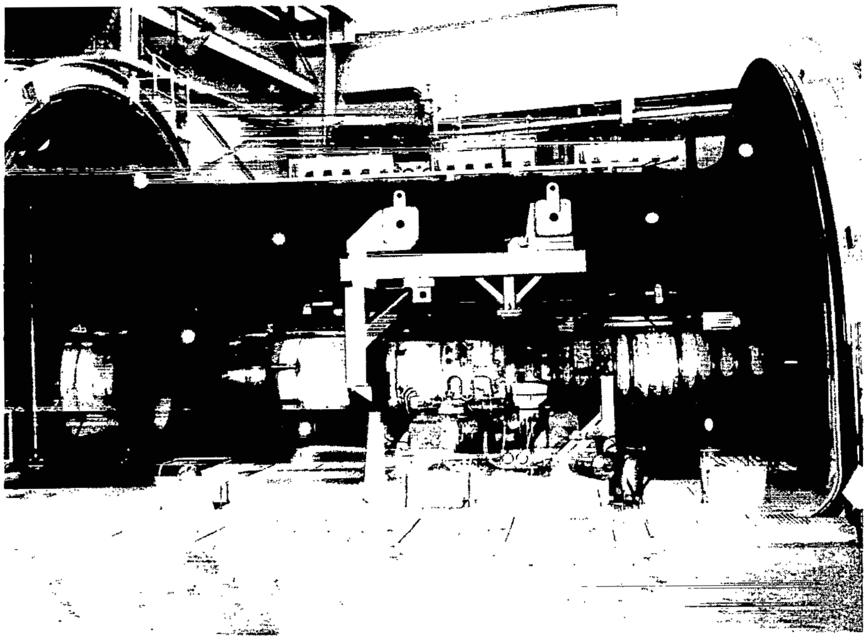


Figure 2. - Engine installation in the altitude test chamber (F100, S/N P680063).

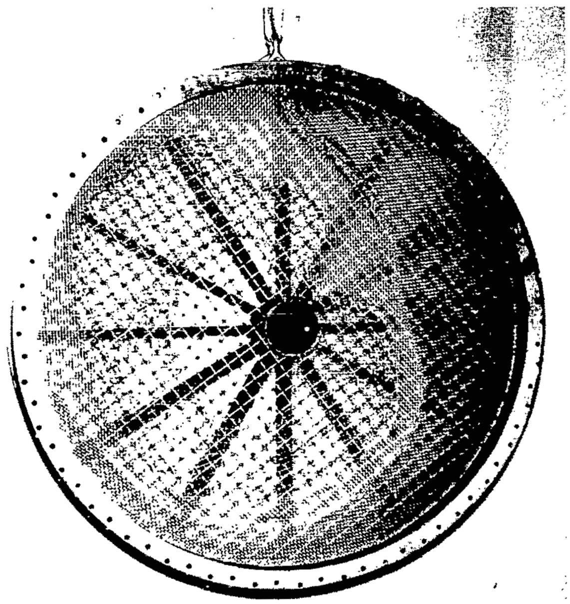


Figure 3. - Distortion screen 1-7F (view looking downstream).

Key to mapping symbols

Border	$\frac{P_{local} - P_{av}}{P_{av}}$
8/7	16
7/6	14
6/5	12
5/4	10
4/3	8
3/2	6
2/1	4
1/0	2
0/A	0
A/B	-2
B/C	-4
C/D	-6

