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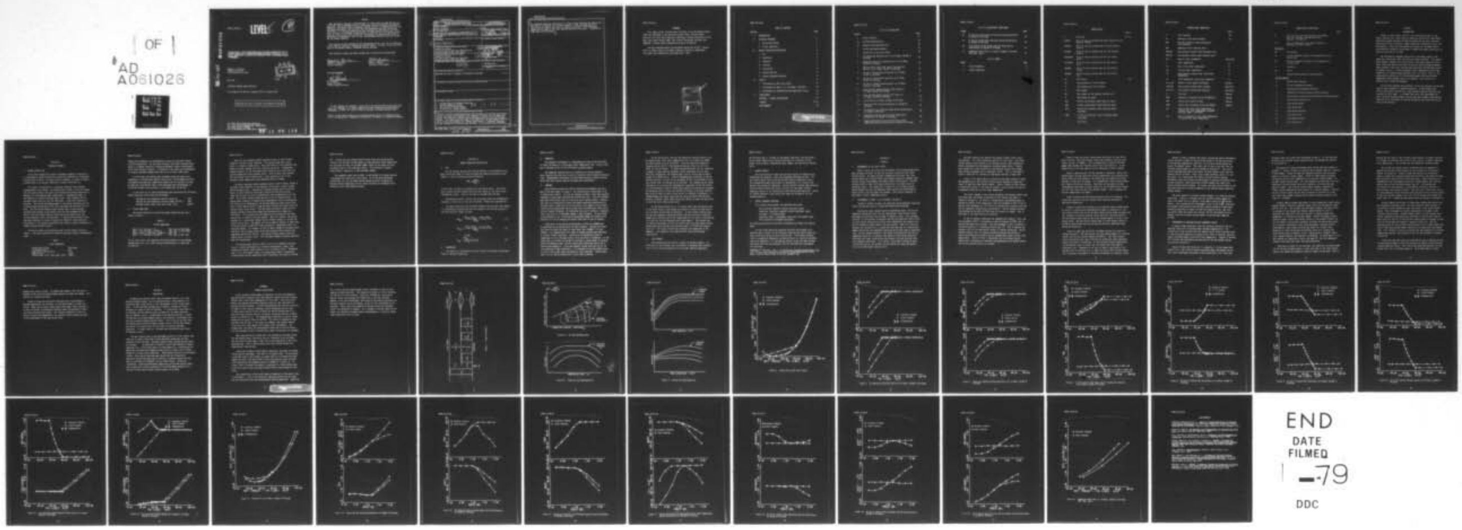

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# OPERATING AND PERFORMANCE CHARACTERISTICS OF A DUCT BURNING TURBOFAN ENGINE WITH VARIABLE AREA TURBINES

Edward K. Norvaisis
Propulsion Branch
Turbine Engine Division



May 1978

TECHNICAL REPORT AFAPL-TR-78-29

Final Report for Period 1 January 1977 to 1 March 1978

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# REPORT DOCUMENTATION PAGE

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| | | |
|---|---|---|
| 1. REPORT NUMBER<br>AFAPL-TR-78-29 | 2. GOVT ACCESSION NO. | 3. RECIPIENT'S CATALOG NUMBER<br>9 |
| 4. TITLE (and Subtitle)<br>6 OPERATING AND PERFORMANCE CHARACTERISTICS OF A DUCT BURNING TURBOFAN ENGINE WITH VARIABLE AREA TURBINES. | | 5. TYPE OF REPORT & PERIOD COVERED<br>Final Report.<br>1 January 1977 - 1 March 78 |
| | | 6. PERFORMING ORG. REPORT NUMBER |
| 7. AUTHOR(s)<br>10 Edward K./Norvaisis | | 8. CONTRACT OR GRANT NUMBER(s) |
| 9. PERFORMING ORGANIZATION NAME AND ADDRESS<br>Propulsion Branch<br>Air Force Aero Propulsion Laboratory/TBP<br>Wright-Patterson AFB, Ohio 45433 | | 10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS<br>16 Project No. 3066<br>Task No. 306611<br>Work Unit No. 30661108 |
| 11. CONTROLLING OFFICE NAME AND ADDRESS<br>Propulsion Branch<br>Air Force Aero Propulsion Laboratory<br>Wright-Patterson AFB, Ohio 45433 | | 12. REPORT DATE<br>11 May 1978 17 |
| | | 13. NUMBER OF PAGES<br>52 |
| 14. MONITORING AGENCY NAME & ADDRESS(if different from Controlling Office)<br>12 52p. | | 15. SECURITY CLASS. (of this report)<br>Unclassified |
| | | 15a. DECLASSIFICATION/DOWNGRADING SCHEDULE |

16. DISTRIBUTION STATEMENT (of this Report)

Approved for public release, distribution unlimited.

62203F

17. DISTRIBUTION STATEMENT (of the abstract entered in Block 20, if different from Report)

18. SUPPLEMENTARY NOTES

19. KEY WORDS (Continue on reverse side if necessary and identify by block number)

1. Turbine Engine Performance Analysis
2. Variable Geometry Turbine
3. Duct Burning Turbofan Engine
4. Variable Cycle Engine Analysis

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

Several variable geometry/variable cycle concepts for aircraft gas turbine engines are being investigated by the industry in search for new propulsion systems with potential operating and performance benefits over conventional engines. This paper describes some of the operating and performance characteristics of one such concept in a fighter application: a separate flow duct burning turbofan with variable area turbines. An existing cycle performance

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SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered) and component matching simulation of a fixed turbine turbofan was extensively modified to incorporate the variable geometry. Data was generated for a representative fighter type cycle at several important flight conditions. Comparison was made to the fixed turbine version of the cycle. Installation drags were not calculated.

## FOREWORD

This report covers in-house work initiated in the Performance Branch (TBA) and completed in the Propulsion Branch (TBP), Turbine Engine Division, Air Force Aero Propulsion Laboratory, Wright-Patterson Air Force Base, Ohio under Project 3066, "Gas Turbine Technology" and Work Unit 30661108, "Turbine Engine Integration Analysis Procedures."

The work reported herein was performed during the period 1 January 1977 to 1 March 1978 by the author, Edward K. Norvaisis (AFAPL/TBP). This report was submitted by the author April 1978.

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## LIST OF ILLUSTRATIONS (CONCLUDED)



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## NOMENCLATURE

| | | UNITS |
|---|---|---|
| A | Flow area | sq. in. |
| A/AMIN | Ratio of area to minimum area-high pressure or low pressure turbine | |
| A4/AMIN | Ratio of area to minimum area for high pressure turbine | |
| A5/AMIN | Ratio of area to minimum area for low pressure turbine | |
| A28/A28DS | Ratio of area to design area for duct nozzle throat | |
| A29/A29DS | Ratio of area to design area for duct nozzle exit | |
| A8/A8DS | Ratio of area to design area for core nozzle throat | |
| A9/A9DS | Ratio of area to design area for core nozzle exit | |
| FN | Net thrust | lbs |
| HP | High pressure; as in HP turbine | |
| LP | Low pressure; as in LP turbine | |
| M | Mach number | |
| M6 | Mach number at low pressure turbine exit | |
| N | Shaft physical speed | |
| %N1 | Percent low pressure shaft physical speed | |
| %N2 | Percent high pressure shaft physical speed | |
| %N1C | Percent low pressure shaft corrected speed ($\%N1/\sqrt{\theta 2}$) | |
| %N2C | Percent high pressure shaft corrected speed ($\%N2/\sqrt{\theta 2T}$) | |
| $\eta$ | Efficiency | |

## NOMENCLATURE (CONTINUED)

| | | UNITS |
|---|---|---|
| P | Total pressure | psi |
| P3 | Combustor inlet total pressure | psi |
| PRF | Fan total pressure ratio (average on illustrations) | |
| PRC | Compressor total pressure ratio | |
| PRT-HPT | High pressure turbine total pressure ratio | |
| PRT-LPT | Low pressure turbine total pressure ratio | |
| SFC | Specific fuel consumption | (lb/hr/lb) |
| T | Total temperature | °R |
| T2 | Engine face total temperature | °R |
| T21 | Fan exit total temperature | °R |
| TIT | High pressure turbine rotor inlet total temperature | °R |
| $\Delta T$ | Total temperature rise across combustor | °R |
| $\Delta h$ or $\Delta H$ | Change in total specific enthalpy | Btu/lbm |
| $\Delta h/T$-HPT | High pressure turbine work function | Btu/lb°R |
| $\Delta h/T$-LPT | Low pressure turbine work function | Btu/lbm°R |
| W | Flow rate | lbm/sec |
| WACC | Compressor corrected airflow ($W\sqrt{\theta 21}/\delta 21$) | lbm/sec |
| WAF | Engine (fan) actual airflow | lbm/sec |
| WAFC | Engine (fan) corrected airflow ($W\sqrt{\theta 2}/\delta 2$) | lbm/sec |
| $\theta 2$ | Ratio of fan inlet total temperature to standard sea level temperature (standard sea level temp. = 518.67°R) | |
| $\theta 21$ | Ratio of compressor inlet total temperature to standard sea level temperature | |

## NOMENCLATURE (CONCLUDED)

| | | UNITS |
|---|---|---|
| $\delta 2$ | Ratio of fan inlet total pressure to standard sea level pressure (standard sea level pressure = 14.696 psi) | |
| $\delta 21$ | Ratio of compressor inlet total pressure to standard sea level pressure | |

Subscripts

| | |
|---|---|
| AV | fan average |
| ID | inside (diameter) portion of fan supplying air to engine core |
| OD | outside (diameter) portion of fan supplying air to outer duct |
| b | combustor |
| DS | design |
| T | turbine (high pressure or low pressure) |

Station Numbers

| | |
|---|---|
| 2 | Engine (fan) entrance |
| 21 | Fan exit/compressor entrance |
| 3 | Compressor exit/combustor entrance |
| 4 | Combustor exit/high pressure turbine entrance |
| 5 | High pressure turbine exit/low pressure turbine entrance |
| 6 | Low pressure turbine exit |
| 8 | Core nozzle throat |
| 9 | Core nozzle exit |
| 23 | Duct burner entrance |
| 24 | Duct burner exit |
| 28 | Duct nozzle throat |
| 29 | Duct nozzle exit |

# SECTION I

# INTRODUCTION

Fighter aircraft today, and will to an increasing extent in the future, operate over a wide range of conditions and mission requirements. Propulsion systems designed for these aircraft must be able to produce high thrust levels at subsonic combat conditions, supersonic cruise, and acceleration. They must also operate efficiently at low power levels associated with subsonic cruise where a fighter can spend a great deal of time and burn much of its fuel.

The afterburning turbofan engine with a low bypass ratio is the propulsion system most often utilized for fighter aircraft. This engine type, however, demands low airflow levels at low power subsonic cruise. Since inlets and nozzles are sized to pass the high airflows at maximum power settings, high inlet spillage drag and nozzle aft-end drags result at subsonic cruise. At the high power levels, depending on the flight condition, this engine cannot operate at all its limits. That is, the engine may be operating at a fan corrected speed limit with turbine temperature below maximum levels, for example. This results in a certain loss in engine performance.

By utilizing variable area turbines, it will be possible to alleviate some of these problems of turbofan operation. To more effectively utilize the turbine variability, a duct burning separate flow cycle was selected for this study. In a mixed flow cycle, the requirement for static pressure balance between the duct and core streams at the mixing plane may limit the degree of turbine variability and cycle flexibility that can be achieved.

# SECTION II

# TECHNICAL APPROACH

## 1. ENGINE DESCRIPTION

An existing steady-state engine performance computer simulation of a fixed geometry turbofan engine was extensively modified to incorporate variable area turbines. An engine station schematic is shown in Figure 1. A description of the engine simulation is given in Section III.

The engine configuration is a two-spool separate flow turbofan incorporating a duct burner. No afterburning is done in the core stream. The duct nozzle throat area varies only during duct burning operation to maintain the intermediate power engine match. Intermediate power is the highest dry (nonduct burning) power level. Both turbines utilize variable stator vanes to vary the flow areas - a 40% area variation is possible. The turbine performance maps in the engine simulation are for a single stage high pressure (HP) and a two stage low pressure (LP) turbine. The core engine exhaust nozzle throat area is also variable. Variable area turbines allow control over the flow parameter $W\sqrt{T}/PA$ while a variable exhaust nozzle throat area allows control over the turbine expansion ratios (turbine work). A detailed description of how the engine operates is given in Section IV, "Results", and a detailed description of the turbine calculations and limits is given in Section III, "Engine Simulation Description".

The actual engine cycle which was used in this study is given in Table 1. The cycle parameters are those at sea level static, intermediate power.

TABLE 1

CYCLE PARAMETERS

| | |
|---|---|
| Engine Face Airflow | 200 lb/sec |
| Fan Pressure Ratio | 3.5 |
| Compressor Pressure Ratio | 5.75 |
| Bypass Ratio | 1.10 |
| Turbine Rotor Total Inlet Temp. (TIT) | 3430°R |

These cycle parameters are representative values for advanced fighter aircraft application. For any given aircraft, the actual optimum engine cycle is a complex function of engine performance, mission requirements, and aircraft design variables. In addition to the above cycle parameters, all engine rotational speeds were taken to be at their 100% levels.

In this study, two engines were simulated using the above cycle parameters, a variable area turbine version and a fixed geometry version. Performance and operating comparisons between the two versions can then be made and a preliminary idea of the advantages and disadvantages of utilizing variable area turbines in this engine type can be had.

In the course of all engine performance data generation the following engine operating limits were used throughout.

| | | |
|---|---|---|
| Maximum fan and compressor physical speed (N) | - | 100% |
| Maximum fan and compressor corrected speed ($N/\sqrt{\theta}$) | - | 100% |
| Maximum turbine rotor inlet total temp. (TIT) | - | 3430°R |

## 2. FLIGHT CONDITIONS

The flight conditions at which the engine simulation was run is given in Table 2.

TABLE 2

FLIGHT CONDITIONS

| | |
|---|---|
| Mach = 0.0, Altitude = 0.0 | - Max. Aug. to Part Power |
| Mach = 0.9, Altitude = 30,000 Ft | - Max. Aug. to Part Power |
| Mach = 0.6→2.0, Altitude = 36,000 Ft | - Max. Aug. |
| Mach = 1.6, Altitude = 50,000 Ft | - Max. Aug. to Min. Aug. |

At sea level static, the important operating condition is the maximum augmentation (max. aug.) point because maximum thrust is often used on take-off.

There are two important engine operating points at the 0.9 Mach, 30,000 ft altitude flight condition. The maximum power (max. aug.) point is a subsonic combat point. Here high thrust is very important and fighter aircraft engines are often sized by this point. Conventional turbofan engines are fan corrected speed limited at this condition with turbine temperature below maximum levels. With variable area turbines and core exhaust nozzle throat area, it was possible to run at the maximum turbine temperature with consequent thrust increase.

