[redacted]

ANALYSIS OF EFFECT OF TAIL-PIPE-BURNER DESIGN PARAMETERS ON THRUST AUGMENTATION

By Eugene J. Manganiello

Lewis Flight Propulsion Laboratory

INTRODUCTION

The tail-pipe-burning method of thrust augmentation for turbojet engines consists of introducing and burning fuel between the turbine and the exhaust nozzle of the engine. The increased temperature of the exhaust gases results in increased jet velocity and hence increased thrust. Tail-pipe burning, or afterburning or reheat, as it is sometimes designated, is not only an augmentation device for improving the take-off and high-speed performance of aircraft, but also the complete configuration may be considered as a distinct engine type for flight at supersonic speeds.

A theoretical analysis of tail-pipe burning is reported in reference 1 wherein generalized charts are presented that permit convenient estimation of tail-pipe-burning performance for various design and operating conditions. In this paper, results of the investigation of reference 1 are reviewed and extended with particular attention to the effect of burner design parameters on augmented and normal engine performance. Consideration is also given to the correlation of tail-pipe-burner blow-out limits with flight operating conditions.

METHODS

Schematic diagrams of normal and tail-pipe-burning engine configurations are shown in figure 1. The two engines are the same except, of course, for their tail pipes. In the normal configuration (fig. 1(a)), the turbine-outlet gas is diffused slightly to the exhaust-cone-exit plane and flows to the jet nozzle through a simple tail pipe of a length dictated by the airplane installation.

In the tail-pipe-burner configuration (fig. (1(b)), the turbine-outlet gas is diffused to the burner-inlet plane where fuel is injected. In some designs fuel is injected at various positions in the diffuser. Flame holders are located downstream of the fuel-injection nozzles to furnish the stagnation regions and the turbulence necessary for combustion, and a suitable length of

[redacted]

tail pipe is provided to permit completion of combustion before reaching the exhaust nozzle.

The tail-pipe-burner-inlet velocities must be sufficiently low to avoid excessive pressure losses and to insure satisfactory combustion. Accordingly, the system requires more diffusion and a tail pipe of greater area than the normal engine. The exhaust nozzle must also be larger than that of the normal engine because of the increased gas volume associated with the higher temperature and must be adjustable (either two-position or continuously variable) in order to provide for operation under both normal and augmented conditions.

Calculations were made to investigate the effect on engine performance of the following tail-pipe-burner and engine design parameters:

1. Diffuser efficiency η_d, considered herein as adiabatic efficiency on energy basis between turbine-outlet and burner-inlet stations, $\dfrac{\dfrac{R\gamma}{\gamma-1}\, t_t \left[\left(\dfrac{p_b}{p_t}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]}{\frac{1}{2}\left(V_t^2 - V_b^2\right)}$

2. Burner-inlet velocity V_b

3. Burner drag coefficient C_D, defined as total frictional pressure drop across tail-pipe burner divided by burner-inlet dynamic head

4. Burner-outlet gas temperature T_b

5. Exhaust-nozzle velocity coefficient C_V, defined as ratio of actual to theoretical jet velocity and equal to square root of exhaust-nozzle efficiency (on energy basis)

6. Turbine-outlet velocity V_t

7. Turbine-outlet pressure or engine compressor pressure ratio

The additional symbols used herein are:

p static pressure

R gas constant

t static temperature

γ ratio of specific heats

Subscripts:

b burner inlet

t turbine outlet

The effects of these design parameters were calculated for a range of flight Mach numbers at sea level and 35,000 feet altitude.

The engine assumptions used in the calculations are:

Compressor pressure ratio at sea level and flight Mach number of zero (At other flight conditions the pressure ratio was varied to meet the condition of constant rotative speed, that is, constant work input per pound of air; for example, at sea level and a Mach number of 2, the pressure ratio is 2.4.) . . . 4

Compressor polytropic efficiency, $\dfrac{\gamma-1}{\gamma}\,\dfrac{\log\left(\frac{P_2}{P_1}\right)}{\log\left(\frac{T_2}{T_1}\right)}$ 0.80

Turbine polytropic efficiency, $\dfrac{\gamma}{\gamma-1}\,\dfrac{\log\left(\frac{T_t}{T_4}\right)}{\log\left(\frac{P_t}{P_4}\right)}$ 0.85

Turbine-outlet temperature corresponding to turbine-inlet temperature of 1960° R, °R 1650
Combustion efficiency 0.96
Primary combustion-chamber pressure drop divided by combustion-chamber-inlet pressure 0.03
Exhaust-nozzle velocity coefficient (normal engine) 0.975
Engine inlet-diffuser polytropic efficiency,

$$\frac{\gamma-1}{1}\,\frac{\log\left(\frac{P_1}{P_0}\right)}{\log\left(1+\frac{\gamma-1}{2}M^2\right)}:$$

Flight Mach number up to 1 (For flight Mach numbers above 1, the diffuser efficiency was reduced 0.1 per unit increase in Mach number; for example, at a Mach number of 2, the efficiency was 0.75.) 0.85

The additional symbols are:

M Mach number

P total pressure

T total temperature

Subscripts:

0 free stream

1 compressor inlet

2 compressor outlet

4 turbine inlet

The foregoing assumptions are, for the most part, fairly conservative and represent an average of the performance of various present-day engines. The inlet-diffuser efficiency values, which are representative of the performance of convergent-divergent-type diffusers, are conservative compared to values currently being obtained experimentally with other types of supersonic diffuser.

The performance of the normal engine for the different flight conditions was calculated by step-by-step methods and the performance of the tail-pipe-burner configuration was calculated from the normal engine performance by the methods of reference 1. Dissociation was taken into account in the calculations of fuel consumption for the tail-pipe-burner configuration, and the combustion efficiency was assumed to be 0.96 as for the primary engine combustion chamber. The normal engine was assumed to have no tail-pipe pressure losses; that is, the exhaust-nozzle-inlet total pressure was taken equal to the turbine-outlet total pressure. Inasmuch as the calculations were made for constant turbine-outlet temperature it is implicitly assumed that the exhaust-nozzle area is adjusted to the proper value at all operating conditions.

The data and calculations involved in the correlation of tail-pipe-burner blow-out limits are based upon the results of experimental investigations with a current turbojet engine and tail-pipe burner.

[redacted]

RESULTS AND DISCUSSION

The effects of design parameters on augmented and normal engine performance are presented in figures 2 to 9 and the information pertaining to blow-out limits is given in figures 10 and 11.

