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THEORETICAL PERFORMANCE OF TURBOJET ENGINE FOR LIGHT SUBSONIC AIRCRAFT

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SUMMARY

The performance of a turbojet engine that is being studied at the Lewis Research Center for possible use in a light subsonic airplane is presented. The variation of specific fuel consumption with thrust at both cruise (Mach 0.65 and 25 000 ft (7620 m)) and sea-level static conditions was calculated using component maps estimated by Lewis component specialists. At nominal cruise conditions, thrust is 346 pounds (1540 N) and specific fuel consumption is 1.25 pounds fuel per hour per pound of thrust (0.127 kg/(hr)(N)). Compressor pressure ratio is four while turbine inlet temperature is 1560° R (867° K). A minimum specific fuel consumption of 1.15 pounds fuel per hour per pound thrust (0.117 kg/(hr)(N)) is obtained at 75 percent rated thrust and a turbine inlet temperature of 1330° R (739° K). At 25 percent rated thrust, specific fuel consumption is 1.59 pounds fuel per hour per pound thrust (0.1625 kg/(hr)(N)).

With the design value of exhaust nozzle area, the equilibrium operating line for sea-level static conditions intersects the estimated compressor surge line at 72 percent rotational speed. Thus, some form of variable geometry is required to obtain surge margin below this speed. When compressor exit bleed was used, adequate surge margin was obtained; but very high values of turbine inlet temperature were required (2300° R (1278° K)) at a thrust setting of 6 percent of maximum sea-level thrust).

A two-position exhaust nozzle is a better solution. A setting slightly less than the frontal area of the turbine provides adequate low speed surge margin and low values of turbine inlet temperature which infer a fast acceleration capability from idle to maximum thrust.

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INTRODUCTION

As part of the gas-turbine technology program at the NASA Lewis Research Center, studies are being made to examine turbojet and turbofan engines for lightweight subsonic aircraft. These engines offer such potential advantages as compactness, light weight, and greater simplicity as compared with reciprocating or turboprop engines. In addition, improved aircraft performance in terms of cruise speed and rate of climb could be realized.

In reference 1, a parametric study was made of turbojet and turbofan design-point performance with turbine inlet temperature restricted to 1560° F (867° K) and below to avoid the requirement of exotic turbine materials or turbine cooling. The study showed that the specific fuel consumption at Mach 0.65 and 25 000 feet (7620 m) varied from a low of 0.9 pounds fuel per pound thrust per hour (0.092 kg/(N)(hr)) for a high-bypass-ratio turbofan to a high of about 1.3 pounds fuel per pound thrust per hour (0.133 kg/(N)(hr)) for a low-pressure-ratio turbojet. The gas-generator diameters of the turbofan engines were smaller than those of the turbojet. The fan diameter, however, dominated. A multiple-shafting arrangement was required for the high-bypass-ratio engines (BPR = 3 to 5). Such a requirement, using gears or two spools, would increase engine complexity, which must be weighed against the performance improvement thus obtained.

In reference 2, the off-design performance of three geared-fan turbofan engines of design bypass ratio 2.5 was calculated. While off-design performance at cruise was satisfactory, operation at sea-level static conditions resulted in excessive values of turbine inlet temperature and surge of both the fan and the compressor when inlet and exhaust nozzle areas were fixed at their design values. Acceptable solutions were to incorporate into the design of the geared turbofan engine a variable inlet area, a variable primary nozzle area, a variable secondary nozzle area, or some combination of these.

Based on the performance and geometry considerations of reference 1 for turbojet engines, a particular engine was selected for further study by the Lewis Research Center. Component specialists at Lewis have designed the compressor, combustor, turbine, and nozzle, and estimated the off-design performance maps of these components. The Curtiss-Wright Corporation is currently under contract to make engineering drawings of this turbojet engine with the intent that the engine will be built and tested at a later time. To assist Curtiss-Wright in their task, design and off-design performance of the engine was calculated based on the component maps supplied by the Lewis specialists. At the design cruise conditions of Mach 0.65 and 25 000 feet (7620 m), compressor pressure ratio is four and turbine inlet temperature is 1560° R (867° K). The variations of specific fuel consumption, turbine inlet temperature, and compressor surge margin as thrust is reduced from its maximum value are calculated for cruise and sea-level static operation. Compressor exit bleed and a two-position exhaust nozzle were studied as means of obtaining adequate surge margin and acceptable turbine temperatures at thrust settings down to idle (6 percent of maximum sea-level static thrust).

RESULTS AND DISCUSSION

The performance at cruise and sea level static conditions was calculated by matching the components according to the procedures given in reference 3 and calculating the thrust and specific fuel consumption with the charts of reference 4. The inlet map was obtained from reference 5. The compressor, combustor, turbine, and nozzle maps were estimated by Lewis component specialists.

Component performance maps are presented in figures 1 and 2. All symbols are defined in appendix A. The compressor map is shown in figure 1. At design airflow and equivalent speed, pressure ratio is four and efficiency is 0.805. A maximum efficiency of 0.885 is obtained at 90 percent equivalent speed. (The operating lines shown in figure 1 will be discussed later.) The inlet performance (ref. 5) is shown in figure 2(a). At 100 percent equivalent flow, inlet recovery is 0.95 at Mach 0 and 0.99 at Mach 0.65.

At the design combustor velocity of 100 feet/second (30.5 m/sec), combustor efficiency is 0.96 and pressure loss is 6.5 percent of combustor inlet pressure (fig. 2(b)). Turbine performance is shown in figure 2(c). At the design value of the blade-speed-jet-velocity parameter, turbine efficiency is 0.85. Also shown is the variation of turbine equivalent weight flow with turbine pressure ratio and equivalent speed. The flow and gross thrust parameter of a simple convergent nozzle as a function of nozzle pressure ratio is shown in figure 2(d).

Cruise Performance

The performance at Mach 0.65 and 25 000 feet (7620 m) is shown in figure 3 for an exhaust nozzle area of 0.417 feet^2 (0.0387 m^2). At 100 percent thrust (346 lb or 1540 N), specific fuel consumption is 1.25 pounds fuel per hour per pound thrust (0.127 kg/(hr)(N)) and turbine inlet temperature is 1560° R (867° K). A minimum specific fuel consumption of 1.15 pounds fuel per hour per pound thrust (0.117 kg/(hr)(N)) is obtained at 75 percent thrust and a turbine inlet temperature of 1330° R (739° K). The cruise equilibrium operating line is shown on the compressor map in figure 1. The operating line goes through the maximum efficiency operating point and is well removed from the estimated compressor surge line.