The other important engine operating point at the 0.9 Mach, 30,000 ft altitude flight condition is the subsonic cruise point, usually around 30% of maximum thrust. Conventional turbofan engines pass low airflow levels at this condition resulting in high inlet and aft-end drags. These low airflows are due to rematching of the turbomachinery at a low RPM. The variable turbine engine was run in the following fashion. As engine power (thrust) is reduced below intermediate levels by decreasing the turbine inlet temperature, the fan and compressor match points were held at their design levels by appropriately varying the turbine and core exhaust nozzle throat areas. This was done to maintain engine airflow, thereby reducing installation drags and yielding better propulsive efficiency. This constant fan and compressor match operation was maintained until a turbine limit was reached. When this happens, the engine operates in a conventional manner with no further changes in turbine and exhaust nozzle area. The engine can also be made to operate in other ways since the turbine variability allows essentially independent control of the shaft speeds. For example, at the 0.9 Mach, 30,000 ft altitude cruise condition, the fan match point can be held constant while allowing the compressor match point to drop. This would increase bypass ratio and reduce overall pressure ratio. This and other methods of engine operation were not investigated.

The maximum power points at Mach 0.6 to 2.0 at 36,000 ft altitude represent engine performance for aircraft acceleration. Again, the conventional engine cannot operate to all its engine limits. Throughout most of this Mach number range, the conventional engine is operating below maximum turbine temperature and at high Mach, fan speed is rolling off. Turbine and core exhaust nozzle throat areas were appropriately varied in the variable turbine engine so that maximum turbine temperature was achieved over most of the Mach number range and fan speed was kept at design levels in the high Mach regime. This resulted in greater thrust levels, especially in the high Mach regime.

The augmented power hook at Mach 1.6 and 50,000 ft altitude contains a representative supersonic cruise point. As in the above mentioned acceleration run, the variable turbine engine was able to operate at maximum turbine temperature and fan speed resulting in increased thrust relative to the fixed geometry engine.

# SECTION III

# ENGINE SIMULATION DESCRIPTION

## 1. FAN

The fan average and fan core (ID) performance is represented by two maps of the form shown in Figure 2a where the normalized pressure ratio (PRN) is a ratio of pressure ratios defined by

$$PRN = \frac{PRF-1}{PRF_{DS}-1}$$

In this form, the maps are normalized to the design point. The design point values are those values which the user inputs as sea level static intermediate values. Entry to the maps is through %N1C and PRN.

With pressure ratio, airflow, and efficiency known the thermodynamic conditions at the fan exit can be calculated. Bleeds are accounted for.

The outer portion of the fan which exhausts into the duct (OD stream) is calculated by assuming that the fan average is an airflow average of the core and duct stream, so that:

$$PRF_{OD} = \frac{(PRF_{AV})\ WAF_{AV} - (PRF_{ID})\ WAF_{ID}}{WAF_{OD}} \tag{1}$$

$$T21_{OD} = \frac{(T21_{AV})\ WAF_{AV} - (T21_{ID})\ WAF_{ID}}{WAF_{OD}} \tag{2}$$

$$\eta F_{OD} = \frac{PRF_{OD}^{\ 0.286} - 1}{(T21_{OD} - T2)\ /T2} \tag{3}$$

## 2. COMPRESSOR

The compressor calculations are very similar to the fan calculations. There is one map (Figure 2a).

3. COMBUSTOR

The combustor performance is represented by a map of the form shown in Figure 2b where $\Delta T$ is the burner total temperature rise. Entry to the map is through $\Delta T$ and P3 with efficiency $\eta_b$ being output.

The combustor pressure drop is a function of a design pressure drop (input by the user) and a ratio of corrected flow to design corrected flow. Thermodynamic conditions are calculated using gas tables routines which take into account fuel/air mixtures and dissociation.

4. TURBINES

Both turbine routines use similar calculation procedures and use a map of the form shown in Figure 3. The high pressure turbine map is for a single stage turbine. For the low pressure turbine, a map that actually represents a two-stage high pressure turbine was used. This will probably mean that efficiencies can be in error by up to several points. However, this does not substantially affect the engine operating characteristics and the overall trends are still valid even if the numbers are not exact. A more serious problem was the fact that the LP turbine work limit could only be roughly estimated. An indication that the LP turbine is approaching a work limit is a high Mach number at the discharge. In this study, it was decided to use a limiting discharge Mach number of around 0.6 at Station 6 in lieu of knowing the actual load limit of the LP turbine.

Going back to the turbine calculations, a value for A/AMIN must be specified at the design point for both turbines. Setting this value too close to either the upper or lower area limit (A/AMIN = 1.4 and A/AMIN = 1.0, respectively) will lead to reduced turbine area variation at certain flight conditions. Deciding on an appropriate value is something of an art and some initial trial and error is necessary. It was found that at 0.9 Mach and 30,000 ft altitude as engine power is reduced below intermediate power, the HP turbine area closed down while the LP turbine area opened up. Since the subsonic cruise point was one of the more important flight conditions, it was decided to set the A/AMIN values so that no area limit would be reached at this condition. These values were 1.25 for the HP turbine and 1.15 for the LP turbine.

At the design point, the fan and compressor operating points, the HP turbine rotor inlet total temperature (TIT), and values for A/AMIN for both turbines must be specified. Using the specified compressor operating point and TIT, the program calculates values for $\Delta h/T$ and $\Delta h/N^2$ for the HP turbine. It should be mentioned here that $\Delta h$ represents a change in total specific enthalpy; static enthalpies are not used. With this information, the program enters the HP turbine map to obtain values for $W\sqrt{T}/PA$ and $\eta_T$. From the thermodynamic conditions at the HP turbine entrance, a value for the parameter $W\sqrt{T}/P$ is calculated so that the turbine area A at design can be obtained. The program now has enough information to calculate the thermodynamic properties at the HP turbine exit and proceeds to the LP turbine where these calculations are repeated. At the design point, values for turbine efficiency $\eta_T$ for both turbines must also be input so that a scale factor, as defined in Reference 1, is formed between the input values and the values obtained from the maps. Actually, however, these input values were chosen to equal the map values so that the scale factors were unity.

At off-design points (i.e., all points other than the design point at sea level static), the program can operate in either a fixed turbine or a variable turbine mode of operation. In the variable turbine mode, the fan and compressor operating points (pressure ratio and rotational speed) and a turbine rotor inlet temperature TIT are input to the program. This sets the fan and compressor work requirements. The program uses an iterative procedure to find the turbine areas and core exhaust nozzle throat area which maintain the required levels of $W\sqrt{T}/PA$ and turbine work $\Delta h/T$. The actual calculation procedures are summarized in the Appendix. This appendix also summarizes the fixed turbine mode calculations.

## 5. DUCT BURNER

The calculations are similar to those of the main burner. In addition to a dry pressure loss calculation, a momentum (hot) pressure loss is also calculated based on Rayleigh heat addition equations.

An efficiency map is included in the engine simulation, the map being a set of curves showing duct burning efficiency as a function of duct burner inlet pressure, temperature, Mach number, and duct burner fuel/air ratio.

## 6. EXHAUST NOZZLES

The calculations for both core and duct nozzles are based on one-dimensional gas dynamics. Both are convergent-divergent nozzles. The duct nozzle throat area varies during duct burning operation so that the intermediate power engine match is maintained--it is fixed at the design value during dry operation. During variable turbine mode operation, the core nozzle throat area varies to allow for the proper turbine expansion ratios--it is fixed during fixed turbine mode operation. During all engine operation, both nozzle exit areas are allowed to vary for optimum expansion.

## 7. ENGINE COMPONENT MATCHING

For a proper engine match, the following must hold:

Fan rotational speed=LP turbine rotational speed

Compressor rotational speed=HP turbine rotational speed

Fan power = LP turbine power

Compressor power + horsepower extraction = HP turbine power

Flow continuity throughout the engine

All matching equations are basically expressions of these five requirements.

For any given operating condition for which the program is to generate data, certain initial estimates regarding the engine match are made inside the program to start calculations. Since these estimates result in unmatched components (matching errors), an iterative procedure taken from Reference 1 is used whereby several passes are made through the engine refining the estimates until all the matching requirements are satisfied.

Reference 1. McKinney, John S., Simulation of Turbofan Engine (Parts I and II); Air Force Aero Propulsion Laboratory, Wright-Patterson Air Force Base; Technical Report AFAPL-TR-67-125, November 1967.

# SECTION IV
# RESULTS

## 1. PERFORMANCE AT SEA LEVEL STATIC

Figure 4 shows the thrust-SFC performance from maximum augmentation down through part power for both the variable turbine and the fixed turbine engine. At power settings intermediate and above, the fixed turbine engine has slightly better performance because of somewhat lower turbine cooling requirements and better turbine efficiencies. Variable turbines suffer a small efficiency penalty due to variable vane tip clearance losses. Part power SFC is better for the variable turbine engine because the fan and compressor match points were maintained at design levels as engine power was reduced. This kind of operation is described in detail immediately below.

## 2. PERFORMANCE AT MACH = 0.9, ALTITUDE = 30,000 FT

Figures 5 through 13 depict the operating and performance characteristics of both the variable turbine and fixed turbine engines.

During part power operation below intermediate power, the fan and compressor operating points for the conventional fixed turbine engine move down an operating line as shown in Figures 5 and 6. With variable turbines, it was possible to maintain the fan and compressor match points at their design levels while turbine inlet temperature was reduced until the LP turbine exit Mach number limit was reached. When this limit was reached, the variable turbine engine was then operated in a conventional manner and the fan and compressor operating points move down an operating line. As was stated previously, the reason for maintaining the fan and compressor match was to maintain high engine airflow (high propulsive efficiency and lower installation drag losses) and high pressure ratio (high thermal efficiency). At a typical cruise power setting of about 30% maximum thrust, the variable turbine engine has about 20% greater airflow and 30% higher pressure ratio.

Variable turbines also benefit the engine at power levels above intermediate. For flight conditions where $\theta 2$ is less than 1, the fixed turbine engine cannot operate at its maximum turbine inlet temperature without exceeding fan and compressor corrected speed limits. Turbine area variability allows operation at the maximum turbine inlet temperature without exceeding fan or compressor limits. This is indicated in Figure 7 where the TIT increase at high power settings is about 270°.

Figure 7 also shows that at a typical cruise power setting of about 30% maximum thrust, the variable turbine engine operates at about 200° lower TIT than the fixed turbine engine. Since a fighter aircraft spends a significant portion of its time at subsonic cruise, engine life may be increased by operation at this lower temperature.

The actual changes in the turbine parameters and exhaust nozzle areas that occur when the engine is operated in this fashion are shown in Figures 8 through 12. As thrust is reduced below intermediate power by decreasing turbine inlet temperature, the flow parameter $W\sqrt{T}/P$ at the HP turbine inlet decreases by $\sqrt{T}$ (W and P are essentially constant since the fan and compressor match points are constant). The HP turbine area must then decrease to maintain $W\sqrt{T}/PA$ since the turbine is choked. This is indicated in the upper portion of Figure 8.

At the LP turbine, conditions are somewhat more complex. Here, the flow parameter $W\sqrt{T}/P$ is increasing so that to maintain $W\sqrt{T}/PA$, the LP turbine area must also increase. The flow parameter at the LP turbine is increasing because P is decreasing faster than $\sqrt{T}$. This is because the pressure ratio of the upstream HP turbine is rapidly increasing. The HP turbine pressure ratio is rapidly increasing because the turbine is extracting essentially constant work at progressively lower turbine inlet temperatures. While all this is happening, the core exhaust nozzle throat area is opening up to decrease the backpressure and thereby permit the turbine pressure ratios to increase.

Figure 8 shows the actual turbine area variations for both the HP and LP turbine. Note that neither turbine has reached a physical area limit (1.0 and 1.4 A/AMIN). This is because the LP turbine exit Mach number (M6) reached the limiting value of 0.6 (see Figure 7). As was stated earlier, this was taken to be an indication of limit loading.

Figure 9 shows how $\Delta h/T$ for the turbines is changing. Below intermediate power settings where the turbine temperatures are decreasing and match points are held constant, the turbine work $\Delta h$ remains relatively constant so that $\Delta h/T$ increases. In a conventional fixed turbine engine, $\Delta h/T$ remains relatively constant. Note that at high power settings (intermediate and above) the $\Delta h/T$ values for the variable turbine engine are less than those for the fixed turbine engine. This is because the turbine temperatures are higher.

Figure 10 shows how the turbine pressure ratios are changing. The trends are similar to the $\Delta h/T$ trends since these are essentially equivalent expressions.

Figure 11 shows the variations in core and duct nozzle throat areas. As thrust is reduced below intermediate by means of reducing TIT, the core nozzle throat area of the variable turbine engine opens up permitting the increases in turbine pressure ratios (and $\Delta h/T$) mentioned earlier. The core nozzle throat area of the fixed turbine engine is fixed at the design level. The duct nozzle throat areas for both engines behave in a similar fashion and vary only during duct burning, remaining fixed during dry operation.

Figure 12 shows the variations in exhaust nozzle exit areas for optimum expansion. The duct nozzles for both engines behave in a similar fashion but the core nozzles behave differently. The fixed turbine engine core nozzle exit area is decreasing during part power operation because airflows are decreasing. In the variable turbine engine which is maintaining higher airflows, the core nozzle exit area remains relatively high. Depending on the type of engine installation, this may mean that the aft-end drag is lower for the variable turbine engine with consequent improvement in installed performance at subsonic cruise.

Figure 13 finally presents the overall uninstalled engine performance (Thrust and SFC) at Mach 0.9 and altitude 30,000 ft for the two engines. At the maximum augmentation condition, the variable turbine engine has about 4.3% greater thrust. This small gain is due to the fact that the variable turbine engine is able to operate at maximum TIT. The installation drag penalties should be similar for both engines since both are passing similar airflows. Often, this flight condition, which is a combat point, is a sizing point for fighter aircraft turbine engines. Therefore, since the variable turbine engine is putting out greater thrust at this condition, lower engine and aircraft weights may result with using variable turbine engines. Of course, this gain may be offset by the fact that addition of variable geometry in the turbines means inherently greater weight.

The other benefit of variable turbine engines is in reducing subsonic cruise SFC. Figure 13 indicates that the variable turbine engine has about 5.4% better SFC at a typical cruise power setting of 30% maximum thrust. This uninstalled SFC improvement is due to the higher propulsive and thermal efficiencies of the variable turbine engine at cruise power caused by operating at higher airflows and pressure ratios. If inlet and nozzle drags were taken into account, the SFC improvement would be greater since the variable turbine engine is passing about 20% more airflow.

3. PERFORMANCE AT ACCELERATION AND SUPERSONIC CRUISE

At these flight conditions, both the variable turbine and the conventional engine are at maximum power (maximum augmentation). The variability of the turbines and core exhaust nozzle throat area were used to make the engine operate at as many of its limits as possible--physical speed, corrected speed, and turbine inlet temperature. Figures 14 through 20 illustrate the performance and operation of the two engines during acceleration conditions.