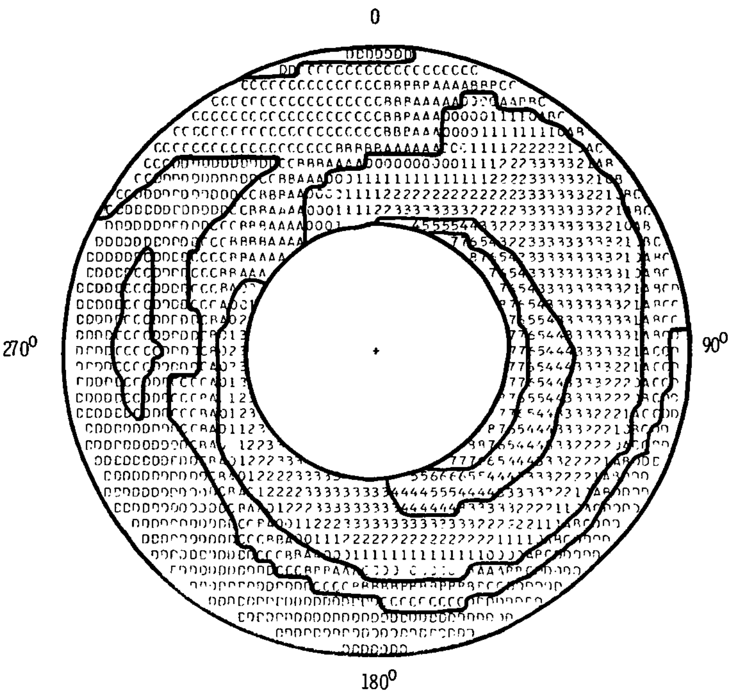


Figure 4. - Distortion pattern produced by screen RA518-21AA 1-7F at engine inlet for flight Mach number of 0.9 and altitude of 7380 meters (24 200 ft); maximum power.

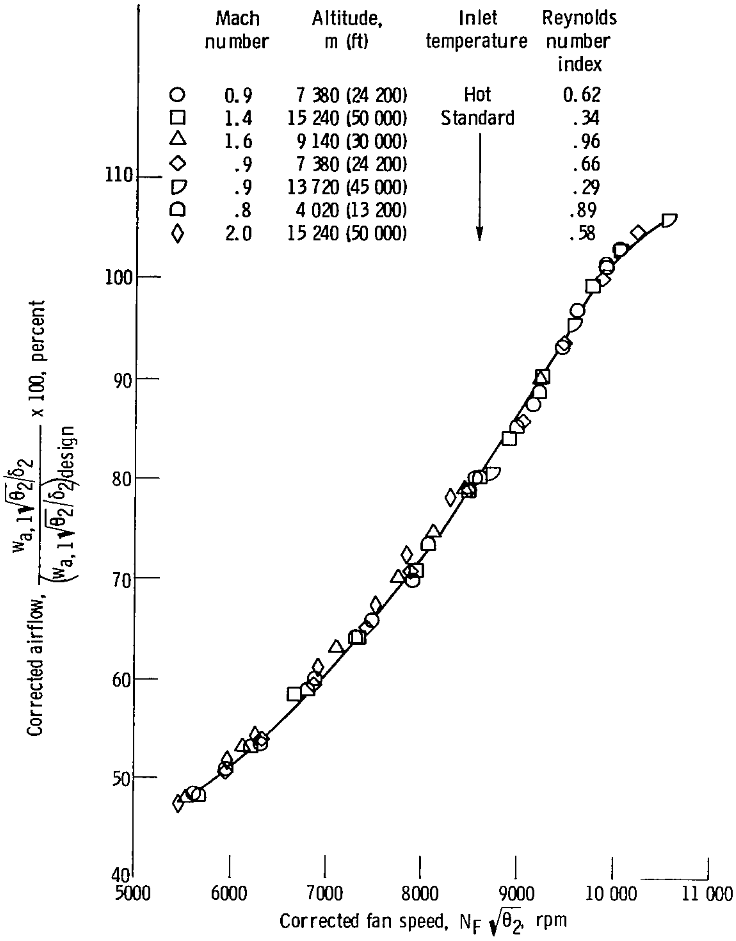


Figure 5. - Corrected airflow as function of corrected fan speed.

