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INFLUENCE OF HIGH-TURBINE-INLET-TEMPERATURE ENGINES IN A METHANE-FUELED SST WHEN TAKEOFF JET NOISE LIMITS ARE CONSIDERED

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ABSTRACT

Afterburning turbojet, nonafterburning turbojet, and duct-burning turbofan engine designs were optimized with respect to payload and direct operating cost for a Mach 3.0 cruise SST both with and without takeoff noise constraints. Only jet noise was considered, and none of the engines had noise suppressors. Design turbine-inlet temperature was varied from 2200° to 3100° F (1204° to 1704° C). Without noise restrictions, improvements occurred as the turbine-inlet temperature was raised. The duct-burning turbofan was best at low turbine temperatures, but the nonafterburning turbojet was superior at the upper temperatures. When jet exhaust noise restrictions are imposed, the duct-burning turbofan cycle was superior to the other cycles, with high-turbine-inlet temperature offering little benefit. High-turbine-inlet temperatures offer no benefits to turbojet cycles with noise restrictions.

CONTENTS

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SUMMARY

The benefits that can be obtained from designing high-turbine-inlet-temperature afterburning-turbojet, nonafterburning-turbojet, and duct-burning-turbofan-cycles were analyzed by using these engines on a fixed-ramp gross-weight methane-fueled Mach 3 SST. Airport and community noise restrictions during aircraft takeoff and climb may limit the use of high-design turbine-inlet temperature that are possible for methane-fueled supersonic aircraft by using the heat-sink capacity of methane to cool the turbine blades. The benefits are evaluated in terms of the number of passengers that can be carried and direct operating cost of an SST when analytically flying a 3500-nautical-mile (6482-km) range mission. The engines studied have design turbine-inlet temperatures ranging from 2200° to 3100° F (1204° to 1704° C). Payload improvements and direct operating cost reductions resulted with increased turbine-inlet temperature for all engine cycles when noise restrictions were not considered. For low-design turbine-inlet temperatures, the duct-burning turbofan was able to carry the most passengers, but the less expensive afterburning turbojet yielded the lowest direct operating cost. At higher temperatures, the nonafterburning turbojet was superior in both payload and direct operating cost.

When airport and community noise restrictions were considered, the duct-burning-turbofan was superior to the other cycles, but it provided only a marginal payload and direct operating cost improvement with increasing turbine-inlet temperature. Raising the temperature in the afterburning or nonafterburning turbojets provided no benefits. The noise calculations considered only jet exhaust noise of engines with no suppression devices.

The optimum values of the engine design parameters (compressor pressure ratio, fan-pressure ratio, and bypass ratio) as well as engine size and appropriate engine weights are affected by both turbine-inlet temperature and noise restrictions.

INTRODUCTION

Previous studies (refs. 1 and 2) have indicated that liquid methane has potential for use as a fuel for future commercial supersonic transports using afterburning turbojet engines. Among the advantages of methane are its availability, economy, high heat of combustion, and high heat-sink capacity relative to kerosene fuels. The greater heat-sink available with liquid methane indicates that turbine-inlet temperatures higher than those currently being considered may be permissible if the additional heat-sink capacity of methane is used to cool the turbine blades. High-turbine-inlet temperatures mean potentially higher cycle thermal efficiencies and engine thrust-to-weight ratios.

References 1 and 2 pointed out the benefits of high temperatures and mentioned, but did not evaluate, the effect on takeoff jet noise generation. High-turbine-inlet temperatures are undesirable from the engine noise standpoint because they usually produce a higher jet exhaust velocity. Jet velocity is the primary factor in jet noise. Reference 3 illustrated that jet noise limits during takeoff and climb can significantly compromise airplane performance.

During takeoff and climb when a high thrust level is required, exhaust jet noise is usually high enough to mask compressor and fan noise. However, at very low thrust settings, exhaust noise is much lower, and fan and compressor noise may be the predominate noise source. This situation can exist during takeoff after power cut back at the 3-mile (4.8-km) point as well as during approach. It is possible that this problem area can be handled through proper fan and compressor design and the use of inlet and duct sound suppression treatment. This particular problem is not considered in this report.

The purpose of this study is to determine what benefit will be obtained from the high-turbine-inlet temperature permitted by methane fuel. This is done with and without consideration of airport and community jet noise restrictions during airplane takeoff and climb. The method used is to determine the improvement that might be obtained in two overall airplane figures of merit, namely, (1) payload (or number of passengers) and (2) direct operating cost in cents per seat statute mile (cents/km). The airframe is arbitrarily selected as a fixed-sweep, arrow-wing SCAT15F configuration.

The afterburning turbojet, nonafterburning turbojet, and the duct-burning turbofan are the three engine cycles investigated. The turbine-inlet temperature is varied from 2200° to 3100° F (1204° to 1704° C). The compressor pressure ratio, bypass ratio, and fan-pressure ratio are optimized for each turbine inlet temperature, both with and without airport and community noise restrictions.

Noise restrictions are imposed on the engines because the problem of airport and community noise during airplane takeoff and climb is of major concern to the airports and the public. The airport and community noise restrictions used in this study are those defined in reference 4. Approach noise levels are also defined, but these restrictions are not considered herein. The so-called airport noise is measured at the start of takeoff roll, 1500 feet (457 m) from the centerline of the aircraft and at the angle of maximum noise. The noise level at this point should not exceed 116 perceived noise level in decibels (PNdB). For the community noise, during airplane climb a point on the ground directly beneath the flight path and at a distance of 3 statute miles (4.8 km) from the point of brake release is considered. After the engine power is reduced for a 500-foot-per-minute (152-m/min) rate of climb, the maximum noise at this point should not exceed 105 PNdB.

Noise suppression devices of the exhaust jet are not used in the data presented in order to better emphasize the influence of the primary engine parameters. It is entirely possible that noise suppression devices will change the results of this study. The data presented can be considered as the two extreme cases. The best possible case assumed no airport or community noise restrictions, and the worst possible case assumed noise restrictions without suppression devices. Thus, data obtained by using various degrees of suppression, which is beyond the scope of thus study, would most likely occur somewhere between the two extremes.

METHOD OF ANALYSIS

The effect of increasing turbine-inlet temperature with or without noise restrictions

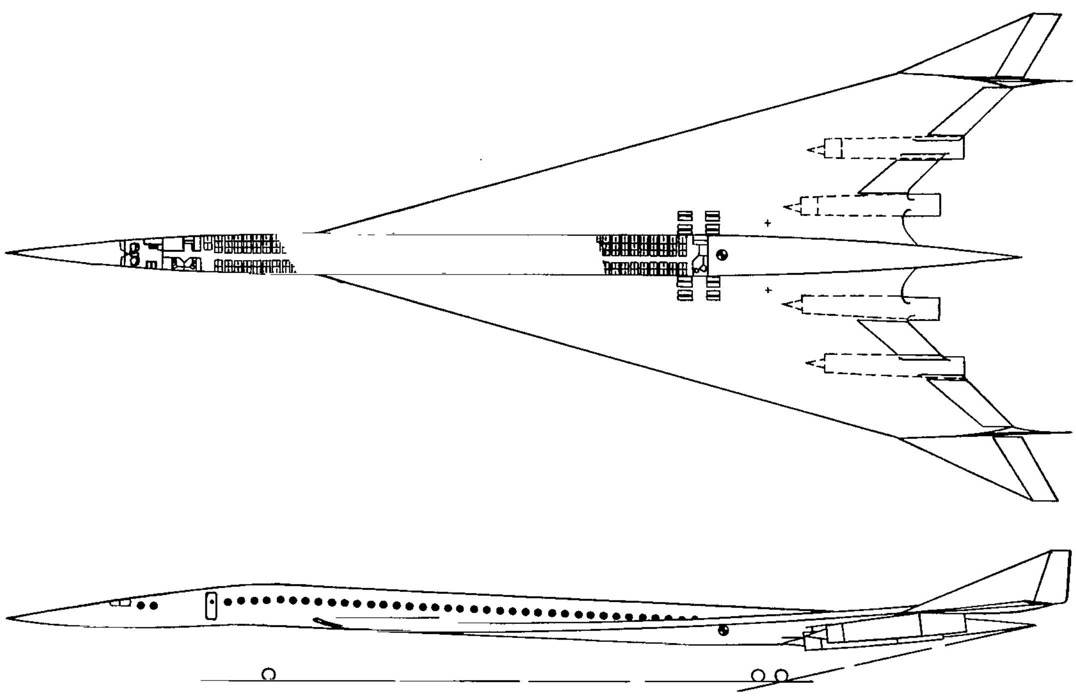


Figure 1. - Supersonic transport.

was determined by analysis of the results obtained from analytically flying a fixed-wing airplane over a standard mission profile. The engines used methane as a fuel. The airplane was similar to the one shown in figure 1. The airplane ramp gross weight was fixed at 460 000 pounds (208 652 kg). The engine size was varied in order to maximize the payload or to meet airport and community noise restrictions within certain takeoff performance constraints. The fuselage was fixed in maximum cross-sectional area, and the length was varied in order to accommodate different numbers of passengers. A comparison was made among the afterburning turbojet, nonafterburning turbojet, and duct burning turbofan engines.

Mission

The mission requirements (same as ref. 1) observed in this study were:

Range, n mi; km	3500; 6482
Cruise Mach number	3.0
Maximum sonic-boom overpressure limit, lb/ft^2; N/m^2	
Climb	2.0; 95.76
Cruise	1.5; 71.82
Minimum climb-acceleration thrust-to-drag ratio	1.4
Minimum second segment climb angle, deg	1.7
Maximum lift-off distance, ft; m	4450; 1460

A typical flight plan in Mach number and altitude coordinates is presented in figure 2. The flight path in all cases was fixed up to Mach 1.0. At higher Mach numbers, the maximum sonic-boom overpressure limit for climb and acceleration dictated the flight path. Even though the airplane had a constant 460 000 pound (208 652 kg) ramp gross weight, the weight at the sonic threshold, fuselage size, and altitude affect the sonic-boom overpressure; therefore, the flight path varied from one case to another so the sonic-boom overpressure on the ground did not exceed the limit.

No firm minimum climb-acceleration thrust-to-drag ratio requirement exists today, but many authorities believe it should be at least 1.4 on a standard day. On completion of the climb and acceleration phase, cruise began at the altitude that maximized the Breguet cruise factor. During cruise, altitude increased so that the Breguet factor was maintained constant at its maximum value. This resulted in minimum fuel consumption during the cruise phase of the flight. To simplify calculations, it was assumed that descent time and range remained constant for all cases at 25 minutes and 400 nautical miles (741 m), respectively, with fuel consumption calculated at engine idle conditions.