Figure 14 shows the thrust and SFC comparison. At the lowest Mach number of 0.6, the variable turbine engine has about 7½% greater thrust. This thrust improvement decreases to zero around Mach 1.2-1.4 and then increases again up to about 34% improvement at Mach 2. At the high Mach condition, the variable turbine engine suffers its greatest SFC penalty of only about 2%.

The reasons for these performance gains can be seen from Figures 15 through 17. Figure 15 shows fan physical and corrected speed as a function of Mach number. Figure 16 shows the compressor physical and corrected speed as a function of Mach number. Figure 17 shows engine corrected airflow and turbine inlet temperature as a function of Mach number. At low Mach numbers (below about 1.2), both engines are operating at fan and compressor corrected speed limits and, therefore, also have the same physical speeds. The variable turbine engine, however, operates at higher turbine temperatures which results in a thrust increase. Even the variable turbine engine was not able to operate at maximum TIT at the very lowest Mach numbers because the HP turbine reached a physical area limit of 1.4 A/AMIN.

At Mach numbers greater than about 1.2 the conventional fixed turbine engine reaches a compressor physical speed limit, and fan physical speed and TIT roll off. The variable turbine engine, however, was able to run to maximum TIT and maximum fan physical speed (Figures 15 and 17). Corrected fan speed still rolls off but this is because engine face temperature is increasing with Mach number. Because fan speed is maintained at higher levels, the engine airflow is also greater (Figure 17). The net effect of being able to operate at higher TIT and airflow levels is to increase performance by the large percentage indicated earlier. Unfortunately, these higher airflows may result in a larger inlet, depending on inlet design and sizing point, which will increase the installation drags at subsonic cruise. There may be a tradeoff, therefore, between subsonic and supersonic performance. At about Mach 1.2-1.4, both engines are operating at nearly the same TIT, fan, and compressor speeds so that both have about the same performance.

Referring to Figure 16, at high Mach, the compressor corrected speed for the variable turbine engine is lower than that of the fixed turbine engine even though the compressor physical speeds are the same. This is because the fan speed of the variable turbine engine is greater resulting in a higher fan pressure ratio which in turn results in higher compressor inlet temperatures. This brings down the compressor corrected speed.

The turbine and exhaust nozzle variations that are necessary to operate the variable turbine engine in this fashion are illustrated in Figures 18 through 20. Figure 18 shows the turbine area variations. Figure 19 shows the corrected turbine work variations and Figure 20 shows the core nozzle throat and exit area variations. These variations are determined by the simultaneous variations in Mach number, fan, and compressor physical and corrected speeds, and TIT. The turbine area variations are those necessary to keep $W\sqrt{T}/PA$ relatively constant in both turbines since they are choked. The core nozzle throat area varies to allow operation at the required turbine pressure ratios and work levels. At the very low Mach numbers (0.6-0.8), the HP turbine area reached the upper limit of 1.4 A/AMIN and the engine could not operate at maximum TIT.

At Mach numbers below about 1.4 where both engines have similar fan and compressor operating points, the $\Delta h/T$ values for both turbines in the variable turbine engine are lower than those in the fixed turbine engine. This is because the turbine temperatures are higher in the variable turbine engine. At Mach numbers above 1.4, both engines have similar compressor operating points resulting in similar work requirements ($\Delta h$), but the TIT in the variable turbine engine is greater so that $\Delta h/T$ for the variable HP turbine is smaller (Figure 19). The fan operating point for the variable turbine engine is at a higher speed than that of the fixed turbine engine resulting in a higher work requirement ($\Delta h$). This increase in fan $\Delta h$ is greater than the increase in LP turbine temperature causing a new increase in $\Delta h/T$ for the variable LP turbine (Figure 19).

Figure 20 shows that the core nozzle throat area is opening up with increasing Mach number to allow the turbines to operate at the required turbine expansion ratios. This figure also shows the variation in core nozzle exit area for optimum expansion. Up to about Mach 1.4, both engines have similar values. At higher Mach numbers, the flow rate is greater in the variable turbine engine because of higher fan speeds. This results in a larger exit area.

Figure 21 shows the thrust-SFC characteristics from minimum to maximum augmentation for the Mach 1.6, altitude 50,000 ft flight condition. Again the variable turbine engine has higher thrust because it is able to operate at maximum fan physical speed and maximum TIT while the fixed turbine engine cannot. At a typical supersonic cruise condition of about 50% augmentation, the variable turbine engine has about 9% SFC improvement at the same thrust level.

# SECTION V

# CONCLUSIONS

Variable area turbines offer some performance benefits for a duct-burning turbofan engine. On an uninstalled basis, these benefits are relatively small (4-8% thrust or SFC improvement) at most flight conditions. At the high Mach maximum power condition, substantial thrust increases in excess of 25% are possible. If installation losses were to be included, the SFC benefits could increase for low power operation such as subsonic cruise. As mentioned in the previous section, a trade-off may have to be made between subsonic and supersonic performance even with this variable turbine engine concept. This is due to the fact that increasing engine airflow to obtain greater thrust at high Mach numbers may result in a larger inlet (if the inlet is sized by high Mach operation). A larger inlet will increase the installation drag losses at subsonic cruise.

The next logical step in the performance analysis of this engine type would be to examine these data to help determine guidelines for more realistic turbine designs. New maps based on these turbine designs should be included. It was mentioned in Section II that other methods of engine operation are possible which were not investigated (e.g., maintaining fan match while allowing the compressor match point to drop at the 0.9 Mach, 30,000 ft altitude cruise condition). These methods should be investigated. Finally, inlet and aft-end drag maps should also be incorporated. Realistic aft-end drag maps may be difficult to obtain because not much data is available for concentric nozzle installations. Using this information, installed performance comparisons can be made against other engine types and a better assessment of the performance potential of a variable turbine duct burning turbofan engine can be made.

## APPENDIX

## TURBINE CALCULATIONS

In the variable turbine mode of operation, the fan and compressor operating points (pressure ratio and rotational speed) and high pressure turbine rotor inlet total temperature TIT are input. Using this information, the program calculates required values for $\Delta h/T$ and $\Delta h/N^2$ for the high pressure turbine. It should be mentioned here that $\Delta h$ represents a change in total specific enthalpy; static enthalpies are not used. It then starts with an initial estimate for the HP turbine area A (it uses the value from the previous run point) and enters the map to obtain values for $W\sqrt{T}/PA$ and $\eta_T$. The thermodynamic conditions at the HP turbine entrance also yield a value for $W\sqrt{T}/PA$. This value is compared to that obtained from the map; if the values are unequal, the program chooses a new value for A and repeats this process until convergence. The program then calculates the thermodynamic properties at the HP turbine exit and proceeds to the LP turbine where these calculations are repeated.

Upon reaching the LP turbine exit, the program has obtained values for the turbine areas, flow rate, and for a pressure in the tailpipe. The core nozzle throat area is then set to allow operation at this tailpipe pressure. This is what actually lets the turbines extract the required work from the gas.

During the variable turbine mode of operation, two types of turbine limits must be considered. One limit is a loading limit. As was stated in the main body of this report, an upper limit on LP turbine exit Mach number was used. The other limit is an actual physical area variation limit governed by the bounding values for A/AMIN of 1.0 and 1.4. If either limit is reached, the engine is operated in a fixed turbine mode using the turbine areas and exhaust nozzle throat area attained at this point.

The conventional fixed turbine mode of operation is the same as that in Reference 1. The fan and compressor operating points are not input but are calculated from the turbomachinery matching equations. Typically, TIT is input and the program makes initial estimates for the fan and compressor operating point. The program also makes an initial estimate for the $\Delta h/T$ for both turbines. With this information, the program obtains values from the maps for $W\sqrt{T}/PA$ and $\eta_T$ for both turbines. However, since the thermodynamic conditions at the turbine entrances and the work required can be calculated, values for $\Delta h/T$ and $W\sqrt{T}/PA$ can be obtained. These values are compared to the map values and "matching errors" are generated if unequal. As is stated in the main body of this report, an iterative procedure taken from Reference 1 is used to reduce these "matching errors" to zero.

