Measurement of XB-70 Propulsion Performance Incorporating the Gas Generator Method

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The ability to accurately calculate in-flight propulsion performance is essential to any flight test program which has as its objective the determination of aircraft drag characteristics. Calculation of the engine-inlet performance for the XB-70 is achieved through extensive instrumentation of the inlet duct and engines. Engine net thrust is calculated using the "gas generator method" which was pioneered and developed by General Electric Company. Presented is an explantion of how the gas generator method is used to calculate the propulsion performance for the XB-70, the fundamental logic assumptions used in the gas generator method, and the instrumentation associated with it.

Nomenclature

A_8 = primary nozzle throat area
A_{e9} = effective secondary nozzle exit area
C_{f8} = primary nozzle flow coefficient
C_{fg} = gross thrust coefficient
C_{V9} = secondary nozzle velocity coefficient
dh/dt = rate of change of altitude with respect to time
dV_0/dt = rate of change of velocity with respect to time
F_G = gross thrust
F_{GI} = ideal gross thrust
F_N = net thrust
F_{NE} = net propulsive effort
F_R = ram drag
$H_{A(\ )}$ = enthalpy of the air at the station indicated by subscript
$H_{G(\ )}$ = enthalpy of the gas at the station indicated by subscript
HP_{ex} = power extracted from engine
J = work equivalent of heat
N = engine compressor rotor speed
P_0 = freestream static pressure
$P_{T\sec}$ = total pressure of secondary airflow
$P_{T(\ )}$ = total pressure at station indicated by subscript
$P_{S(\ )}$ = static pressure at station indicated by subscript
Q_L = lower heating value of fuel
$T_{T\sec}$ = total temperature of secondary airflow
$T_{T(\ )}$ = total temperature at station indicated by subscript
V_0 = freestream velocity
V_9 = gas velocity of primary airflow at secondary nozzle exit area
$V_{9\sec}$ = gas velocity of secondary airflow at secondary nozzle exit area
W_{A2} = primary airflow at station 2
W_B = bleed air from compressor
W_{FM} = main fuel flow
W_{FR} = reheat fuel flow
W_{FT} = total fuel flow
$W_{G(\ )}$ = primary stream weight flow at station indicated by subscript
$W_{\rm LOST}$ = turbopump airflow (lost from primary stream to secondary stream)
W_t = air vehicle gross weight
$W_{\sec}$ = secondary airflow
η_{BM} = main combustion efficiency
η_{BR} = afterburner combustion efficiency
$\theta_{(\ )}$ = relative absolute total temperature at station indicated by subscript
$\delta_{(\ )}$ = relative absolute total pressure at station indicated by subscript

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Definition of Propulsion Performance

THE XB-70 is powered by six YJ-93 engines. Each of the two two-dimensional mixed compression inlets supplies air for three engines (see Fig. 1). The air vehicle propulsion performance of the XB-70 is defined as the net propulsion effort. This term is equal to the engine net thrust less the inlet additive drag, bypass drag, and inlet boundary-layer bleed drag. The engine net thrust is the gross thrust of the primary and secondary airstreams through the ejector nozzle less the freestream momentum of the primary and secondary airflow (see Fig. 2). The bypass and inlet bound-

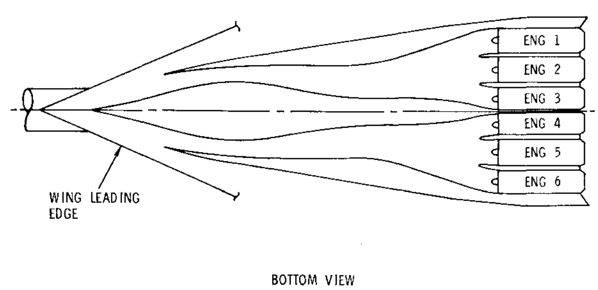


Fig. 1 XB-70 inlet engine configuration.

ary-layer bleed drag is defined as the difference between the freestream total momentum and the exit total momentum of the airflow.

Additive drag or spillage drag is determined from theoretical data and empirical data obtained from wind-tunnel tests. In determining the additive drag, the freestream ambient pressure, effective inlet Mach number, and the inlet mass flow ratio are taken into account.

