COMPARISON OF VARIOUS METHODS OF THRUST AUGMENTATION

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INTRODUCTION

The previous papers covered both theoretical and experimental results on the performance of various thrust-augmentation methods. Computations, guided by the experimental results, are used to compare the relative efficiency, additional weight, and applicability of each thrust-augmentation method over a range of flight conditions. The methods considered are tail-pipe burning, water injection at the compressor inlet, a combination of tail-pipe burning plus water injection, and bleedoff with water injection. The tail-pipe-burning plus water-injection method, although not previously discussed, is also considered to be of interest and is included to show what might be expected of this method. Because rocket-assist units have been used to a large extent in assisting the take-off of conventional aircraft, their performance is compared with that of the other methods on the basis of liquid consumption. A more complete description of the systems, methods of analysis, and results are presented in reference 1.

THRUST-AUGMENTATION ANALYSIS

The comparison of augmentation methods was made for altitudes of sea level and 35,000 feet and for flight Mach numbers of 0, 0.85, 1.50, and 2.50. The thrust augmentation of the engine was determined from step-by-step calculations of the performance of both the normal and the augmented engine. The efficiencies chosen were polytropic and are for the compressor, 0.80; for the turbine, 0.85; for the exhaust nozzle, 0.95; and for the inlet diffuser, 1.00, 0.85, 0.80, and 0.70 for flight Mach numbers of 0, 0.85, 1.50, and 2.50, respectively. The primary combustion-chamber total-pressure loss was assumed to be 3 percent of the combustion-chamber-inlet total pressure. The drag coefficient of the tail-pipe burner (ratio of total-pressure loss to inlet velocity head) was assumed to be 0.5. Two fixed engines, one having a constant compressor work input of 85 Btu per pound of air and the other having twice the work input or 170 Btu per pound of air, were assumed to operate over the entire range of altitudes and flight speeds. These values of work input give compressor pressure ratios at sea-level static conditions of about 4 and 11 and correspond to one- and two-stage centrifugal compressors, respectively. The exhaust-nozzle area was assumed to be adjusted for all cases to give tail-pipe temperatures of 1650° R for the engine with the low-pressure-ratio compressor and 1500° R for the engine with the high-pressure-ratio compressor. These values give turbine-inlet temperatures of about 1960 and 2100° R, respectively.

With tail-pipe burning, the tail-pipe area was assumed to be double the normal tail-pipe area in order to establish a reasonable burner-inlet velocity; charts, which account for the effects of dissociation, were used to calculate the temperatures for various tail-pipe fuel-air ratios. Calculations were made for fuel-air ratios up to stoichiometric.

For the water-injection calculations, the component efficiencies were altered to bring agreement between theoretical and experimental results by the same methods as previously discussed in the fifth paper on the analysis of water injection. For each flight condition, calculations were made with varying amounts of water injected at the compressor inlet to the point where the compressor-outlet air was saturated.

With bleedoff the amount of augmentation for a given liquid injection increases rapidly with an increase in the amount of air bled off. Bleeding off large quantities, however, increases the mass flow of air through the compressor and also the pressure ratio across the turbine and may result in large decreases in the efficiencies of these components. It was found from theoretical considerations that by maintaining the area of the primary-engine exhaust nozzle the same as for normal engine operation at sea-level static conditions the change in the operating conditions of these components was very small (less than the change with only water injection at the compressor inlet. For each flight condition, bleedoff was considered for the case where just sufficient water was injected at the compressor inlet to saturate the air at the compressor outlet. In all cases, the turbine-outlet temperature was maintained at the assumed value by adjusting the amount of air bled off for each amount of water injected in the combustion chamber. The ratios of water flow injected in the primary combustion chamber to bleedoff flow were calculated theoretically. The values obtained gave lower values of the bleedoff flow for a given liquid flow than were obtained experimentally. The bleedoff or auxiliary burner was assumed to operate with a stoichiometric fuel-air ratio in all cases.

The rockets were assumed to have a specific impulse of 190 pounds-seconds per pound for all conditions of altitude and flight speed.

Based upon these assumptions, the values of augmentation to be presented are somewhat higher than the values obtained experimentally at the present time from the various methods. The relative values of the maximums are, however, believed to be indicative of what may be expected in actual practice.

RESULTS AND DISCUSSION

Thrust Augmentation

On the basis of the given assumptions and methods of analysis, figure 1 shows a comparison of the thrust-augmentation methods. The ratio of augmented thrust to normal thrust is plotted against the ratio of total liquid consumption of the augmented engine to fuel consumption of the normal engine. The results are for the engine with the low-pressure-ratio compressor. Because the engine was assumed to operate at a constant rotor speed, the compressor pressure ratio changes with change in flight conditions and with the injection of water. The range of pressure ratios obtained with the low-pressure-ratio compressor (fig. 1(a)) is from 4.3 to 5.0 at sea-level static conditions. The pressure ratio of 4.3 is obtained with the normal engine and the pressure ratio of 5.0 is obtained with water injection.

With water injection, a thrust ratio of 1.32 can be obtained at a liquid ratio of 5.0 when injecting sufficient water to saturate the air at the compressor outlet. The condition for which just sufficient water is injected to saturate the air at the compressor inlet is represented by the lowest cross (fig. 1(a)). The most economical method is tail-pipe burning, which can provide a maximum thrust ratio of 1.55 at a liquid ratio of 4.0; this thrust ratio is comparable to 1.32 with water injection at this liquid ratio. The circle (on the tail-pipe-burning curve fig. 1(a)) represents a stoichiometric fuel-air ratio and is the maximum thrust ratio that can be obtained with tail-pipe burning. Large increases in thrust can be obtained by adding water injection to the tail-pipe-burning method when the over-all fuel-air ratio remains at stoichiometric. A thrust ratio of 2.05 at a liquid ratio of 8.0 can be obtained by this method when saturating the air at the compressor outlet. The cross on the water-injection curve (fig. 1(a)) represents the condition for which just sufficient water is injected to saturate the air at the compressor inlet.

Bleedoff is less efficient than tail-pipe burning or the combination of tail-pipe burning and water injection, inasmuch as a higher liquid ratio is required for the same thrust ratio. A higher ratio is possible, however, with bleedoff. For example, a thrust ratio of 2.60 can be obtained with bleedoff as compared to a maximum of 2.05 for the combination of tail-pipe burning and water injection. Higher values of the thrust ratio are not possible with bleedoff because stoichiometric fuel-air ratio was reached in the primary combustion chamber. If the amount of water injected at the compressor inlet is limited to that amount required to saturate the air at the compressor inlet instead of the outlet, the maximum thrust ratio of 2.20 is represented by the cross below the curve for bleedoff (fig. 1(a)).