The fuel reserve for the mission allows for (1) an additional 7 percent of the total

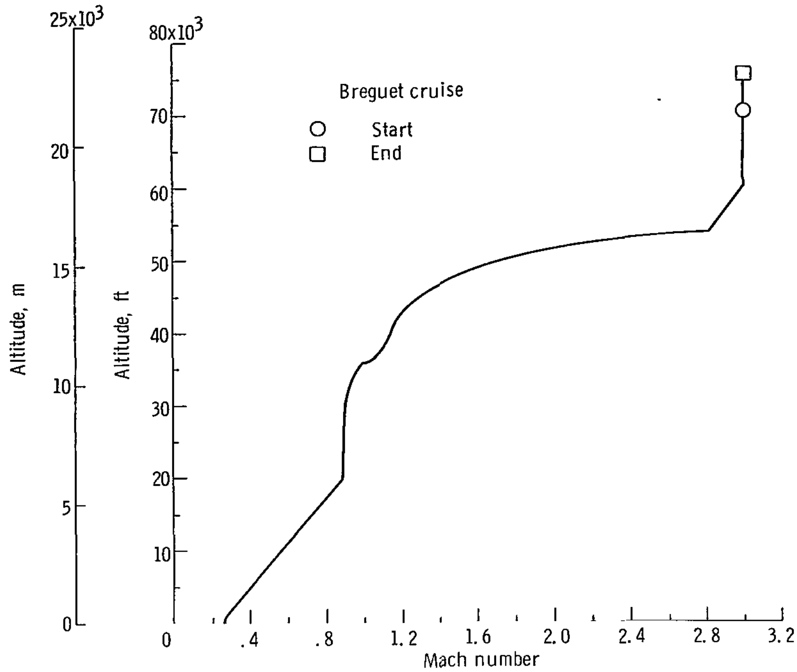


Figure 2. - Typical flight profile. Aircraft ramp gross weight, 460 000 pounds (208 652 kg); sonic-boom overpressure, 2.0 pound per square foot (95.76 N/m^2); climb, 1.5 pounds per square foot (71.82 N/m^2) cruise; takeoff wing loading, 50 pounds per square foot (2395 N/m^2).

mission fuel, (2) an extension of 261 nautical miles (483 km) to an alternate airport at cruise altitude and Mach number, and (3) a 30-minute hold at 15 000 feet (4572 m) altitude at Mach 0.6. An additional fuel allowance was incorporated in the mission fuel for a 25-minute idle prior to takeoff as well as a 1-minute period of maximum augmentation power application prior to takeoff roll.

Airframe Characteristics

The aerodynamic parameters used for this study were based on wind-tunnel data supplied by the NASA Langley Research Center for the SCAT15F. The SCAT15F is an advanced fixed-sweep, arrow-wing SST configuration similar to the one depicted in figure 1.

The weights of the major components that comprise the empty weight of the airplane were estimated by empirically established relations. These estimates are based on preliminary designs for similar configurations by industry and are consistent with references 1 to 3. Errors in weight estimates should not influence the comparison of one engine relative to another, even though on an absolute basis some error in overall airplane figure of merit would be indicated.

Details of the low-speed aerodynamics and weight assumptions are discussed in appendix A.

Engines

Three types of engines were considered. The performance and weight of each engine was calculated for a specific range of design variables. The range of variables covered in analytically finding the optimum cycle combination was as follows:

Turbine inlet temperature, ^{o}F; ^{o}C	2200 to 3100; 1204 to 1704
Compressor-pressure ratio	7 to 19
Fan-pressure ratio	1.5 to 3.5
Turbofan bypass ratio	1.0 to 3.1

The maximum augmentation gas temperature (duct burner or afterburner) considered was 3100^{o} F (1204^{o} C). For the duct-burning turbofan, the bypass ratio is defined as the bypass duct airflow divided by the gas generator airflow at the design point. As used herein, engine design point refers to sea-level-static operation of the engine at maximum unaugmented thrust setting.

Performance. - In calculating the design and off-design performance, each engine component was matched to satisfy the relations involving the continuity of flow, engine rotational speed, and power balance between the compressor (or fan) and its driving turbine. The procedures used are similar to those discussed in reference 5.

The duct-burning turbofan had a fixed area primary exhaust nozzle and a variable area nozzle for the duct stream. The afterburning and nonafterburning turbojets had a variable area primary exhaust nozzle.

Engine component efficiency, turbine cooling airflow requirements, and inlet and exhaust nozzle performance details that were used to calculate engine performance are discussed in appendix A.

Although the compressor adiabatic efficiency varied as the engine operating conditions and power settings were changed, design values of compressor and turbine adiabatic efficiency remained fixed. There are many factors that will affect these values. As shown in appendix B, an additional study was made where the design compressor and turbine adiabatic efficiency of the afterburning turbojet were independently varied. The effect of this variation on SST payload carrying capability when powered by these afterburning turbojets was determined.

Also shown in appendix B is the effect on the payload carrying capability of the SST when the turbine cooling airflow schedule is changed.

Engine weight. - Engine weight was calculated from empirical equations that relate installed engine weight to the type of engine and the design engine airflow, compressor-pressure ratio, fan-pressure ratio, bypass ratio, and turbine-inlet temperature. The equations are based on a composite of industry data.

For a turbine-inlet temperature of 2200° F (1204° C) and a combination of optimum values of the remaining design parameters for no noise restrictions, engine thrust-to-weight ratios of 5.75 for the duct-burning turbofan and 6.5 for the afterburning turbojet were calculated. In the term engine thrust-to-weight ratio, engine thrust was maximum augmented thrust at sea-level-static conditions; engine weight included the thrust reverser and exhaust nozzles.

For the afterburning and nonafterburning turbojet engines designed to cruise at Mach 3, the installed engine weight for four engines was calculated by the empirical equation presented in appendix A (eqs. (A1)).

There are more variables to consider when the weight of the duct-burning turbofan is calculated. The duct-burning turbofan engine weight is affected by the design airflow, overall compressor-pressure ratio, bypass ratio, fan-pressure ratio, and turbine-inlet temperature. The empirical equations for installed weight of four engines are also presented in appendix A (eqs. (A2)).

Also shown and discussed in appendix A are curves relating engine weight factors to various values of engine design parameters.

The engine weight changed according to the equations in all cases except when perturbed in the study discussed in appendix B. The weight of four 3100° F (1704° C) afterburning turbojet engines was arbitrarily varied to determine the effect on SST payload carrying capability.

Engine operation. - The method of engine operation was dependent on whether noise restrictions were or were not observed.

Without noise restrictions: takeoff power settings were at maximum augmentation (afterburner or duct burner) for the afterburning turbojet and duct-burning turbofan and at maximum dry thrust for the nonafterburning turbojet. After the aircraft was airborne and flight Mach number 0.40 achieved, power on the augmented engines was reduced to the maximum unaugmented condition for improved specific impulse. As the high drag transonic region was approached, maximum augmentation was again applied until the cruise Mach number was approached. Thrust was then reduced such that the aircraft entered cruise at the altitude corresponding to maximum Breguet factor.

With noise restrictions: Part-power engine operation at takeoff and initial climb out was usually required. The afterburning and nonafterburning turbojet engines used the Mode B part-power operation as described in reference 3. Mode B part-power operation involves operation with constant corrected airflow at the compressor face when the turbine-inlet temperature is reduced below its maximum value. This mode of operation was used during takeoff and for power reduction at the point 3 miles (4.8 km) from the start of takeoff roll where community noise was considered. Mode B part-power operation was selected because the engine then produces the lowest jet noise for a given thrust setting. This type of part-power operation was made possible by use of a variable primary exhaust nozzle. In order not to exceed the noise limit, it was necessary to operate the turbojet at less than maximum unaugmented power settings. When the aircraft attained a speed of Mach 0.4, thrust setting was increased to maximum unaugmented. Beyond this point, thrust setting was scheduled as described previously.

The duct-burning turbofan engines considered herein have fixed primary exhaust nozzles. During takeoff and at the 3-mile (4.8-km) point, it was necessary to operate the duct-burning turbofan at less than maximum design turbine-inlet and duct-augmentation temperatures. Both of these temperatures were regulated to minimize noise for a given thrust. At Mach 0.4 and higher speeds, thrust setting was scheduled as described earlier.

Noise Calculations

The procedures followed in this study for calculating the appropriate levels of jet noise are those outlined by the Society of Automotive Engineers (SAE) in references 6 and 7. The method includes effects of atmospheric absorption, ground attentuation, and multiple engines. The calculations are for noise produced by the jet exhaust only and do not include the noise generated by the fan or compressor.

A two-point noise criterion for takeoff and climb was used. It was specified by the Federal Aviation Agency (FAA) in its SST Economic Model Ground rules (ref. 4) for the SST evaluation. The two points selected are at the airport and the surrounding community. An acceptable noise environment is not well defined; therefore, specifying noise levels at given points may not be entirely suitable. A more elaborate specification for an acceptable noise environment should account for the number of noise events, the duration of each event, the maximum sound level, nonstandard atmospheric conditions, and areas exposed. An effective PNdB is being proposed to account for these items. However, these effects are not included in the present noise calculations.

Takeoff noise at the airport varies with engine design parameters (airflow, overall compressor-pressure ratio, bypass ratio, fan-pressure ratio, turbine-inlet temperature, and augmentation temperature), thrust setting, and model of engine operation.

Community noise at 3 miles (4.8 km) from the start of takeoff roll is dependent on other factors in addition to the engine parameters because the aircraft is in flight. Here, the community noise is a function of the low-speed airplane aerodynamics and selected climb procedures as well as the engine type, size, and operation. Several types of flight paths are possible. One, for example, results from a climb at maximum rate, with a minimum increase in speed, and gives a maximum altitude at the 3-mile (4.8-km) point.

There the engine power is reduced so the airplane will maintain a 500 foot-per-minute (152 m/min) rate of climb.

In another climb procedure, the aircraft reaches its takeoff velocity and continues accelerating to a higher velocity with some sacrifice in altitude at the 3-mile (4.8-km) point (ref. 3). The low-speed aerodynamics of this type of aircraft are such that, although the 3-mile (4.8-km) point altitude is decreased in achieving higher speeds, lower thrust is required for the constant 500 foot-per-minute (152 m/min) rate of climb, compensating for the lower altitude. In this study the climb procedure was optimized to minimize community noise.

Engine Sizing

Engine size refers to the design engine airflow when it is corrected to standard sea-level-static conditions at the compressor face. The engine size selected enables the aircraft to carry the largest number of passengers when it was operating within the constraints of various limits.

Without noise restrictions, the engine size selected produced sufficient thrust to meet the operational limits of maximum lift-off distance, minimum second-segment climb angle, minimum climb-acceleration thrust-to-drag ratio, and maximum sonic-boom overpressure. Engine airflows greater than that required to meet these limits were considered to find the engine size that maximized the number of passengers.

With noise restrictions, the additional constrain of airport and community noise limits were considered.

The details of the various operational limits and the steps required to optimize the engine size with noise restrictions are discussed in appendix A.

Direct Operating Cost Estimation

Direct operating cost is probably a better airplane figure of merit than payload. The direct operating cost calculations were performed in the manner described in reference 8; however, uncertainties of airframe and engine pricing are involved in its calculation.

In the direct operating cost calculations, the airframe cost was assumed to be a function of airframe weight. Airframe prices were estimated with development costs included. These prices are based on a production of 200 aircraft. An equation that approximates the airframe prices is shown in appendix A.

The engine price was assumed to be a function of engine size and type. A production schedule of 1200 engines was assumed in the estimation of engine price with the view that each of the 200 four-engine aircraft would eventually require two spare engines. The assumed price also includes development cost and time between engine overhaul (TBO) of 2000 hours. Equations that approximate the engine price are shown in appendix A.

The direct operating cost calculations also assumed that a typical price for liquid-methane fuel delivered to the airplane would be 1.2 cents per pound (2.65 cents/kg).

RESULTS AND DISCUSSION

Effects of Design Variables

The afterburning turbojet, the nonafterburning turbojet, and the duct-burning turbofan cycles were considered separately to determine the engine design parameters that would enable the aircraft, within certain performance limitations, to carry the greatest number of passengers. The effect that design turbine-inlet temperature had on engine design parameters, payload, and direct operating cost was considered with and without takeoff and community noise limits.

Afterburning turbojet. - In addition to design turbine-inlet temperature, other design variables were compressor-pressure ratio, engine airflow, and engine weight.

Figure 3(a) shows the passenger-carrying capability of an SST as a function of design turbine-inlet temperature when the SST is powered by four afterburning turbojet engines. Also shown are values of design compressor pressure ratio, engine airflow, and the respective engine weight (figs. 3(b) to (d), respectively). The figures represent a combination of engine size and design parameters that will permit the maximum number of passengers to be carried when the aircraft ramp gross weight is fixed at 460 000 pounds (208 652 kg). Although the figure shows the optimization of all combinations of variables both with and without noise restrictions, the results obtained without noise restrictions will be presented and discussed first. Next, the results are compared with and without noise at a design turbine-inlet temperature of 2200° F (1204° C). The, the results with noise restrictions are discussed.