Sea-Level Static Performance

The operating line at sea-level static conditions and a nozzle area of 0.417 feet^2 (0.0387 m^2) is seen to intersect the surge line at about 72 percent equivalent speed. The absence of ram pressure ratio shifts the equilibrium operating line away from its cruise location toward the compressor surge margin. One way to provide surge margin down to lower speeds is to provide bleed ports at the compressor exit. The equilibrium operating line for a bleed area of 0.0275 feet^2

(0.00255 m^2) provides surge margin down to at least 50 percent speed. Using this design feature, thrust and turbine temperature would vary as shown in figure 4. The thrust variation is shown in figure 4(a). With the bleeds closed, speed could be reduced to a value slightly greater than 72 percent so as not to surge the compressor. If the bleed were opened here, speed could be reduced to 41.5 percent to attain idle thrust setting. The engine starter would accelerate the engine to some speed below 41.5 percent where the combustor would be lit. The starter would be disengaged prior to the attainment of idle conditions. The variation in turbine inlet temperature is shown in figure 4(b). With bleeds open, the temperature rises quite rapidly as speed is reduced below 65 percent. At idle conditions, turbine inlet temperature is 2300° R (1278° K). The high temperature results from depriving the turbine of some of its working fluid. With fixed bleed area, the amount of bleed varies with speed. At idle, it is 26.2 percent of compressor airflow.

The bleed area of 0.0275 feet^2 (0.00255 m^2) was arbitrary. In figure 5, the effect of bleed area on turbine temperature and compressor airflow is shown for 50 percent speed. A bleed area of 0.0165 feet^2 (0.00153 m^2) results in the minimum turbine inlet temperature of 1805° R(1003° K) at the expense of some surge margin. This point is plotted in figures 4(a) and (b). At speeds below 50 percent as required for idle (fig. 4(A)), the turbine inlet temperature would be still higher. It is concluded that compressor exit bleed is not a feasible solution to the surge and high temperature problem.

A larger exhaust nozzle area would provide surge margin and lower values of turbine inlet temperature. The equilibrium operating line for a nozzle area of 0.775 feet^2 (0.072 m^2) is shown in figure 1. (Turbine frontal area is 0.785 feet^2 (0.0729 m^2).) This results in adequate surge margin. The thrust and turbine temperature variations are shown in figure 6. The larger nozzle setting could be employed at any speed to reduce both thrust and turbine temperature. At idle thrust, turbine temperature is only 1310° R (728° K). With such low equilibrium temperatures, it is likely that fast acceleration from idle to maximum thrust can be achieved. A two-position nozzle appears to be an adequate solution to the surge margin and high temperature problems of a fixed geometry turbojet.

SUMMARY OF RESULTS

The performance of a turbojet engine that is being studied at the Lewis Research Center for possible use in a light subsonic airplane is presented. The variation of specific fuel consumption with thrust at both cruise (Mach 0.65 and 25 000 feet (7620 m)) and sea-level static conditions was calculated using component maps estimated by Lewis component specialists. At cruise, with a compressor pressure ratio of four and a turbine inlet temperature of 1560° R (867° K), thrust was 346 pounds (1540 N) and specific fuel consumption 1.25 pounds fuel per hour per pound thrust (0.127 kg/(hr)(N)). A minimum specific fuel consumption of 1.15 pounds fuel per hour per pound thrust (0.117 kg/(hr)(N)) is obtained at 75 percent rated thrust and a turbine inlet temperature of 1330° R (739° K). At 25 percent rated thrust, specific fuel consumption is 1.59 pounds fuel per hour per pound thrust (0.1625 kg/(hr)(N)).

With the design value of exhaust nozzle area, the equilibrium operating line for sea-level static conditions intersects the estimated compressor surge line at 72 percent rotational speed. Thus, some form of variable geometry is required to obtain surge margin below this speed. When compressor exit bleed was used, adequate surge margin was obtained but very high values of turbine inlet temperature were required (2300° R (1278° K) at a thrust setting of 6 percent of maximum sea-level thrust).

A two-position exhaust nozzle is a better solution. A setting slightly less than the frontal area of the turbine provides adequate low speed surge margin and low values of turbine inlet temperature.

These low equilibrium temperatures should result in a fast acceleration capability from idle to maximum thrust.

Lewis Research Center,
National Aeronautics and Space Administration
Cleveland, Ohio, December 20, 1968,
789-50-01-01-22

APPENDIX A

SYMBOLS

A	area
F_g	gross thrust
g	gravitational acceleration
h	altitude
ΔH	turbine enthalpy change
J	mechanical equivalent of heat
N	rotational speed
P	total pressure
U_m	wheel speed at mean radius of turbine
W	weight flow
γ	ratio of specific heats
δ	ratio of total pressure to sea level pressure
θ	ratio of total temperature to 519° R (379.4° K)
η	adiabatic efficiency

Subscripts

1	compressor inlet
2	compressor exit
B	bleed
i	ideal
N	nozzle

REFERENCES

1. Roelke, Richard J.; and Stewart, Warner L.: Turbojet and Turbofan Cycle Considerations and Engine Configurations for Application in Lightweight Aircraft. NASA TM X-1624, 1968.
2. Kraft, Gerald A.; and Miller, Brent A.: Off-Design Theoretical Performance of Three Turbofan Engines for Light Subsonic Aircraft. NASA TM X-52469, 1968.
3. Dugan, James F., Jr.: Compressor and Turbine Matching. Aerodynamic Design of Axial-Flow Compressors. NASA SP-36, 1965, pp. 469-508.
4. Esgar, Jack B.; and Ziemer, Robert R.: Methods for Rapid Graphical Evaluation of Cooled or Uncooled Turbojet and Turboprop Engine or Component Performance (Effects of Variable Specific Heat Included). NACA TN 3335, 1955.
5. Mossman, Emmet A.; and Anderson, Warren E.: The Effect of Lip Shape on a Nose-Inlet Installation at Mach Numbers from 0 to 1.5 and a Method for Optimizing Engine-Inlet Combinations. NACA RM A54B08, 1954.