The rocket-assist method is the least efficient inasmuch as it requires the highest liquid ratio for a given thrust ratio. The rocket-assist method, however, has no theoretical limit. The specific impulse of 190 assumed for the rockets is an average value for current rockets. Values as high as 220 are obtained on some commercially available units and values of 350 can be obtained theoretically with liquid oxygen and hydrogen. The specific fuel consumption is inversely proportional to the specific impulse. For a thrust ratio of 2.05, which can be obtained by the combination of tail-pipe burning and water injection, the bleedoff and rocket-assist methods require from 2 to 2.5 times the liquid ratio.

The results for the same conditions, except at a flight Mach number of 0.85, are shown in figure 1(b). The compressor pressure ratio was reduced to 3.7 at a flight Mach number of 0.85 as compared to 4.3 at a flight Mach number of 0 because of the higher air temperature at the compressor inlet. The curves obtained are similar to those for a flight Mach number of zero except that greater values of the thrust ratio are obtained for the same values of the liquid ratio. For example, at a liquid ratio of 8.0, a thrust ratio of 2.90 can be obtained with the combination of tail-pipe burning plus water injection as compared to 2.05 at a flight Mach number of zero. Water injection with saturation at the compressor inlet is more effective because of the higher compressor-inlet temperature at the higher flight Mach number. A maximum thrust ratio of 3.70 at a liquid ratio of 31.5 can be obtained with bleedoff.

In order to show the comparison of the various methods at high altitudes, figure 1(c) presents the results for an altitude of 35,000 feet. The flight Mach number for this case is 0.85. The trends are similar to those for sea level. Both the maximum thrust ratio and the thrust ratio for a given liquid ratio are, however, less for each method. A maximum thrust ratio of 1.32 can be obtained with water injection at a liquid ratio of 4.5. Water injection with only sufficient water to saturate the air at the compressor inlet is much less effective at the higher altitudes because of the lower inlet-air temperature. With tail-pipe burning, the maximum thrust ratio is 1.92 at a liquid ratio of 3.5. Adding water injection to tail-pipe burning increases the maximum thrust ratio to 2.36 at a liquid ratio of 7.0. Bleedoff can provide a maximum thrust ratio of 2.88 at a liquid ratio of 26.0.

The results for a flight Mach number of 2.50 at 35,000 feet are given in figure 1(d). With an increase in flight Mach number from 0.85 to 2.50, the maximum thrust augmentation obtained from each method is increased about five times. The effectiveness of water injection is more rapidly increased than the effectiveness of the other methods by an increase in flight Mach number. Saturating the air at the compressor inlet gives a thrust ratio of 2.4 with water injection at a liquid ratio of 9.6. The maximum thrust ratio obtained with water injection is 3.5 at a liquid ratio of 18.6. With tail-pipe burning, the maximum thrust ratio is 5.5 at a liquid ratio of 6.0. Adding water injection to tail-pipe burning until the compressor-inlet air is saturated results in a thrust ratio of 8.2 at a liquid ratio of 15.0 and increasing the water-injection rate results in a maximum thrust ratio of 9.3 at a liquid ratio of 22. The maximum thrust ratio with bleedoff is 10.4 at a liquid ratio of 58. It should be noted that the high thrust ratios at a flight Mach number of 2.50 are primarily due to the low thrust of the normal engine. The high liquid ratios for the methods involving the injection of water are due to the large quantities of water that can be evaporated at the high Mach numbers because of the high inlet-air temperature.

The effect of altitude on the maximum thrust ratio is more clearly shown in figure 2. The maximum thrust ratio of each method is plotted against altitude for a flight Mach number of 0.85. All methods show a moderate decrease in maximum thrust ratio as the altitude is increased to 35,000 feet. Because of the constant air temperature above the tropopause (approximately 35,000 feet), the augmentation remains about constant. The maximum thrust ratio with water injection decreases from 1.59 to 1.32 as the altitude is increased for sea level to 35,000 feet. Tail-pipe burning is affected to a smaller extent by altitude than any of the other methods; the maximum thrust ratio decreases from 2.12 to 1.92. The combination tail-pipe burning plus water injection decreases from 2.90 to 2.36 and bleedoff decreases from 3.70 to 2.88 as the altitude increases from sea level to 35,000 feet.

In order to show the effect of flight Mach number on the augmentation more clearly, the maximum thrust ratio of each method is plotted against flight Mach number in figure 3 for an altitude of 35,000 feet. The flight Mach number ranges from 0.85 to 2.50. All methods show an increase in the maximum thrust ratio as the flight Mach number is increased; although, as shown in figure 1, the increased thrust ratios are obtained at the expense of large increases in the liquid ratios. Increasing the flight Mach number from 0.85 to 2.50 increases the thrust ratio with bleedoff from 2.88 to 10.4 and with the combination of tail-pipe burning plus water injection from 2.34 to 9.3. At a flight Mach number of 2.50, the liquid ratio with bleedoff is about ten times that for tail-pipe burning, and with tail-pipe burning plus water injection, the liquid ratio is about four times that for tail-pipe burning.

The preceeding results are all for an engine having the low-pressure-ratio compressor. The performance of the high-pressure-ratio-compressor engine is presented in figure 4 along with the performance of the low-pressure-ratio-compressor engine for comparison. The thrust ratio is plotted against the liquid ratio for sea level and a flight Mach number of 0.85. For these flight conditions the high-pressure-ratio compressor has a normal pressure ratio of 9.9, which is increased to 14.3 with water injection. In the range of liquid ratios covered by the low-pressure-ratio-compressor engine, there is little difference between values of the thrust ratio for a given liquid ratio for the low- and high-pressure-ratio-compressor engines. Higher values of the thrust ratios are possible with the high-pressure-ratio-compressor engine, but at higher values of the liquid ratio. The greatest increase in thrust ratio is obtained with the methods involving water injection. For example, the maximum thrust ratio with the combination of tail-pipe burning plus water injection increases from 2.90 to 4.36, whereas with tail-pipe burning alone the maximum increases from 2.12 to 2.26. The maximum thrust ratio with bleedoff increases from 3.70 to 4.98.