Figure 3(a) shows that the number of passengers increase by 12.1 percent when the turbine-inlet temperature was raised from a reference value of 2200° to 3100° F (1204° to 1704° C). For this temperature change, the afterburning turbojet had only a small increase of airport noise level (from 122.3 to 123.8 PNdB) because the afterburner was operating at a constant 3100° F (1704° C) gas temperature. As the design turbine-inlet temperature was increased, the optimum design compressor pressure ratio increased from 10 to 19, but the engine airflow decreased 11.1 percent (figs. 3(b) and (c)). These trends in design compressor pressure ratio and airflow help explain the trend in engine weight shown in figure 3(d). Engine weight per pound (kg) per second of design airflow in-

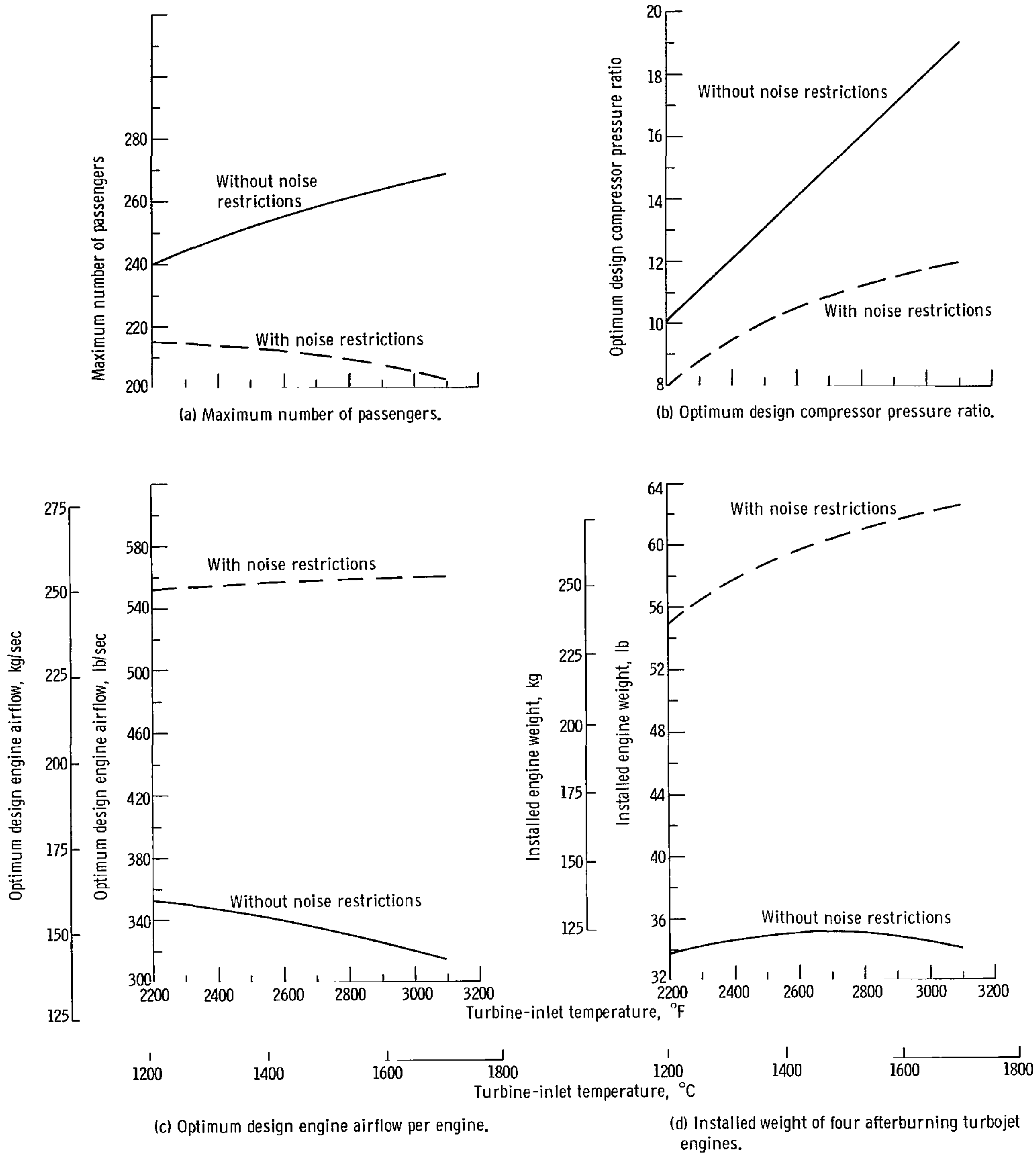


Figure 3. - Effect of design turbine-inlet temperature for afterburning turbojets; ramp gross weight, 460 000 pounds (208 652 kg); Mach 3.0 cruise.

creased continuously with rising design turbine-inlet temperature and compressor pressure ratio. The combined effects of temperature, pressure ratio, and size at turbine-inlet temperatures below 2700° F (1403° C) resulted in an increase in engine weight. Above 2700° F (1483° C), the engine size decrease offset the temperature and pressure ratio effect and the engine weight decreased.

Because the variation in engine weight was slight (fig. 3(d)), the 12.1 percent increase in the number of passengers (fig. 3(a)) must stem mainly from a reduction in fuel weight. The weight of fuel used for the SST mission was affected by engine specific impulse (lb of thrust/(lb of fuel/sec) or N/(kg/sec)) and takeoff engine thrust to airplane ramp gross weight ratio. As shown in appendix C, the total fuel to fly the mission decreased by 10 800 pounds (4 900 kg) as the turbine-inlet temperature was increased from 2200° to 3100° F (1204° to 1704° C). It was the combination of a slight decrease in engine weight and a decrease in fuel weight that accounted for the payload increase.

Also shown and compared in appendix C are figures of engine specific impulse for climb-acceleration and cruise conditions for the afterburning turbojets designed for 2200° and 3100° F (1204° and 1704° C).

The results obtained when takeoff noise limits were imposed are also shown in figure 3. In meeting the noise restriction limits, a 10.4-percent payload penalty resulted at a design turbine-inlet temperature of 2200° F (1204° C) (fig. 3(a)). A significant increase in engine size and weight was the cause of the payload decrease. It was necessary to increase the design engine airflow by 57 percent (fig. 3(c)). The engine weight increased by 62.2 percent (fig. 3(d)).

In figure 4, which uses airport sideline noise and community noise as coordinates,

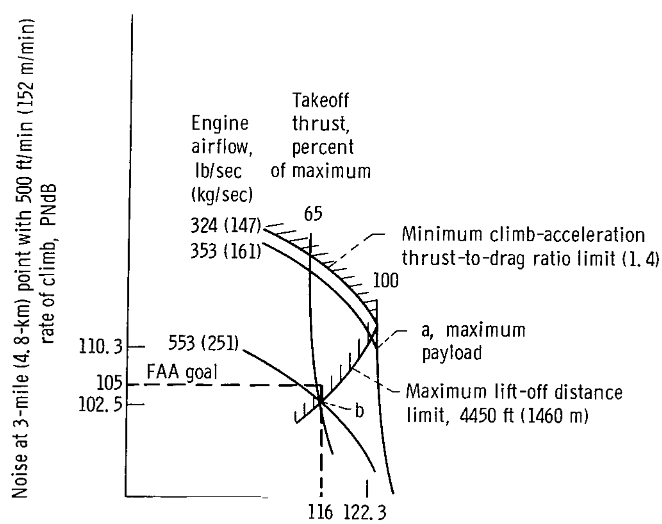


Figure 4. - Effect of engine noise goals on aircraft and engine parameters with four afterburning turbojet engines. Design turbine-inlet temperature, 2200° F (1204° C).

the 353-pound-per-second (161-kg/sec) airflow afterburning turbojet engines allowed the aircraft to carry maximum payload (point (a)). This size engine produced sufficient thrust to meet the lift-off distance and climb-acceleration constraints for the mission. However, these engines produced 122.3 PNdB airport sideline noise and 110.3 PNdB community noise levels, which were considerably over the respective maximum limits. The jet noise, which is much more sensitive to jet velocity than to gas mass flow (airflow plus fuel flow), was reduced by engine operation at less than maximum thrust. This was accomplished by reducing the afterburner gas temperature until no afterburning was used. The jet noise was further reduced by decreasing the turbine inlet temperature.

Lift-off distance is affected by engine takeoff thrust. The loss in thrust that resulted from the lower jet velocity had to be compensated for by an increase in engine size. An engine size of 553 pounds per second (251 kg/sec) was required to meet the noise and lift-off distance constraints (point (b), fig. 4). The 116 PndB airport sideline noise limit was accomplished with the afterburning turbojet engines operating with no afterburning and the turbine-inlet temperature lowered to 1755° F (957° C) from a design value of 2200° F (1204° C). Operating in this manner, the engines were producing 65 percent of the maximum afterburning thrust.

The payload decrease that came about from increasing engine size was the result of a tradeoff between engine weight and fuel consumption. Larger engines as required by noise and lift-off distance constraints, weighed more and used less fuel during climb and acceleration than did the smaller engines sized for maximum payload. The large engines, which are capable of more thrust, reached the cruise speed and altitude using less time and range than the smaller engines. Because the specific impulse for both engines was nearly the same, the large engines used considerably less fuel during the climb-acceleration part of the mission.

Sufficient cruise thrust was possible with the large engine at a lower power setting. Thus, as shown in appendix C (fig. 25(c)) for the 2200° F (1204° C) design turbine-inlet temperature engine, less afterburning during cruise would result in a specific impulse increase. Nevertheless, the cruise specific impulse advantage of the larger engine did not result in a decrease in fuel consumption; this is due to the longer cruise range the vehicle must fly using the larger engines because less range was used getting up to cruise and the total mission range was fixed at 3500 nautical miles (6482 km). However, an overall fuel savings resulted for the aircraft using the large engines. Despite this, the tradeoff of an engine weight increase for a fuel weight decrease caused a payload reduction of 10.4 percent (fig. 3(a)) when noise restrictions were imposed on the afterburning turbojet engines at a design turbine-inlet temperature of 2200° F (1204° C).

The community noise at the 3-mile (4.8-km) point after power cutback was 102.5 PNdB (point b, fig. 4). Because the noise level is less than the 105 PNdB limit, it would be possible, if desired, to increase the engine thrust until a noise level of

105 PNdB is reached. Instead of a 500 foot-per-minute (152 m/sec) rate of climb, it would be possible to have a 1250 foot-per-minute (381 m/sec) rate of climb at the 3-mile (4.8-km) point and not exceed the noise limit. Or, if the lift-off distance constraint did not determine the engine size, it would be possible to use a smaller engine, thus increasing the payload capability, and still meet the noise goals.

With noise restrictions, as the turbine-inlet temperature was increase to 3100° F (1704° C), the payload decreased (fig. 3(a)) by 5.6 percent.

Figure 3(b) shows that a lower compressor pressure ratio was best when noise restrictions were considered even though the change in exhaust noise brought about by changing the design overall compressor pressure ratio was small. Design compressor pressure ratio had a significant effect on engine weight. Therefore, a decrease in overall pressure ratio would decrease the engine weight, which was a primary factor in determining payload.