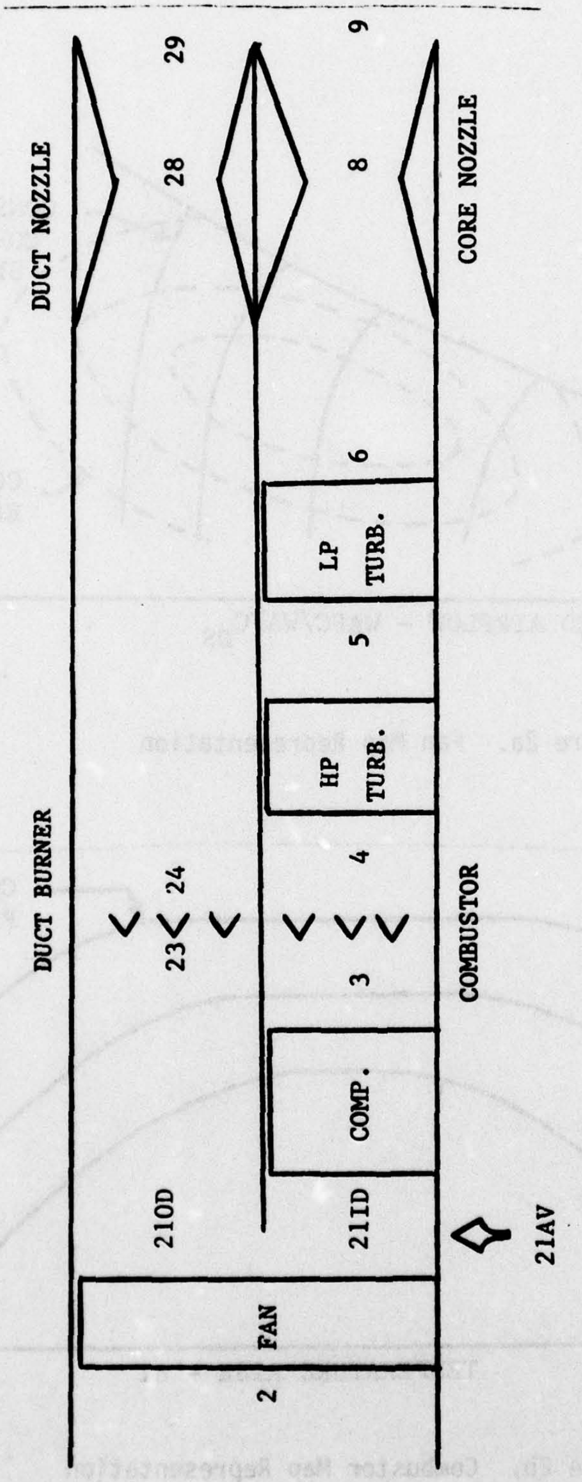


Figure 1. Engine Schematic

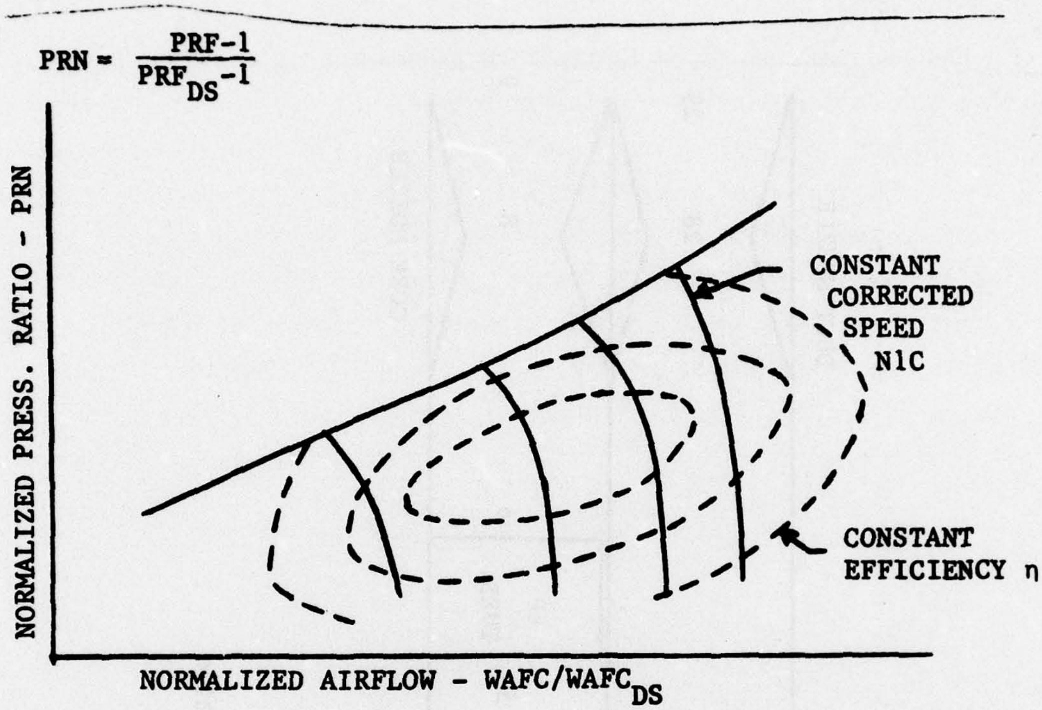


Figure 2a. Fan Map Representation

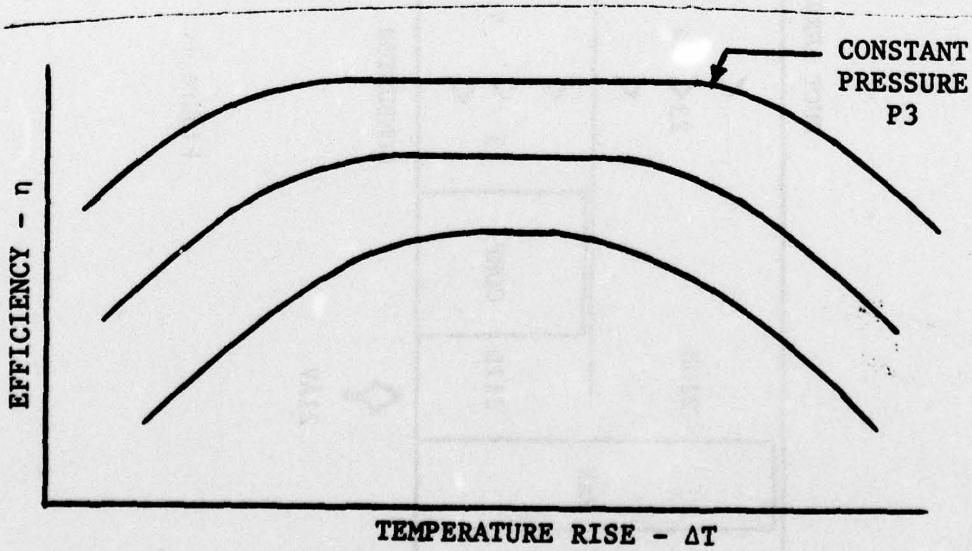


Figure 2b. Combustor Map Representation

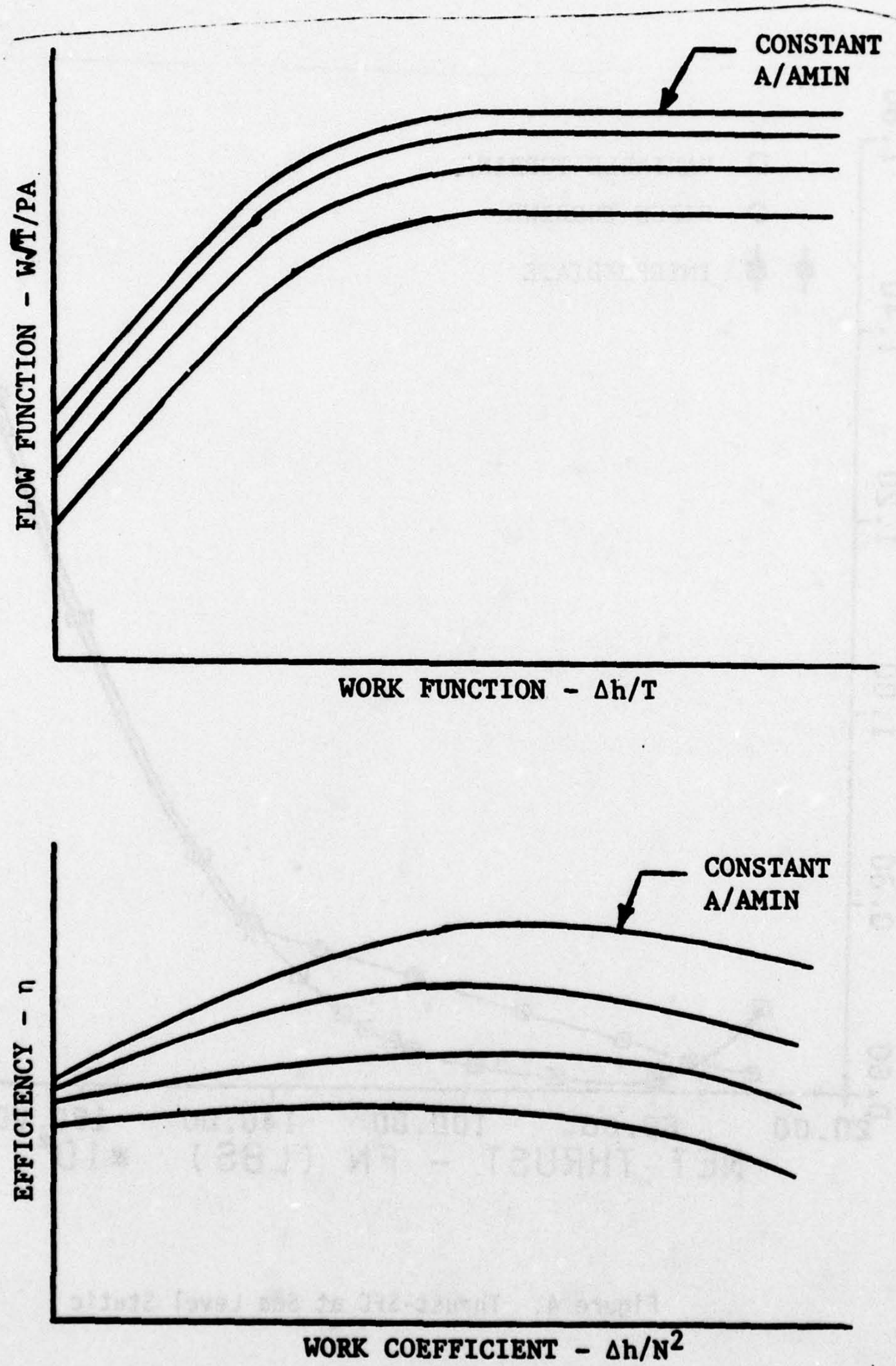


Figure 3. Turbine Map Representation

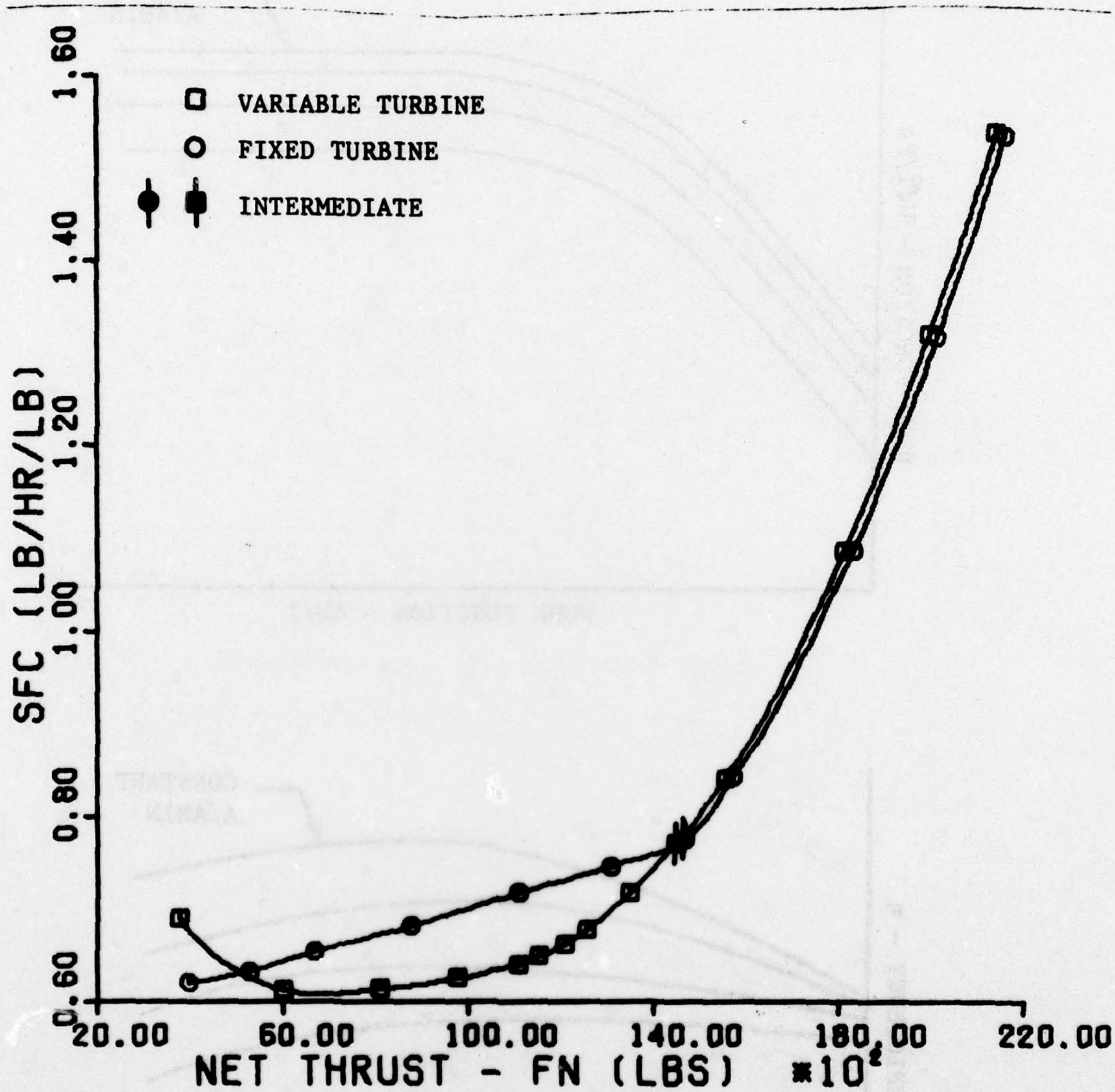


Figure 4. Thrust-SFC at Sea Level Static