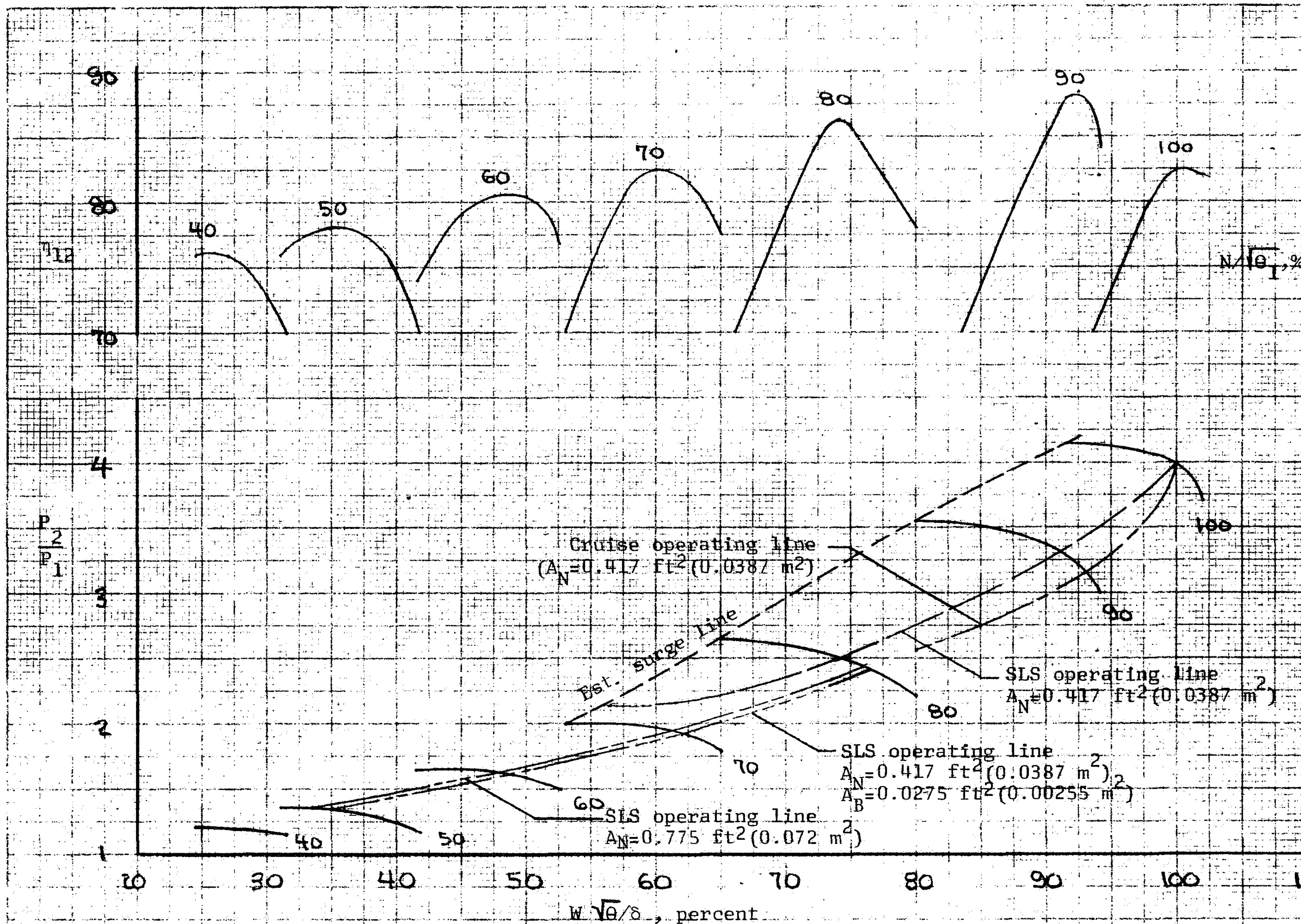
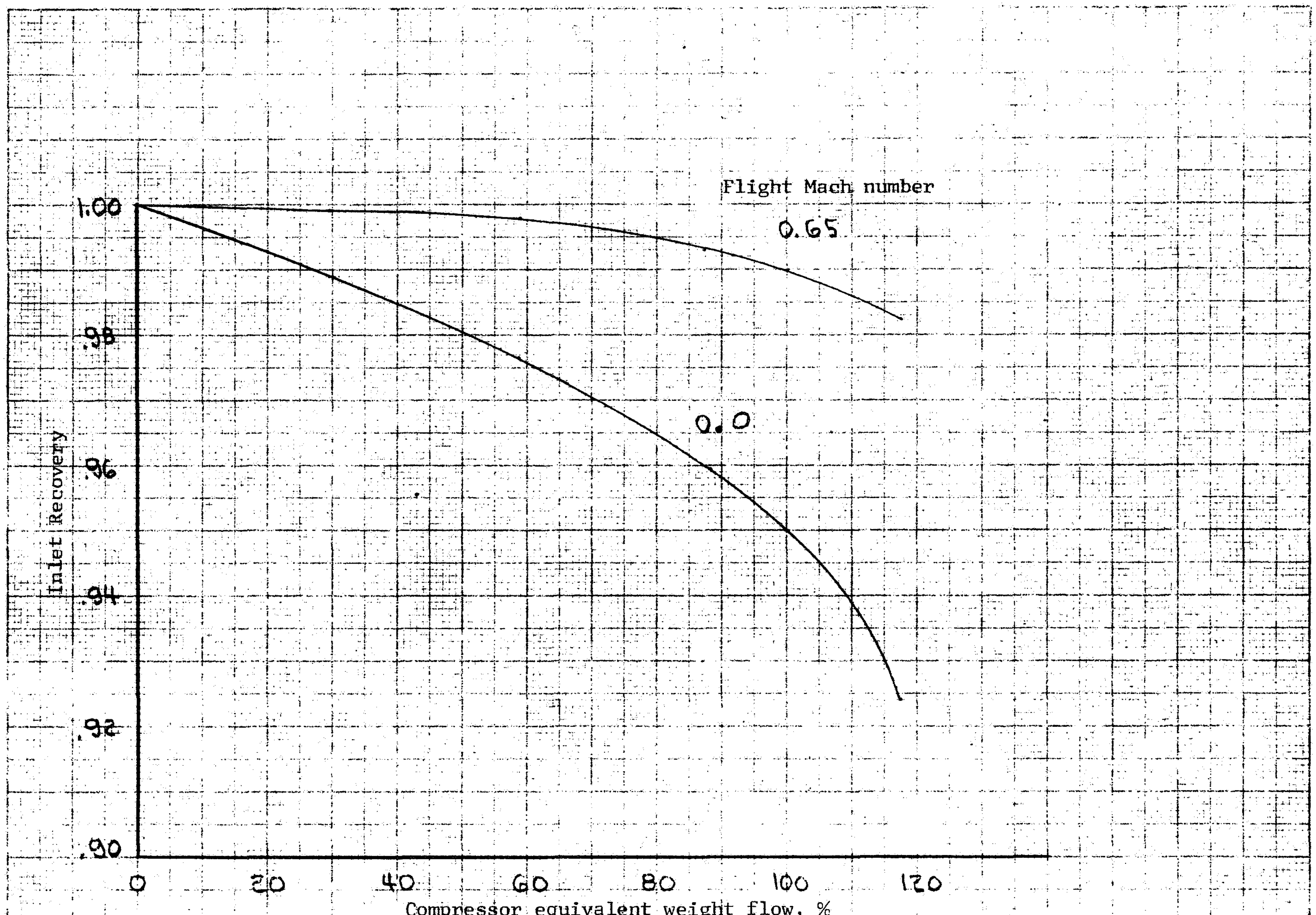