Engine size increased slightly with turbine-inlet temperature (fig. 3(c)) when noise was considered. In order to meet the minimum takeoff thrust requirements, it was necessary to increase the engine airflow 1.5 percent as the turbine-inlet temperature was increased from 2200° to 3100° F (1204° to 1704° C). This engine size increase came about mostly from a decrease in part-power component efficiency and compressor pressure ratio during the takeoff conditions for which the engines were sized. This effect is shown on the compressor map of figure 5. A major effect was that, regardless of the design turbine-inlet temperature, a nearly constant part-power turbine-inlet temperature was required to meet the 116 PNdB noise goal. At the design turbine-inlet temperature,

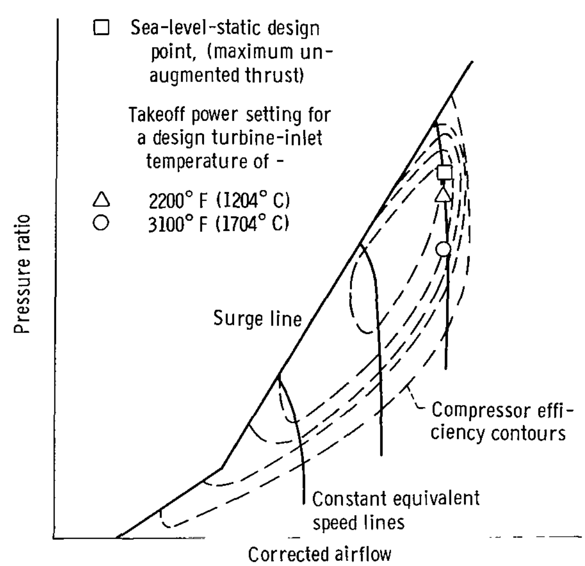


Figure 5. - Afterburning turbojet compressor map showing part-power takeoff operating points.

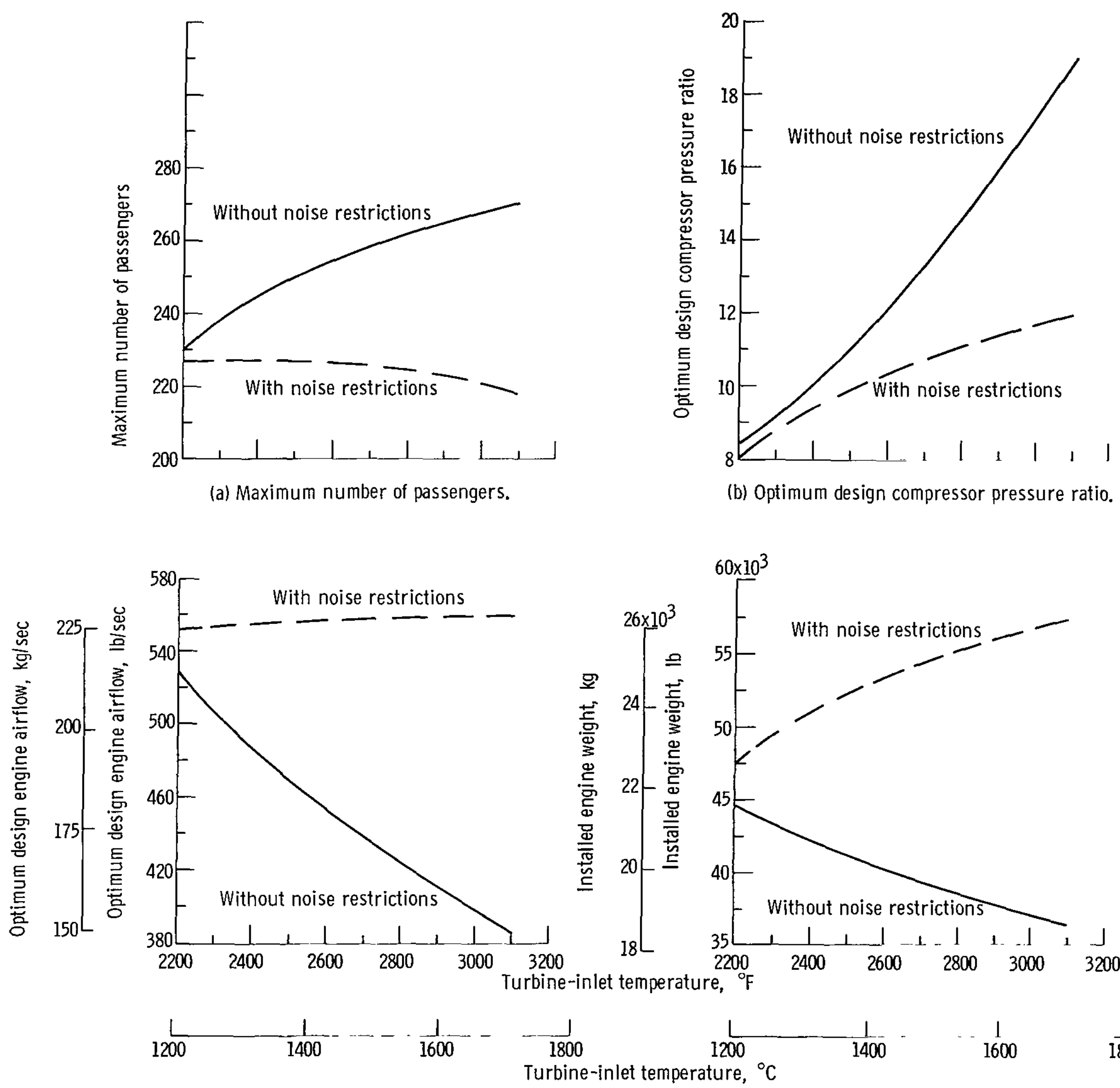
the compressor was operating at a high efficiency and design compressor pressure ratio. As the turbine-inlet temperature was reduced below design for part-power operation, compressor efficiency and pressure ratio decrease. A design turbine-inlet temperature increase meant a larger reduction was necessary to meet the noise goal. Thus, as figure 5 indicates, less desirable part-power operation on the compressor map occurred as the design turbine-inlet temperature was increased. The possibility of selecting an alternate design point that might mitigate this penalty has not been evaluated.

The combination of the effects of increasing design turbine-inlet temperature, compressor pressure ratio, and engine size resulted in an installed-engine weight increase (fig. 3(d)) of 13.5 percent.

As the turbine-inlet temperature of the afterburning turbojet was increased with noise restrictions, the payload decreased because the engine weight increased faster than the fuel weight decreased. Engine performance improvement with increasing turbine-inlet temperature was offset by the engine weight as a result of the large engine size required to meet the noise limit and lift-off distance constraints. In fact, the oversized engines in some cases cruised while using no afterburning and with turbine-inlet temperature reduced below design values. Therefore, little use was made of the high-design turbine-inlet temperature capability.

Nonafterburning turbojet. - The principal design variables of a nonafterburning turbojet are the same as for the afterburning turbojet. The nonafterburning turbojet differed from the afterburning turbojet only in that it could not augment the gas flow from the turbine in an afterburner to produce more thrust. Figure 6(a) shows that, when airport and community noise was not considered, the payload carrying capability of the SST using dry turbojets increased by 17.4 percent as the design turbine-inlet temperature was increased from 2200° to 3100° F (1204° to 1704° C). As shown on figure 6(b), the design compressor-pressure ratio increased from 8.4 to 19.0 as the design turbine-inlet temperature was raised. The compressor-pressure ratio influences the payload primarily through its effect on engine weight, specific impulse, and thrust per unit airflow. As the turbine-inlet temperature was increased, the engine size decreased. Figure 6(c) shows the engine size decreasing by 27 percent. It should be noted, however, that all the nonafterburning turbojet engines studied without noise restrictions were sized for a minimum climb-acceleration thrust-to-drag ratio. Because the nonafterburning turbojet had no means of increasing its thrust except by increasing the engine size for a specific design turbine-inlet temperature, the engine size was increased until a minimum 1.4 climb-acceleration thrust margin was achieved. A 528-pound-per-second (239-kg/sec) airflow engine met this requirement at 2200° F (1204° C) turbine-inlet temperature. As the design turbine-inlet temperature was raised above 2200° F (1204° C), the thrust of the nonafterburning turbojet increased. Consequently, the use of smaller engines was possible.

Without noise restrictions, the installed-engine weight decreased 13.2 percent as the



(a) Maximum number of passengers.

(b) Optimum design compressor pressure ratio.

(c) Optimum design engine airflow per engine.

(d) Installed weight of four nonafterburning turbojet engines.

Figure 6. - Effect of design turbine-inlet temperature for nonafterburning turbojets. Ramp gross weight, 460 000 pounds (208 652 kg); Mach 3.0 cruise.

turbine-inlet temperature was raised as shown on figure 6(d). The engine weight increase that normally would result as the design compressor ratio is raised, was offset by the large engine airflow reduction.

The remaining weight saving that enabled the nonafterburning turbojet powered SST to carry 17.4 percent more payload (fig. 6(a)) came about from a decrease in the fuel that was required to fly the mission. As the turbine-inlet temperature was raised, the resulting smaller engine size approached the size that enables the SST to carry maximum payload. Therefore, as the design turbine inlet temperature was increased, engine operation throughout the mission approached the point where fuel economy was the best and the total amount of fuel used decreased.

Imposing airport and community noise limits of 116 and 105 PNdB, respectively, resulted in lower payload (fig. 6(a)). At a design turbine-inlet temperature of 2200° F (1204° C), the reduction is 1.3 percent. The payload penalty was small at 2200° F (1204° C) design turbine-inlet temperature because the nonafterburning turbojet already had adequate takeoff performance with no augmentation. A 4.7-percent airflow increase (fig. 6(c)) was necessary to meet the noise restrictions and lift-off distance limits. Figure 7 (point (a)) shows that a nonafterburning turbojet, operating at maximum power, produces a 1500-foot (457-m) sideline noise level higher than the goal. The engine is sized to meet the minimum climb-acceleration thrust margin. The nonafterburning turbojet engines sized for this condition have excess thrust during takeoff. Therefore, some thrust loss (to point b, fig. 7) as a result of part-power operation to meet the noise goal is tolerable and the lift-off distance limit is still not exceeded. Compared with the after-

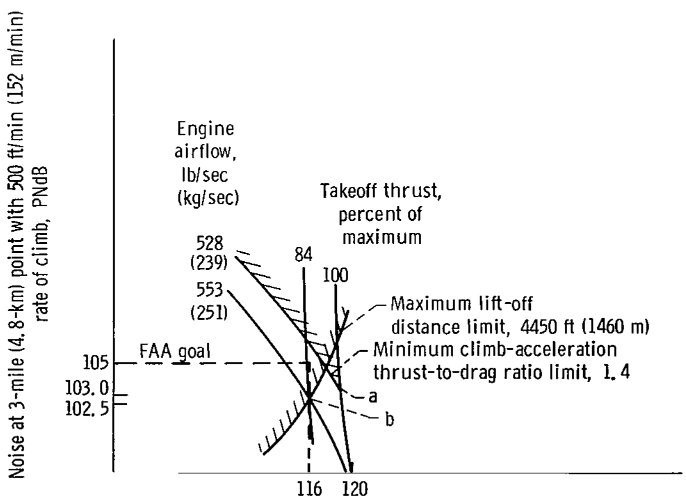


Figure 7. - Effect of engine noise goals on aircraft and engine parameters with four nonafterburning turbojet engines. Design turbine-inlet temperature, 2200° F (1204° C).

burning turbojet engines, a relatively small engine air-increase was necessary to meet both noise goals and lift-off distance limitations.

Engine airflow had a major effect on engine weight. Figure 6(d) shows a 5-percent engine weight increase at 2200° F (1204° C) design turbine-inlet temperature.

As the design turbine-inlet temperature was increased from 2200° to 3100° F (1204° to 1704° C), the payload decreased 4 percent as shown on figure 6(a). Figure 6(b) shows the optimum compressor-pressure ratio increased from 8 to 12. Figure 6(c) shows the engine airflow increasing 1.5 percent as the turbine-inlet temperature was raised from 2200° to 3100° F (1204° to 1704° C). The nonafterburning and afterburning turbojet compressor-pressure ratio and airflow requirements with noise restrictions were the same. The trend in engine weight for increased turbine-inlet temperature is upward as shown on figure 6(d).