Augmented and Normal Engine Performance

Gas temperature and inlet velocity. - The ratio of augmented to normal thrust is plotted in figure 2 against tail-pipe-exit gas temperature for a range of burner-inlet velocities from 200 to 750 feet per second. The results in figure 2(a) are for sea-level altitude, flight Mach number of zero, turbine-outlet velocity of 750 feet per second, diffuser efficiency of 80 percent, burner drag coefficient of 1, and exhaust-nozzle velocity coefficient of 0.975, which is the same as that assumed for the normal engine. The normal thrust used as the basis of augmented ratio is that calculated for the engine with a normal or conventional tail pipe.

The augmented thrust ratio increases with increase in tail-pipe gas temperature as a result of the accompanying increase in jet velocity and decreases with increase in burner-inlet velocity because of increased friction and momentum pressure drop across the burner. At a gas temperature of 3600° R and a burner-inlet velocity of 400 feet per second, the augmented thrust is 1.45 times the normal thrust. At the same temperature but at an inlet velocity of 700 feet per second, the augmented thrust ratio is reduced to 1.2. At high burner-inlet velocities (700 and 750 ft/sec), the maximum augmentation is limited to the end points of the curves because of thermal choking, which limits the maximum temperature that can be realized without affecting the engine operating conditions.

The effect of the tail-pipe burner on engine performance for the condition of no afterburning is shown by the results at tail-pipe gas temperature equal to turbine-outlet temperature, that is, 1650° R. At a burner-inlet velocity of 400 feet per second, the augmented thrust is about 97 percent of the normal engine thrust, and at an inlet velocity of 700 feet per second, the thrust is reduced to 93 percent of the normal engine thrust. These losses are a result of the diffuser inefficiency and the friction drag of the burner and correspond to total-pressure-loss ratios $\frac{\Delta P}{P_b}$ of 0.04 and 0.085 at 400 and 700 feet per second, respectively. These losses in normal thrust and those indicated in subsequent curves are higher than would be obtained in practice for the same design

[redacted]

conditions because they are based on zero pressure loss in the normal engine tail cone and tail pipe.

Thus low burner-inlet velocity is not only desirable for obtaining high thrust augmentation but also for minimizing the loss of normal or nonaugmented thrust. In addition, low velocity is required for satisfactory combustion efficiency and stability as is discussed in the second and third papers of this series.

The effects of tail-pipe gas temperature and burner-inlet velocity are illustrated in figure 2(b) for altitude of 35,000 feet and flight Mach number of 1.50. Again the ratio of augmented to normal thrust is plotted against tail-pipe gas temperature for a range of burner-inlet velocities. The values of the design parameters are the same as in figure 2(a); but the normal engine thrust used as the base for the augmented ratio is changed to the value corresponding to the new flight conditions. Effects similar to those illustrated in figure 2(a) are obtained, however change in inlet velocity results in only about half as much percentage change in the augmented thrust ratio as occurs at sea level and zero flight Mach number. The smaller effects are due to the fact that at a higher pressure ratio across the exhaust nozzle (as exists at the high Mach number condition) a given percentage change in pressure loss produces a smaller change in thrust than at a lower pressure ratio across the nozzle. The higher values of augmentation indicated are due to the higher flight Mach number and not the higher altitude as is illustrated in a subsequent figure of this paper.

Turbine-outlet velocity and diffuser efficiency. - In figure 3, augmented-to-normal thrust ratio is plotted against turbine-outlet velocity for a tail-pipe gas temperature of 3800° R and diffuser efficiencies of 100, 80, and 60 percent. A similar set of curves is included for a gas temperature of 1650° R in order to illustrate the performance at nonburning conditions. These results are for sea-level altitude, zero flight Mach number, burner-inlet velocity of 400 feet per second, drag coefficient of 1, and exhaust-nozzle velocity coefficient of 0.975.

For a diffuser efficiency of 100 percent, both the augmented and normal thrust remain constant with change in turbine-outlet velocity; but for the more realistic values of diffuser efficiency, the performance decreases progressively with increased turbine-outlet velocity and decreased diffuser efficiency. For example, at a diffuser efficiency of 80 percent, the augmented thrust ratio decreases from about 1.48 at 800 feet per second to 1.43 at 1200 feet per second for the afterburning condition and from

0.97 to 0.95 for the nonburning condition. With a diffuser efficiency of 60 percent, the adverse effects of increased turbine-outlet velocity are greater.

The curves of figure 3 illustrate the desirability of designing the turbojet engine with a low turbine-outlet velocity in order to realize high augmentation and to minimize penalties during nonburning operation. Alternatively, if the engine has a high turbine-outlet velocity, the designer should make every effort to obtain a high diffuser efficiency.

Burner drag coefficient. - In figure 4 the augmented-to-normal thrust ratio is plotted against burner drag coefficient for burner-inlet velocities of 200, 400, and 600 feet per second, and for tail-pipe gas temperatures of 3800° (augmented condition) and 1650° R (nonburning condition). These curves are for sea-level altitude, zero flight Mach number, turbine-outlet velocity of 750 feet per second, diffuser efficiency of 80 percent, and exhaust-nozzle velocity coefficient of 0.975.

As might be expected the ratio of augmented to normal thrust is not appreciably affected by increase in burner drag at low burner-inlet velocities. At the higher inlet velocities, however, the adverse effects of high drag coefficient are of significant magnitude; for example, at 600 feet per second, an increase in drag coefficient from 0.5 to 2 reduces the augmented-to-normal thrust ratio from 1.39 to 1.23 for the 3800° R gas temperature condition and from 0.98 to 0.89 for the nonburning condition. At 400 feet per second, which may be considered a desirable design value for burner-inlet velocity, the loss in performance with increase in drag coefficient is about 40 percent as much as at 600 feet per second. Although low burner drag is advantageous for obtaining maximum thrust, some drag is necessary for satisfactory combustion as is discussed in the second and third papers of this series.

Nozzle velocity coefficient. - In figure 5 the ratio of augmented to normal thrust is plotted against nozzle velocity coefficient for sea-level altitude, tail-pipe gas temperatures of 3800° and 1650° R, and flight Mach numbers of 0, 0.75, and 1.50. For these calculations, the turbine-outlet velocity was 750 feet per second; diffuser efficiency, 80 percent; burner-inlet velocity, 400 feet per second; and drag coefficient, 1. The variation in nozzle velocity coefficient applies only to the tail-pipe-burner configuration, that is, the normal engine thrust used as the base of the augmented ratio is calculated for a constant value of the coefficient of 0.975.

The augmented-to-normal thrust ratio decreases linearly with decrease in nozzle velocity coefficient, the decrease being greater at the high than at the low flight Mach numbers. For example, decrease in nozzle coefficient from 0.975 to 0.850 for the augmented condition results in a 13 percent reduction in thrust ratio at 0 Mach number and 23 percent reduction at 1.50 Mach number. For the nonburning condition, the thrust reductions are 13 percent at 0 Mach number and 42 percent at 1.50 Mach number. The percentage decrease in high-speed thrust accompanying decrease in nozzle velocity coefficient is thrust of greater magnitude for the nonaugmented than for the augmented condition. This situation is aggravated by the fact that variable-area exhaust nozzles are more difficult to design for high velocity coefficient in the closed position corresponding to nonburning operation than in the open position corresponding to tail-pipe-burning operation.