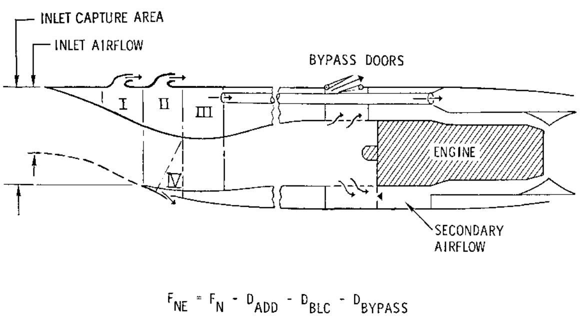
The set of six bypass doors for each inlet are above and just forward of the engines. They are used for inlet-engine airflow matching and shock control. The determination of the bypass airflow and drag are obtained from inflight measurement of door position, bypass plenum static pressures and bypass exit total pressure.

Inlet boundary-layer bleed from the inlet duct walls is divided into four zones. Airflows from the bleed zones are calculated from in-flight measurement of exit total and static pressures. Exit momentum is calculated by knowing the airflow, exit static pressure, temperature, and area. Figure 3 shows the percentage of each of these drags compared to the maximum net propulsive effort at various Mach numbers. At Mach 2.5 and above, the inlet drags accounts for only 10% of the net propulsive effort.

Summary of Gas Generator Method

This method provides a useful tool in analyzing flight test data to determine engine net thrust. Parameters measured in flight are supplemented or modified by known engine characteristics and used to calculate in-flight engine net thrust and SFC. Engine characteristics are obtained from extensive wind-tunnel and test cell runs. Engine thrust can be calculated if the weight flow, pressure, and temperature at the engine inlet and nozzle exit can be determined. Inlet pressure and temperature are obtained from the flight test data. Compressor airflow is obtained from inlet total pressure measurements, engine speed, and compressor inlet temperature. Turbine discharge temperature is calculated from a heat balance across the main engine, taking into account combustion efficiency and parasitic flows. Jet nozzle total pressure is determined from the measured turbine discharge pressure, the estimated tailpipe dry loss, and momentum pressure loss due to heat addition for reheat operation. Jet nozzle inlet temperature is equal to turbine discharge temperature for dry operations, and is calculated from a heat balance across the afterburner on reheat operation using a previously established reheat operation combustion efficiency. Jet nozzle temperature calculated from the measured A_8 and nozzle flow coefficient is also printed out for comparison for dry or reheat operations.

Secondary nozzle flow, pressure, and temperature are obtained from the selected combination of measured test data and known nozzle characteristics. Gross thrust is then calculated for the primary and secondary streams. Nozzle



$F_{NE} = F_N - D_{ADD} - D_{BLC} - D_{BYPASS}$

$F_N = F_G - F_r$

Fig. 2 XB-70 inlet schematic.

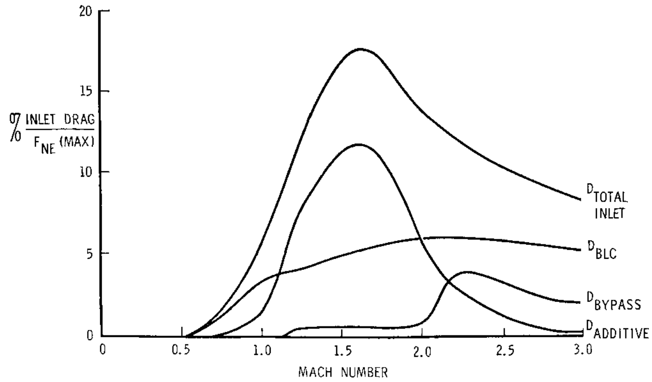


Fig. 3 Percentage of inlet drags of maximum net propulsive effort.

flap leakage, velocity coefficients, and expansion ratio are included in this calculation.

Details of the Gas Generator Method

Instrumentation

The location and type of engine instrumentation is presented in Fig. 4. The pressures P_{T2} and temperature T_{T2} at the engine inlet are obtained directly from in-flight measurements. Total temperature is measured with an externally mounted stagnation temperature probe. Temperature measurements are also taken at the inlet to the Nos. 1 and 3 engines and used as a check of the OAT probe. Total pressure measured at the inlet of engines Nos. 1 and 4 is from four total pressure rakes, each rake having five probes at the center of equal area. The average pressure for the Nos. 5 and 6 engines is established from inlet similarity, i.e., the No. 6 engine inlet is a mirror image of the No. 1 engine and likewise for the Nos. 5 and 2 engines. Validity of the inlet similarity assumption is checked by comparing the No. 4 engine inlet rake data with the No. 3 engine rake data. Another check is made by comparing the pressure level measured with a single probe located on the bullet nose of three engines in the left inlet to those located in the right inlet.