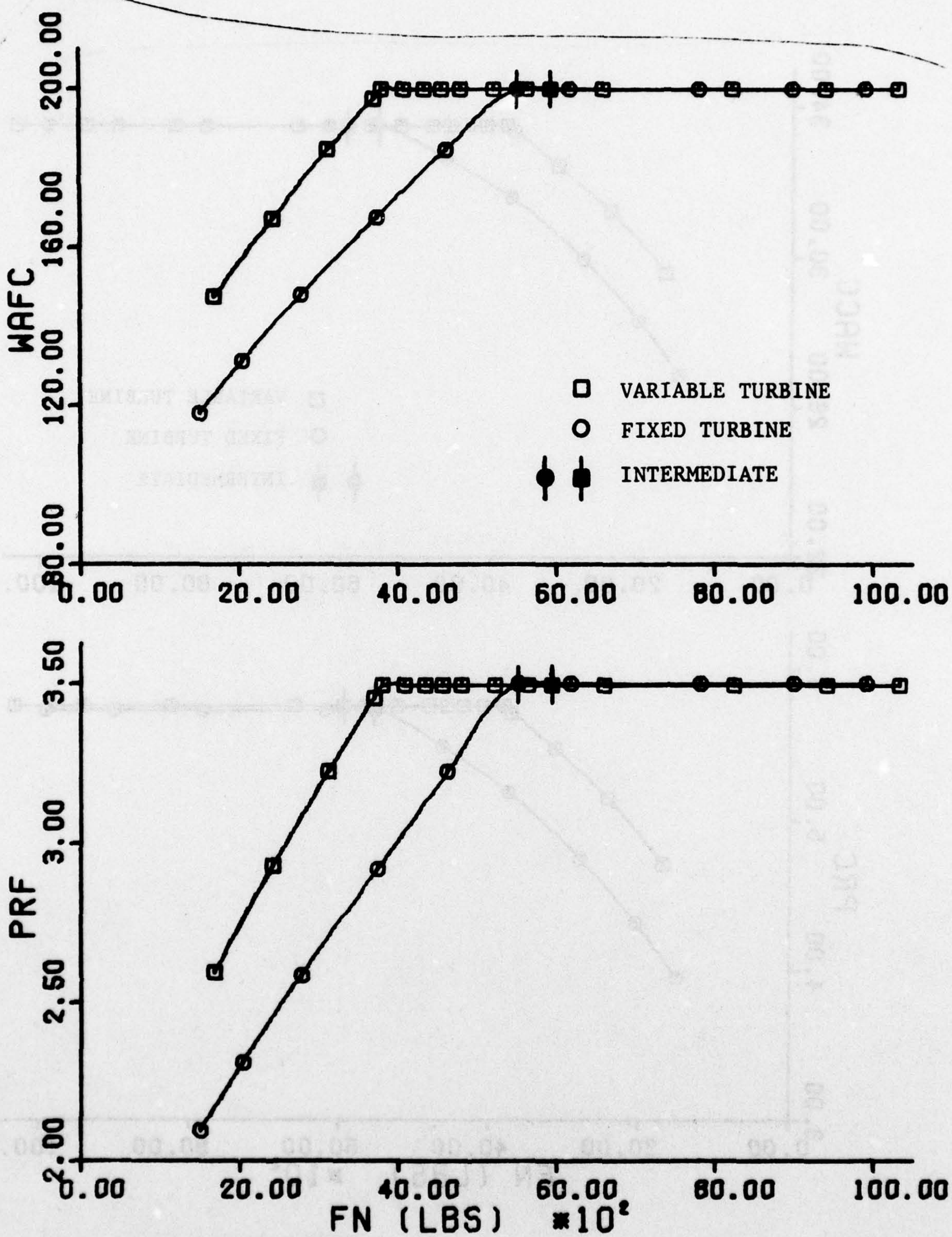


Figure 5. Fan Operating Characteristics at 0.9 Mach, 30,000 Ft Altitude

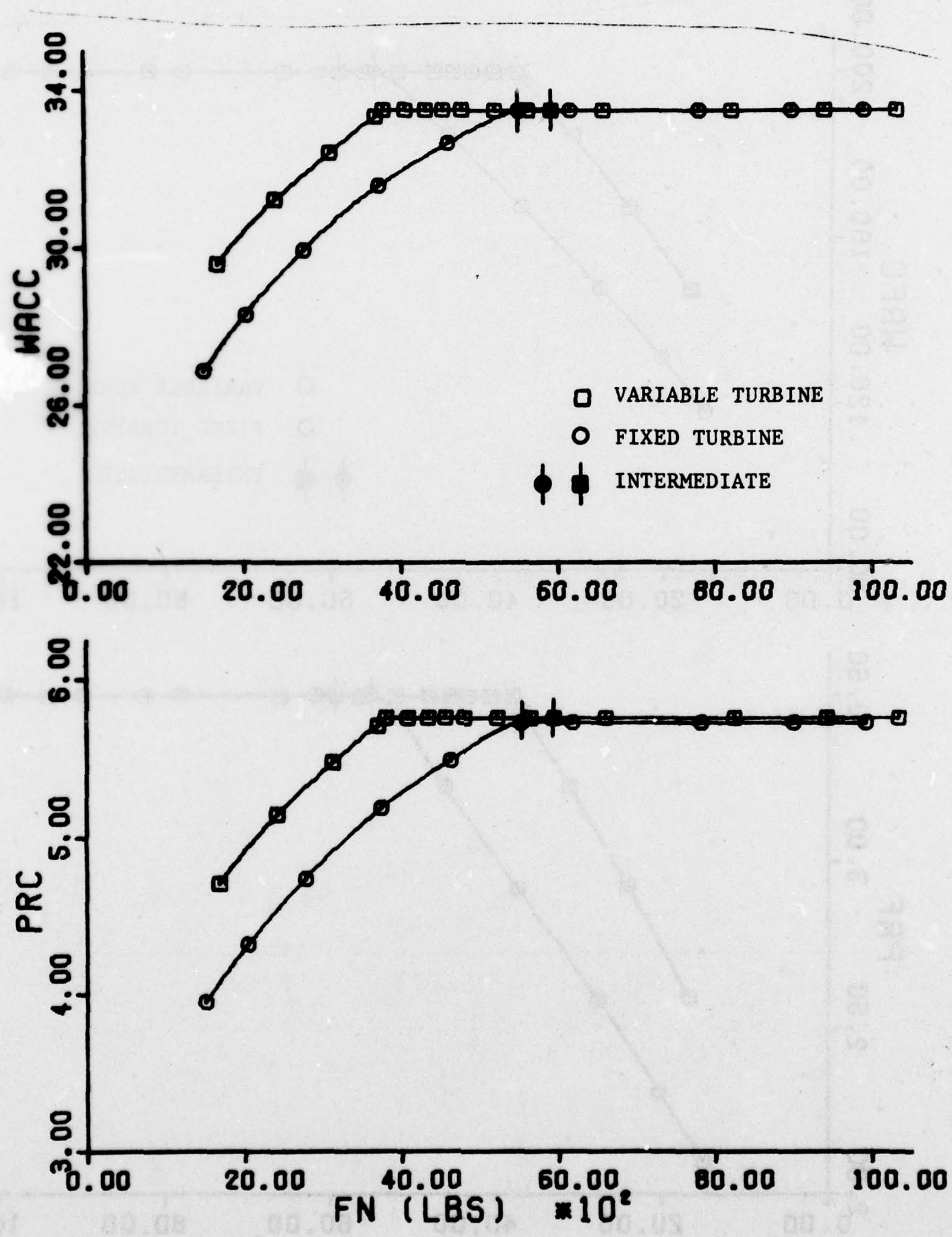


Figure 6. Compressor Operating Characteristics at 0.9 Mach, 30,000 Ft Altitude

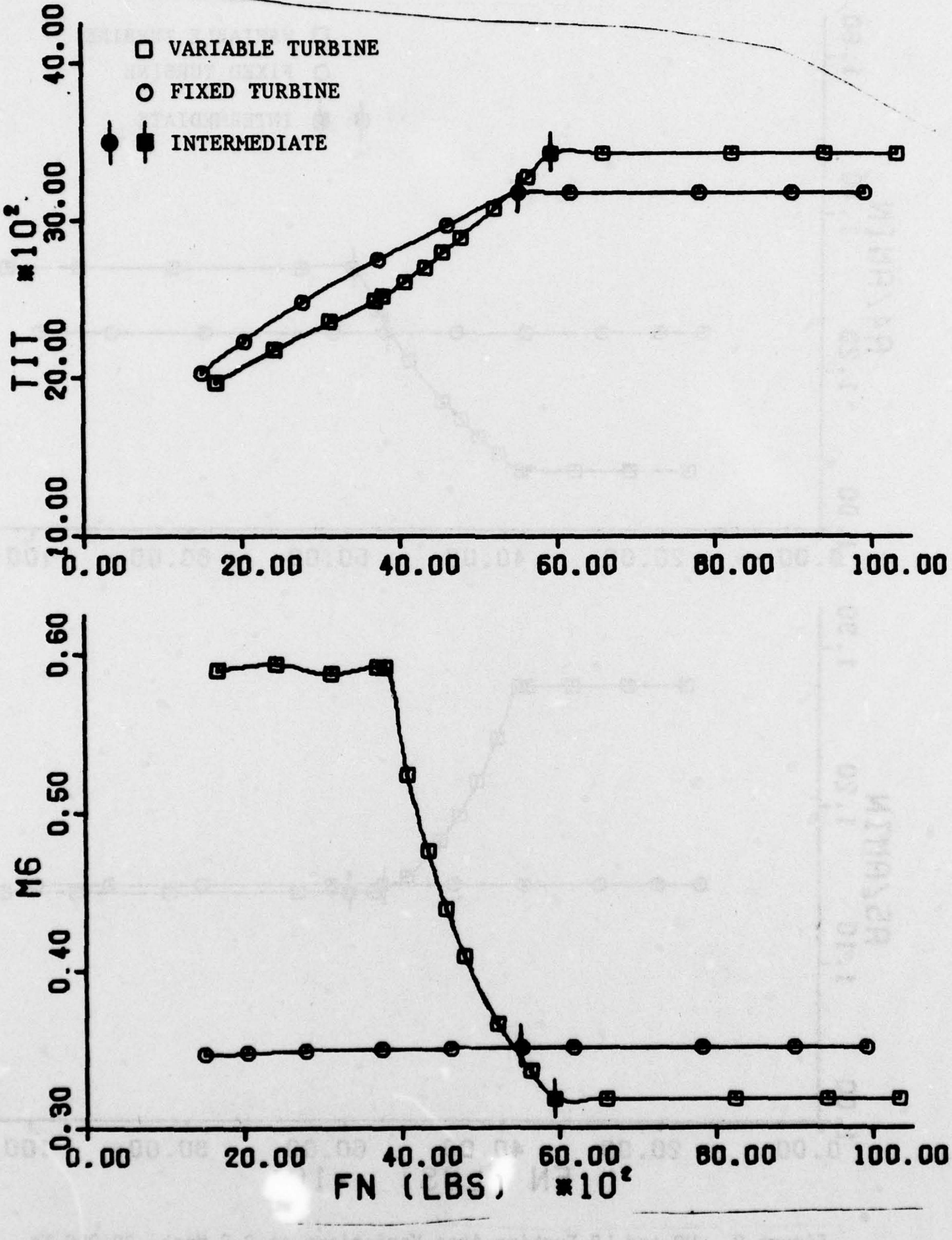


Figure 7. Turbine Rotor Inlet Temp. and LP Turbine Exit Mach No. at 0.9 Mach, 30,000 Ft Altitude