Weight Analysis

In addition to the weight of the liquid consumed by each method, the weight of the added equipment required by each method is also of importance. Figure 5 shows a comparison of the additional weight of equipment required by the thrust-augmentation methods. The additional weight of equipment divided by the additional thrust or the specific weight of augmentation equipment is plotted against the thrust ratio. The values of the additional thrust and the thrust ratio are for sea-level static conditions. The increased weight is the weight of equipment only and does not include any additional liquids. The equipment weight was estimated from the weight of existing experimental equipment by taking into account any modifications required for airplane installation. For all the methods, the specific weight decreases as the thrust ratio increases. The minimum specific weight is approximately the same for all methods (0.05 to 0.07) at the maximum values of thrust ratio. When all the methods are considered at a constant thrust ratio, the water-injection method entails the least additional weight followed by the tail-pipe burning method. The specific weights of bleedoff and of the combination of tail-pipe burning plus water injection are about equal and have the highest values. The specific weight of an average normal engine is included for reference. It is apparent, however, that adding the additional augmentation equipment to an engine is equivalent to adding an additional engine having a very low specific weight, except for the additional liquid consumed.

Both the weight of the liquids consumed and the weight of equipment of the various methods have been compared. Neither of these comparisons is adequate inasmuch as the weight of the liquids consumed is a function of the time of operation and the equipment weight is fixed. Calculations were therefore made of the total propulsive weight (weight of engine, liquid consumed, and auxiliary equipment) for each method. Figure 6 shows the ratio of the total propulsive weight of an augmented engine to the total propulsive weight of a larger unaugmented or normal engine (both engines producing the same thrust) plotted against the thrust ratio of the augmented engine. The ratio shown is for 5 minutes of operation of each engine at an altitude of 35,000 feet and a flight Mach number of 0.85. The normal engine total propulsive weight is 1. Values less than 1 indicate a reduction in the weight of the augmented engine, equipment, and liquids from that of a normal engine for the same value of thrust. For the tail-pipe-burning method the total weight of the augmented engine decreases as the thrust ratio increases and reaches a value of less than seven-tenths the normal engine total weight. For the water-injection and the tail-pipe-burning plus water-injection methods, the total weight first decreases to a minimum and then increases as the thrust ratio increases. With bleedoff for 5 minutes of operation the least total weight occurs at the lowest thrust ratio and is approximately equal to the total weight of a normal engine. For shorter periods of operation with bleedoff, 3 minutes for example, the lowest total weight occurs at the maximum thrust ratio and is about 0.85 times the normal engine total weight.

Load-Range Characteristics

Because of the high thrusts for a given engine size and weight, engines equipped with the thrust-augmentation methods may be desirable for high-speed flight in spite of their high liquid consumption.

A study was therefore made of the performance of the complete airplane for a flight Mach number of 1.50 at which the engine was assumed to be operating in the augmented configuration for the entire time of flight. The details of this study and method of analysis are similar to those in reference 2 in which various engine types are compared in terms of airplane load-range characteristics.

In figure 7 the airplane disposable load per pound of gross weight is plotted against the liquid rate per mile per ton of gross weight. This comparison is for an altitude of 35,000 feet and a flight Mach number of 1.50. The performance is calculated for level flight only and does not include the take-off and climb requirements. The disposable load is equal to the gross weight less the weight of the airplane structure and the engine and may consist of liquid weight or liquid and cargo weight. A wing lift-drag ratio of 7 was assumed and the airplane structure weight was assumed to be 30 percent of the gross weight. The normal engine specific weight at sea-level static conditions was assumed to be 0.45 and the weight of the additional equipment was the same as that presented in figure 5. The engines were assumed to be installed in nacelles. The nacelle and fuselage drag and the drag of the additional equipment were deducted from the engine thrust in calculating the gross weight from the lift-drag ratio. All calculations were based on an engine having a normal thrust at sea-level static conditions of 10,000 pounds and weighing 4500 pounds. The engine frontal area was taken as $12\frac{1}{2}$ square feet. Each point on the curves in figure 7, therefore, represents a different airplane, because the airplane becomes larger as the augmentation and the thrust increase.

Operation with the normal engine is shown by the cross in the lower left corner. The various lines represent the different methods with varying amounts of augmentation. The numbers on the curves are values of the thrust ratio. The ratio of ordinate to abscissa and, therefore, the slope of the slanting lines connecting the points on the curves with the origin represent the maximum range of the airplane when all the disposable load is fuel. These slopes have been labeled directly in terms of range on the outer scale. The normal engine, for example, has a maximum range of 233 miles. The initial point for tail-pipe burning at a thrust ratio of 1 is at a lower disposable load and range than the normal engine because of the loss in thrust and the increase in drag when the engine is equipped for tail-pipe burning. As the augmentation by tail-pipe burning is increased to stoichiometric, the ratio of disposable load to gross weight and the range are increased without much increase in liquid rate per unit gross weight. The liquid rate per ton mile remains nearly constant with augmentation by tail-pipe burning in spite of the increase in specific fuel consumption for the following reason: One of the important factors that affects this quantity is the liquid rate per hour per pound of net thrust where the net thrust is defined as the difference between the engine thrust and the engine nacelle drag. By augmentation the engine thrust is increased without an increase in nacelle drag and hence the percentage increase in net thrust is greater than the percentage increase in engine thrust. This effect tends to offset the increase in fuel rate per pound of engine thrust to give an approximately constant liquid rate per pound of net thrust (and hence gross weight) with increase in augmentation by tail-pipe burning.

At the point of maximum augmentation by tail-pipe burning the gross weight is approximately 3.5 times the gross weight for normal operation. The decrease in range that is obtained with tail-pipe burning by decreasing the gross weight to the same value as for normal operation is also shown (fig. 7). Although the range is decreased about 100 miles by decreasing the gross weight, the range is still considerably greater than with the normal engine. The maximum range with tail-pipe burning is 590 miles. Adding water injection to tail-pipe burning increases the disposable load but the increase in liquid rate per unit gross weight results in a slightly shorter maximum range than with tail-pipe burning alone. Water injection alone results in about the same maximum range as the normal engine. The principal effect of bleedoff is to increase the disposable load per unit gross weight with an increase in liquid rate per unit gross weight and some decrease in maximum range. From these curves, it may be concluded that for a flight Mach number of 1.50, tail-pipe burning is the only method when used for the entire flight that will increase the range over that of a normal engine.