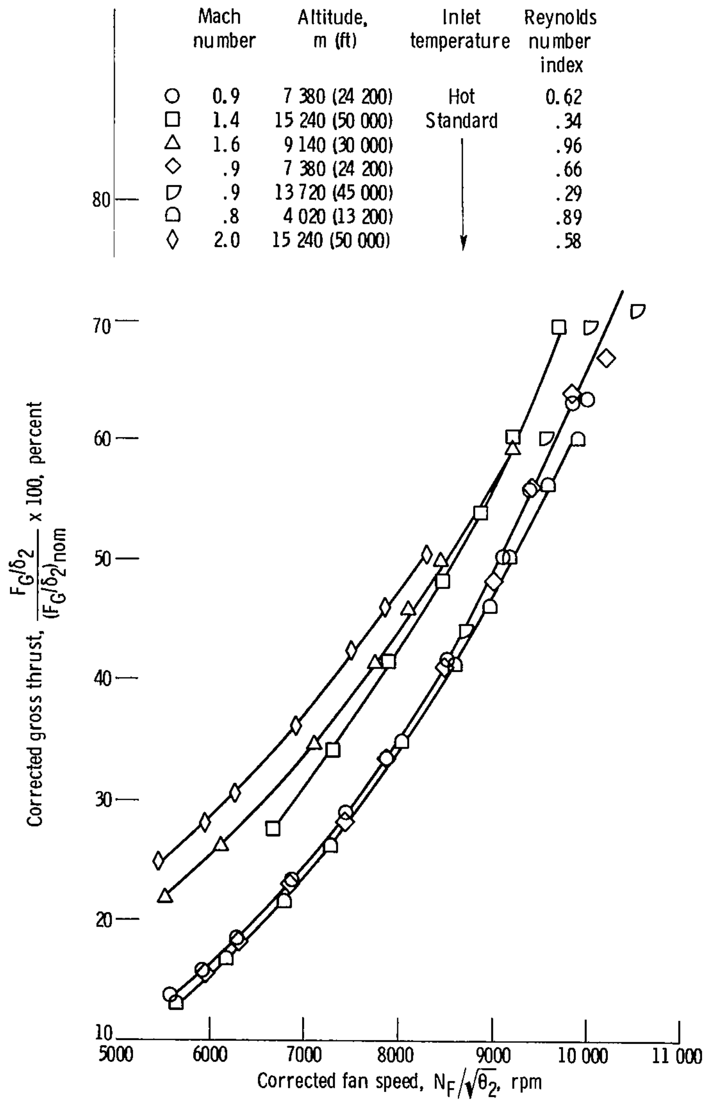


Figure 6. - Corrected gross thrust as function of corrected fan speed.

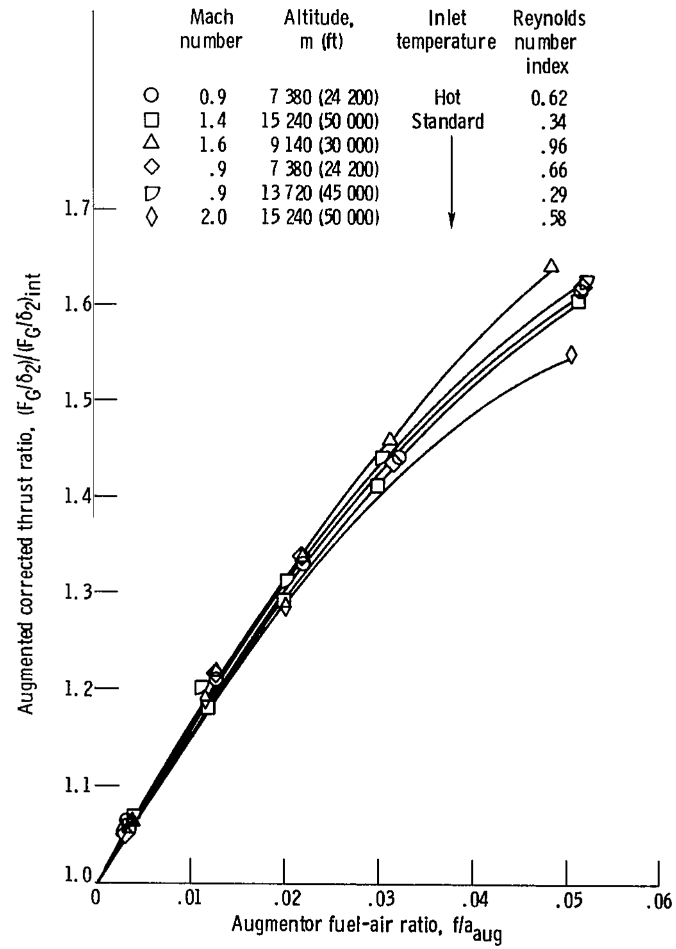


Figure 7. - Augmented corrected thrust as function of augmentor fuel-air ratio.