Flight conditions. - The effect of flight Mach number on tail-pipe-burning performance has already been partially indicated, however, in order to give a more complete and direct representation of the effects of flight operating conditions, figure 6 has been prepared wherein the ratio of augmented to normal thrust is plotted against flight Mach number for altitudes of sea level and 35,000 feet. The tail-pipe design parameters are the reference values used in preceeding figures. Included for references are curves of the thrust of the normal engine configuration divided by the thrust obtained at sea-level altitude and zero Mach number; a subscript 0 has been used to indicate that the base thrust is for the sea-level, zero Mach number condition.

The somewhat wavy curve of sea-level normal thrust is the result of the combined effects of changing air flow, pressure ratio, propulsive efficiency, and inlet-diffuser efficiency that accompany change in flight Mach number. If a higher inlet-diffuser efficiency had been assumed, the decrease in thrust at high Mach number would not have occurred until a higher flight speed. The 35,000-foot curve is lower than the sea-level curve because of the decreased air density at altitude. It does not fall off as rapidly as the sea-level curve at the high Mach numbers because of the lower air temperature and the consequently higher permissible heat addition before the turbine.

The augmented-to-normal thrust ratio increases considerably with increase in Mach number but is not appreciably affected by altitude up to Mach numbers of about 1.0. At higher Mach numbers, the sea-level augmentation is greater than the high altitude augmentation, attaining a value 4 times the normal thrust at a Mach number of 2.0 compared with a value of 2.7 times the normal thrust at 35,000 feet altitude. A large portion of this reduction is due to the decrease in normal thrust for the sea-level high-speed condition.

The corresponding specific fuel consumptions are shown in figure 7 plotted against Mach number for sea level and 35,000 foot altitudes. The normal fuel consumption increases rapidly with increased flight Mach number, varying from 1.1 pounds per hour per pound of thrust at sea level and 0 Mach number to 2.6 pounds per hour per pound of thrust at 2.0 Mach number. The normal consumption is from 9 to 30 percent lower at the 35,000-foot-altitude condition than at sea level because of the lower atmospheric air temperature. The total fuel consumption for the augmented condition at sea level varies from about 2.5 times the corresponding normal fuel consumption at 0 Mach number to 1.25 times the normal consumption at 2.0 Mach number. At an altitude of 35,000 feet, the augmented consumption is about 17 percent lower than at sea level and, at a Mach number of 2.0, is 1.5 times the corresponding normal consumption.

Pressure ratio. - The effect of change in engine compressor pressure ratio is illustrated in figure 8 where the ratio of augmented to normal thrust is plotted against flight Mach number at the 35,000-foot-altitude condition for design pressure ratios of 4 and 8. The engine with the design pressure ratio of 4 is the reference engine used for all previous calculations. The other engine is assumed to have the same component efficiencies and design conditions as the reference engine except for the higher pressure ratio. Similar to the reference engine, the design pressure ratio of 8 pertains to the sea-level, zero Mach number condition. At the 35,000-foot-altitude condition in figure 8, the actual pressure ratio varies from 5.3 at Mach number of 2.0 to 12.4 at Mach number of 0. The corresponding pressure ratios for the reference engine are between 3.0 and 5.6. The tail-pipe-burner conditions are the same as those used in figure 7.

Included for reference is the normal thrust of each engine divided by the normal thrust of the engine with a pressure ratio of 4 at the sea-level, zero Mach number condition, designated by the subscripts 0, 4. The high-pressure engine develops more than 100-percent-higher normal thrust than the low-pressure engine at 0 Mach number and about 60 percent more at 2.0 Mach number. The augmented-to-normal thrust ratio of the high-pressure engine is only between 6 and 11 percent higher than that of the low-pressure engine; however, the actual augmented thrust is much greater because of the higher normal thrust.

The corresponding specific fuel consumptions of the two engines are compared in figure 9. For the normal engine configuration, the high-pressure engine shows a 15-percent-lower fuel consumption than the low-pressure engine at 0 Mach number and a few-percent-lower consumption at 2.0 Mach number. For the augmented condition the high-pressure engine provides about 8-percent-lower fuel consumption than the low-pressure engine at 0 Mach number and slightly higher consumption at 2.0 Mach number. Thus from figures 8 and 9 it appears that higher pressure ratio engines than those in current use are advantageous both for normal and tail-pipe-burning operation for the range of flight Mach numbers considered.

Altitude Limits

The discussion thus far has dealt with the thrust and fuel consumption of tail-pipe burners. Another important performance criterion is the combustion stability or blow-out limits of the burner, which determine the maximum altitude at which the burner will operate. Such information cannot be readily predicted from analysis but must be obtained experimentally in altitude test chambers, altitude tunnels, or in flight. Analysis can, however, provide methods for generalizing the blow-out data and thus reduce the amount of testing required to establish the altitude limits.

Experience with turbojet and ram-jet engine combustion chambers indicated that the combustion blow-out of a given configuration and fuel-air ratio is affected by combustion-chamber-inlet velocity, inlet temperature, and inlet pressure. Tail-pipe-burner blow-out should be affected by the same parameters; accordingly, figure 10 illustrates the variation of tail-pipe-burner-inlet temperature, velocity, and static pressure with flight Mach number and altitude for a current turbojet engine operating at rated engine speed.

Over the range of flight Mach number and altitude, the burner-inlet or turbine-outlet temperature is held constant at the maximum permissible value by varying either the exhaust-nozzle area or the burner fuel flow. The corresponding burner-inlet velocities are substantially constant with flight Mach number but decrease with increased altitude, the change being smaller at high altitudes. Between 20,000 and 40,000 feet, burner-inlet velocity decreases about 6 to 10 percent. This variation is characteristic of the particular engine under consideration. Other engines for which data have been obtained at the Lewis laboratory indicate an even smaller change in burner-inlet velocity with flight conditions and in some cases the change with altitude is in the opposite direction to that shown in figure 10.

The burner-inlet pressure increases with increase in flight Mach number and decreases with increase in altitude. At a Mach number of 1.0, the pressure at 20,000 feet is double that at 40,000 feet. Thus the variation of inlet pressure with flight operating conditions is considerably greater than the variation of the other inlet conditions. It might then be expected that altitude blow-out data for constant engine speed could be correlated simply with burner-inlet pressure.