The large airflow engines, which were sized for takeoff noise conditions, had excess thrust-to-drag margin during climb-acceleration. Consequently, the fuel required for this part of the mission was less. This trend was discussed in the previous section on afterburning turbojets. The payload decrease, which occurred as the turbine-inlet temperature was increased, which occurred as the turbine-inlet temperature was increased, was due to a combination of an engine weight increase that was partially offset by a fuel weight decrease.

Duct-burning turbofan. - The design variables of fan-pressure ratio and bypass ratio were considered in addition to compressor-pressure ratio, engine airflow, and engine weight for the duct-burning turbofan. Figure 8(a) shows the maximum number of passengers the aircraft could carry as a function of turbine-inlet temperature either with or without airport and community noise restrictions.

When airport and community noise restrictions were ignored, the number of passengers increased by 8.2 percent (fig. 8(a)) as the turbine-inlet temperature was increased from 2200° to 3100° F (1204° to 1704° C). During takeoff with maximum duct augmentation, the engine noise at the airport was 117.9 to 121.7 PNdB for the lowest and highest temperatures, respectively. To carry this number of passengers, the duct-burning turbofan engines optimum compressor-pressure ratio, bypass ratio, and fan-pressure ratio were as shown on figures 8(b) to (d).

As the design turbine-inlet temperature was increased, the engine size decreased by approximately 11.5 percent (fig. 8(e)). Engine weight, of course, changed as the design variables and size were changed. The weight of the installed duct-burning turbofan engines decreased by 8.4 percent (fig. 8(f)) as the turbine-inlet temperature was increased overs its range.

The 8.2-percent payload increase (fig. 8(a)) was due to decreases in both engine weight and fuel weight. As the turbine-inlet temperature was increased, the engine-thrust-per-pound of airflow increased resulting in an engine thrust-to-engine-weight in-

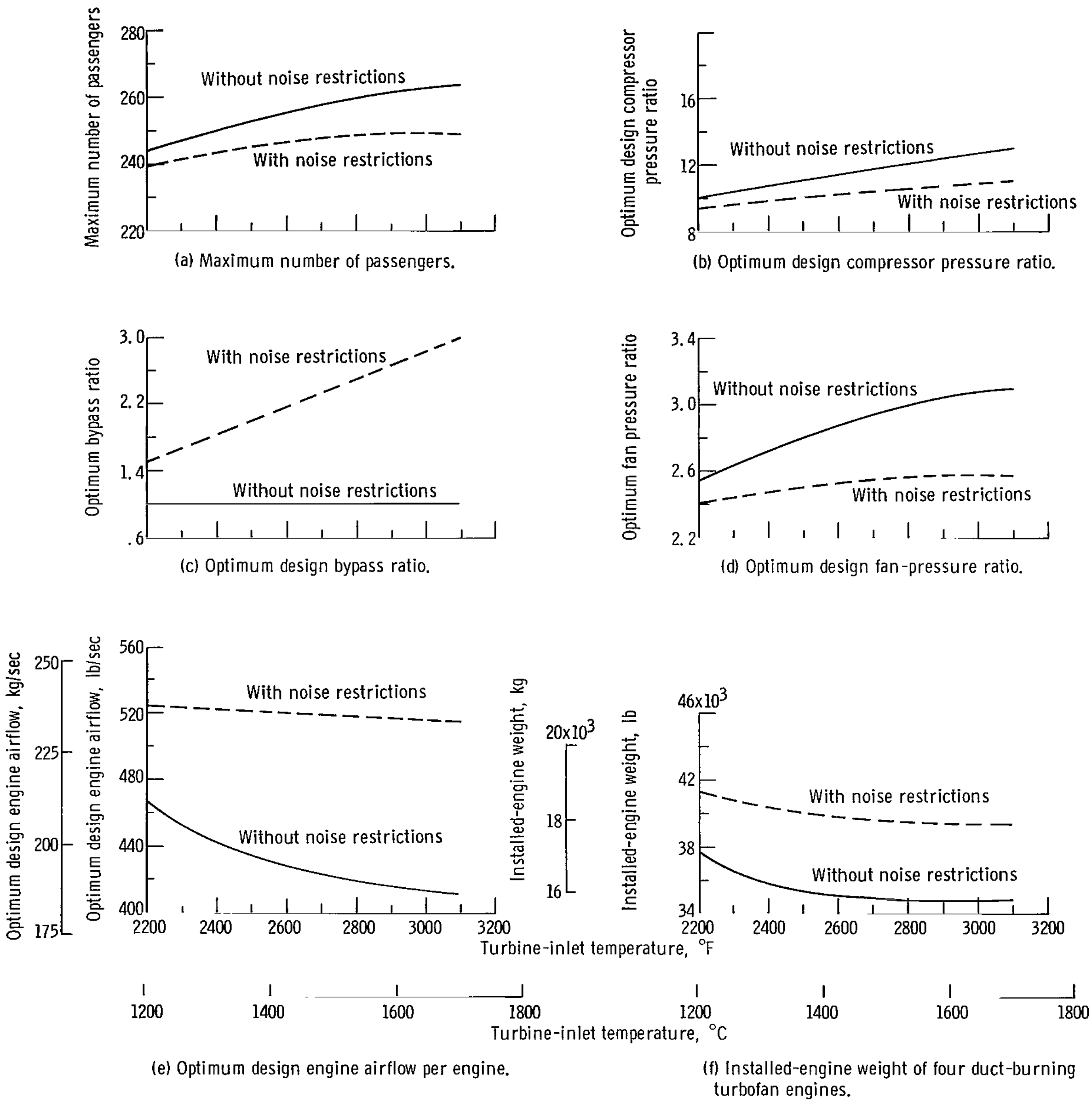


Figure 8. - Effect of design turbine-inlet temperature for duct-burning engines. Ramp gross weight, 460 000 pounds (208 652 kg); Mach 3.0 cruise.

crease from 5.75 to 6.4. Thus, it was possible to meet the mission constraints with a smaller engine size, which, in turn, reduced the engine weight.

The specific impulse improvement with increasing turbine-inlet temperature for the duct-burning turbofan was very little; therefore, the decrease of fuel to fly the mission because of specific impulse changes were small. The fuel savings was mostly the result of an increase in engine thrust to airplane ramp gross weight for maximum payloads as the turbine-inlet temperature was raised. This meant the time required to reach the cruise speed and altitude decreased and a fuel savings was the result.

When airport and community noise limits were imposed on the duct-burning turbofan powered SST, the number of passengers decreased by 2.1 percent at 2200° F (1204° C) turbine-inlet temperature (fig. 8(a)). The decrease in the payload carrying capability was again the result of a trade-off of a larger and heavier engine that is capable of producing more thrust at maximum power, which, in turn, somewhat decreased the fuel required to fly the mission.

At 2200° F (1204° C) design turbine-inlet temperature, very little decrease in the compressor-pressure ratio (fig. 8(b)) was noted. It decreased from 10.0 to 9.4 when the engines with and without noise considerations were compared. Figures 8(c) and (d) show that, although the design bypass ratio increased from 1.0 to 1.45, the design fan-pressure ratio decreased from 2.55 to 2.4.

Bypass ratio, fan-pressure ratio, and duct-burner augmentation gas temperature are the parameters that most strongly control the exhaust noise of the duct-burning turbofan engines. Unlike the turbojet, the duct-burning turbofan has the capability of separately controlling the energy level of each stream (duct and primary). This can be done by changing the combination of design bypass ratio, fan-pressure ratio, and compressor-pressure ratio. Changing these engine design parameters, in conjunction with scheduling fuel flow to either the primary on duct-burner combustors, allows a controllability of jet velocity that enables the duct-burning turbofan engine to produce the greatest thrust for a given noise level. The fan-pressure ratio affects duct-stream noise by changing the duct-stream velocity. The bypass ratio has a small effect on duct noise because the noise level is generally only slightly affected by mass flow changes. However, increasing either or both parameters affects the primary-stream noise by increasing the energy extracted from the engine to drive the fan. This reduces the noise generated by the primary stream. Therefore, an increase of the design turbine-inlet temperature, which would generally cause a primary-stream noise level increase, is offset by a combined increase of design bypass and fan-pressure ratio (figs. 8(c) and (d)). The lowest noise level for the duct-burning turbofan occurs when the engine design parameters and duct- and primary-stream gas temperatures produce an optimum combination such that the two streams are producing approximately the same noise. Some reduction of the gas temperature (part-power operation) was generally required to meet the noise levels.

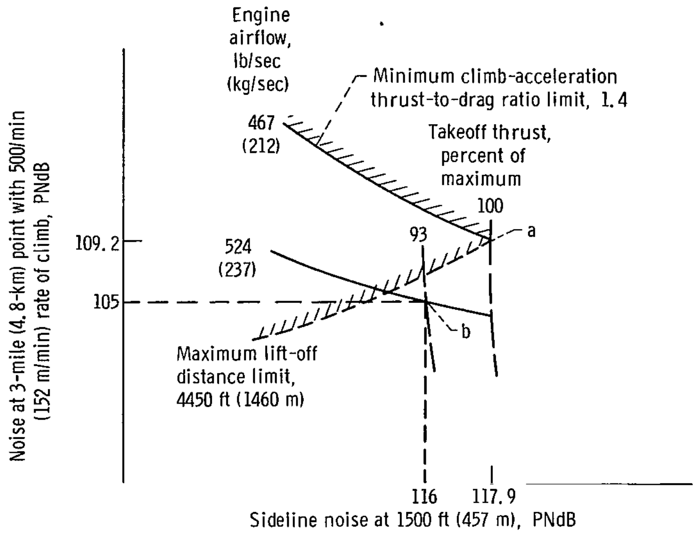


Figure 9. - Effect of engine noise goals on aircraft and engine parameters with four duct-burning turbofan engines. Design turbine-inlet temperature, 2200° F (1204° C).

Operation at less than maximum power to meet the noise limit requirements necessitated an increase in engine size. Figure 9 shows airport sideline noise and community noise as a function of engine airflow and percent of maximum thrust. A constraint of maximum lift-off distance and minimum climb-acceleration thrust-to-drag ratio is also shown. The duct-burning turbofan engines, designed for 2200° F (1204° C) turbine-inlet temperature and operating at maximum thrust, had sufficient thrust to meet both lift-off distance and minimum climb-acceleration thrust-to-drag ratio (point (a) on fig. 9). With this engine airflow (467 lb/sec or 212 kg/sec) the SST was able to carry the maximum number of passengers. However, the noise limits were exceeded.

Engine operation at 93 percent of maximum thrust was necessary to meet the airport noise limit constraint. The loss in thrust from part-power operation was compensated for by a 11.9-percent increase (point (b) on fig. 9) of engine airflow. This engine size was required to produce sufficient thrust during part-power operation at the 3-mile (4.8-km) point where a 500-foot-per-minute (152-m/mm) rate of climb had to be maintained and still not exceed the community noise limit constraint. Ample thrust was available to meet the lift-off distance and the climb-acceleration thrust to drag constraint.

The engine weight increased by 10 percent (fig. 8(f)) when airport and community noise restrictions were imposed on the engines designed for 2200° F (1204° C) turbine-inlet temperature.

As the design turbine-inlet temperature was increased to 3100° F (1704° C) with noise restrictions, the number of passengers the SST could carry increased by 4.3 percent (fig. 8(a)). The design compressor-pressure ratio increased from 9.4 to 11.0

(fig. 8(b)). As the turbine-inlet temperature was increased from 2200° to 3100° F (1204° to 1704° C), the design bypass ratio increased from 1.50 to 3.00 (fig. 8(c)) and the design fan-pressure ratio increased from 2.40 to 2.58 (fig. 8(d)).