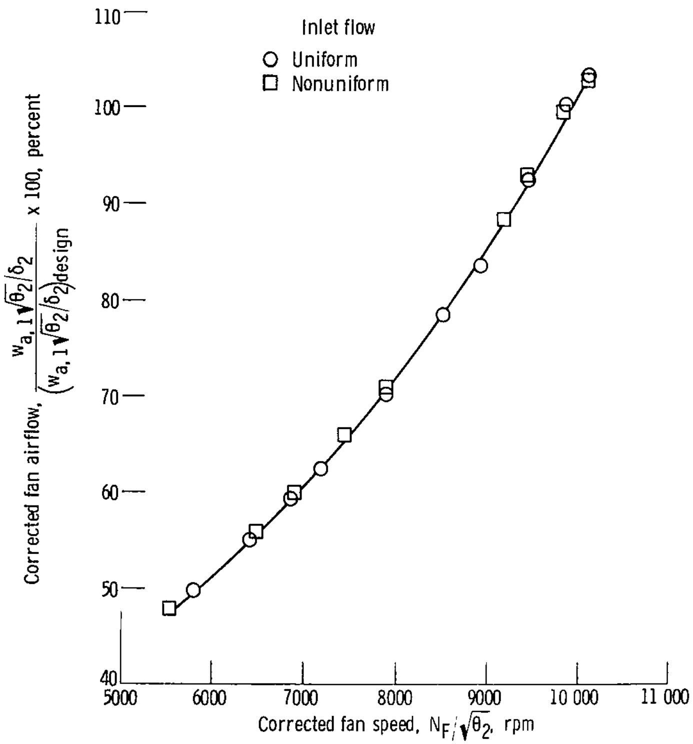


Figure 8. - Corrected airflow as function of corrected fan speed with and without uniform inlet flow. Ram pressure ratio, 1.66; altitude, 7380 meters (24 000 ft).

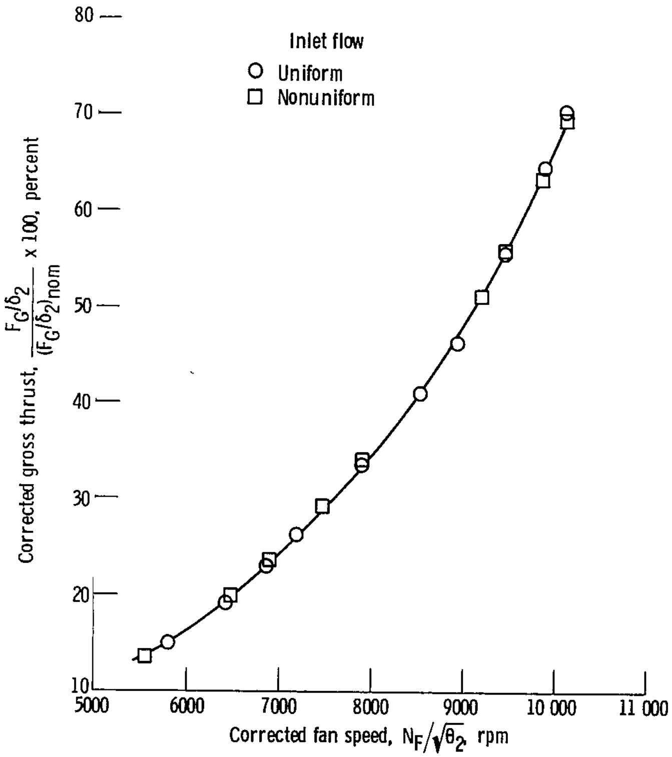


Figure 9. - Corrected gross thrust as function of corrected fan speed with and without uniform inlet flow. Ram pressure ratio, 1.66; altitude, 7380 meters (24 200 ft).