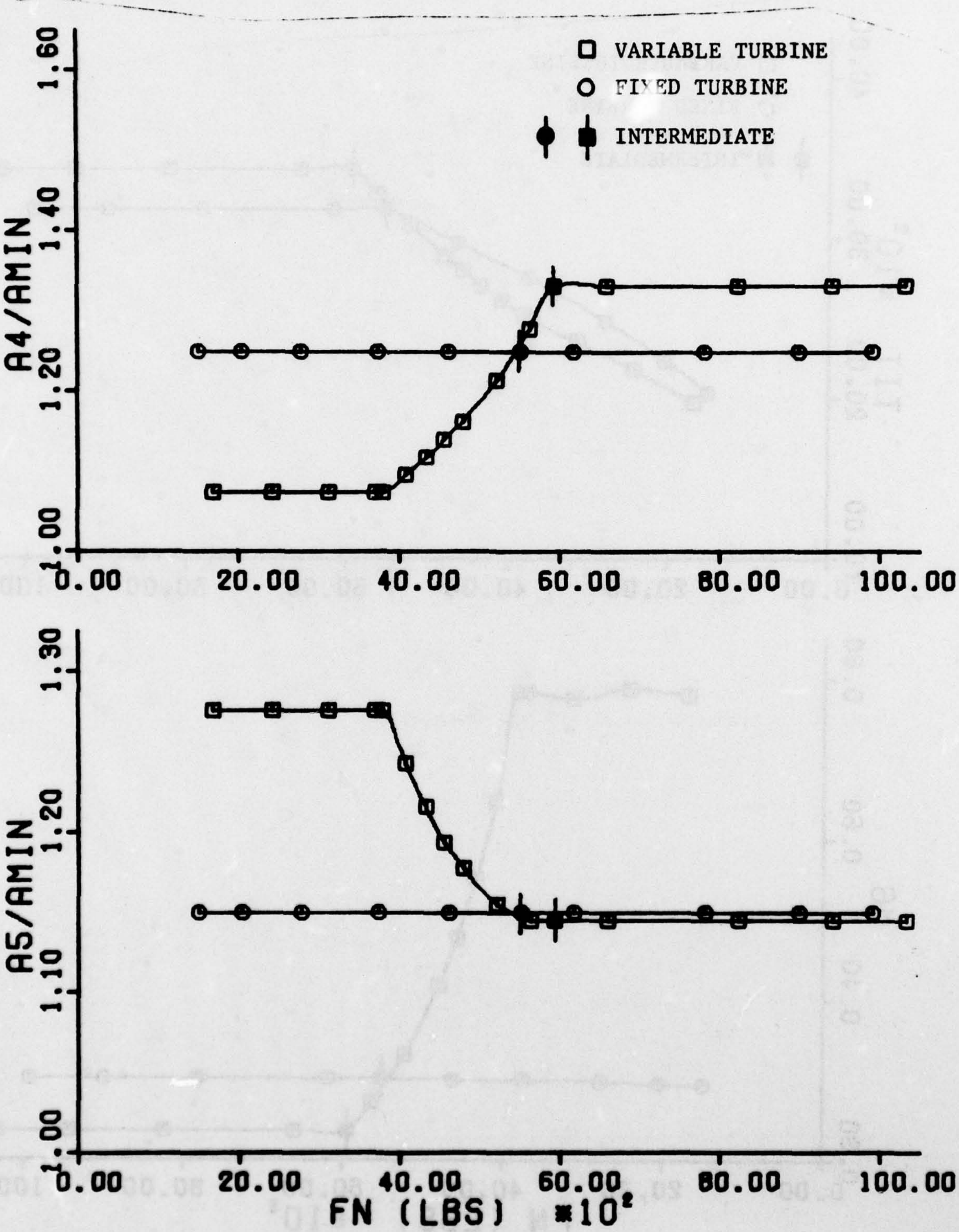


Figure 8. HP and LP Turbine Area Variations at 0.9 Mach, 30,000 Ft Altitude

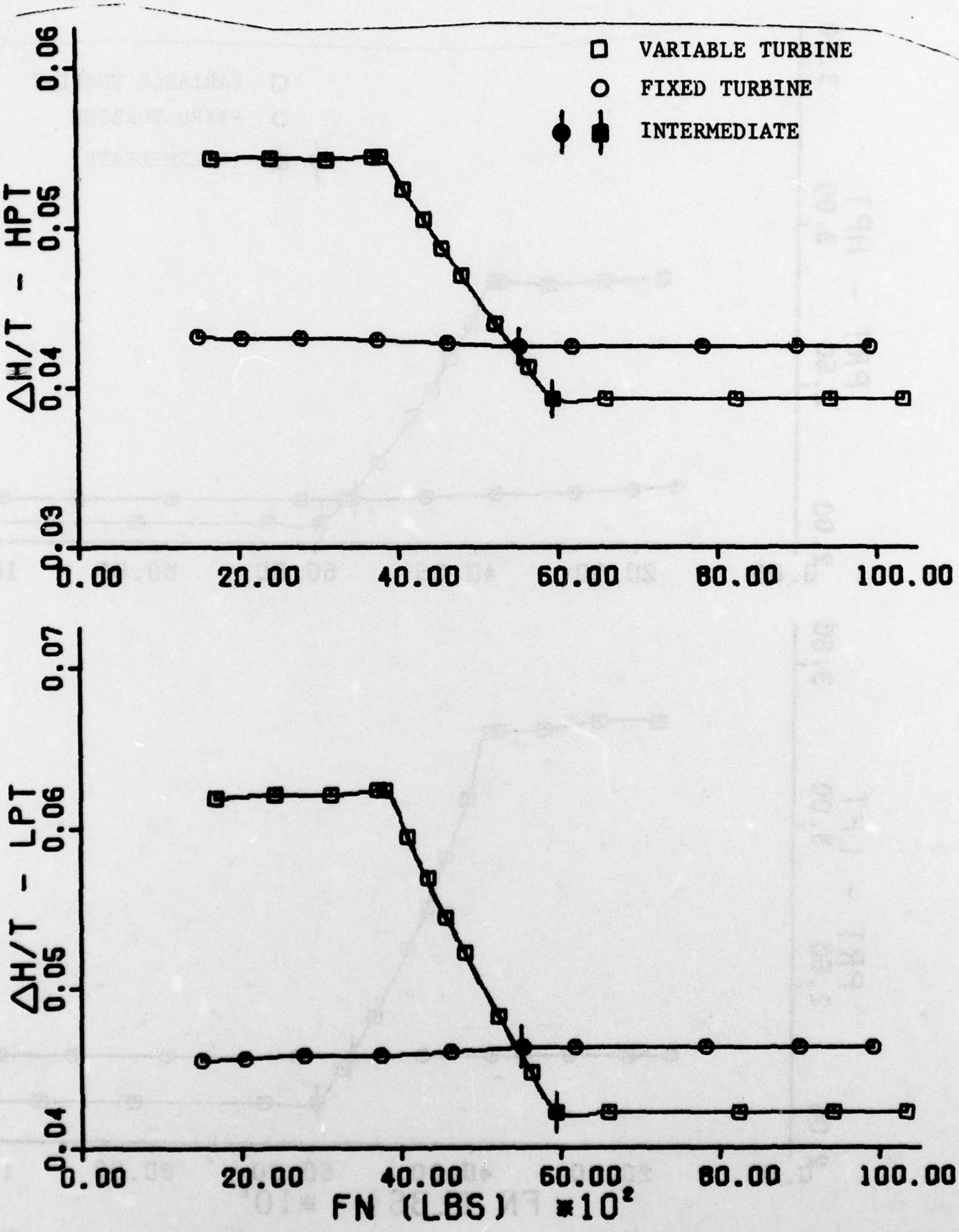


Figure 9. HP and LP Turbine Work Functions at 0.9 Mach, 30,000 Ft Altitude

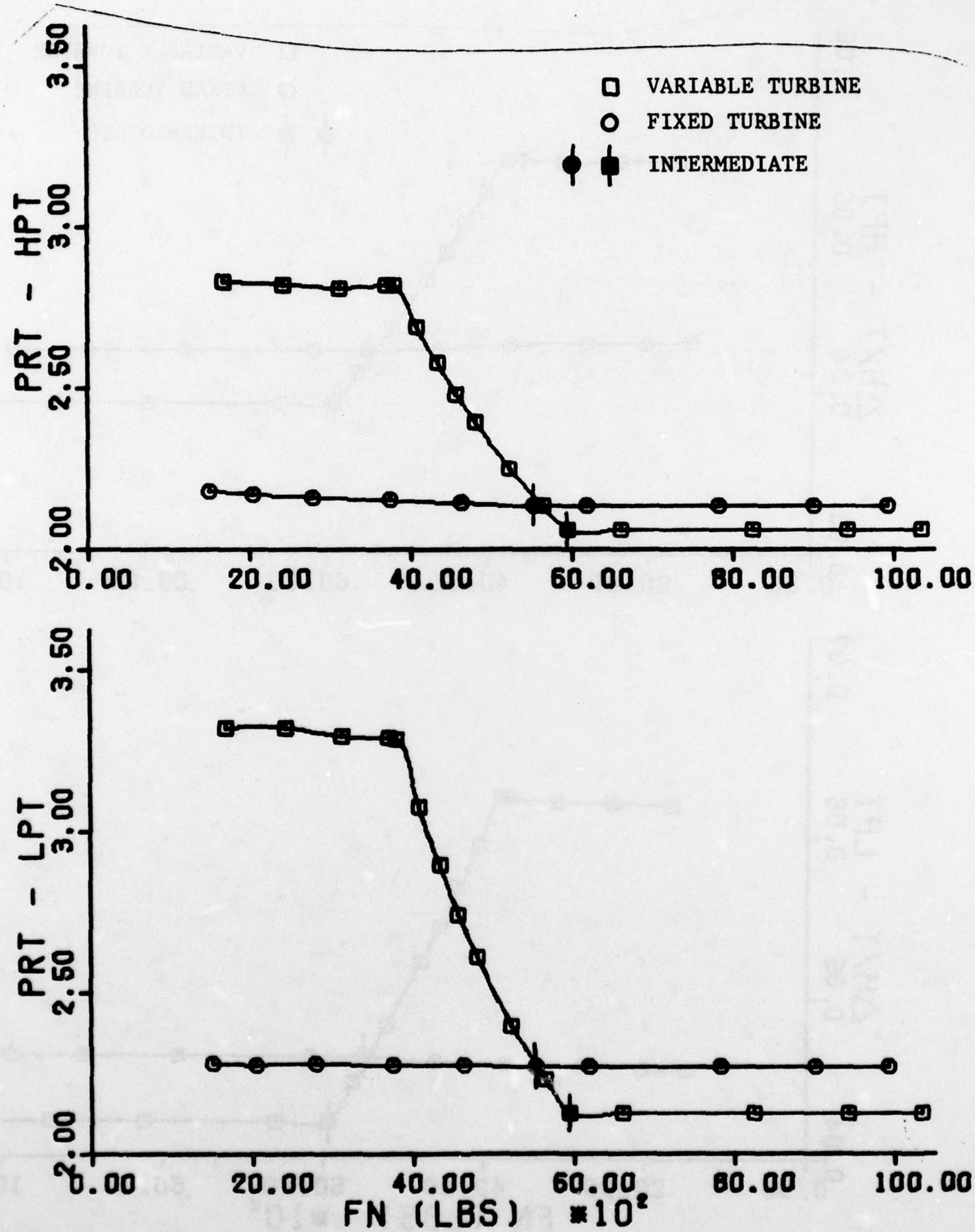


Figure 10. HP and LP Turbine Pressure Ratios at 0.9 Mach, 30,000 Ft Altitude

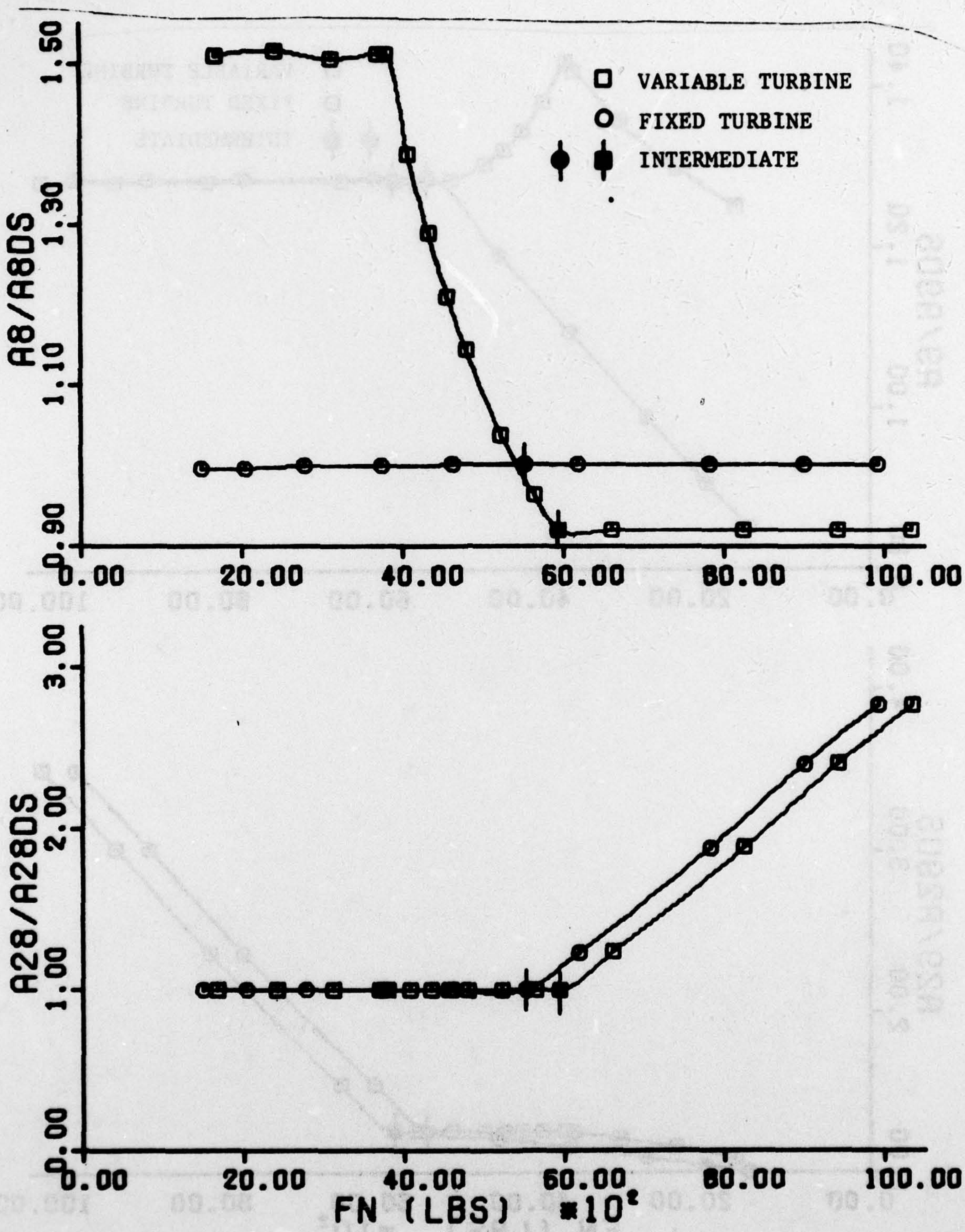


Figure 11. Core and Duct Exhaust Nozzle Throat Areas at 0.9 Mach, 30,000 Ft Altitude

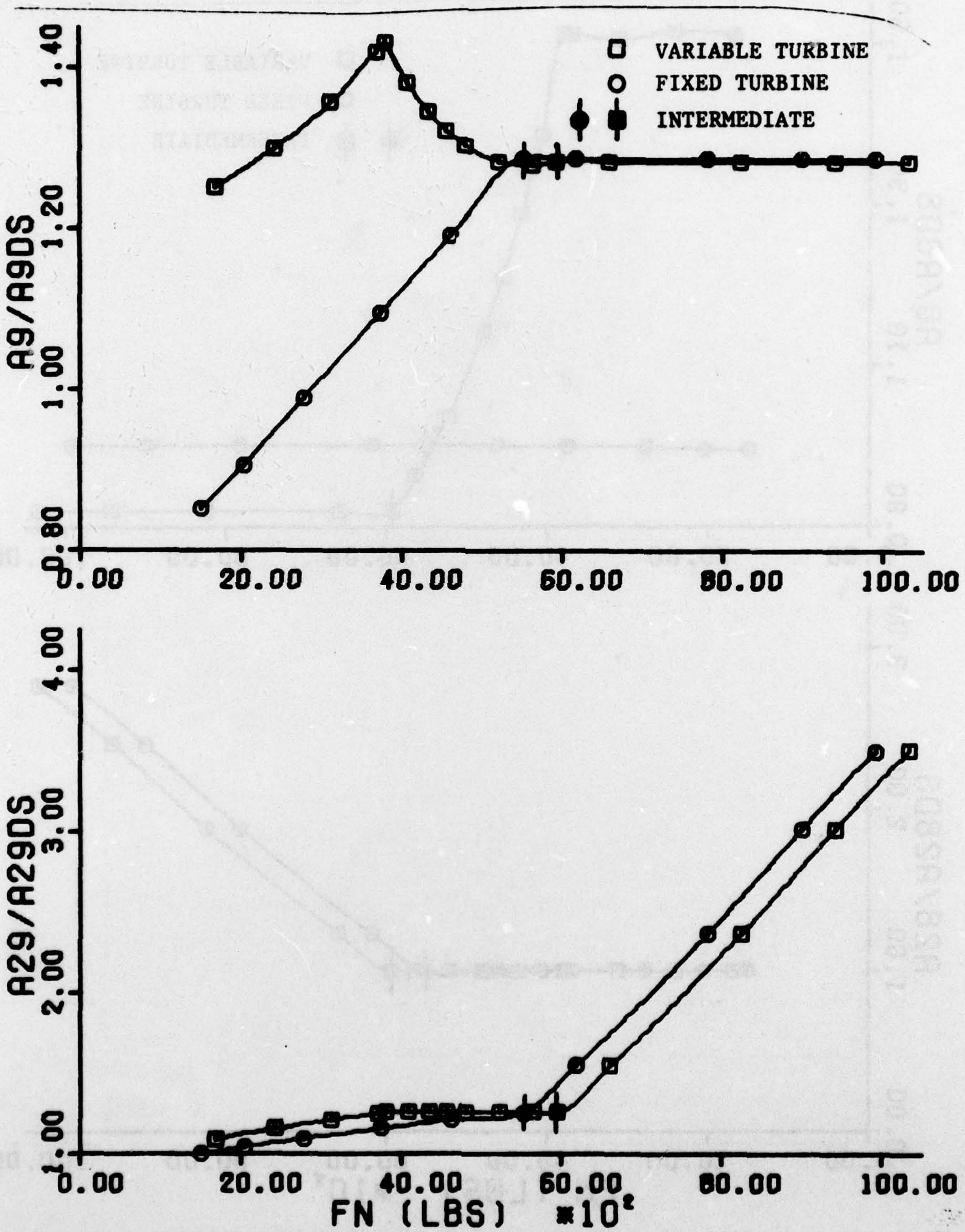


Figure 12. Core and Duct Exhaust Nozzle Exit Areas at 0.9 Mach, 30,000 Ft Altitude