These results are based on rather conservative estimates of engine performance. In order to determine whether the tail-pipe-burning or the tail-pipe-burning plus water-injection method increases the range of a normal engine when the engine is much more efficient, these methods are compared for two different engines in figure 8. The results for engine A are the same as those presented in figure 7. For engine B both the compressor and turbine efficiencies were increased 5 percent and the inlet-diffuser efficiency was increased from 0.80 to 0.965. Somewhat lower values for the engine weight and frontal area were also assumed. The maximum range of the engine B without tail-pipe burning is 750 miles. As the augmentation increases, the maximum range increases to 1000 miles. Maintaining the same gross weight for the augmented engine as for the normal engine gives a maximum range of 940 miles for engine B with augmentation.

Although this increase in range with the addition of tail-pipe burning is not as great as for engine A, it appears safe to conclude that even with a highly efficient engine the addition of tail-pipe burning increases the maximum range.

SUMMARY OF RESULTS

From a theoretical comparison of various methods of thrust augmentation, it may be stated that either bleedoff or rocket assist offers the possibility of large thrust increases at the expense of high values of specific liquid consumption. For small increases in thrust, water injection offers the advantage of extreme simplicity and light weight. Tail-pipe burning offers the advantages of light weight with thrust increases intermediate between those for water injection and bleedoff and of the lowest values of specific liquid consumption. Tail-pipe burning may, however, involve some loss of thrust during unaugmented operation. The combination of tail-pipe burning plus water injection permits a flexible system, either providing large amounts of augmentation with a moderate specific liquid consumption or smaller amounts with a low specific liquid consumption. For continued operation at supersonic speeds, tail-pipe burning appears to be the only augmentation method of those considered that increases the maximum range over that obtained with the normal engine.

REFERENCES

1. Hall, Eldon W., and Wilcox, E. Clinton: Theoretical Comparison of Several Methods of Thrust Augmentation for Turbojet Engines. NACA RM No. E8H11, 1948.

2. Cleveland Laboratory Staff: Performance Ranges of Application of Various Types of Aircraft-Propulsion System. NACA TN No. 1349, 1947.

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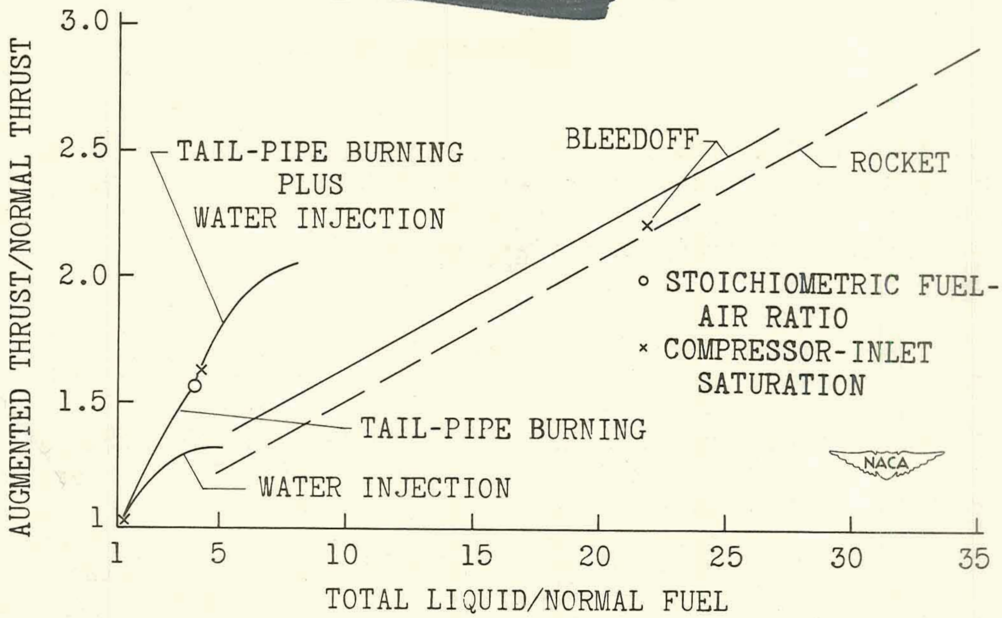
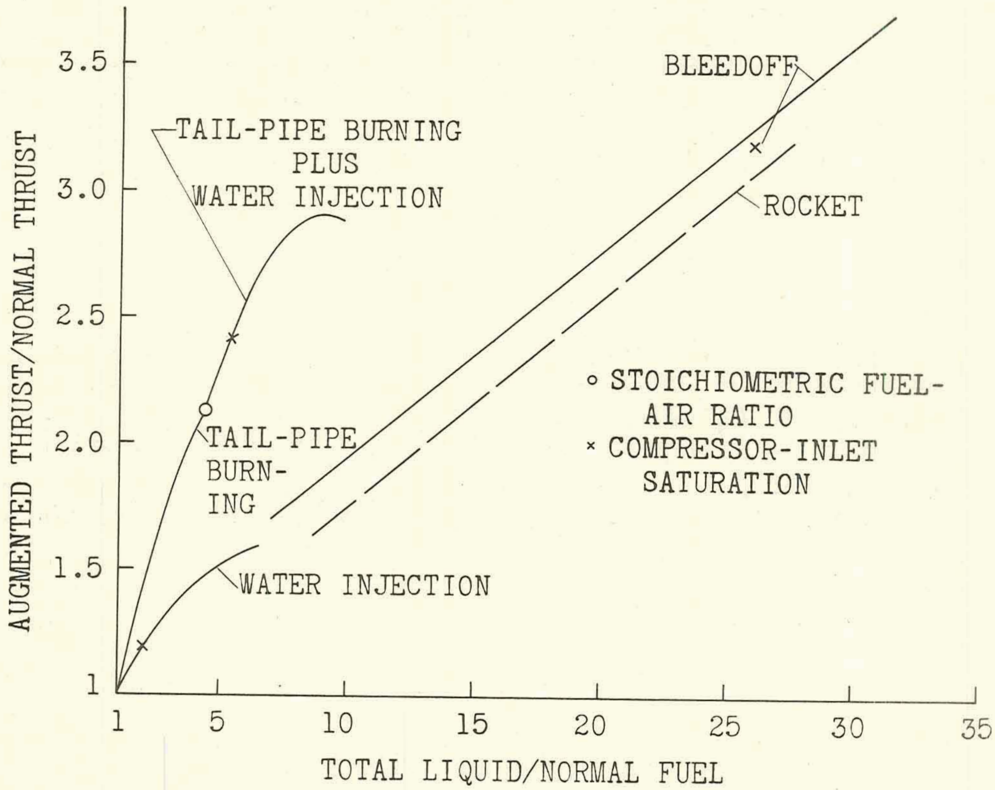


(a) Altitude, sea level; flight Mach number, 0; low-pressure-ratio compressor (pressure ratio, 4.3 to 5.0).