Such a correlation is illustrated in figure 11 for a typical tail-pipe burner on the same engine used in figure 10, again operating at rated engine speed and constant turbine-outlet temperature. Lines of constant burner-inlet pressure as obtained from normal engine-performance characteristics are plotted on coordinates of flight Mach number and altitude. Each line in figure 11 represents the combinations of altitude and Mach number at which the particular pressure is obtained in the tail pipe. The data points represent experimentally determined blow-out limits for the specific tail-pipe burner; for example, at a Mach number of 0.3, blow-out occurred at 32,000 feet altitude, and at a Mach number of 0.97, blow-out occurred at an altitude of 41,000 feet.

Higher altitude limits than these have been obtained with other tail-pipe burners; however, the data for this particular burner serve to illustrate general trends. Because blow-out is sensitive to small differences in operational technique, the data do not delineate a definite curve of altitude limit but indicate a band of altitude (5000 to 8000 ft wide) in which blow-out may occur. Similar bands of blow-out limits are generally obtained in testing other burners.

The data tend to fall within a band of constant pressure lines, in this case between 20 and 25 inches of mercury. It thus appears that if the altitude blow-out limit for a tail-pipe burner is obtained at one flight Mach number, the limits for other Mach numbers can be predicted from a knowledge of flight operating characteristics of the engine. Tail-pipe fuel-air ratio has considerable effect on altitude limits; in these tests the fuel-air ratio did not, however, have to be varied appreciably to maintain constant turbine-outlet temperature over the range of flight operating conditions. Similar data will have to be obtained with other engines and other tail-pipe burners before this method can be unreservedly accepted.

SUMMARY OF RESULTS

This theoretical investigation indicated the desirability of designing tail-pipe burners with low burner-inlet velocity, low burner drag, high diffuser efficiency, and high exhaust-nozzle velocity coefficient. These design criteria are considered essential not only for obtaining high augmentation, but also for minimizing the loss in normal engine performance during non-burning operation. Low turbine-outlet velocity was shown as a favorable engine design characteristic for tail-pipe-burning application, and higher pressure ratios than those currently used appeared to be advantageous for flight Mach numbers up to at least 2.0. Thrust augmentation increased considerably with increased flight Mach number but it was not appreciably affected by altitude except at Mach numbers above 1 where augmentation decreases with increased altitude. The total specific fuel consumption during tail-pipe-burning operation is about 2.5 times the normal consumption at sea level and 0 flight Mach number but was only 1.5 times the normal consumption at 35,000 feet altitude and a Mach number of 2.0.

REFERENCE

1. Bohanon, H. R., and Wilcox, E. C.: Theoretical Investigation of Thrust Augmentation of Turbojet Engines by Tail-Pipe Burning. NACA RM No. E6L02, 1947.

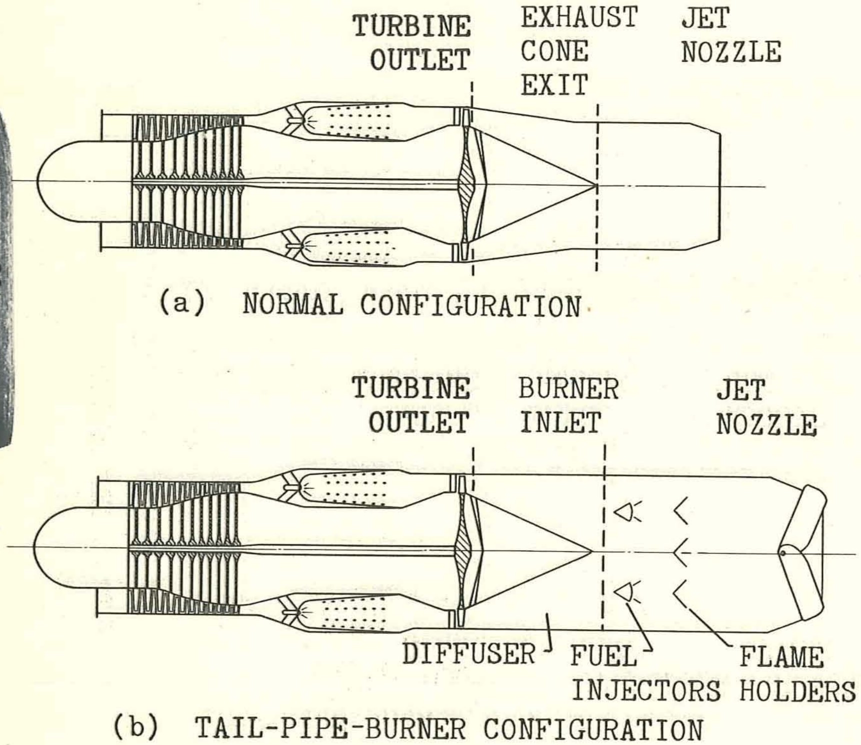
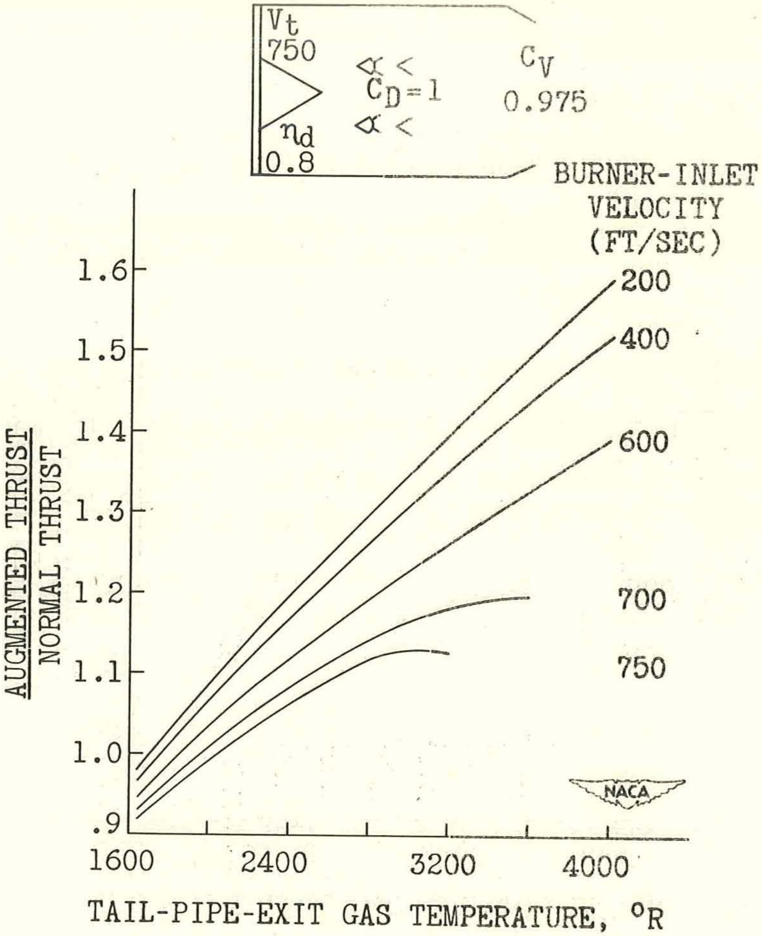


Figure 1. Normal and tail-pipe-burner engine configuration.