Primary Airflow W_{A2}

The engine inlet airflow is calculated by uncorrecting the engine face corrected airflow:

$$W_{A2} = [W_{A2}(\theta_2)^{1/2}/\delta_2](P_{T2}/14.7)[(519)/T_{T2}]^{1/2}$$

where P_{T2} and T_{T2} are measured data. The engine corrected airflow can be determined from established compressor characteristics, once the corrected speed is known. A large amount of engine test experience has provided the following engine corrected speed vs corrected airflow characteristics (Fig. 5).

Corrected airflow data for the engine have been acquired on two independent measuring systems: the standard factory

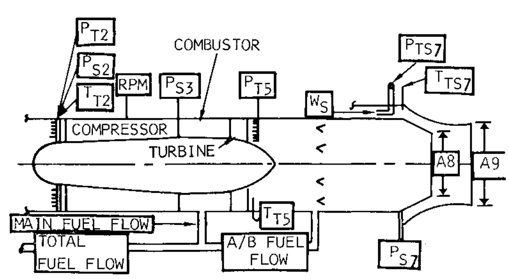


Fig. 4 Engine performance instrumentation.

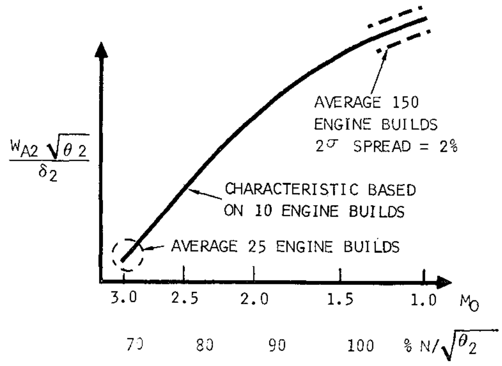


Fig. 5 Engine compressor corrected airflow curve.

bellmouth that was calibrated with standard American Society of Mechanical Engineers flow nozzles during the full-scale compressor testing at the General Electric facility in Lynn, Mass.; a choked venturi during the development and qualification testing at the Arnold Engineering Development Center (AEDC) in Tullahoma, Tenn. Excellent agreement was obtained between the two independent systems.

An alternate method of obtaining the compressor airflow is from the measurement of the total and static pressures at the engine face. Knowing the pressure, temperature, and area the airflow can be calculated, these two methods have shown agreement within ±3%.

Secondary Airflow

Flow characteristics

$$[W(T_T)^{1/2}]/P_{\text{in sec}} = f(P_{\text{in}}/P_{\text{out}})_{\text{sec}}$$

established by calibration tests run at AEDC and the North American Rockwell test facility at Santa Susana, define the amount of secondary flow through the heat shroud passage. By measuring secondary pressure, temperature, and pressure drop across the heat shroud, the secondary flow rate can be computed.

An alternate method of calculating the secondary airflow is to use the nozzle air handling characteristics shown in Fig. 6. These characteristics were established by $\frac{1}{4}$-scale model testing at NASA Lewis Research Center, and full-scale testing at AEDC. They yield levels of secondary airflow which agree quite well with the calibrated heat shroud method.

The gas flow at the nozzle exit is calculated by adding the secondary airflow to the primary gas flow.

$$W_{G9} = W_{G8} + W_{\text{sec}}$$

where

$$W_{G8} = W_{A2} + W_{FT} - W_{\text{LOST}}$$

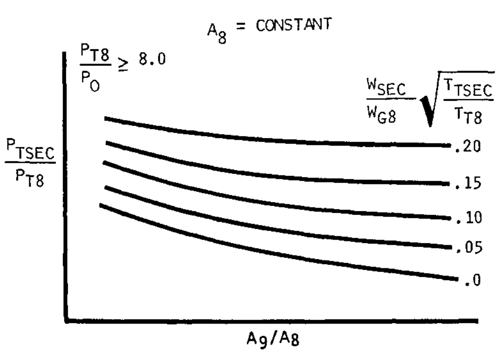


Fig. 6 Nozzle pumping characteristics.