At takeoff, a turbine-inlet temperature less than design was necessary to meet the airport noise constraints. However, the duct burner was operating at its maximum augmentation gas temperature of 3100° F (1704° C). The engine designed for a 2200° F (1204° C) turbine-inlet temperature was operating at 2093° F (1147° C), but the engine designed for a 3100° F (1704° C) turbine-inlet temperature was operating at 2920° F (1606° C) during takeoff.

Figure 8(e) shows the design engine airflow decreasing 2 percent when the design turbine-inlet temperature was raised from 2200° to 3100° F (1204° to 1704° C). The decrease of the duct-burning turbofan airflow and changing design characteristics resulted in a decrease of installed engine weight by 4.3 percent as shown on figure 8(f).

As shown in appendix D, a 3707-pound (1681-kg) decrease of engine and fuel weight allowed the SST to carry 4.3 percent more passengers when the turbine-inlet temperature was raised from 2200° to 3100° F (1204° to 1704° C). The fuel required for climb-acceleration and reserves decreased while the fuel needed for cruise increased as the turbine inlet temperature was raised. The fuel and engine weight decrease was then offset by an increase in the weight of additional passengers and baggage, fuselage length, and associated equipment.

Comparison of Engine Types

Payload. - A comparison of the number of passengers the methane-fueled SST could carry when the afterburning turbojet, nonafterburning turbojet, or duct-burning turbofan engine cycles were used is shown on figure 10 (from figs. 3(a), 6(a), and 8(a)). Without noise consideration, the number of passengers increased by 11 percent by increasing the turbine-inlet temperature from 2200° to 3100° F (1204° to 1704° C). Although effect of the cycle was not great, the 11-percent payload increase came about from using a duct-burning turbofan, which was superior at lower values, and the nonafterburning turbojet, which was superior at the higher values of turbine-inlet temperature.

With takeoff and community noise restrictions, the duct-burning turbofan did significantly better than either turbojet type. This was the case for all temperatures considered. By increasing design turbine-inlet temperature from 2200° to 3100° F (1204° to 1704° C), the number of passengers for the duct-burning turbofan powered SST increased by 4 percent. Thus, the benefits of high-turbine-inlet temperature were markedly affected by the takeoff noise limits. The major difference between the cases with and without noise restrictions was the consequence of noise restrictions forcing the use of larger

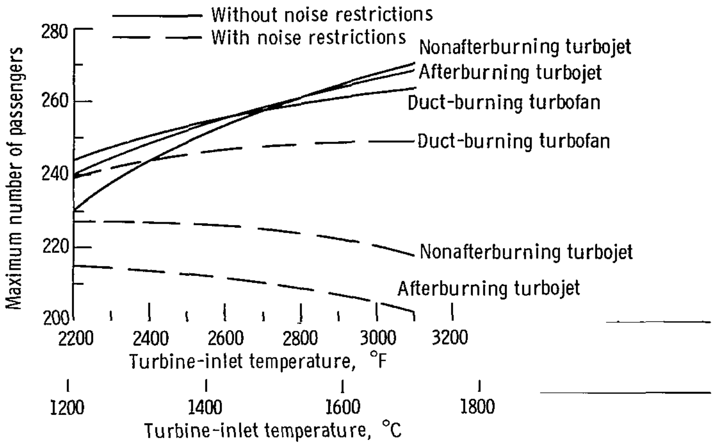


Figure 10. - Payload carrying comparison for various turbine-inlet temperatures. Ramp gross weight, 460 000 pounds (208 652 kg); minimum sea-level-static thrust to gross weight ratio, 0.32; Mach 3.0 cruise.

engines operating at part throttle during takeoff. The differences could be minimized (curves without noise restrictions approached) by development of effective jet noise suppressors having little thrust and weight penalty.

To explain the difference in payload carrying capability with and without noise consideration for the three engine types used, an SST weight summary is shown on table I. The reference design turbine-inlet temperature of 2800° F (1538° C) was used. The engine design characteristics were those that maximize payload with and without the airport and community noise constraints.

Without noise restrictions, the fuel required for the mission was very nearly the same whether the afterburning turbojets or the duct-burning turbofans were used. The nonafterburning turbojet engines used approximately 9000 pounds (4080 kg) less fuel. The installed-engine weights of the afterburning turbojet and duct-burning turbofan were very nearly equal. The mission fuel weight saved by using the nonafterburning turbojet was offset by its greater weight. A large nonafterburning turbojet engine was required to maintain a satisfactory climb-acceleration thrust to drag ratio. The engine weight plus fuel weight for the three engine types was very nearly equal. Thus, the SST payload capability was nearly the same regardless of the engine type when the design turbine-inlet temperature was 2800° F (1538° C) and noise restrictions were ignored.

With airport and community noise restrictions and using engines designed for 2800° F (1538° C) turbine-inlet temperature, a 7.7-percent payload increase resulted when changing from afterburning to nonafterburning turbojet engines. An 18.7-percent payload increase occurred when changing from afterburning turbojet to duct-burning turbofan engines. Engine and fuel weight changes affected the aircraft payload carrying capability.

The nonafterburning turbojet engines weigh approximately 8 percent (4830 lb or

TABLE I. - AIRCRAFT WEIGHT SUMMARY

[Design turbine-inlet temperature, 2800° F (1530° C)].

(a) U. S. Customary units

	Without noise restrictions			With noise restrictions		
	After-burning turbojet	Nonafter-burning turbojet	Duct-burning turbofan	After-burning turbojet	Nonafter-burning turbojet	Duct-burning turbofan
Weight summary:						
Fuel:						
Takeoff and climb	64 770	63 663	72 140	47 580	49 152	64 830
Cruise	86 130	77 987	81 720	92 100	89 595	89 470
Letdown	3 690	4 875	2 240	6 200	6 155	1 650
Reserves	30 570	30 823	29 930	33 400	33 256	28 850
Total fuel load	185 160	177 348	186 030	179 280	178 156	184 800
Installed engines	35 050	42 917	34 800	61 000	56 170	40 310
Passengers and baggage	52 200	52 400	52 000	42 000	45 000	49 800
Associated equipment (electronics, passenger and crew furnishings, service and emergency equipment, air conditioning, etc.)	37 260	37 349	36 900	31 340	33 288	35 810
Fuselage	29 040	29 038	28 840	25 420	26 460	28 150
Wing and vertical tail	82 880	82 880	82 880	82 880	82 880	82 880
Fuel system, including insulation	7 330	6 978	7 460	7 150	7 114	7 320
Landing gear	20 060	20 060	20 060	20 060	20 060	20 060
Hydraulic and electronic systems	6 830	6 830	6 830	6 830	6 830	6 830
Surface controls	4 200	4 200	4 200	4 040	4 040	4 040
Total ramp weight including total fuel	460 000	460 000	460 000	460 000	460 000	460 000
Sizes and dimensions:						
Wing planform area, sq ft	9 200	9 200	9 200	9 200	9 200	9 200
Fuselage outside diameter, in.	125	125	125	125	125	125
Fuselage length, ft	278	278	276	248	248	271
Seat pitch, in.	34	34	34	34	34	34
Aspect ratio	1.71	1.71	1.71	1.71	1.71	1.71
Number of seats abreast	5	5	5	5	5	5
Engine design corrected airflow, lb/sec	332	424	420	558	558	519
Engine overall pressure ratio (design)	16	14.8	12	11.25	11.25	10.5
Fan-pressure ratio (design)	---------	---------	3	---------	---------	2.55
Design engine bypass ratio	---------	---------	1.0	---------	---------	2.50

(b) SI units

Weight summary:						
Fuel:						
Takeoff and climb	29 379	28 877	32 722	21 581	22 295	29 406
Cruise	39 067	35 374	37 066	41 776	40 640	40 583
Letdown	1 674	2 211	1 016	2 812	2 790	748
Reserves	13 866	13 982	13 576	15 150	15 085	13 086
Total fuel load	83 986	80 444	84 380	81 319	80 810	83 823
Installed engines	15 894	19 467	15 785	27 669	25 478	18 284
Passengers and baggage	23 678	23 768	23 587	19 051	20 412	22 589
Associated equipment (electronics, passenger and crew furnishings, service and emergency equipment, air conditioning, etc.)	16 901	16 941	16 738	14 216	15 099	16 217
Fuselage	13 172	13 171	13 082	11 530	12 003	12 769
Wing and vertical tail	37 594	37 594	37 594	37 594	37 594	37 594
Fuel system, including insulation	3 325	3 165	3 384	3 243	3 227	3 320
Landing gear	9 099	9 099	9 099	9 099	9 099	9 099
Hydraulic and electric systems	3 098	3 098	3 098	3 098	3 098	3 098
Surface controls	1 905	1 905	1 905	1 833	1 833	1 833
Total ramp weight including total fuel	208 652	208 652	208 652	208 652	208 652	208 652
Sizes and dimensions:						
Wing planform area, m^2	955	855	855	855	855	855
Fuselage outside diameter, m	3.175	3.175	3.175	3.175	3.175	3.175
Fuselage length, m	84.73	84.73	84.12	75.59	78.33	82.60
Seat pitch, m	.8636	.8636	.8636	.8636	.8636	.8636
Aspect ratio	1.71	1.71	1.71	1.71	1.71	1.71
Number of seats abreast	5	5	5	5	5	5
Engine design corrected airflow, kg/sec	151	192	191	253	253	235
Engine overall pressure ratio (design)	16	14.8	12	11.25	11.25	10.5
Fan-pressure ratio (design)	-----------	-----------	3	-----------	-----------	2.55
Design engine bypass ratio	-----------	-----------	1.0	-----------	-----------	2.50

2191 kg) less than the afterburning turbojet engines (table I). The optimum design compressor-pressure ratio and size that was required to meet airport and community noise goals and lift-off distance limits were the same for both engine types. The engine weight difference was a result of additional hardware that was required to include an afterburner on the afterburning turbojet engines.

The nonafterburning turbojet powered SST used 1124 pounds (509 kg) less fuel to fly the mission. Thus, the combined effects of less engine and fuel weight enabled the nonafterburning turbojet powered SST to carry 15 more passengers than the SST using afterburning turbojet engines.

With noise restrictions, the duct-burning turbofan engines weigh 20 690 pounds (9385 kg) less than the afterburning turbojet engines. However, a fuel weight increase of 5520 pounds (2504 kg) was needed to fly the mission. This fuel weight increase offset part of the engine weight advantage. But, a 15 000-pound (6778-kg) net weight advantage for the duct-burning turbofan enables the SST to carry 39 more passengers. The weight trends that occurred at a design turbine-inlet temperature of 2800° F (1538° C) were typical for other design turbine-inlet temperatures.

Total fuel-weight differences were generally small compared with engine weight differences. However, large fuel-weight differences occurred when certain segments of the mission were examined using different engine types. These fuel-weight differences are examined in detail in appendix E.