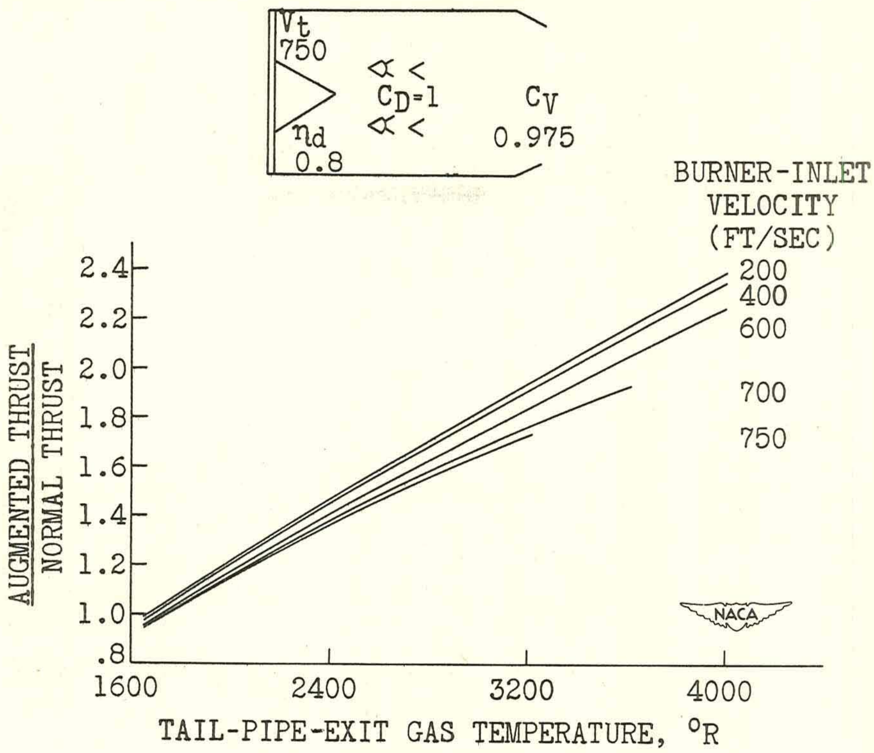


(a) Altitude, sea level; flight Mach number, 0.

Figure 2. - Effect of tail-pipe-exit gas temperature and burner-inlet velocity on augmentation.

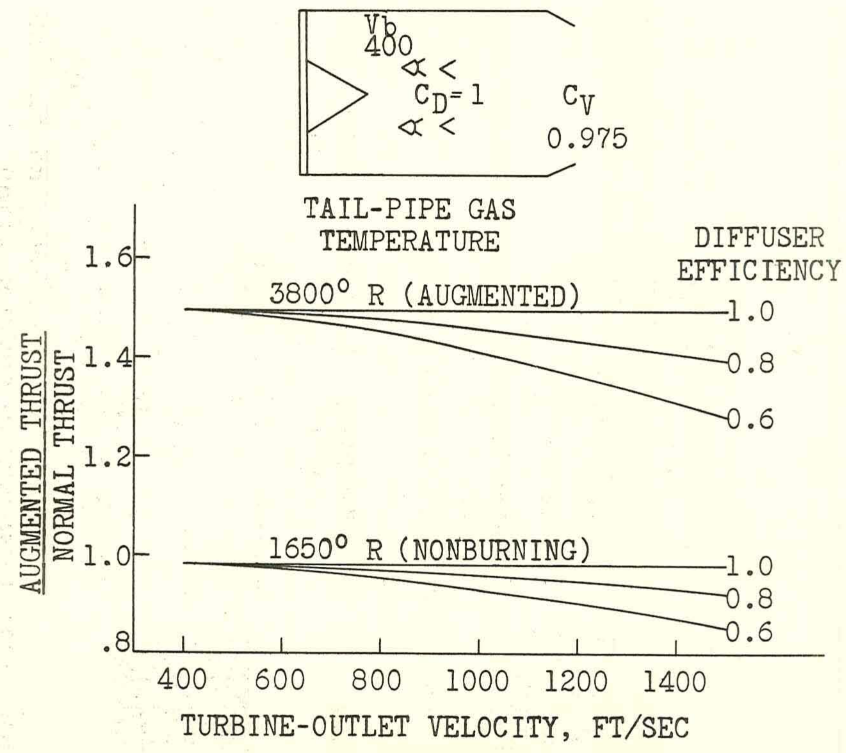
[redacted]

1041A



(b) Altitude, 35,000 feet; flight Mach number, 1.50.

Figure 2. - Concluded. Effect of tail-pipe-exit gas temperature and burner-inlet velocity on augmentation.



ALTITUDE, SEA LEVEL; FLIGHT MACH NUMBER, 0

Figure 3. - Effect of turbine-outlet velocity and diffuser efficiency on augmentation.

[redacted]