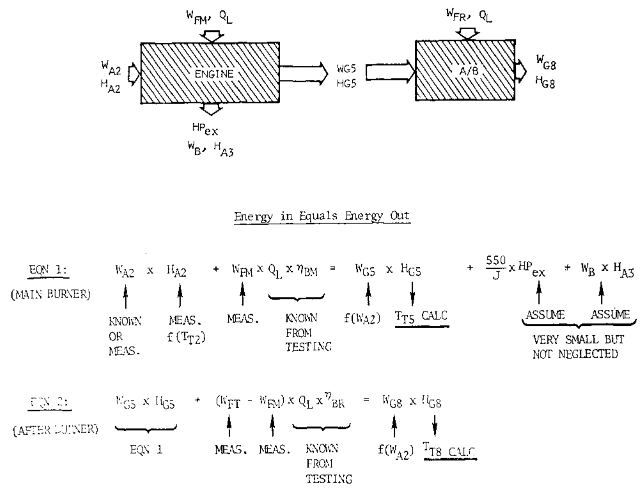


Fig. 7 Temperature calculated from conservation of energy.

Exit Temperature T_{T8}

Having established all the conditions (T_{T2}, P_{T2}, and W_{A2}) at the engine inlet as well as the engine exit flow, the exit gas temperature can be computed. Basically, the exit temperature is calculated by considering the engine to be a black box where the energy leaving the box is equal to the sum of the energy entering the box as shown in Fig. 7.

Equation 1 in Fig. 7, is used to calculate the gas temperature at turbine discharge T_{T5}, which is equal to the exhaust nozzle inlet temperature T_{T8}, during nonafterburner engine operation. The calculated gas temperature is compared to the control system thermocouple harness indications as a check on the accuracy of the calculation. During afterburner operation, Eq. 2 in Fig. 7, which treats the afterburner as a second black box in series with the first, yields the exhaust nozzle inlet temperature T_{T8}. The initial estimates of main and afterburner combustion efficiencies were established from J47 and J79 engine experience and refined by component test data taken in NASA, General Electric, and AEDC testing facilities. During the development of the YJ-93-3 engine, a considerable amount of additional test data has been collected on the engine using gas sample analysis and balanced cycle calculations.

Redundancy Check on T_{T8}

The measured primary jet nozzle area provides a check on the calculated temperature through the calculation of continuity of flow from the following relation.

$$W_{G8}(T_{T8})^{1/2}/P_{T8}\, A_8\, C_{f8} = \text{const} \quad \text{for} \quad P_{T8}/P_0 > 2$$

Exit Pressure P_{T8}

The jet nozzle inlet total pressure is determined by measuring the pressure at the turbine discharge with two 7-element total pressure rakes. The nozzle inlet total pressure is calculated by accounting for the tailpipe pressure loss and during

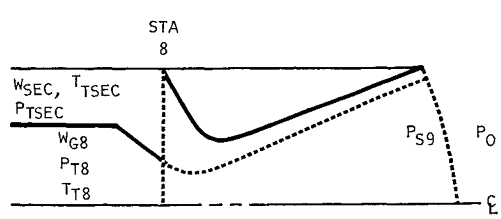


Fig. 8 Ejector nozzle schematic.

afterburning operation, for the heat addition momentum pressure loss as shown in the following equation.

$$P_{T8} = P_{T5}\,(P_{T6}/P_{T5})(P_{T7}/P_{T6})$$

Redundancy Check on P_{T8}

During the development test program the following correlation was established:

$$P_{T8}/P_{S7} = f(A_8)$$

Since both P_{S7} and A_8 are measured, this correlation provides a redundancy check on P_{T8}.

Gross Thrust F_G

With the inlet condition to the jet nozzle thus established (Fig. 8), the gross thrust can now be calculated knowing the nozzle performance characteristics.