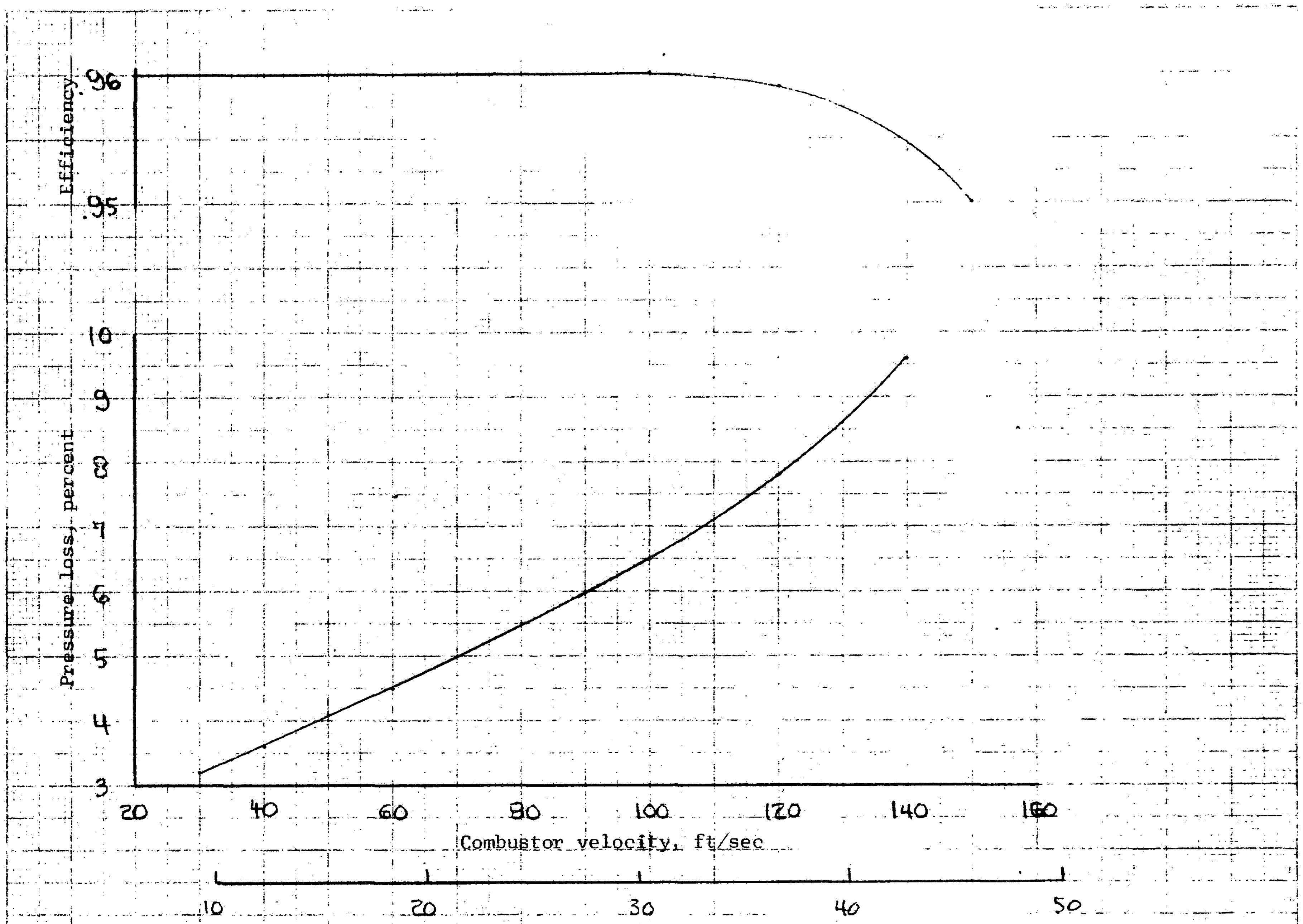