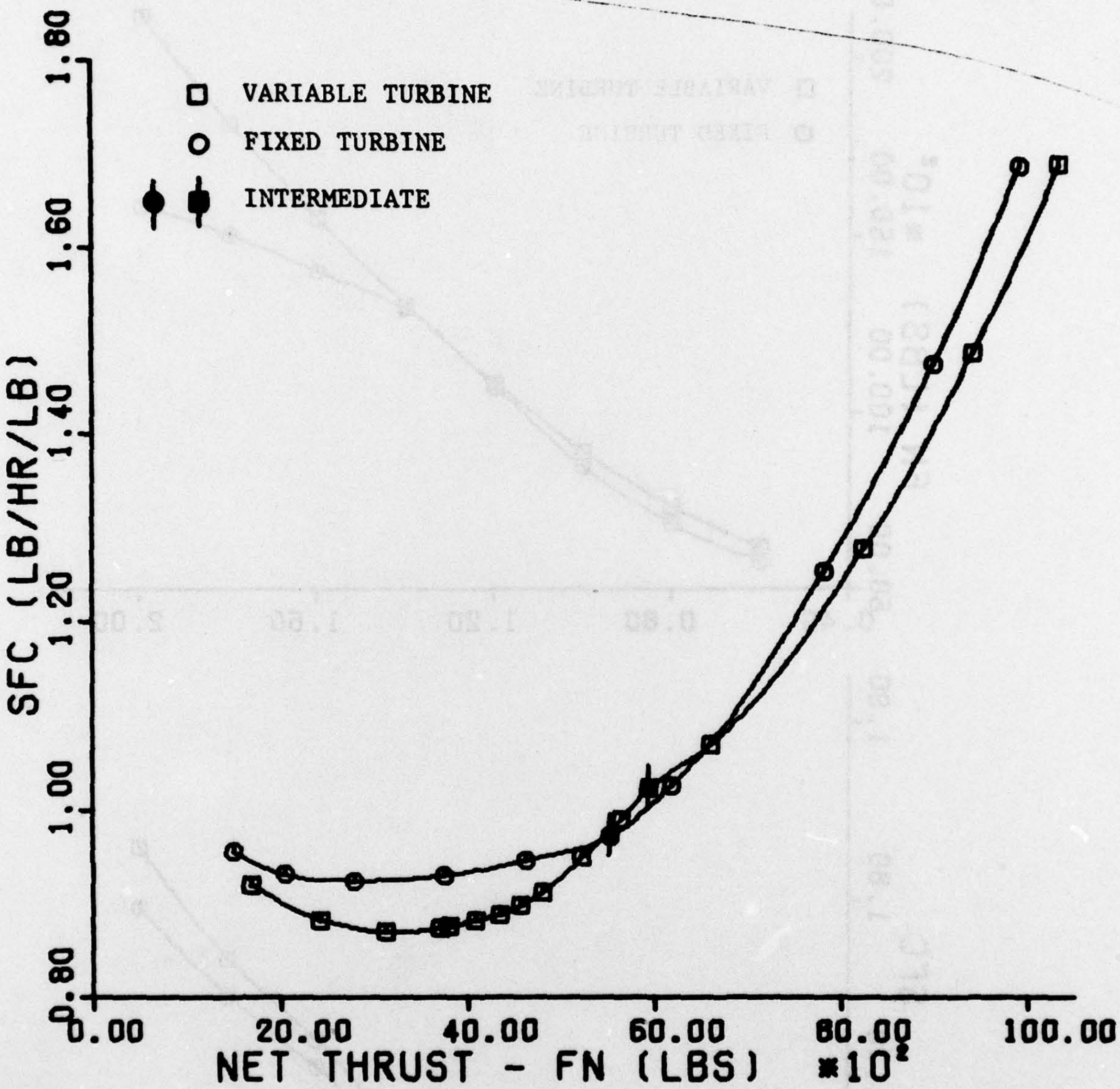


Figure 13. Thrust-SFC at 0.9 Mach, 30,000 Ft Altitude

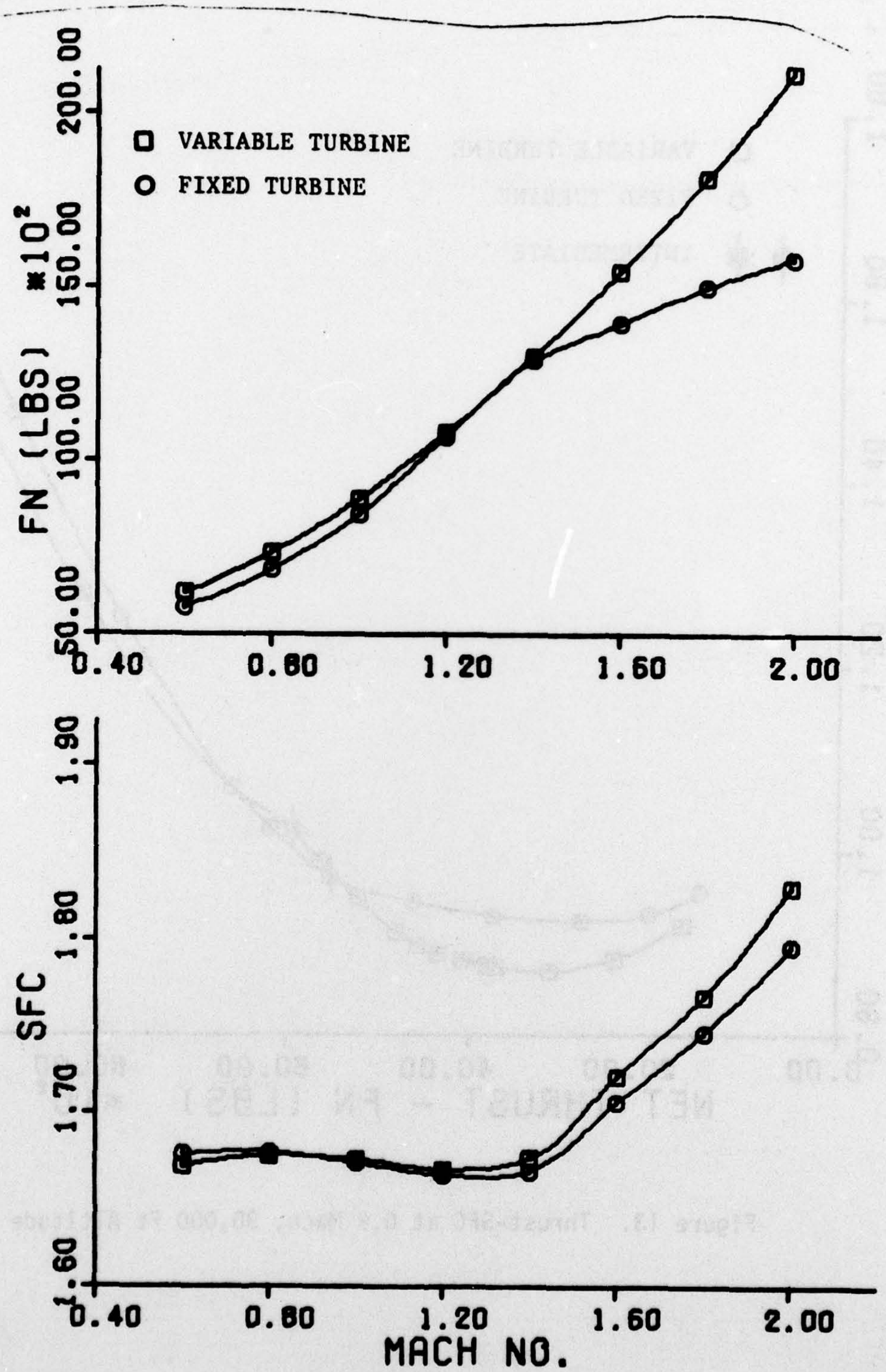


Figure 14. Thrust and SFC During Acceleration at 36,000 Ft Altitude

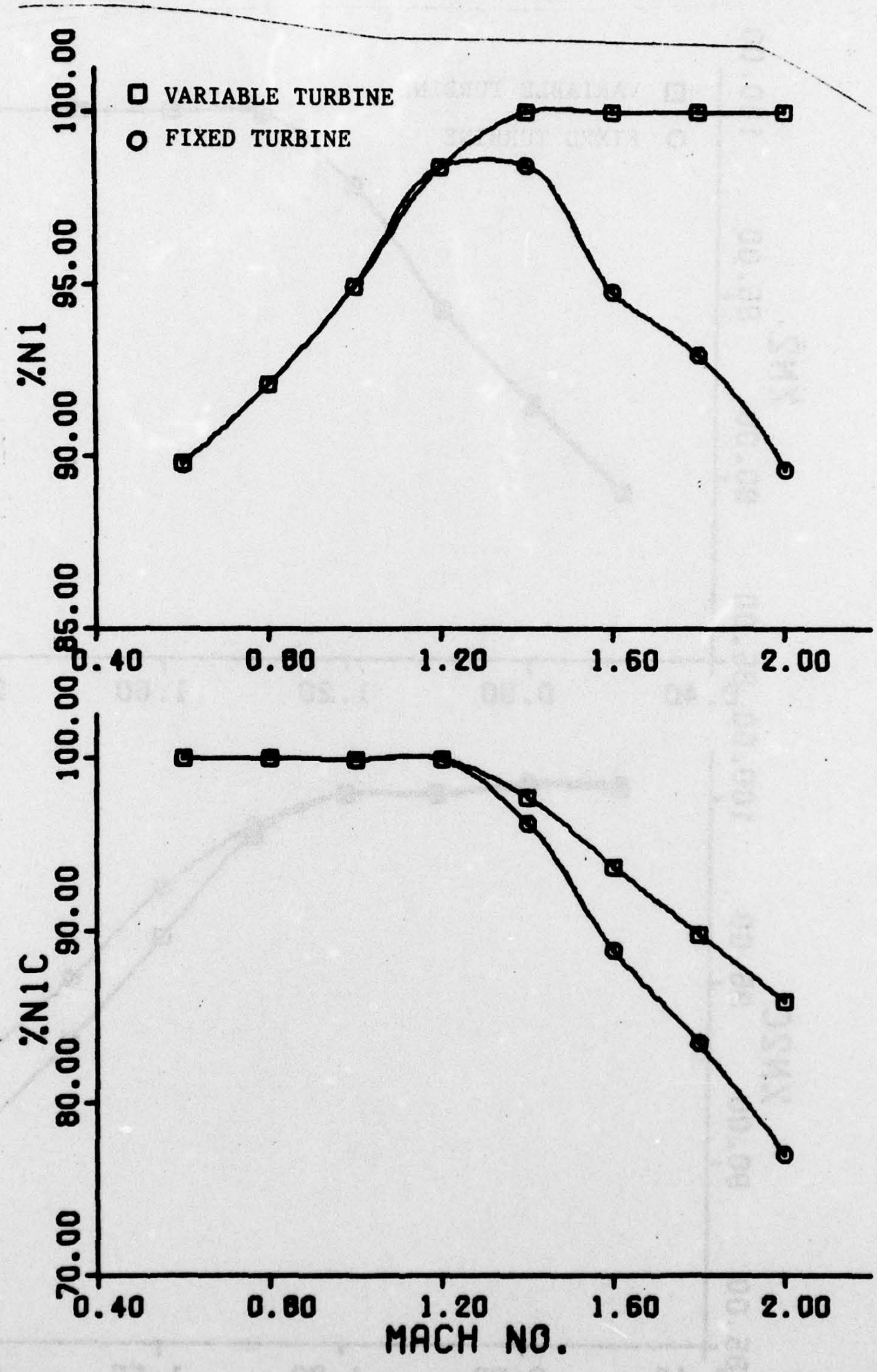


Figure 15. Fan Physical and Corrected Speed During Acceleration at 36,000 Ft Altitude

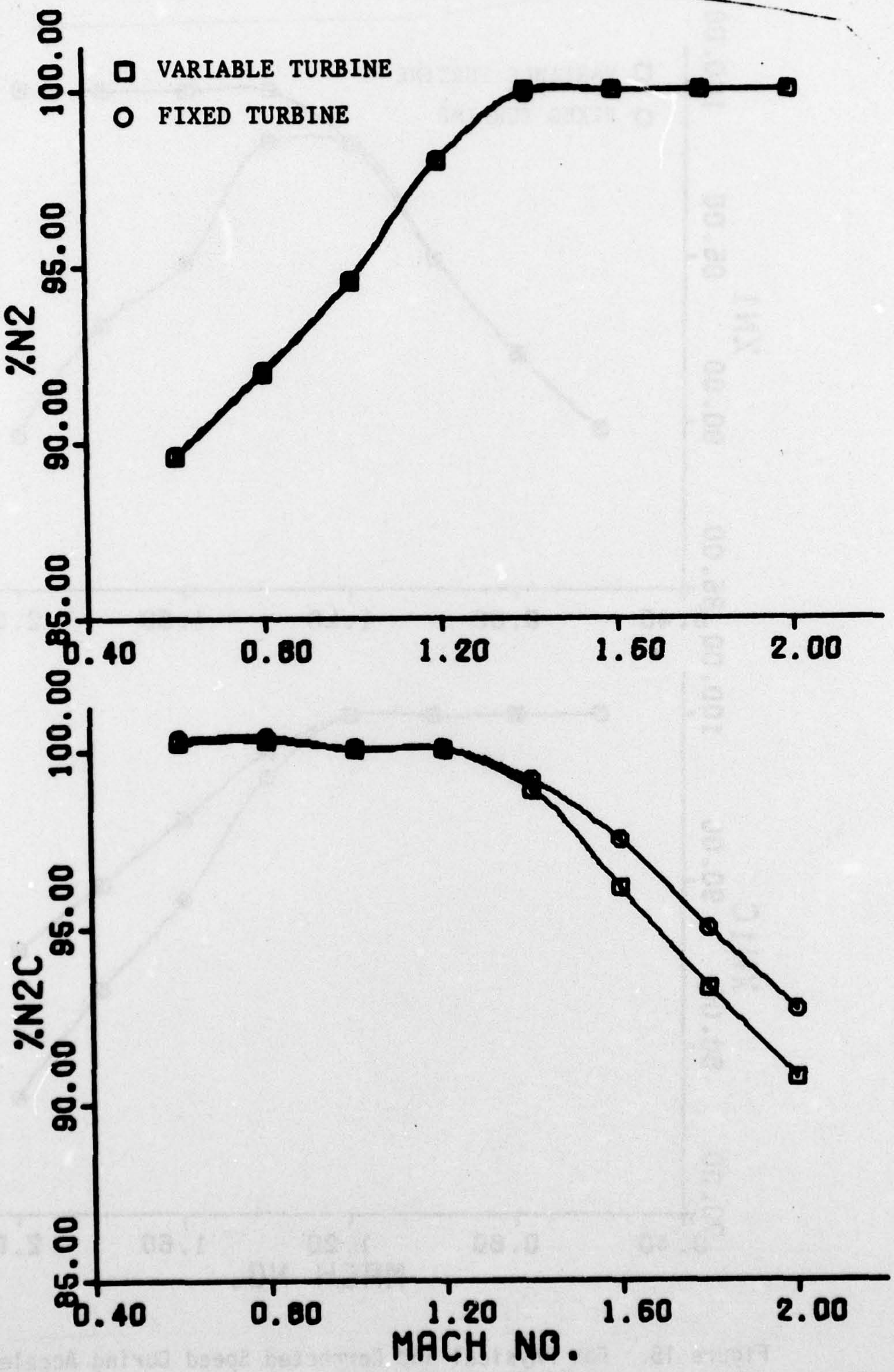


Figure 16. Compressor Physical and Corrected Speed During Acceleration at 36,000 Ft Altitude

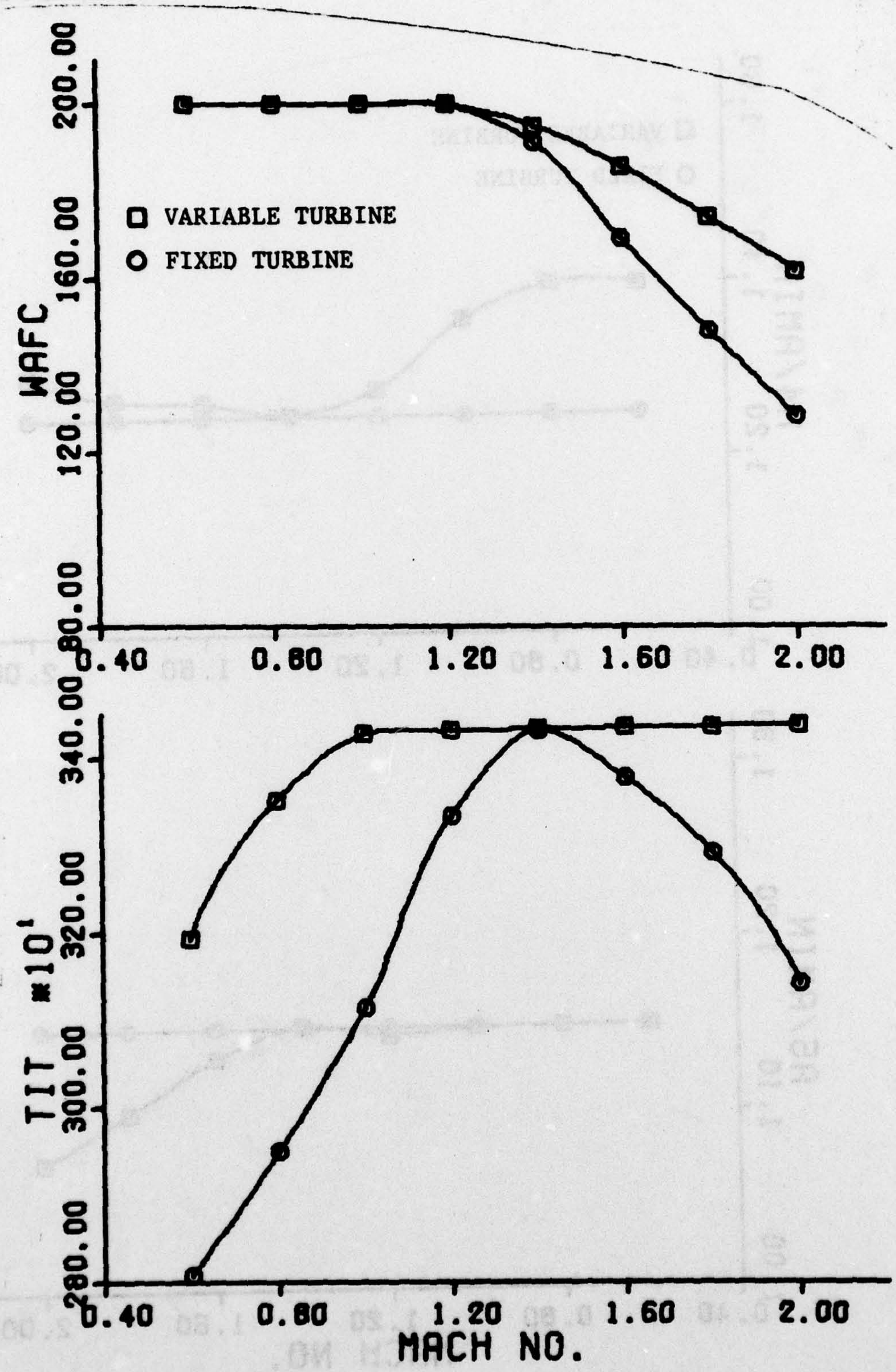


Figure 17. Engine Corrected Flow and Turbine Rotor Inlet Temperature During Acceleration at 36,000 Ft Altitude