Velocity Coefficient C_{V9}

This procedure accounts for primary and secondary nozzle leakage. After leakage corrections are made, the ideal thrust of the nozzle is determined by coexpanding the primary and secondary stream independently to the exit area. This expansion calculation is repeated until the nozzle static pressure is identical for both streams and the combined exit area of both streams is equal to the physical exit area. The actual nozzle thrust is obtained from the ideal momentum thrust by multiplying it by a velocity coefficient and adding the pressure times area thrust term.

$$F_G = [(W_{\mathrm{sec}}V_{9\mathrm{sec}} + W_{G8}V_9)/g]C_{V9} + (P_{S9} - P_0)A_{e9}$$

For the J93 nozzle, these velocity coefficients have been determined from $\frac{1}{4}$-size-scale model tests which were conducted at NASA Lewis Research Center and verified by full scale engine tests at AEDC. These coefficients account for nonaxial flow, friction, and profile effects.

Gross Thrust Coefficient C_{fg}

At low nozzle area ratios and secondary pressure ratios, this expansion calculation cannot be performed because of problems in balancing static pressure at the nozzle exit. This procedure is therefore only used for nozzle pressure ratios P_{T8}/P_0, greater than 5. At pressure ratios less than 5, thrust is calculated by multiplying the ideal thrust F_{GI}, by nozzle thrust coefficients which are functions of secondary flow ratios, nozzle area ratios, and nozzle pressure ratios. These thrust coefficients have been determined by $\frac{1}{4}$-scale and full-scale tests and from factory test cell runs.

$$F_G = F_{GI} \times C_{fg}$$

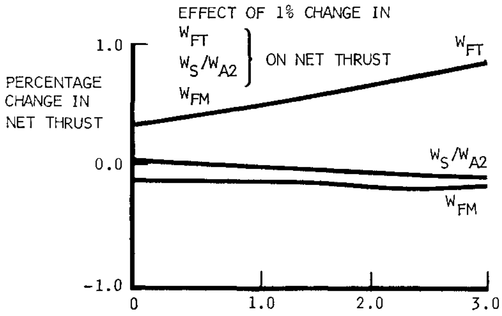


Fig. 9 Effect of 1% change in W_{FT}, W_S/W_{A2}, and W_{FM} on net thrust.

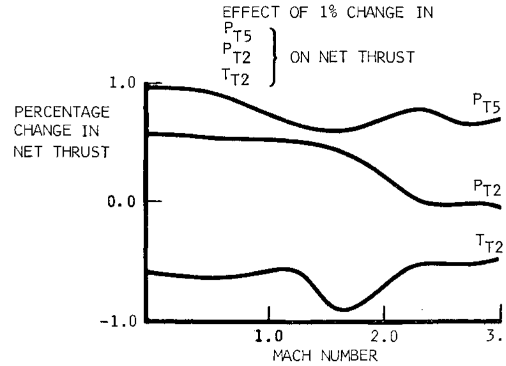


Fig. 10 Effect of 1% change in P_{T5}, P_{T2}, and T_{T2} on net thrust.

Net Thrust

With the gross thrust, engine airflow, and fuel flow established, the net thrust and specific fuel consumption is calculated using conventional methods.

$$F_n = F_G - (W_{A2} + W_{\mathrm{sec}})V_0/g$$

and

$$SFC = W_{FT}/F_n$$

Influence Coefficients for Engine

A study has been made of the effect on engine net thrust of 1% error in the measured parameters used in the gas generator method. Figures 9 and 10 show the effect of a 1% change on engine net thrust as a function of Mach number at maximum afterburner power setting. The Mach number and altitude combination used represents a typical XB-70 flight profile. It is important to note that there is less than a 1 to 1 correspondence between a 1% change in any input parameter and the resulting change in engine net thrust. The gas generator method is a highly accurate method of calculating engine net thrust because an error in the measurement of any one parameter is either compensated for by the correct measurement of the other parameters or does not greatly influence the calculated net thrust.

Validation of Net Propulsive Effort

In order to assess the accuracy of the gas generator method of calculating engine thrust, three approaches were used. First, comparison of the calculated thrust to the measured thrust from static ground thrust runs; second, an in-flight incremental thrust maneuver was made where the change in excess thrust was calculated by two independent methods. One using the gas generator method and the other by accounting for the change in energy of the aircraft; third, a statistical analysis of the degree of accuracy obtainable in calculating the net propulsive effort with the present instrumentation accuracy.