Figure 1. - Performance of 4-stage axial compressor for small turbojet engine.

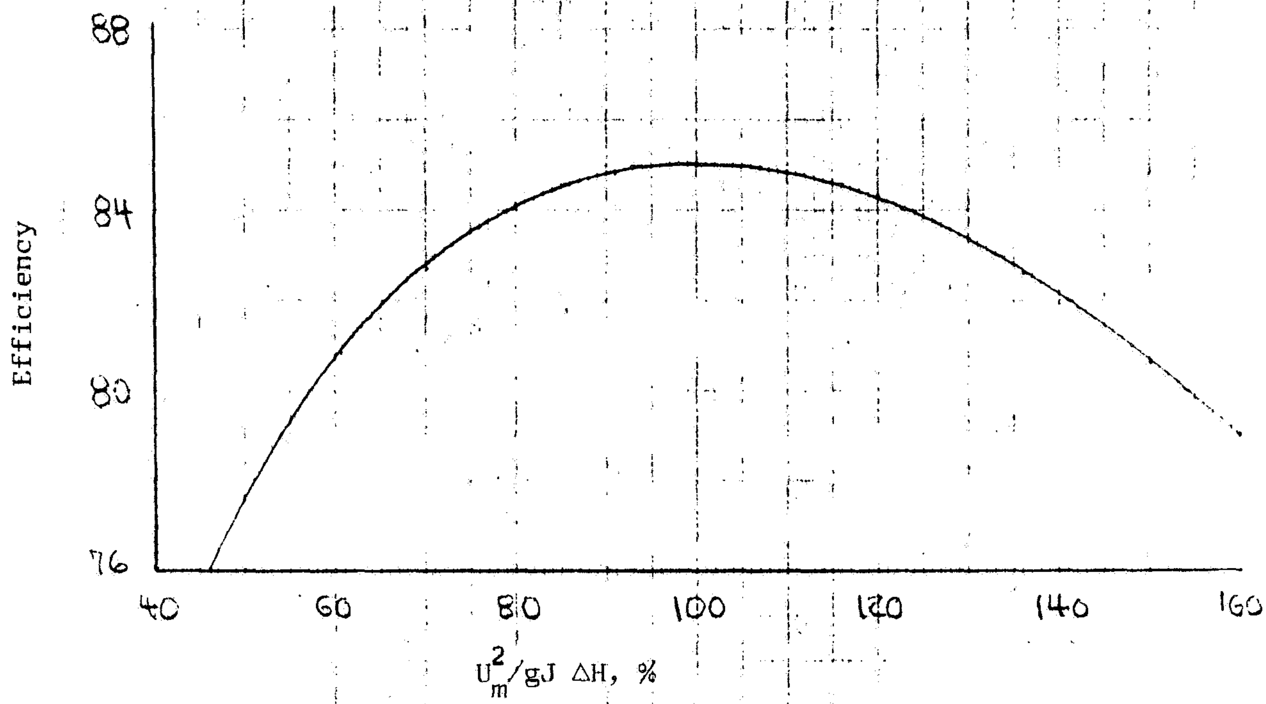


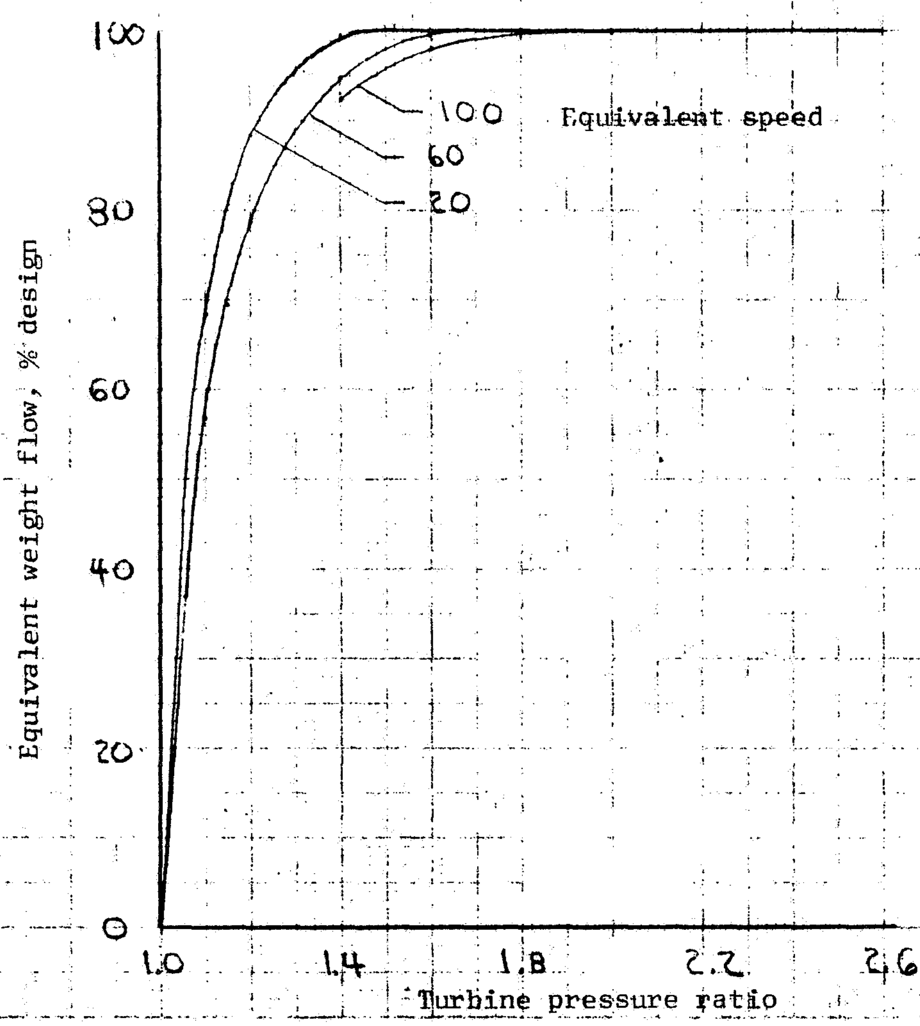
(a) Inlet

Figure 2. - Component performance.



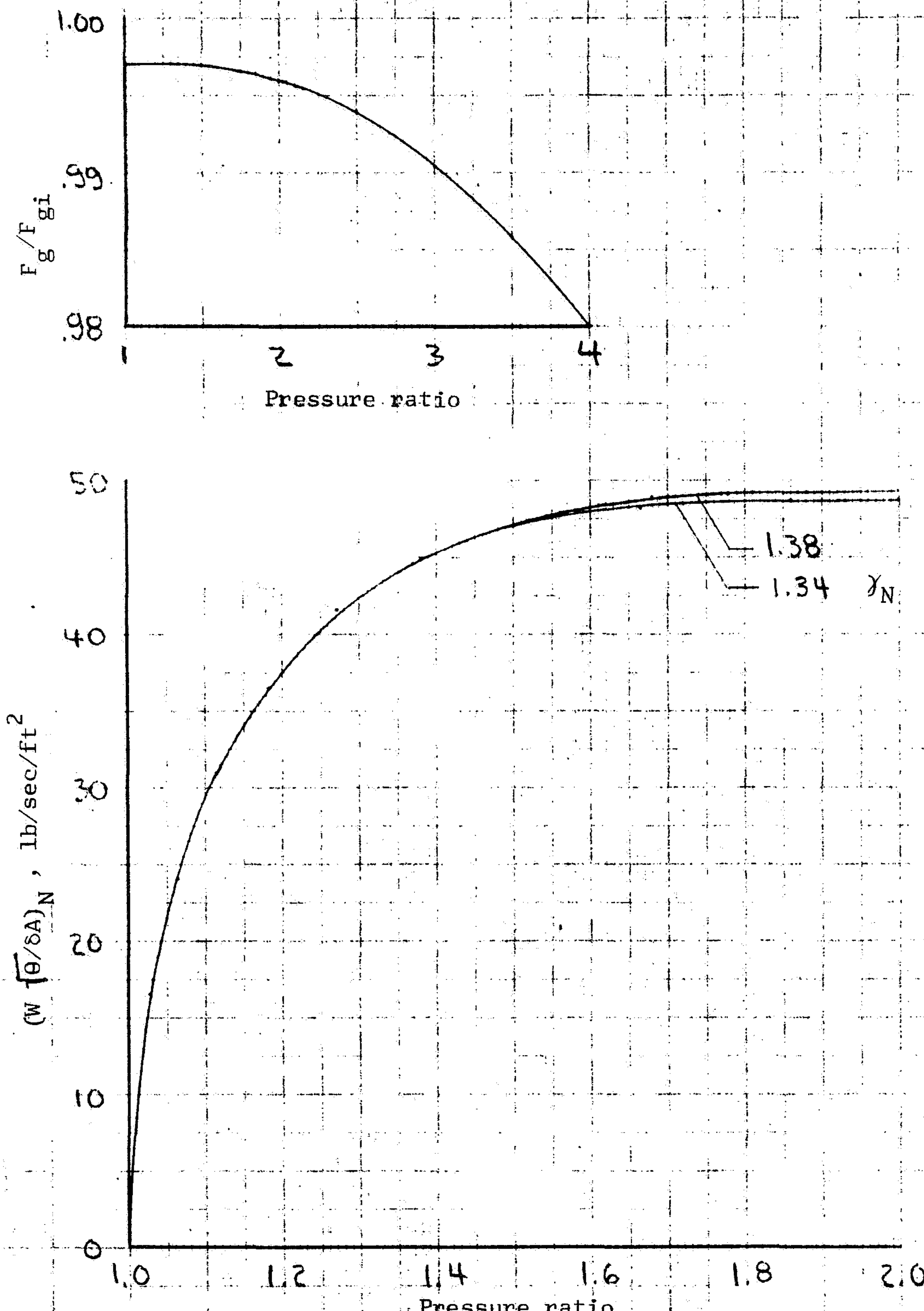
(b) Combustor
Figure 2. - Cont.