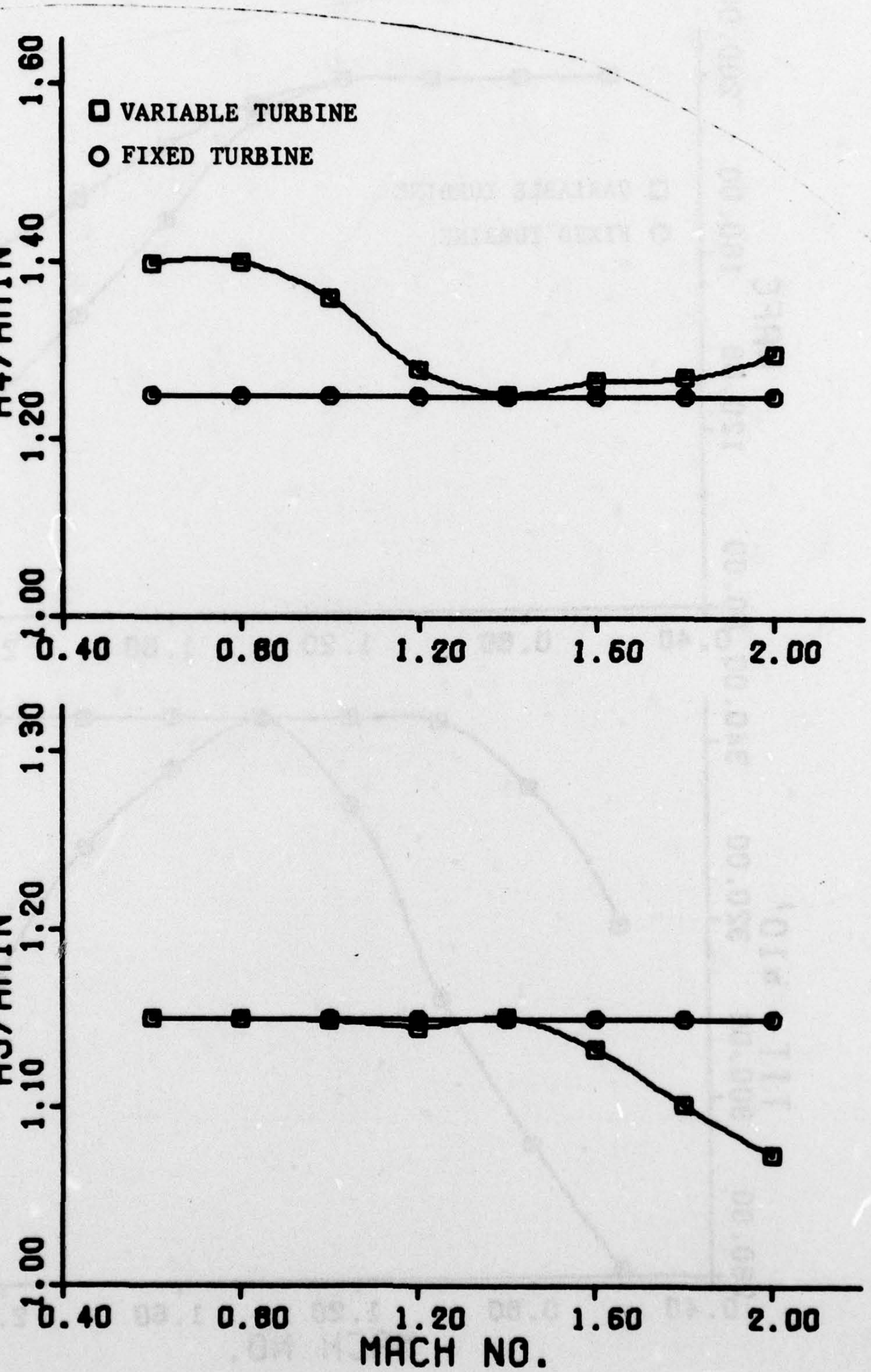


Figure 18. HP and LP Turbine Area Variations During Acceleration at 36,000 Ft Altitude

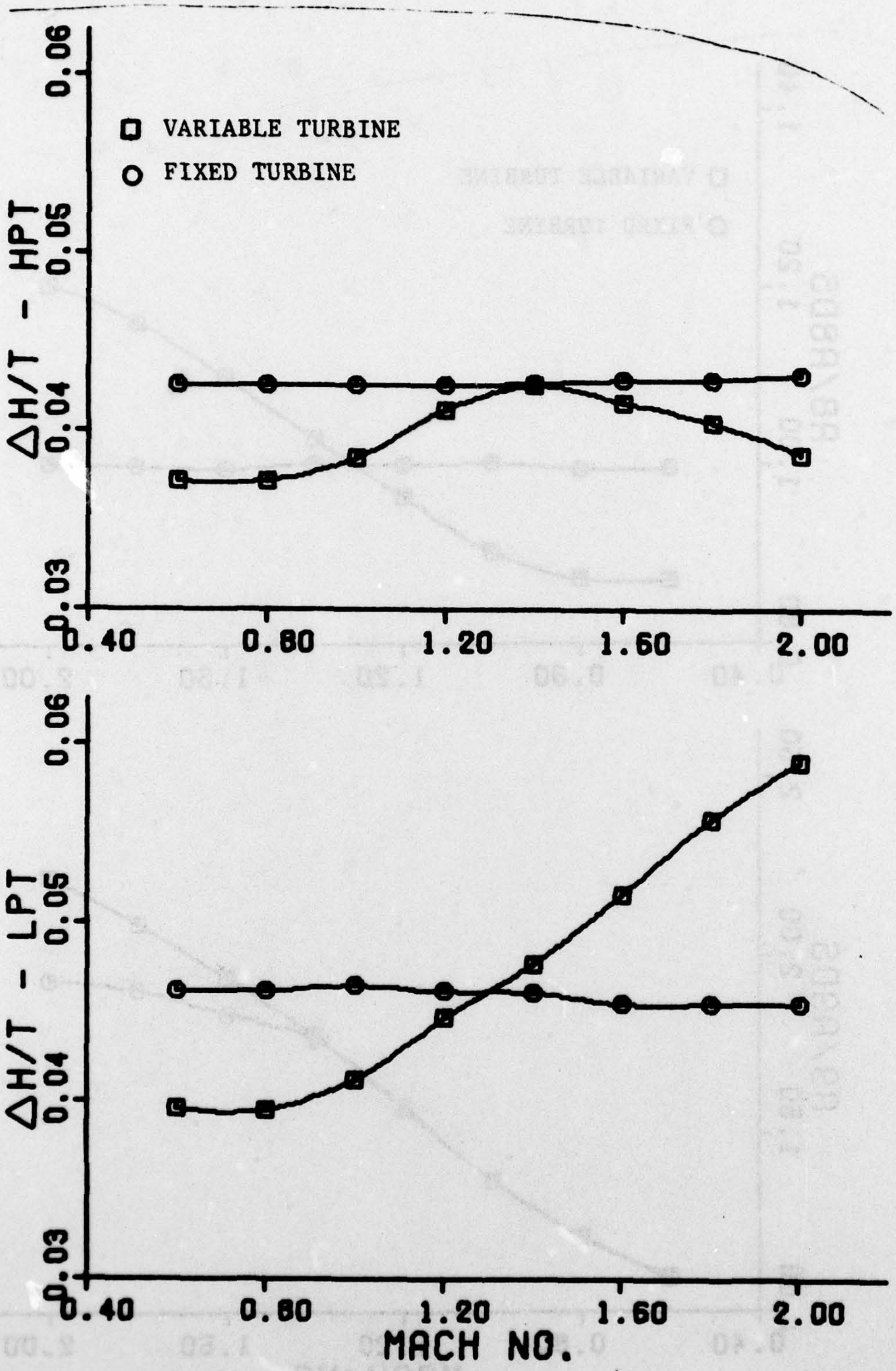


Figure 19. HP and LP Turbine Work Functions During Acceleration at 36,000 Ft Altitude

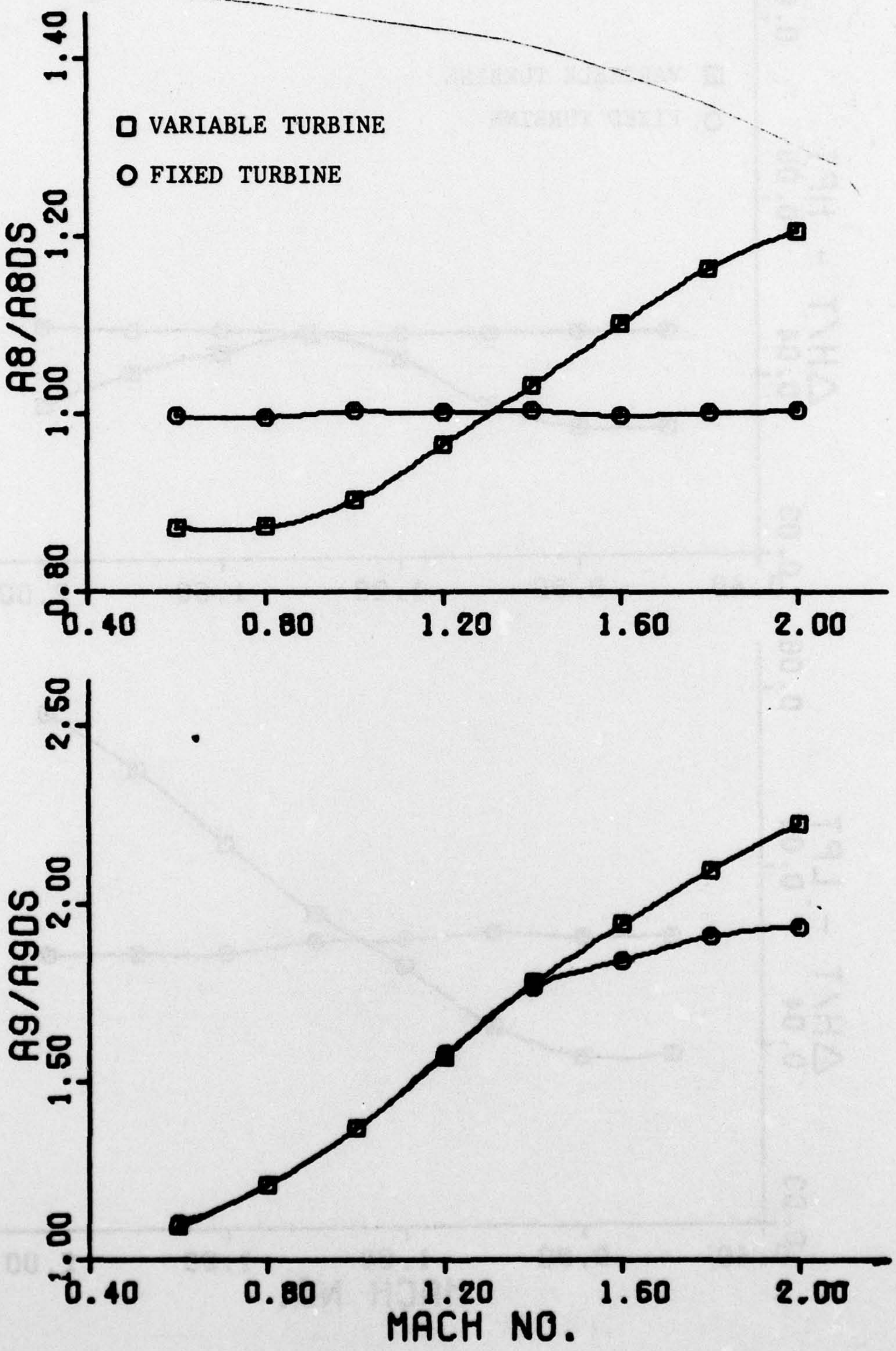


Figure 20. Core Exhaust Nozzle Throat and Exit Areas During Acceleration at 36,000 Ft Altitude

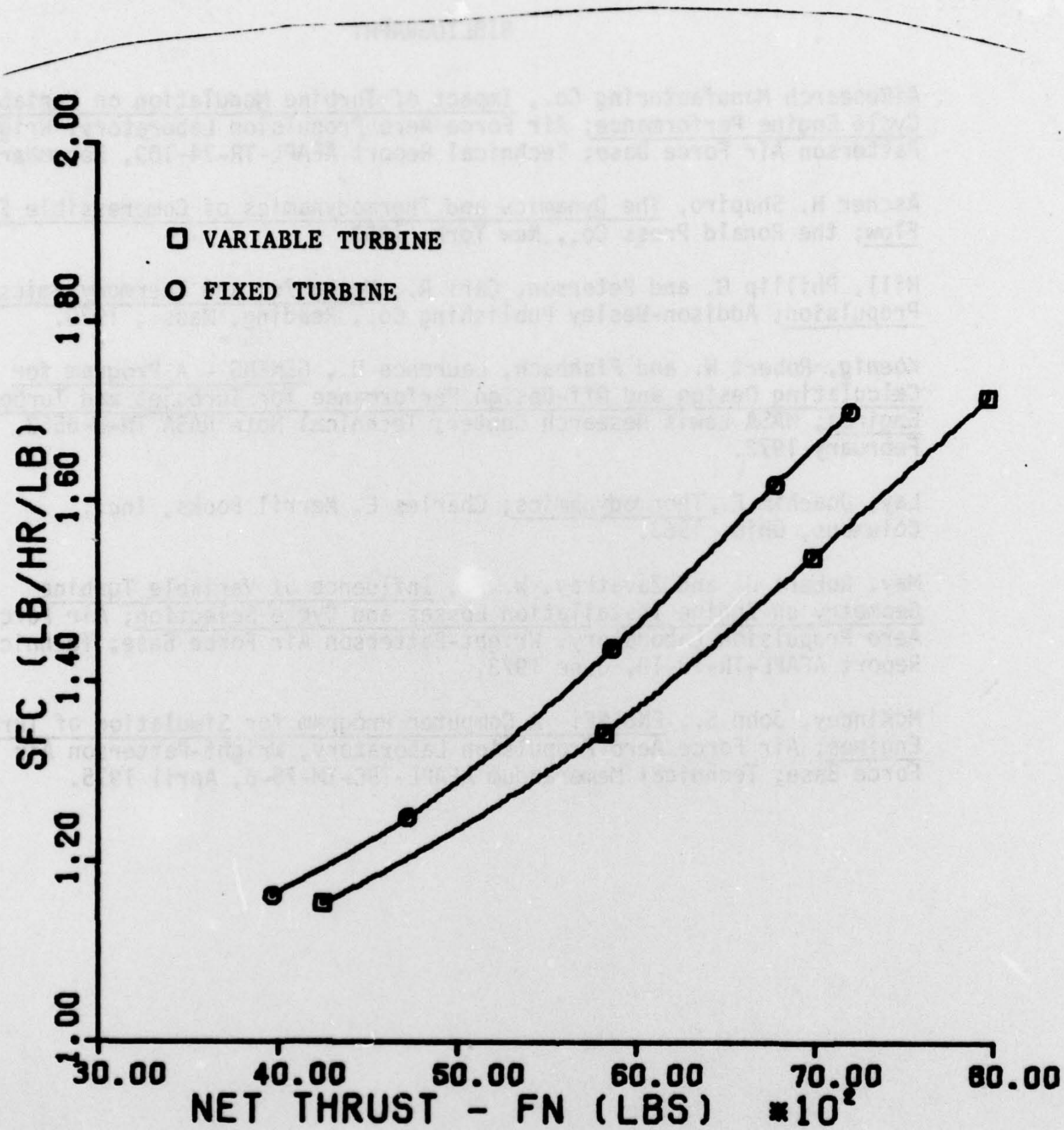


Figure 21. Augmented Thrust-SFC at 1.6 Mach, 50,000 Ft Altitude (Min.-Max. Aug.)

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