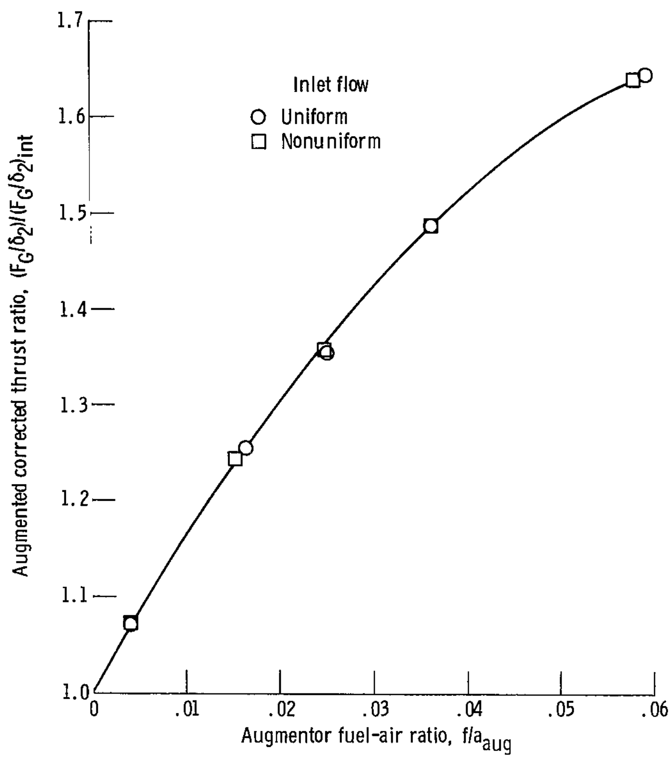
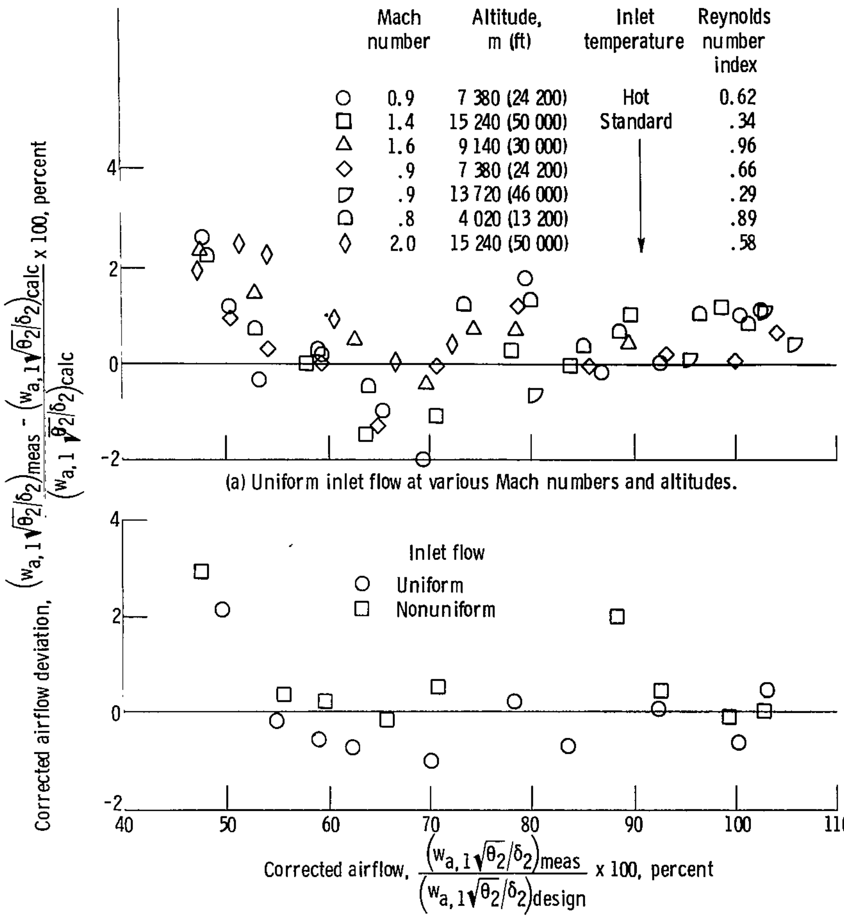