(c) Turbine

Figure 2. - Cont.



(d) Exhaust nozzle

Figure 2. - Concluded

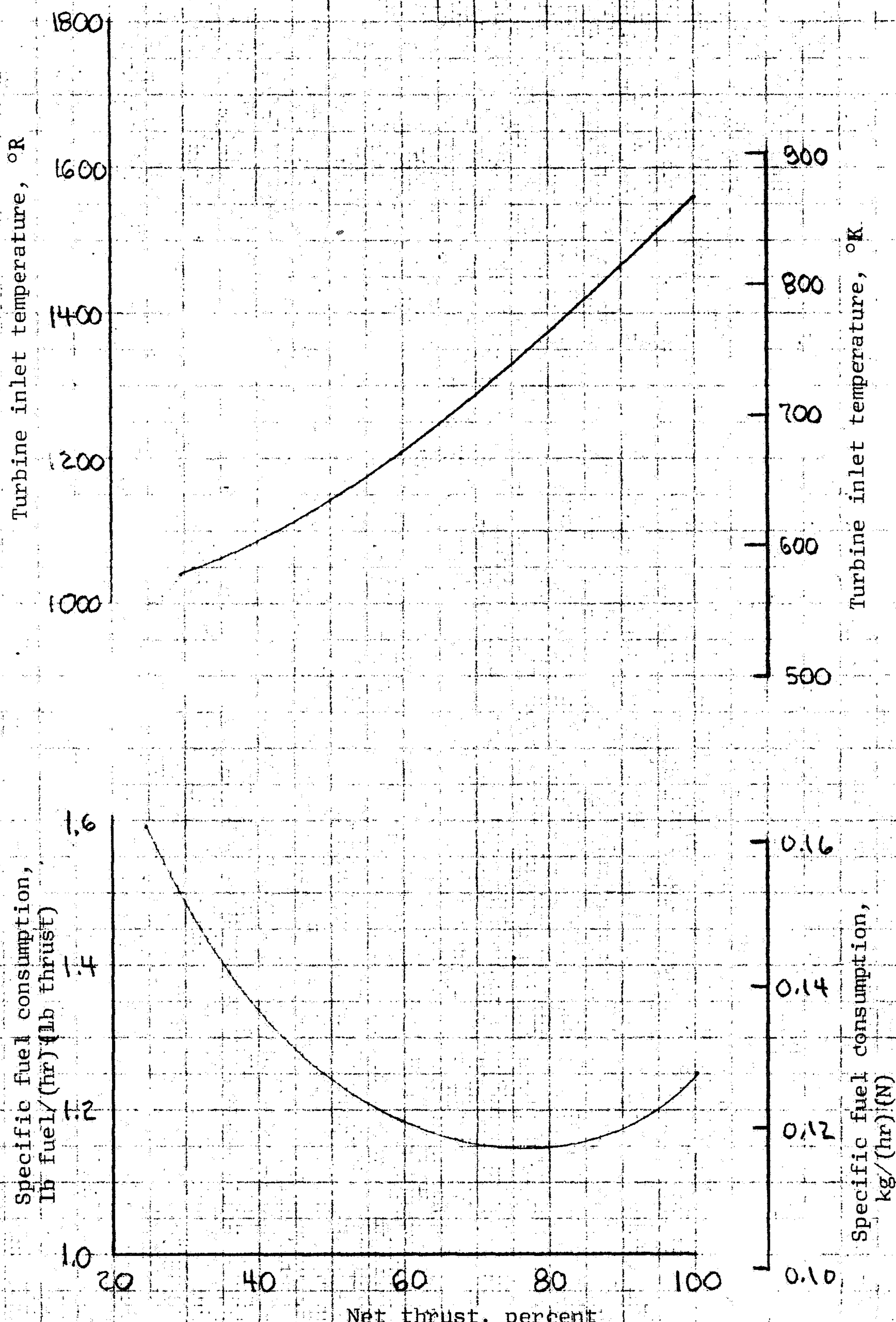
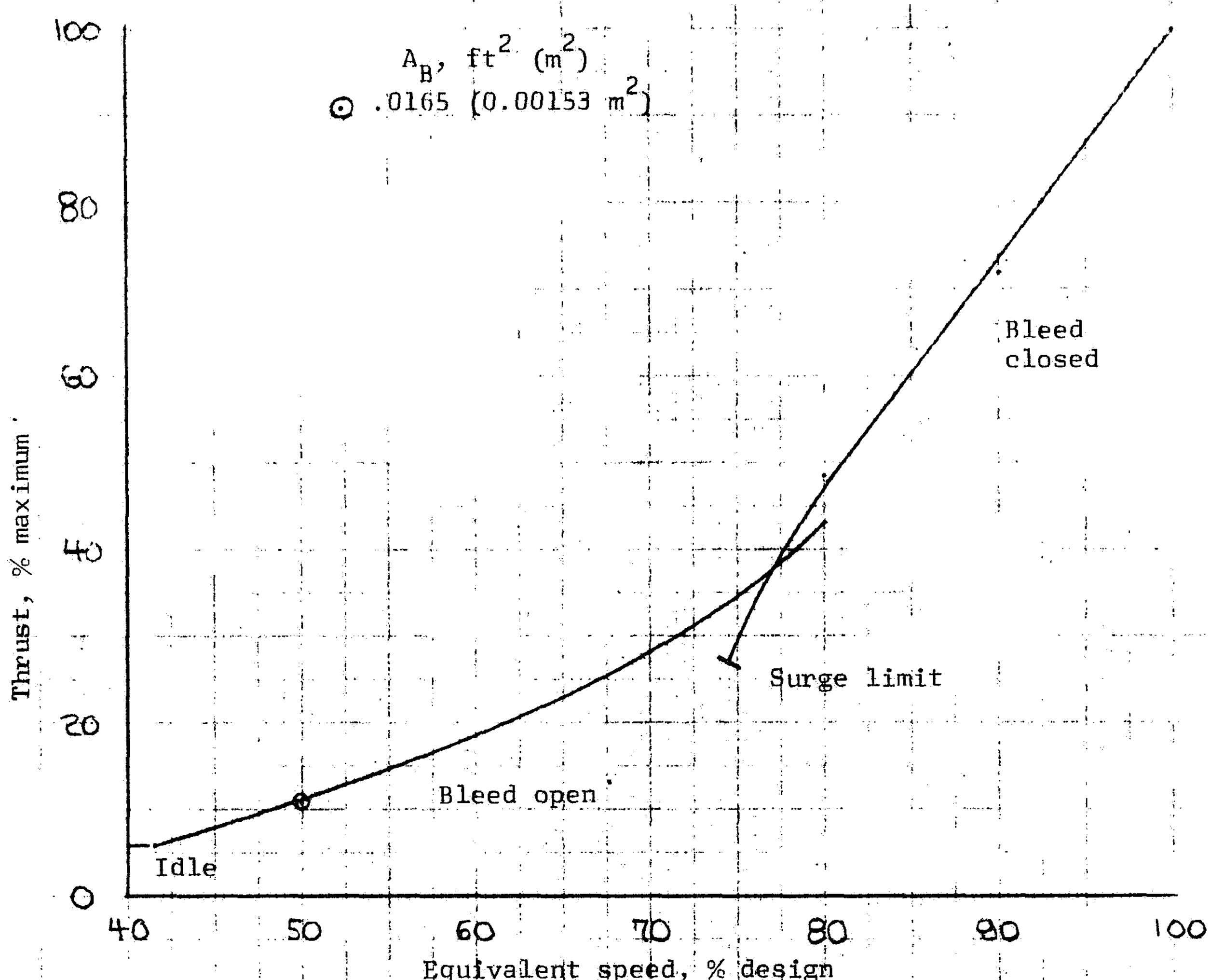