Figure 1. - Variation of ratio of augmented to normal thrust with ratio of total liquid to normal fuel.



(b) Altitude, sea level; flight Mach number, 0.85; low-pressure-ratio compressor (pressure ratio, 3.7 to 4.6).

Figure 1. - Continued. Variation of ratio of augmented to normal thurst with ratio of total liquid to normal fuel.

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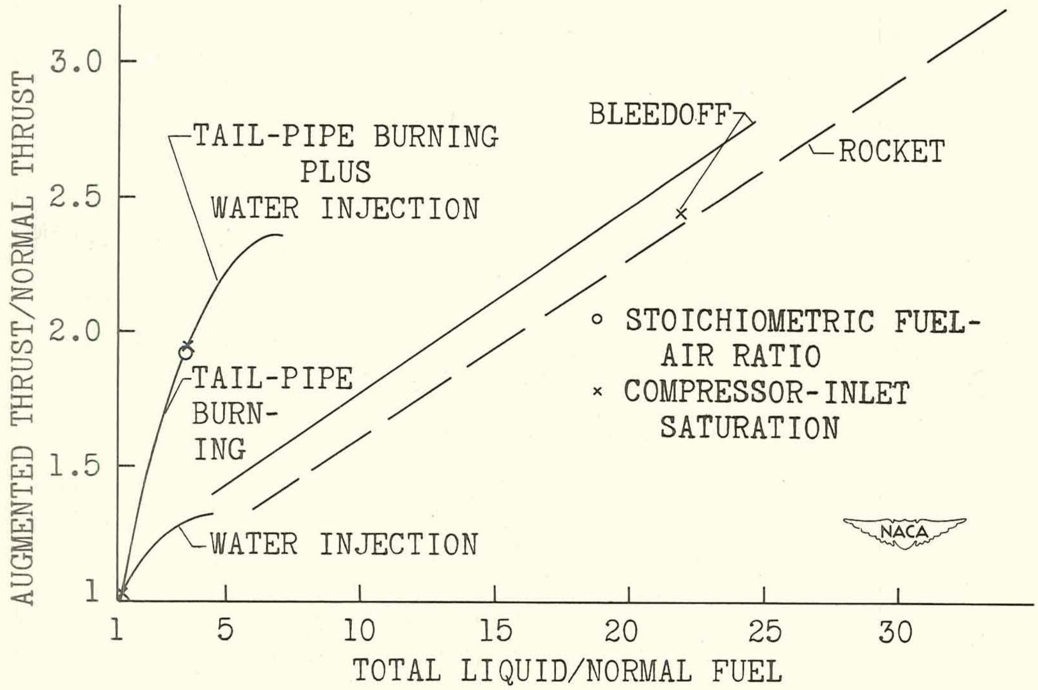
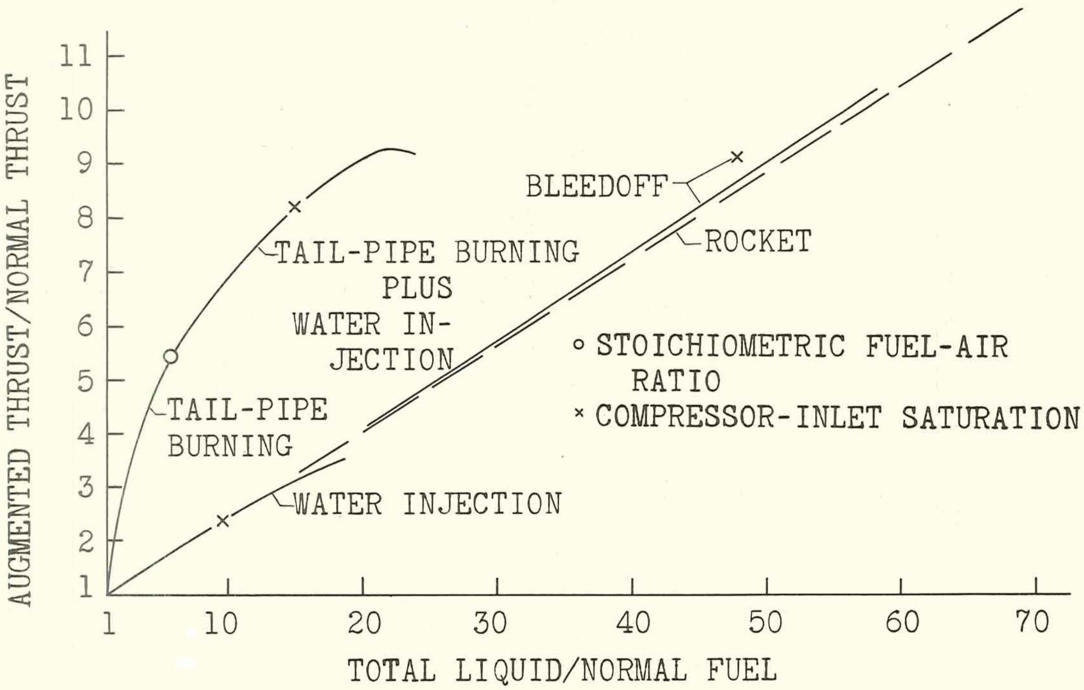


(c) Altitude, 35,000 feet; flight Mach number, 0.85; low-pressure-ratio compressor (pressure ratio, 5.1 to 6.0).

Figure 1. - Continued. Variation of ratio of augmented to normal thrust with ratio of total liquid to normal fuel.



(d) Altitude, 35,000 feet; flight Mach number, 2.50; low-pressure-ratio compressor (pressure ratio, 2.6 to 4.1).

Figure 1. - Concluded. Variation of ratio of augmented to normal thrust with ratio of total liquid to normal fuel.