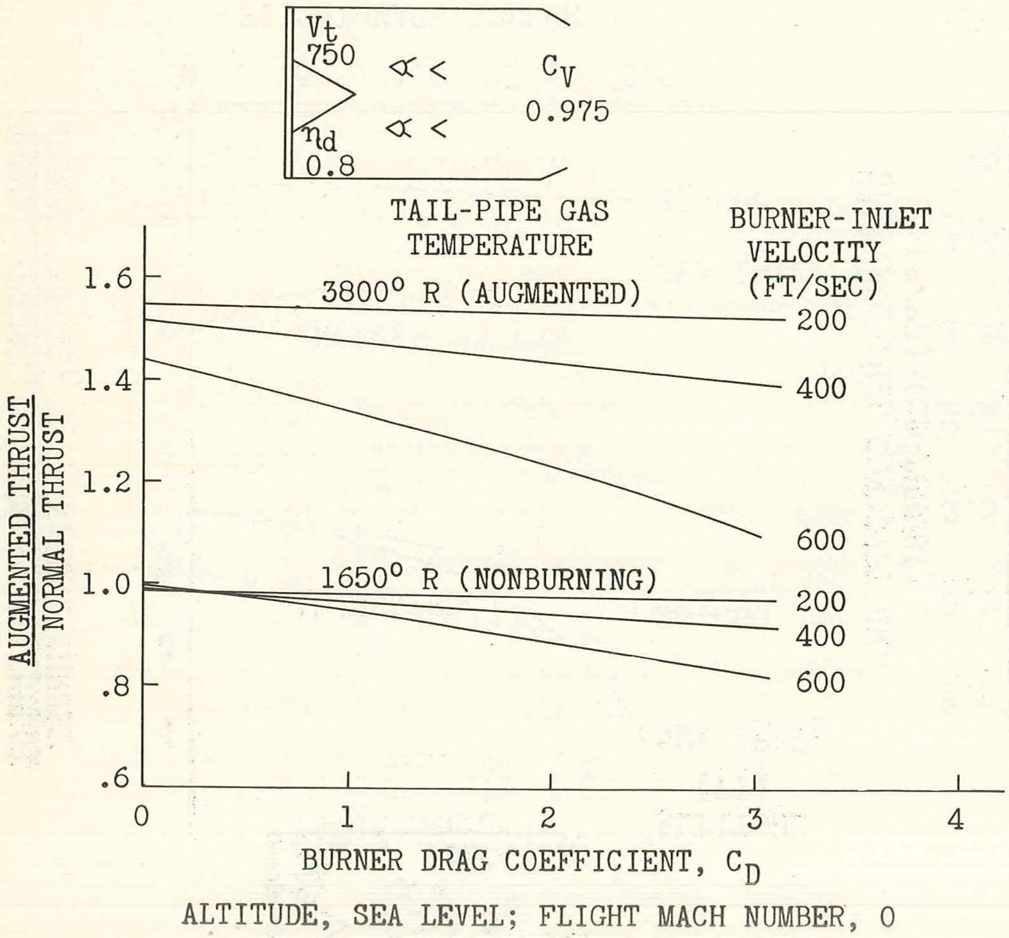


Figure 4. - Effect of burner drag and burner-inlet velocity on augmentation.

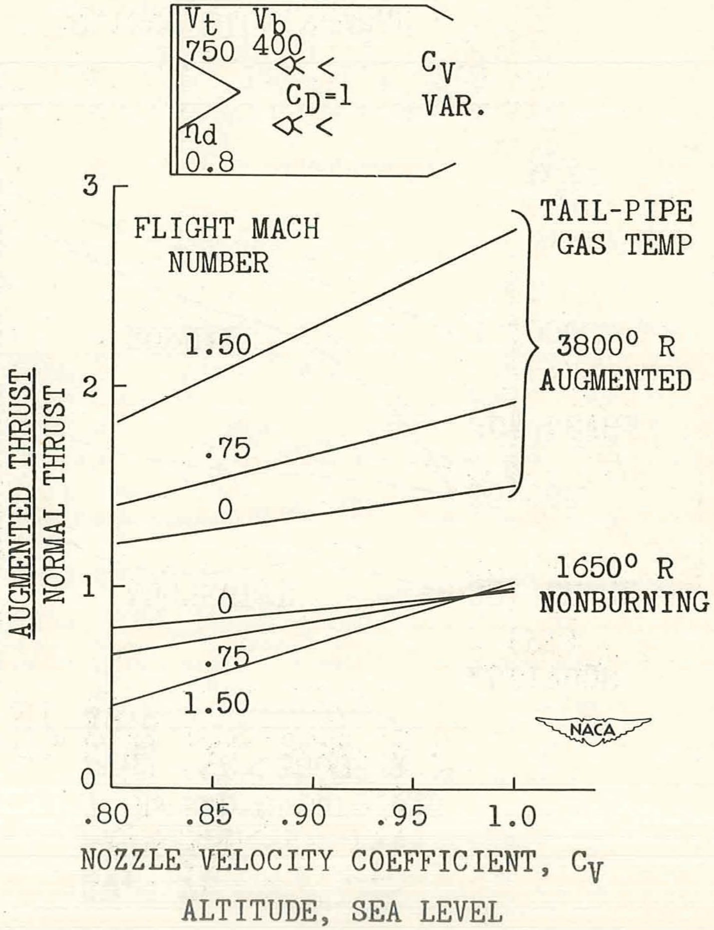


Figure 5. - Effect of tail-pipe-burner nozzle velocity coefficient on augmentation.

[redacted]

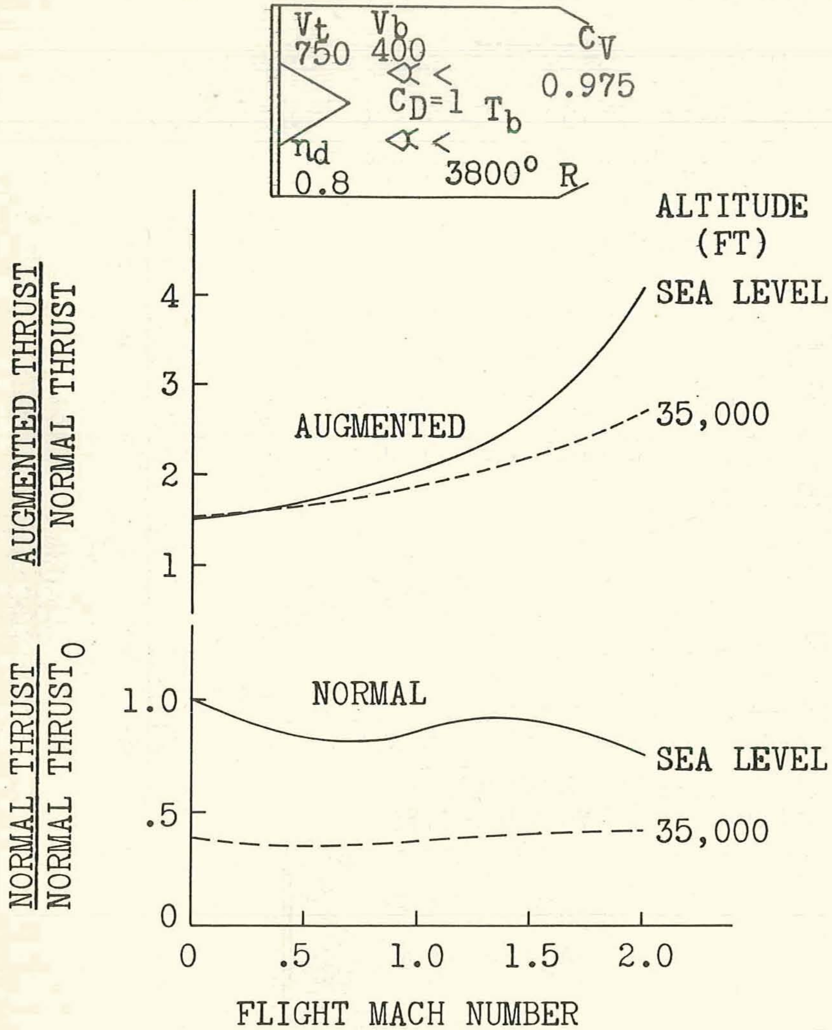


Figure 6. - Effect of flight conditions on augmentation.