Direct operating cost. - Figure 11 shows the effect that increasing the turbine-inlet temperature has on direct operating cost. Without noise restrictions, the direct operating cost decreased by 14 percent when the turbine-inlet temperature was increased from 2200° to 3100° F (1204° to 1704° C). The afterburning turbojet is superior at lower

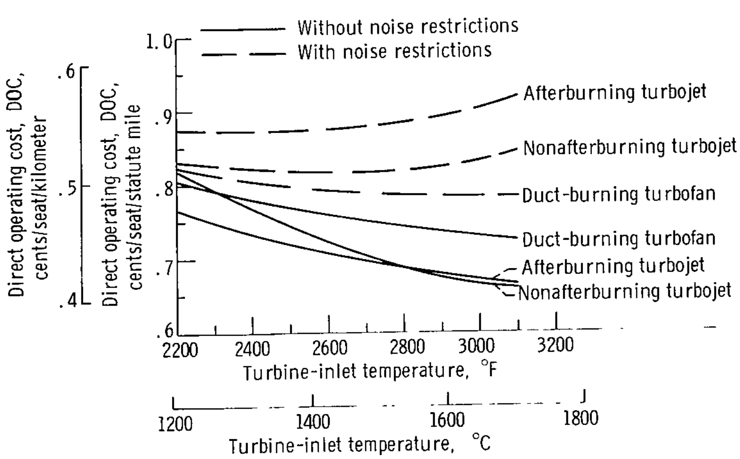


Figure 11. - Direct operating cost comparison for various turbine-inlet temperatures. Ramp gross weight, 460 000 pounds (208 652 kg); Mach 3.0 cruise.

values of turbine inlet temperature, and the nonafterburning turbojet is superior at higher values. The duct-burning turbofan powered SST direct operating cost was approximately 7 percent greater than the afterburning turbojet powered SST because of higher duct-burning turbofan engine cost.

Noise restrictions decreased the payload carrying capacity of the SST. This was reflected in an increase of the direct operating cost. The duct-burning turbofan does significantly better than either turbojet type. This was the case for all turbine-inlet temperatures considered. A reduction of 4.7 percent in direct operating cost was the result of increasing the turbine-inlet temperature from 2200° to 2900° F (1204° to 1594° C) as shown on figure 11. Design turbine-inlet temperatures beyond 2900° F (1594° C) offered no further gain. Figure 11 indicates there was no point in designing the afterburning engines for turbine-inlet temperatures beyond 2200° F (1204° C) and the nonafterburning turbojet engine beyond 2500° F (1372° C) when takeoff noise restrictions were imposed.

Whether or not any improvement in direct operating cost could be realized would depend on trends of engine price and time between engine overhaul as the design turbine-inlet temperature is raised. In calculating the direct operating cost, engine price and time between overhaul (TBO) were assumed to be independent of design turbine-inlet temperature because engine metal temperatures were independent of design turbine-inlet temperature. Because it is possible that engine price will increase and TBO will decrease as the design turbine-inlet temperature is increased, the effects of engine price and TBO on direct operating costs were calculated. Figure 12 shows that, in order to nullify the 4.7-percent direct operating cost advantage of a design turbine-inlet tempera-

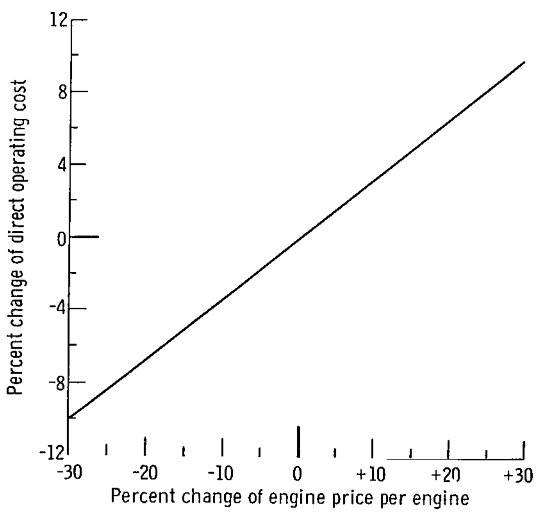


Figure 12. - Effect of change in engine price on direct operating cost, reference unit engine price, $1 530 000.

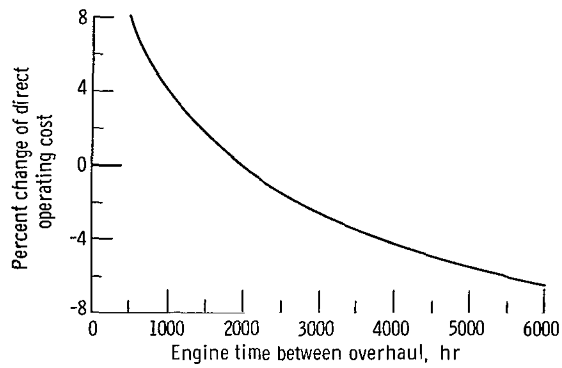


Figure 13. - Effect of time between overhaul on direct operating cost.

ture increase from 2200° to 2900° F (1204° to 1594° C) for the duct-burning turbofan engine with noise restrictions, the engine price would have to increase about 14 percent.

Figure 13 shows the effect time between overhaul has on direct operating cost of an SST. A reduction in TBO from 2000 to about 1000 hours would nullify the 4.7-percent direct operating cost reduction. Although 2000 hours TBO is a standard value used for initial operation (ref. 8) of a new engine, as operational hours build up, the historical trend has been for TBO to increase greatly. Figure 13 indicates that if TBO were to increase to 6000 hours, the direct operating cost would decrease about 6 percent.

CONCLUDING REMARKS

An analytical study was made to determine what benefits are obtained from increasing design turbine-inlet temperature with and without consideration of airport and community noise during aircraft takeoff and climb. High turbine-inlet temperatures were assumed possible because of the high heat-sink capability of liquid methane, which was used to provide additional turbine cooling. Afterburning turbojets, nonafterburning turbojets, and duct-burning turbofan engines were considered in a fixed, arrow-wing configuration designed to cruise at Mach 3. It is expected that the engine trends would be valid for other configurations. An effort was made to make consistent comparisons among the engines, but the differences displayed are small enough that errors in estimating component weight, performance, and cost could affect the relative standing of the various types.

High values of turbine-inlet temperature led to an improved payload carrying capability and a lower direct operating cost for the SST when no takeoff engine noise restrictions were observed. These improvements would be approached if a jet noise suppressor could be developed with little weight or thrust penalty. An 11-percent payload improvement, which, in turn, decreased the direct operating cost by 14 percent, was the result of raising the design turbine-inlet temperature from 2200° to 3100° F (1204° to 1704° C). At low design turbine-inlet temperature, the duct-burning turbofan was slightly superior in number of passengers. But because the duct-burning turbofan engines were more costly than the turbojets, the afterburning turbojet powered SST had a lower direct operating cost. At the high turbine-inlet temperatures, the nonafterburning turbojet was superior. The payload and direct operating cost improvements were a result of lower airflow engines using less fuel to meet the mission requirements as the design turbine-inlet temperature was raised.

Imposing noise restrictions resulted in performance and economic penalties that had a marked effect on the benefit of high turbine-inlet temperature. Only the turbofan powered vehicle benefited from increased turbine-inlet temperature, yielding a 4-percent payload improvement that lowered the direct operating cost by 4.7 percent. The direct operating cost of the SST powered by either turbojet engine increased as the turbine-inlet temperature was raised from 2200° to 3100° F (1204° to 1704° C). In all cases, noise restrictions necessitated the use of part-power operation during takeoff and climb-acceleration beyond the community 3-mile (4.8-km) point. Thus, engine sizes larger than those that enabled the aircraft to carry maximum payload were required to meet the noise and lift-off distance constraints. In fact, for both the nonafterburning and afterburning turbojet engines, the resulting oversized engines sometimes cruised at reduced turbine-inlet temperature, so that little use of the high-temperature capability could be made during the mission.

The optimum engine overall compressor-pressure ratio, bypass ratio, and fan-pressure ratio appear to be dependent not only on turbine-inlet temperature, but also on the level of noise restrictions. Thus, consideration should be given to proposed noise restriction levels because it appears that future engine designs may well be dependent on these noise levels.

Jet exhaust noise at the airport and community during takeoff and climb was the only noise source considered. At low throttle settings, sources other than jet exhaust noise may be controlling. Thus, prior to selecting an engine type or design to meet the noise restriction limits, all noise sources should be accounted for. The study illustrates the importance of jet exhaust noise on payload carrying capability, direct operating cost, and engine design for an SST configuration. The economic penalties that are indicated by the noise restrictions possibly justify further efforts to develop jet noise suppression devices.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, July 11, 1968,
120-15-02-02-22.

APPENDIX A

DETAILS ON DESIGN INPUT ASSUMPTIONS

Airframe Characteristic Details

Aerodynamics. - The assumed low-speed aerodynamic data for takeoff and initial climb are presented in figure 14. A maximum takeoff angle of attack of 11° (as limited by dragging the tail on the ground) was assumed. With this angle of attack, the lift coefficient at lift-off was thereby limited to a maximum value of 0.50. The lift-off velocity was maintained at 169 knots throughout the study. This lift-off velocity was achieved by using the maximum lift coefficient in conjunction with a takeoff using loading (i.e., ramp gross weight divided by wing planform area) of 50 pounds per square foot (2394 N/m^2). A lift-off velocity of 169 knots is similar to that of present day subsonic jets. At lift-off, the lift-to-drag ratio was approximately 5.0 (fig. 14). The higher lift-to-drag ratios were not achieved until after lift-off when higher velocities allowed lower angles of attack.

A representative value of the Mach 3 cruise lift-to-drag ratio would be 9.2. The drag incorporated into this ratio is the total drag and includes engine drag. The airplane is designed and area ruled to minimize the drag and sonic-boom overpressure profiles.

The aerodynamic data were modified in order to evaluate the effect of changes in fuselage length. The associated change in airplane drag was assumed equal to the net

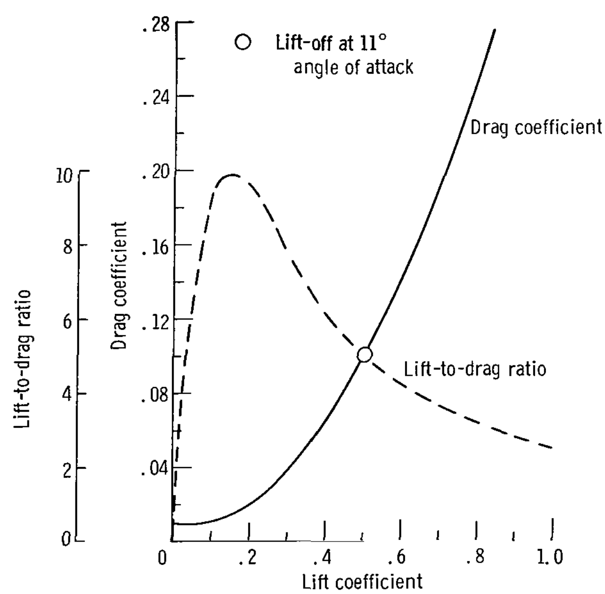


Figure 14. - Low-speed aerodynamics.

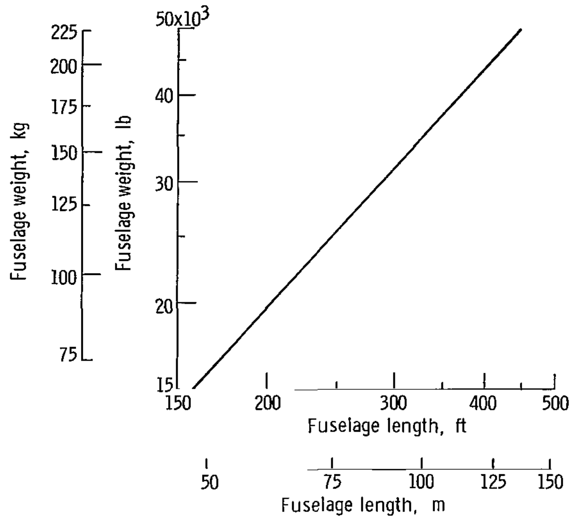


Figure 15. - Variation of fuselage weight with length (ref. 1).

change of the fuselage drag when viewed as an isolated element. However, the effect on the total airplane drag was nearly negligible.