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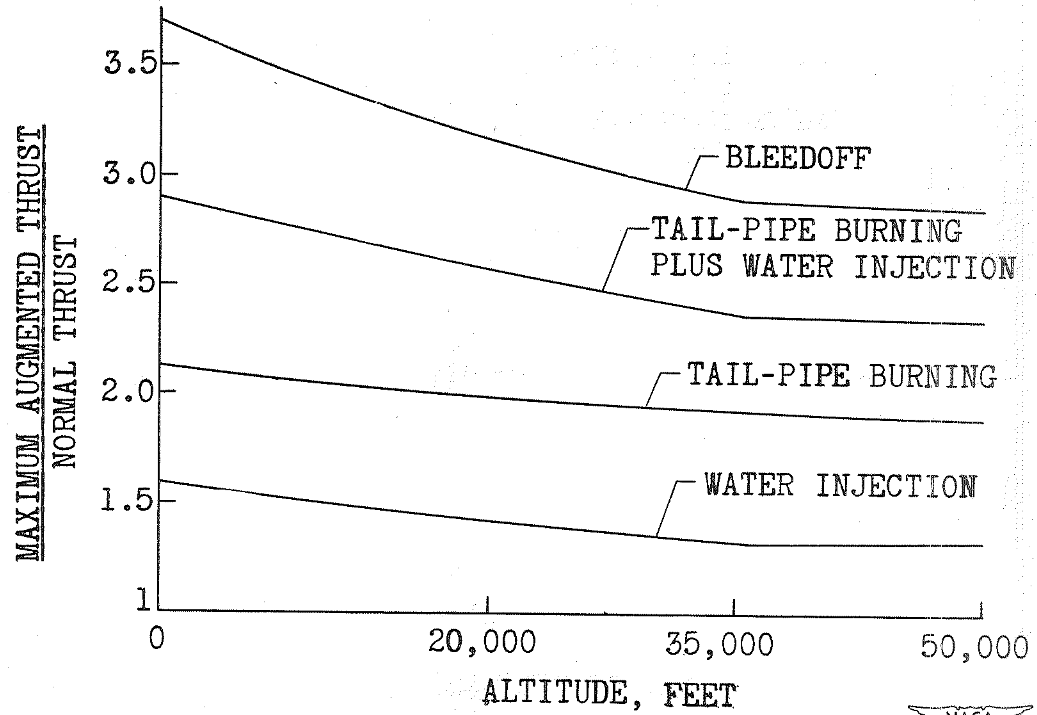

Figure 2. - Effect of altitude on maximum available thrust augmentation. Flight Mach number, 0.85; low-pressure-ratio compressor (pressure ratio, 3.9 to 6.0).

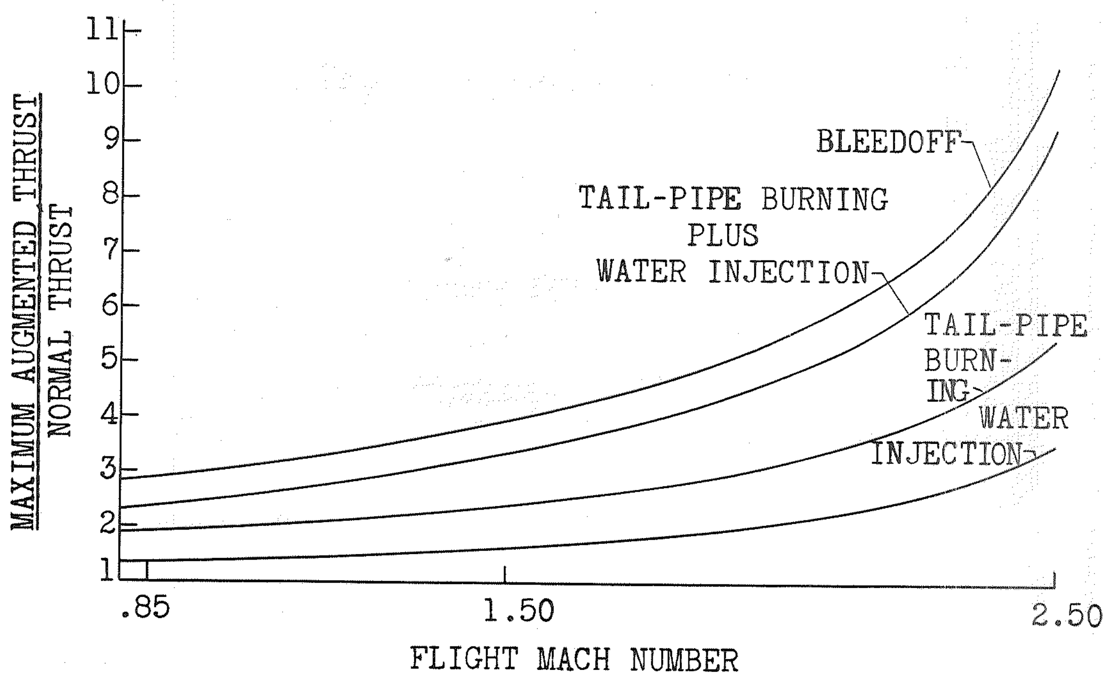

Figure 3. - Effect of flight Mach number on maximum available thrust augmentation. Altitude, 35,000 feet; low-pressure-ratio compressor (pressure ratio, 2.6 to 6.0).

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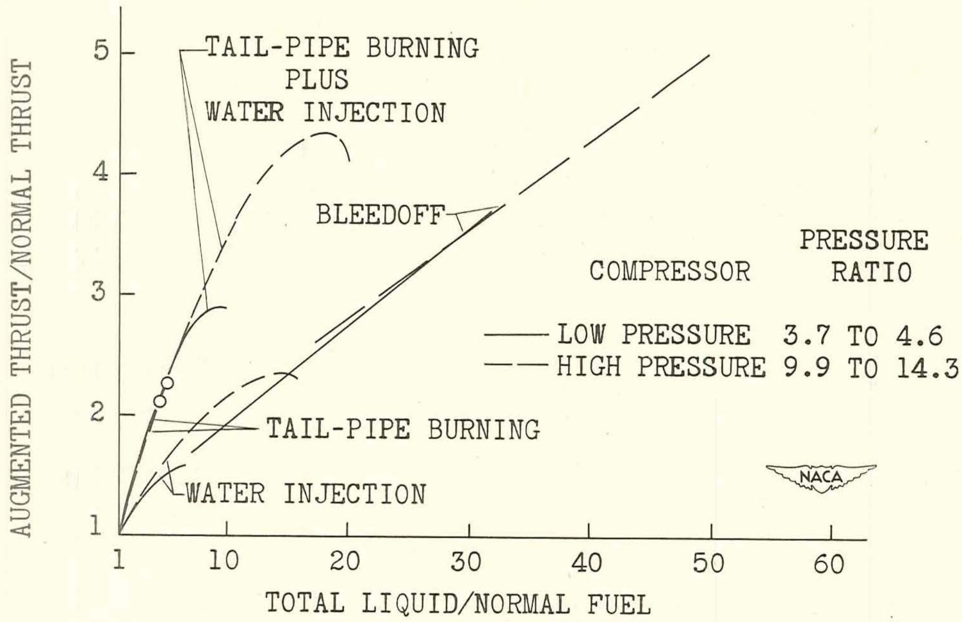

Figure 4. - Effect of compressor pressure ratio on thrust augmentation and ratio of total liquid to normal fuel. Altitude, sea level; flight Mach number, 0.85.