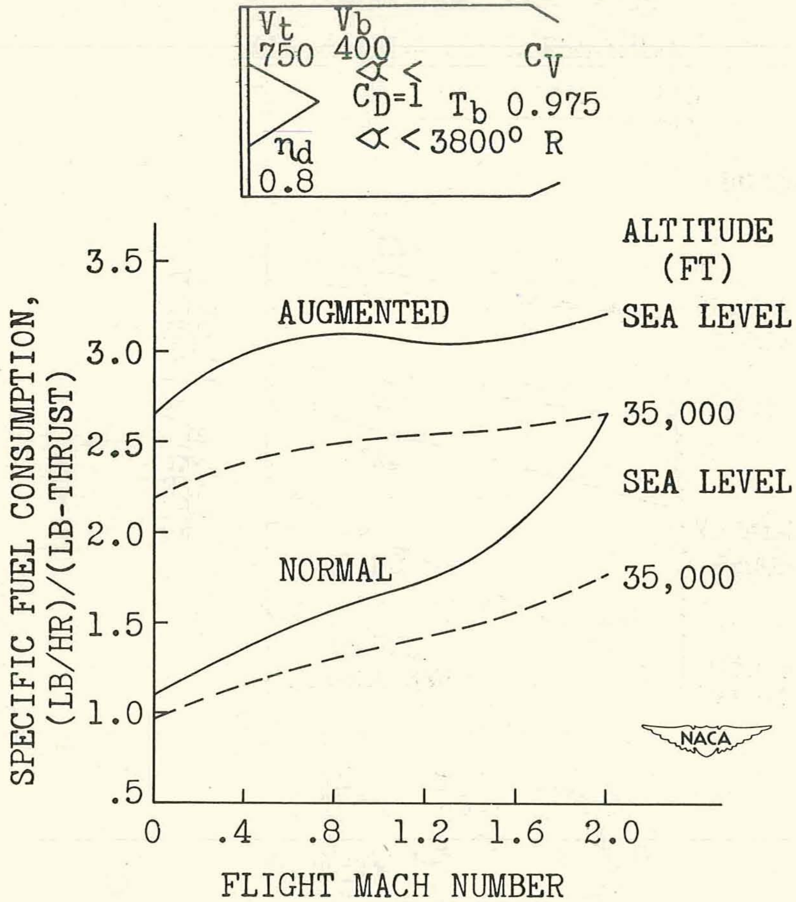


Figure 7. - Effect of flight conditions on fuel consumption.

[redacted]

1041A

[redacted]

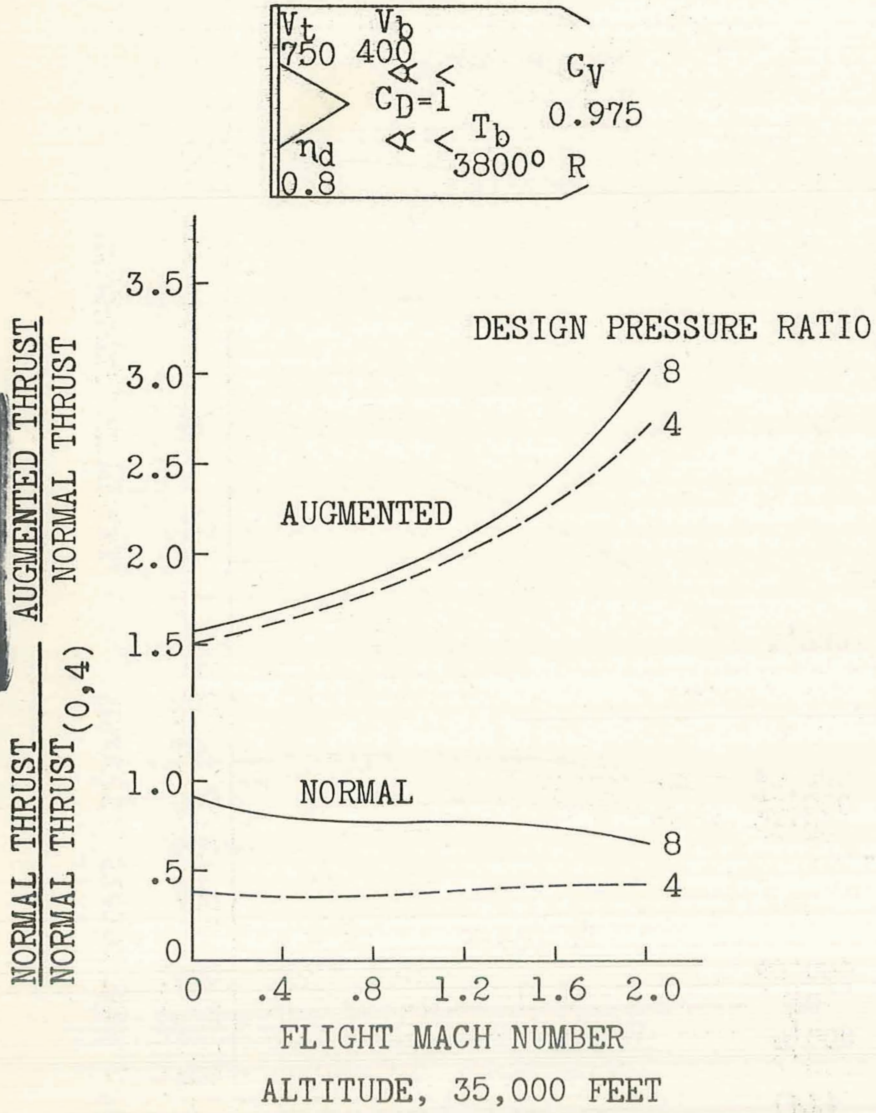


Figure 8. - Effect of compressor pressure ratio on augmentation.