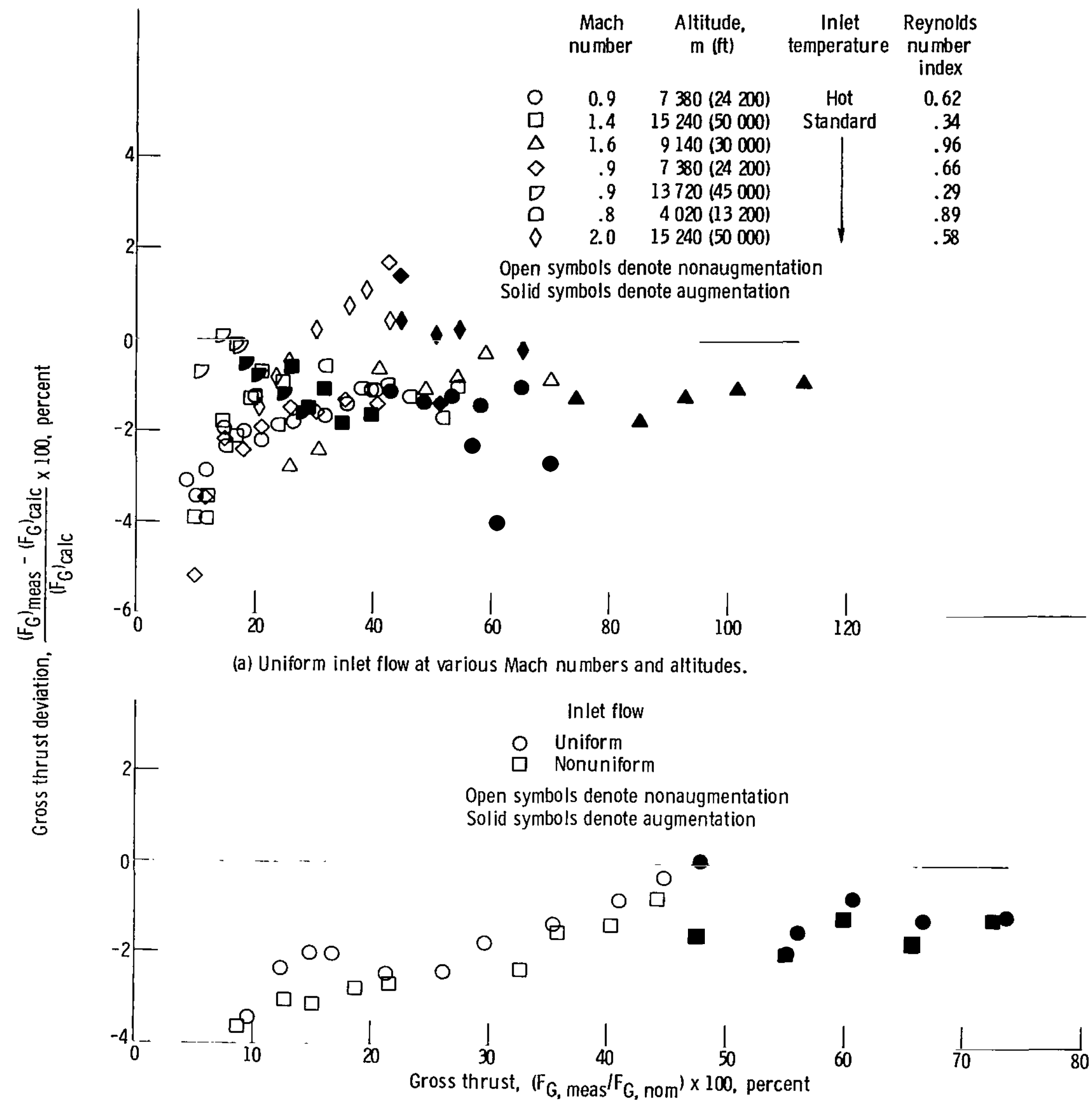
Figure 10. - Augmented corrected thrust as function of augmentor fuel-air ratio with and without uniform inlet flow. Ram pressure ratio, 1.66; altitude, 7380 meters (24 200 ft).



(a) Uniform inlet flow at various Mach numbers and altitudes.

(b) Uniform and nonuniform inlet flow at similar conditions. Ram pressure ratio, 1.66; altitude, 7380 meters (24 200 ft).

Figure 11. - Corrected airflow comparison.



(a) Uniform inlet flow at various Mach numbers and altitudes.

(b) Uniform and nonuniform inlet flow at similar conditions. Ram pressure ratio, 1.66; altitude, 7380 meters (24 200 ft).

Figure 12. - Gross thrust comparison.

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16. Abstract

An airflow and thrust calibration of an F100 engine, S/N P680063, was conducted at the NASA-Lewis Research Center in coordination with a flight test program at the NASA-Dryden Flight Research Center to study airframe-propulsion system integration characteristics of turbofan-powered high-performance aircraft. The tests were conducted with and without augmentation for a variety of simulated flight conditions with emphasis on the transonic regime. Test results for all conditions are presented in terms of corrected airflow and corrected gross thrust as functions of corrected fan speed for nonaugmented power and an augmented thrust ratio as a function of fuel-air ratio for augmented power. Comparisons of measured and predicted data are presented along with the results of an uncertainty analysis for both corrected airflow and gross thrust.

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