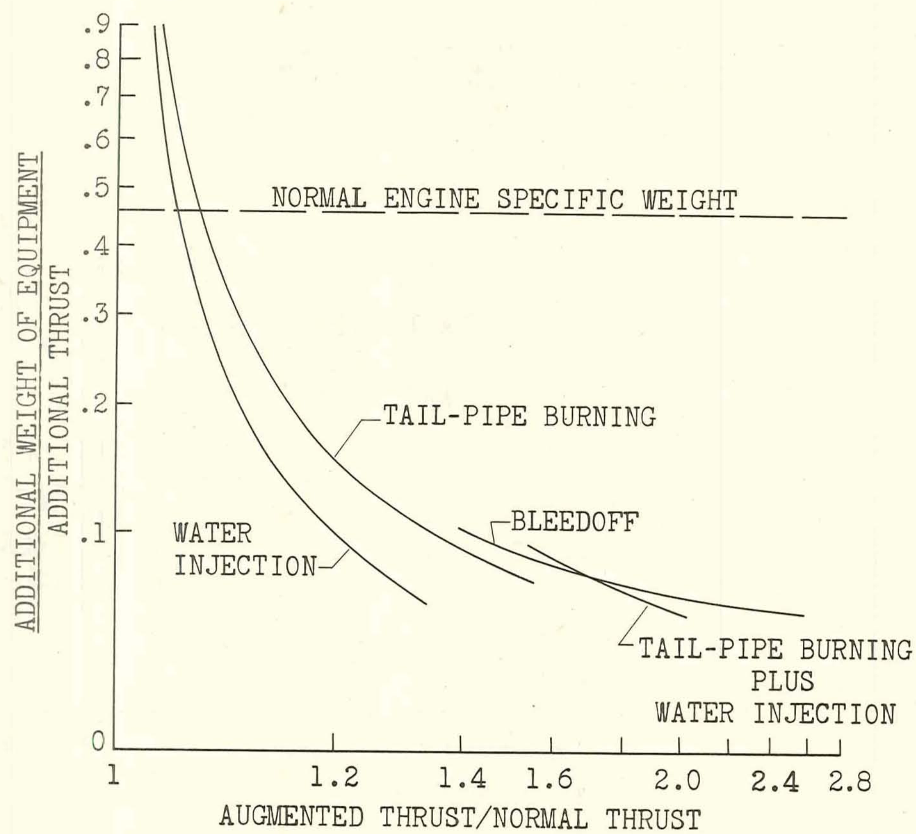

Figure 5. - Variation of specific weight of augmentation equipment with ratio of augmented to normal thrust. Altitude, sea level; flight Mach number, 0; low-pressure-ratio compressor.

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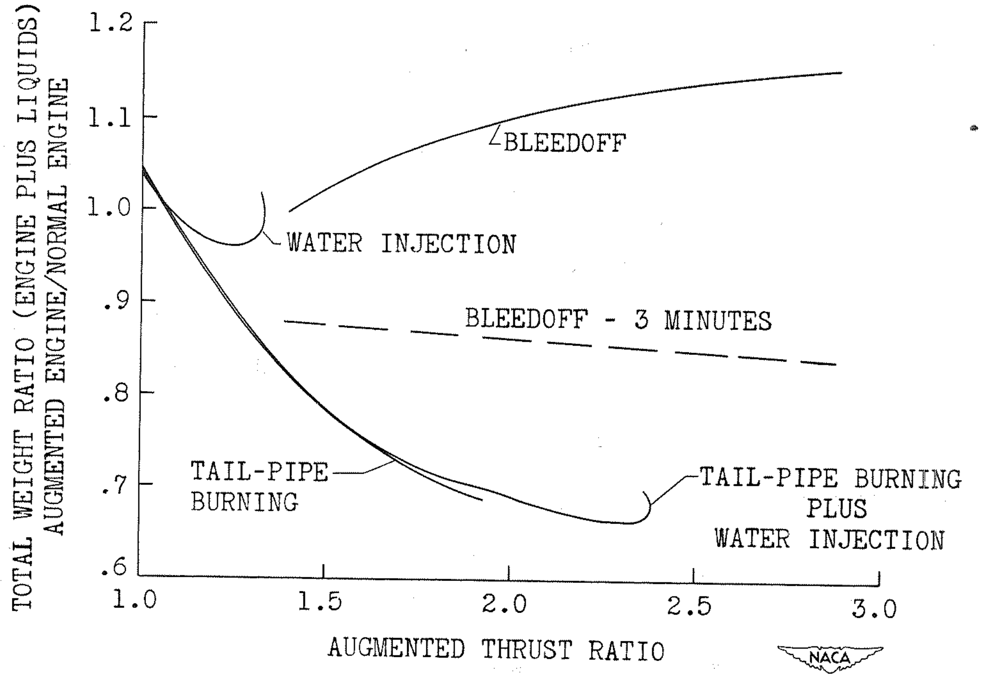

Figure 6. - Effect of thrust ratio on total propulsive weight for given flight time. Altitude, 35,000 feet; flight Mach number, 0.85; time of operation, 5 minutes. (Normal engine with same thrust as augmented engine.)