Ground Thrust Runs

The Edwards Air Force base thrust stand was used to obtain the measured engine thrust of the XB-70. At static conditions, the net propulsive effort is essentially equal to the gross thrust of the engines. There is a small base drag which was accounted for. The ground thrust runs permitted an assessment of the accuracy of the engine gross thrust as determined by the gas generator method. Figure 11 shows the comparison between the calculated thrust and the measured thrust. The data are for six engines running at power settings between military and maximum afterburner. The tests were run on three days spread over a one-month period. After the second day's running, two engines received FOD.

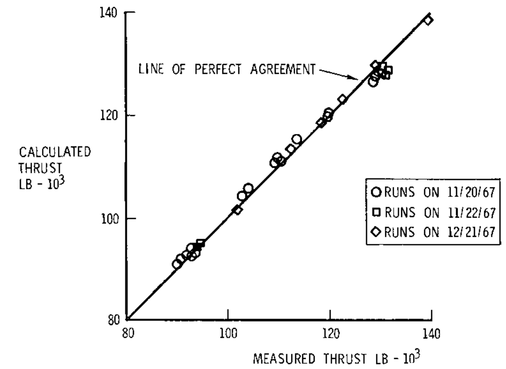


Fig. 11 Comparison of calculated thrust to measured thrust.

Therefore, the data has been taken over a period of time and is not for the same six engines.

The calculated thrust is within ±2% of the measured thrust for all data shown. The rms deviation is ±1.1%. These results not only demonstrate the accuracy of the gas generator method, but show that the instrumentation is capable of measuring the many required parameters with acceptable accuracy.

In-Flight Incremental Thrust Comparison

There is obviously no way to make an absolute in-flight comparison between the actual thrust and the calculated thrust. However, some conclusions can be made from the incremental thrust maneuvers flown on recent XB-70 flights. During a climb and descent maneuver, the change in excess thrust is calculated from the gas generator method and is also calculated by accounting for the change in the energy of the aircraft. Figure 12 shows a time history of one of the incremental thrust maneuvers flown by the XB-70. Following a stabilized speed-power run, the throttles were advanced to maximum afterburning, and a constant Mach number climb was initiated. After a stabilized climb has been achieved, the power was reduced to military on four of the six engines and a descent, still at constant Mach number, was made. The net propulsive effort of the XB-70 was calculated during stabilized portions of the climb and descent at the same altitude, at the indicated points presented in Fig. 12. The difference between the two net propulsive efforts is the change in the excess thrust. The change in the excess thrust was also calculated from the aircraft performance using the following equations.

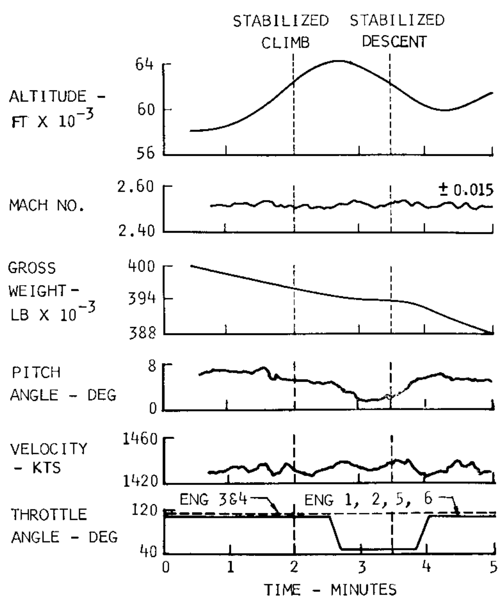


Fig. 12 Time history of incremental thrust maneuver.

Table 1 Summary of the method

	Example: W_{FT}
1) Assume maximum measurement error for key parameter.	±800 lb/hr
2) Vary that parameter for one engine and determine error that results in aircraft F_{NE}.	0.58%
3) Assuming same error in all 6 engines, determine error in aircraft F_{NE}. rms error = [6 × (error for 1 engine)^2]$^{1/2}$	1.4%
Repeat steps 1, 2, and 3 for 26 parameters.	
4) Determine over-all accuracy of aircraft F_{NE} by accounting for all errors of engine and inlet parameters similar to step 3.	