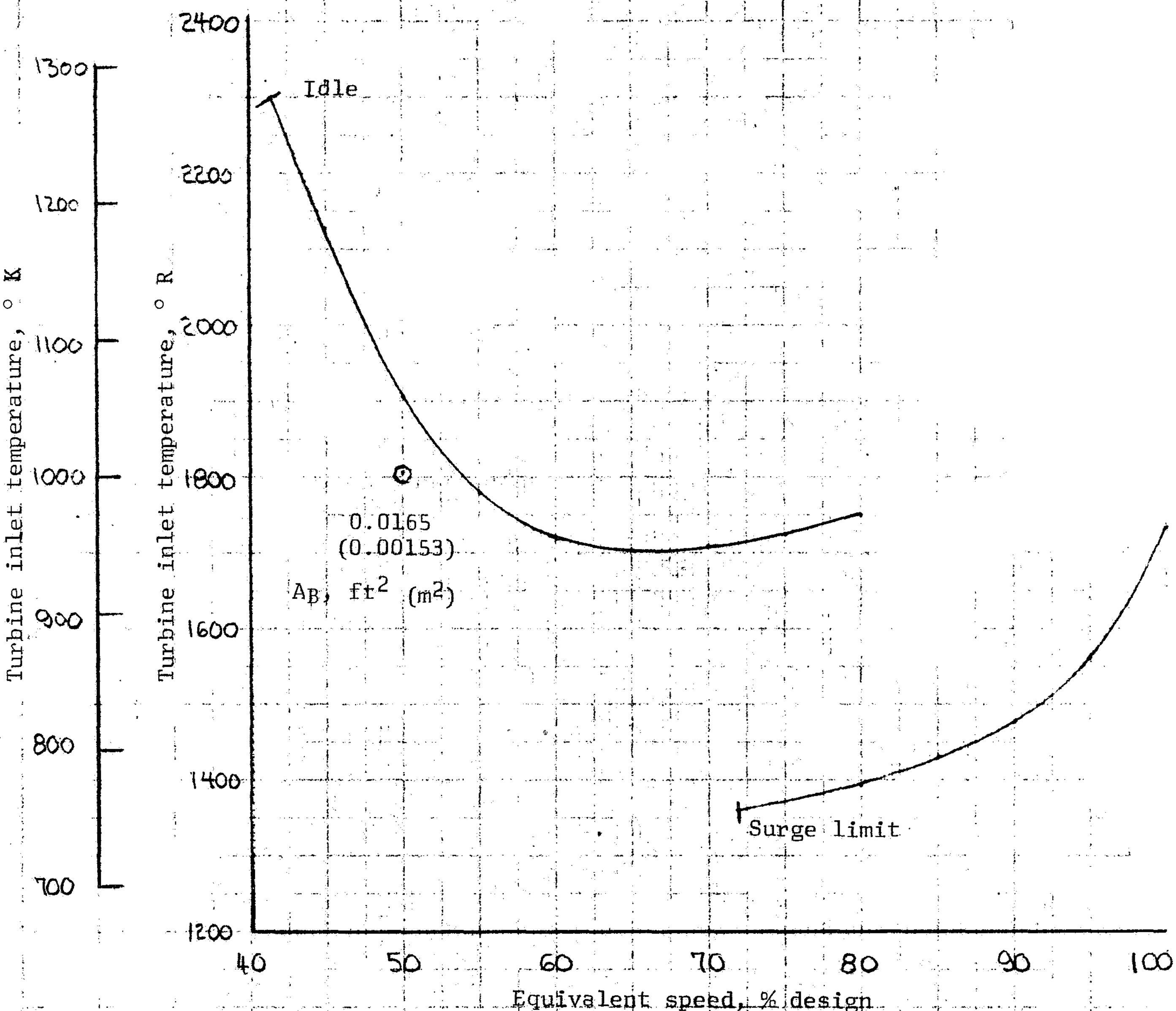
Figure 3. - Cruise performance.