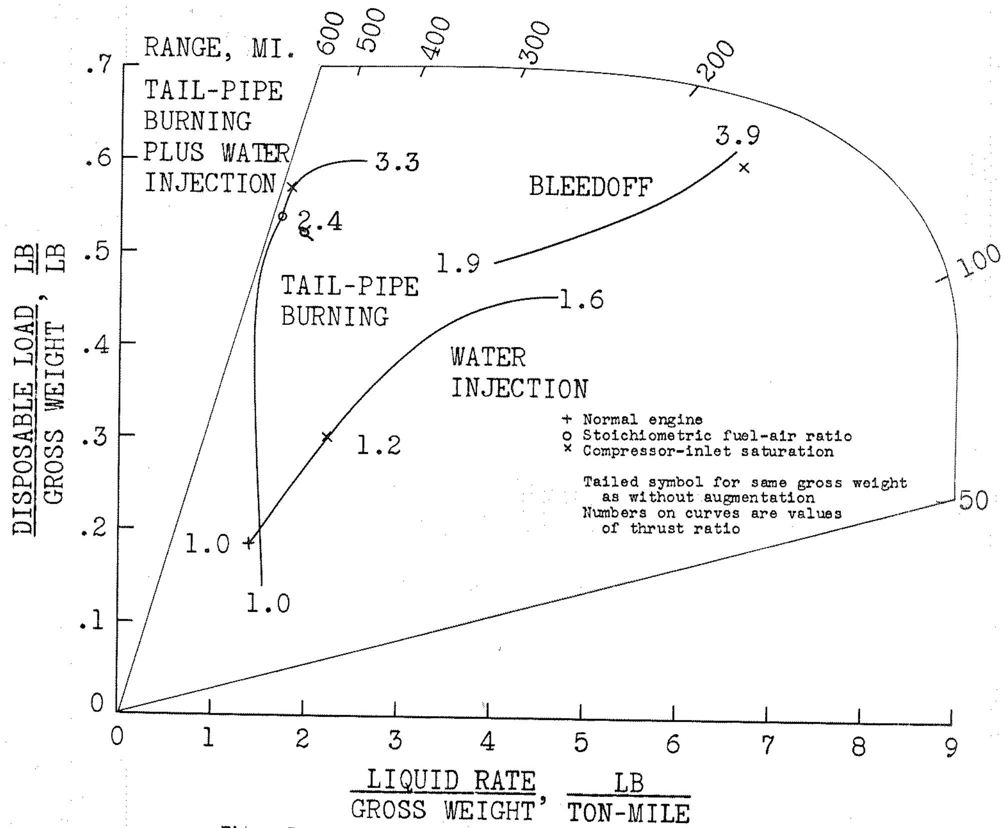

Figure 7. - Load-range characteristics of airplane powered by turbojet engines with various methods of thrust augmentation. Altitude, 35,000 feet; flight Mach number, 1.50.

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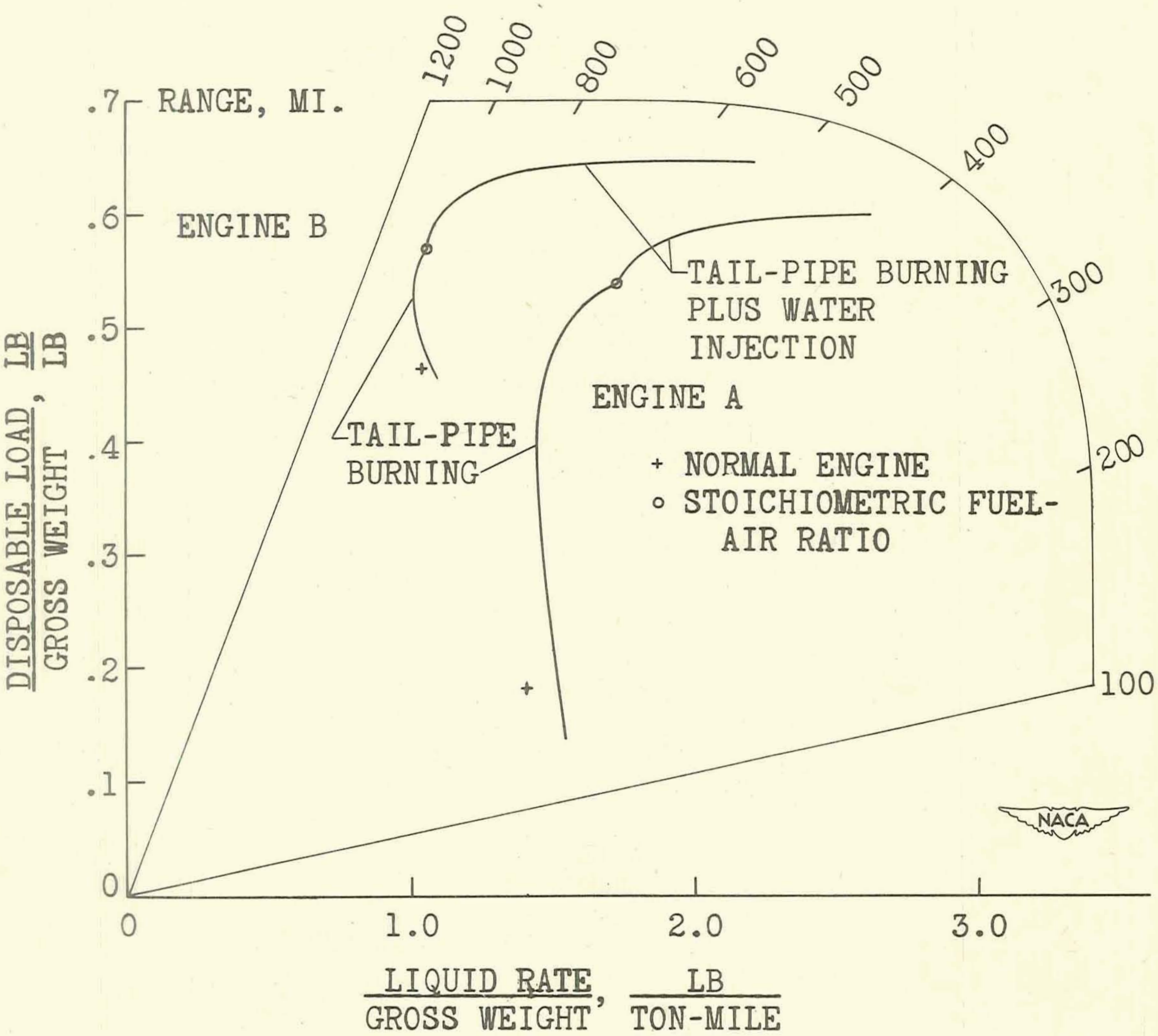

Figure 8. - Comparison of load-range characteristics of airplane with two different engines. Altitude, 35,000 feet; flight Mach number, 1.50.