$$\text{Accuracy} = \text{rms of all errors} = [(\underset{W_{FT}}{1.4})^2 + (\underset{P_{T0}}{0.21})^2 + (\underset{P_{BLC\ I}}{0.14})^2 + \ldots]^{1/2} = \pm 3.0\%$$

Aircraft Performance Method

$$\text{THRUST})_{\text{excess}} = (W_t/V_0)(dh/dt) + (W_t/g)(dV_0/dt) \quad (1)$$

$$\Delta\text{THRUST})_{\text{excess}} = \left[\frac{W_t}{V_0}\frac{dh}{dt} + \frac{W_t}{g}\frac{dV_0}{dt}\right]_{\text{climb}} - \left[\frac{W_t}{V_0}\frac{dh}{dt} + \frac{W_t}{g}\frac{dV_0}{dt}\right]_{\text{descent}} \quad (2)$$

The terms W_t, V_0, dH/dt, and dV_0/dt are determined from flight data.

Net Propulsive Effort Method

$$\text{THRUST})_{\text{excess}} = F_{NE} = D_{A/C} \quad (3)$$

$$\Delta\text{THRUST})_{\text{excess}} = [F_{NE} - D_{A/C}]_{\text{climb}} - [F_{NE} - D_{A/C}]_{\text{descent}} \quad (4)$$

For stabilized flight at the same altitude, Mach number, and external configuration, the aircraft drag is essentially the same during descent and climb. A small correction is made for change in pitch angle and weight.

$$\Delta\text{THRUST})_{\text{excess}} = F_{NE})_{\text{climb}} = F_{NE})_{\text{descent}} \quad (5)$$

Therefore, using Eqs. 2 and 5 the change in the excess thrust can be calculated by two independent methods. Preliminary analysis of the data from past flights have shown good agreement. The two methods agreed within $\frac{1}{2}$% when the difference between the descent and climb excess thrust was the largest. At this point, the change in excess thrust was approximately 50% of the aircraft drag. The close agreement in the two methods gives increased confidence that the gas generator method is an accurate method of calculating engine net thrust.

Statistical Analysis of Net Propulsive Effort

To obtain an indication of how accurately the net propulsive effort can be calculated, a typical point at Mach 2.5 was chosen, 27 key parameters were varied individually, and the effect on the net propulsive effort was determined. The key parameters were varied by the estimated accuracy that each parameter could be measured. This accuracy was determined from past experience and from redundancy and symmetry checks. The key parameter included engine parameters and parameters used in determining the inlet drags.

As an example, the maximum error that could exist in the total fuel-flow measurement is 800 lb/hr. An error greater than this would be replaced by an estimated value. The validity of the total fuel flow is determined by three basic checks. These are the comparison of adjacent and symmetrical engines, the comparison of the sum of the main plus afterburner fuel flow, and the comparison of the calculated nozzle total temperature T_{T8}, from the fuel flow path and area path.

Once the maximum error has been established for the key parameters, its effect on the aircraft F_{NE} is determined. As an example, an 880 PPH error in total fuel flow for a single engine results in a 0.58% error in the aircraft net propulsive effort. Assuming this error in 6 engines, the resulting effect in the aircraft net propulsive effort is 1.4%.

The other 26 key parameters are varied and the resulting error in the aircraft net propulsive effort is determined. The over-all accuracy of the aircraft net propulsive effort is determined by accounting for all of these errors. The results show that at a typical Mach 2.5, the net propulsive effort can be calculated within $\pm 3\%$. A summary of the method used is Table 1.

Conclusions

The gas generator method is an accurate method of determining in-flight engine thrust on the XB-70. Using the gas generator method, an error in the measurement of any one parameter is either compensated for by the correct measurement of the other parameters or does not greatly influence the calculated net thrust. The generator method also provides several redundancy checks to insure accurate results in calculating engine net thrust. In addition, the XB-70 has a statistical advantage with six engines of averaging out any measurement errors. Also, having six engines provides for adjacent and symmetrical engine checks to validate the accuracy of the measured parameters. The accuracy of using the gas generator in calculating the aircraft net propulsive effort has been proved by the ground thrust runs which have shown agreement within $\pm 2\%$, by the in-flight incremental thrust maneuvers which has added confidence to the ability to calculate in-flight net thrust, and by the statistical approach which confirms that the net propulsive effort can be calculated within $\pm 3\%$ at high Mach numbers.