$M_o = 0.65$ h=25,000 ft (7620 m) $A_N = 0.417$ ft^2 (0.0387 m^2)



(a) Effect of compressor exit bleed on thrust variation. A_B = 0.0275 ft^2 (0.00255 m^2)

Figure 4. - Performance at sea-level static conditions. Nozzle area = 0.417 ft^2 (0.0387 m^2)



(b) Effect of compressor exit bleed on turbine temperature variation. A_B = 0.0275 ft^2 (0.00255 m^2)

Figure 4. - Concluded

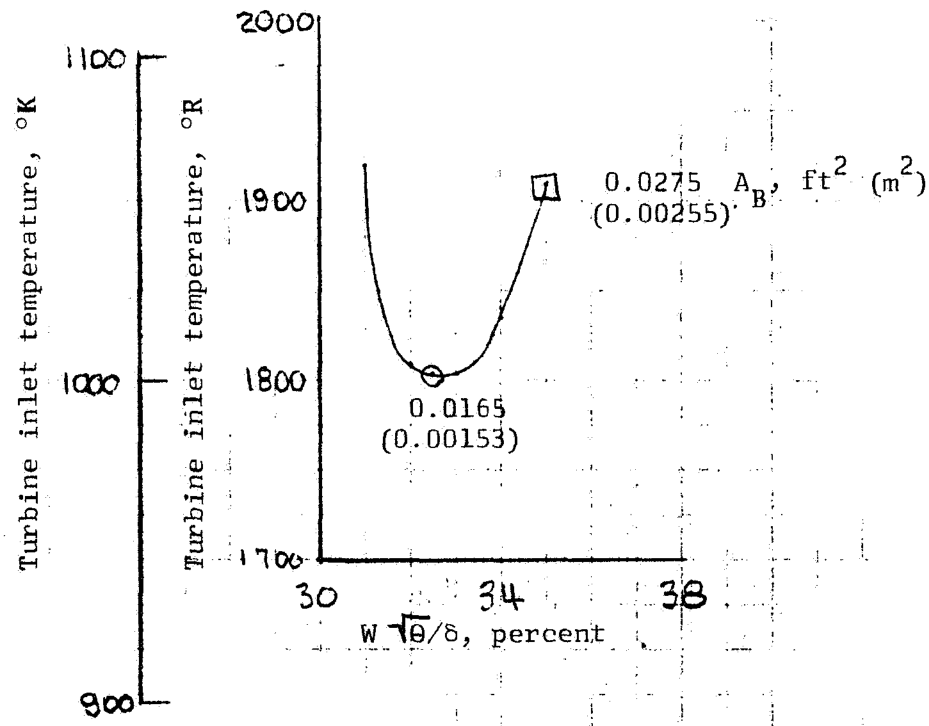


Figure 5. - Selection of bleed area. $N/\sqrt{\theta}$ = 50%
A_N = 0.417 ft^2 (0.0387 m^2)

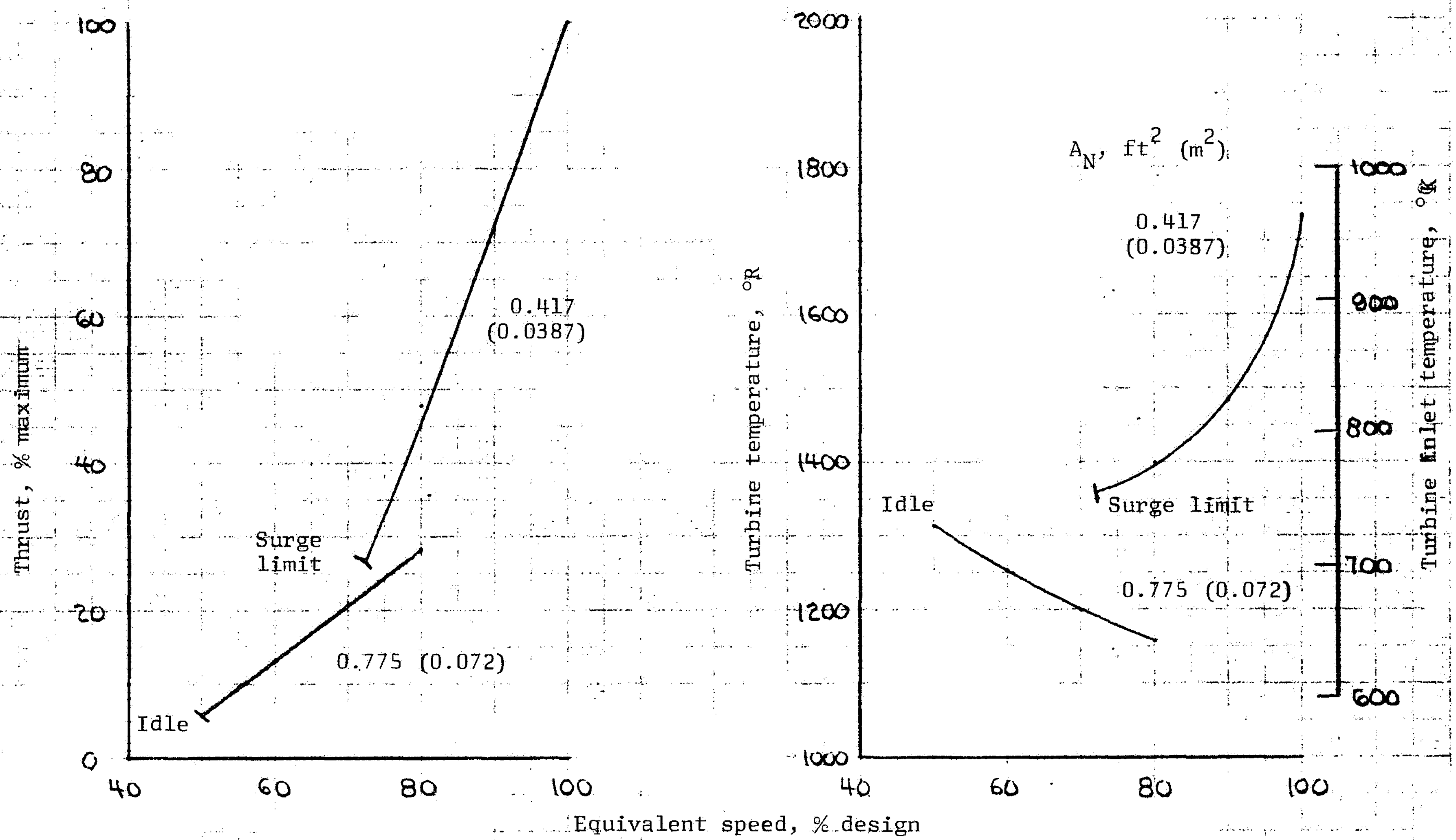


Figure 6 - Performance at sea level static conditions. Two-position exhaust nozzle area.