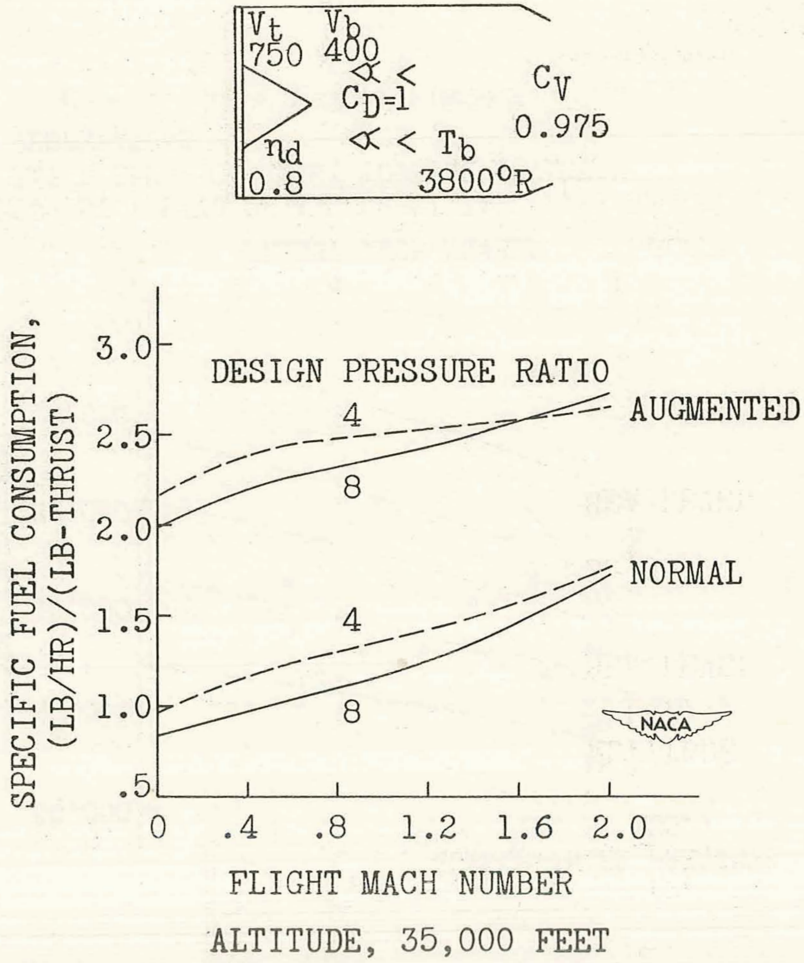


Figure 9. - Effect of compressor pressure ratio on fuel consumption.

[redacted]

[redacted]

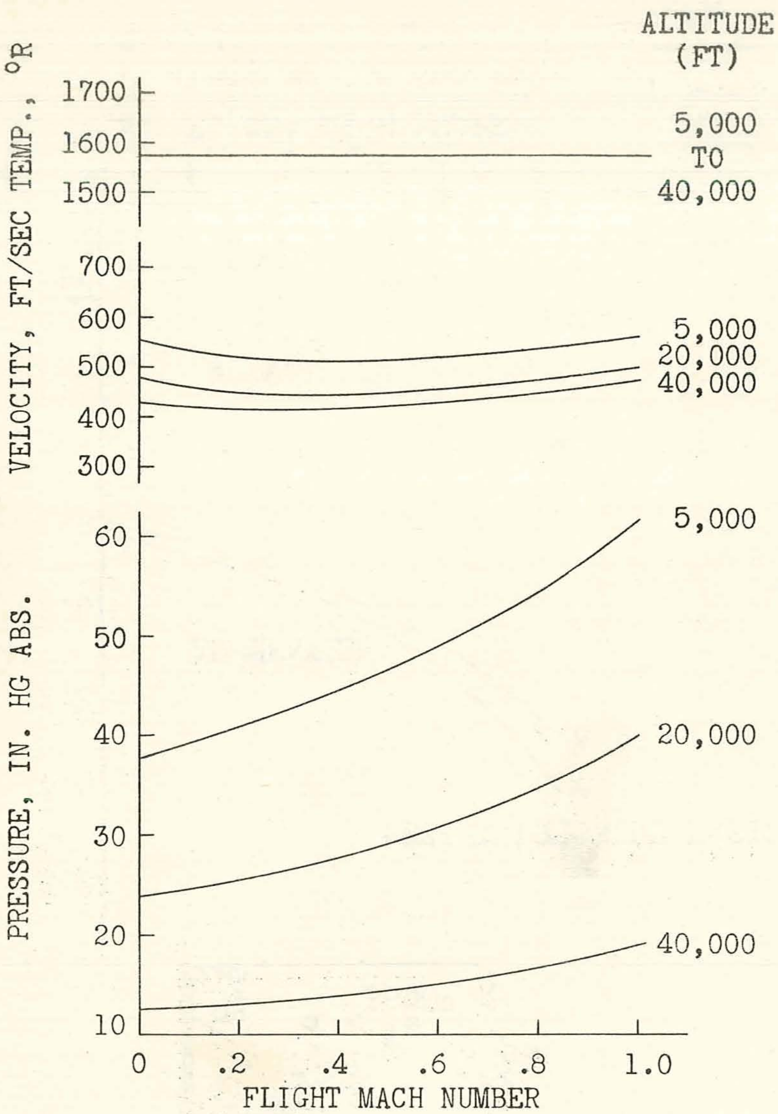
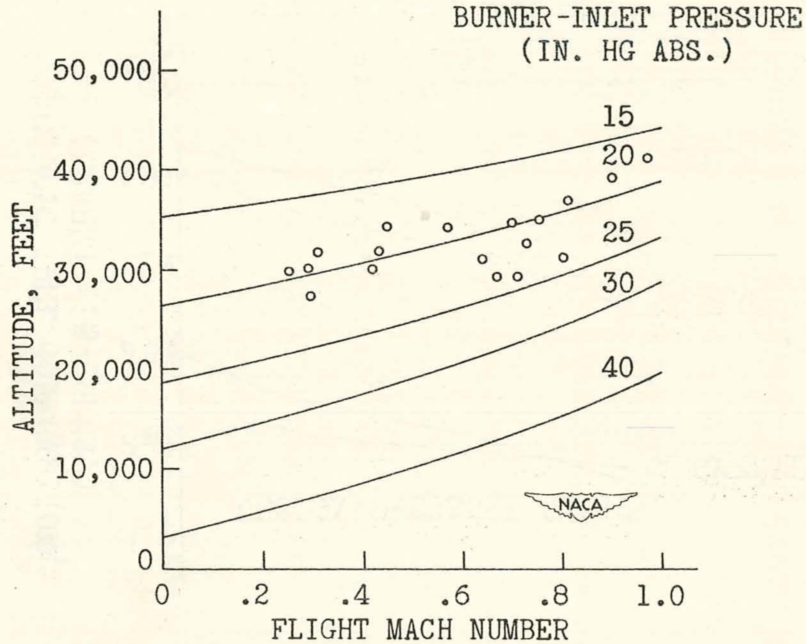


Figure 10. - Tail-pipe-burner conditions for turbojet engine.



BLOW-OUT LIMITS OF A CURRENT TAIL-PIPE BURNER
RATED RPM - CONSTANT TURBINE-OUTLET TEMPERATURE

Figure 11. - Blow-out limits of a typical tail-pipe burner.

[redacted]