Aircraft weight. - Changes in the various engine parameters resulted in a change in the number of passengers the SST carried. A 34-inch (0.8636-m) seat pitch with 5 abreast seating was used. The fuselage weight was varied with length according to the curve presented in figure 15 (from ref. 1). The fuselage weight term does not include the weight of the passengers furnishings and services, emergency equipment, airconditioning, etc., all which are functions of the number of passengers aboard. In this analysis, each additional passenger was considered to require 116 pounds (52.6 kg) of furnishings and other equipment plus 200 pounds (90.7 kg) for passenger and baggage.

Engine Performance Details

Component efficiencies and pressure losses. - The assumed design point component efficiencies for the three engine types are shown in table II. The adiabatic efficiency of the compressors and turbines was the same for each engine at the design point. It is realized that many factors would change these design values, thus, a study was made of the sensitivity to payload of changes in design compressor and turbine adiabatic efficiency.

Primary combustion, turbine, afterburner and duct-burner combustion efficiencies as well as primary combustor pressure loss were assumed to remain constant for all design and off-design operating conditions.

TABLE II. - ENGINE COMPONENT EFFICIENCIES

	After-burning turbojet	Nonafter-burning turbojet	Duct-burning turbofan
Fan efficiency at engine design point	-----	-----	0.85
Compressor efficiency at engine design point	0.87	0.87	.87
Primary combustor efficiency	.98	.98	.98
Primary combustor pressure loss	.06	.06	.06
Turbine efficiency (high- or low-pressure turbines)	.88	.88	.88
Afterburner efficiency	.93	-----	-----
Duct-burner efficiency	-----	-----	.93
Inlet total pressure recovery at Mach 3.0	.851	.851	.851
Exhaust nozzle thrust coefficient at Mach 3.0:			
Maximum augmentation	.966	-----	.966
Minimum augmentation and maximum nonaugmented	.977	.977	.977

The friction pressure losses in the duct and across the duct-burner flame holders of the duct-burning turbofans were calculated as a function of the duct airflow velocity. The duct area was designed so that a 0.15 Mach number resulted at the fan exit and duct-burner entrance section when the engine was operating at the design point. This resulted in approximately a 3-percent pressure loss in the duct and 2-percent across the duct burner. The pressure loss across the afterburner of the afterburning turbojets was calculated as a function of afterburner entrance Mach number. At the design point, approximately a 2 percent pressure loss occurred. Pressure losses in the duct burner and afterburner, due to acceleration of the gases resulting from heat addition, were also accounted for.

Turbine cooling. - The turbine-inlet temperatures used in this study were higher than the temperature that turbine blades can withstand without cooling. Present commercial subsonic jets have blade metal temperatures of approximately 1450° F (788° C). An extension of the blade technology is envisioned for this study, and a blade metal temperature of 1740° F (949° C) was assumed. The excellent heat-sink capacity of methane was used to maintain this blade metal temperature limit as the turbine-inlet temperature was increased. A methane-air heat exchanger was used to precool the compressor discharge air bled to the turbine. A compressor bleed-flow schedule for various turbine-inlet temperatures is shown in figure 16. For the turbine cooling method assumed in this study, this bleed schedule will maintain an approximately constant turbine blade metal temperature of 1740° F (949° C). A turbine-inlet temperature of 3100° F (1704° C) very nearly uses all the available methane heat-sink capacity for turbine cooling.

Inlet. - The inlets are assumed to be of the mixed compression inlet type. These inlets are sized to capture the entire free-stream tube at the Mach 3 cruise condition.

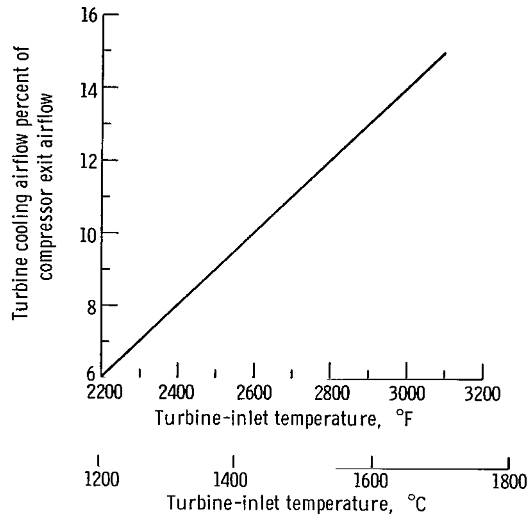


Figure 16. - Compressor bleed airflow schedule for turbine cooling.

Spillage drag below Mach 3, nacelle wave and friction drag, and inlet boundary-layer bleed drag associated with dumping the inlet boundary-layer control bleed overboard are taken into account in calculating installed-engine performance. A schedule of the inlet pressure recovery included in the engine performance is shown on figure 17. This pressure recovery as a function of flight Mach number was used for all engines considered.

Exhaust nozzle. - Figure 18 shows the nozzle thrust coefficient that was used in calculating installed-engine performance. The coefficient, was adjusted to include nozzle boat-tail drag and was a function of both free-stream Mach number and engine power setting. The schedule of thrust coefficient with flight Mach number is representative of a high-performance variable-geometry convergent-divergent ejector exhaust nozzle.

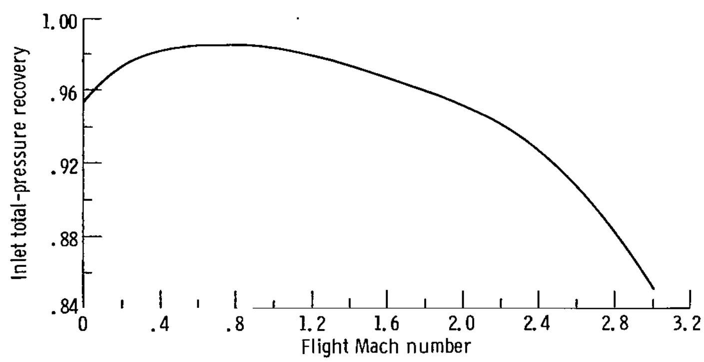


Figure 17. - Inlet pressure recovery.

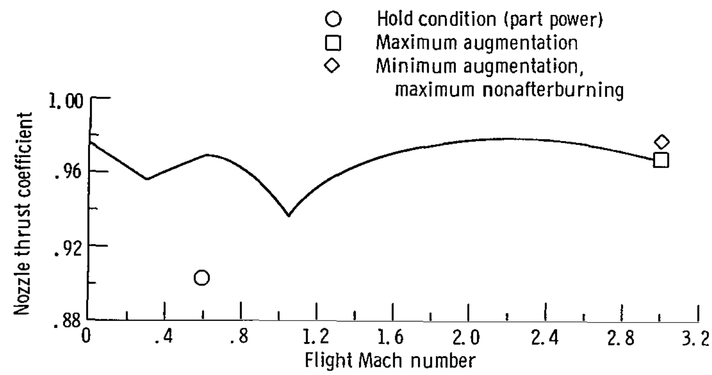


Figure 18. - Nozzle thrust coefficient.

Engine Weight Details

Afterburning and nonafterburning turbojets. - The installed weight for four engines is calculated by the following empirical equation:

$$\text{Installed weight of four turbojet engines (lb)} = \Bigg\{4W_{a_{ref}}\left[17.9(\mathrm{OPR})_{fac} + 6.9\right] + 1200 + 1.33\left(T_{4_{des}} - T_{4_{ref}}\right)\Bigg\}\left(\frac{W_{a_{des}}}{W_{a_{ref}}}\right)^{1.2} \tag{A1a}$$

$$\text{Installed weight of four turbojet engines (kg)} = \Bigg\{4W_{a_{ref}}\left[17.9(\mathrm{OPR})_{fac} + 6.9\right] + 544 + 1.084\left(T_{4_{des}} - T_{4_{ref}}\right)\Bigg\}\left(\frac{W_{a_{des}}}{W_{a_{ref}}}\right)^{1.2} \tag{A1b}$$

The overall compressor-pressure ratio weight factor $(\mathrm{OPR})_{fac}$ as a function of design compressor-pressure ratio is shown in figure 19 for afterburning and nonafterburning turbojet engines. The increase in the turbojet engine weight is a result of several factors. Some of the most important factors are added compressor and turbine stages and heavier rotating and static parts as a result of the higher pressure and temperature levels. The lower weight factor shown for the nonafterburning turbojet results from re-

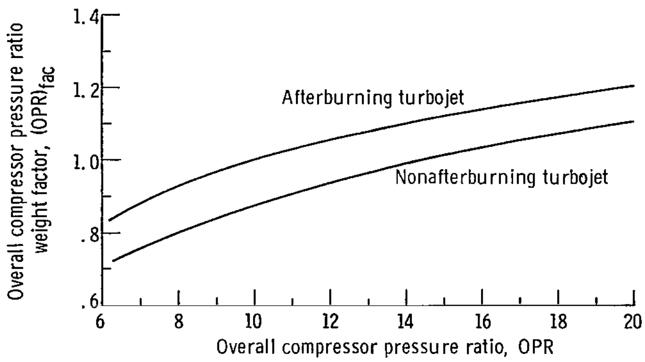


Figure 19. - Turbojet overall compressor pressure ratio weight factor for Mach 3.0 cruise application; reference overall compressor pressure ratio, 9.92.

moval of additional fuel pumps and hardware that are required for afterburning. Figure 19 is applicable only for turbojet engines designed to cruise at Mach 3.0.

An increase in turbine-inlet temperature means a larger heat exchanger is required to pass additional cooling airflow. Also, more rigid construction of the combustor and combustor liner, turbine cases and exhaust structure is necessary. The heat exchangers for 4 reference turbojet engines weigh 1200 pounds (544 kg). The reference turbine-inlet temperature $T_{4_{ref}}$ is 2200° F (1204° C). To include the affect of increased design turbine turbine-inlet temperature on the heat exchanger and engine structural weight, 1.33 times the difference between the design turbine-inlet temperature $T_{4_{des}}$ and the reference value was used $\left(1.084 \text{ when } T_{4_{des}} \text{ is in } {}^{\circ}\mathrm{C}\right)$.

The equation given was used to calculate the weight of four complete pods including inlet, nacelle, gas generator and accessories, methane-air heat exchanger, and ejector nozzle with reverser. The weight for the turbojet is scaled to other engine sizes by using the design to reference airflow ratio $W_{a_{des}}/W_{a_{ref}}$ raised to the 1.2 power where the reference engine airflow is 475 pounds per second (215 kg/sec).

Duct burning turbofan. - The empirical equation for installed-engine weight is as follows:

$$\text{Installed weight of four duct-burning turbofan engines (lb)} = \Big\{4W_{a_{ref}}\big[14.98(\mathrm{OPR}$$

$$+ \mathrm{BPR})_{fac}(\mathrm{FPR})_{fac} + 6.3\big] + 380.0 + 0.286\left(T_{4_{des}} - T_{4_{ref}}\right)\Big\}\left(\frac{W_{a_{des}}}{W_{a_{ref}}}\right)^{1.1} \tag{A2a}$$

$$\text{Installed weight of four duct-burning turbofan engines (kg)} = \left\{4W_{a_{ref}}\left[14.98(\mathrm{OPR} + \mathrm{BPR})_{fac}(\mathrm{FPR})_{fac} + 6.3\right] + 172.0 + \frac{458}{(\mathrm{BPR}_{des} + 1.0)} + 0.233\left(T_{4_{des}} - T_{4_{ref}}\right)\right\}\left(\frac{W_{a_{des}}}{W_{a_{ref}}}\right)^{1.1} \tag{A2b}$$

The turbofan overall compressor pressure ratio and bypass ratio weight factor $(\mathrm{OPR} + \mathrm{BPR})_{fac}$ increases as the design compressor-pressure ratio increases as is illustrated in figure 20(a). The effect of the design bypass ratio is also shown in figure 20(a). The weight factor decreased at a constant pressure ratio when the bypass ratio is increased over the range considered in the study. This is because the gas generator becomes smaller for a given airflow size as the design bypass ratio is increased.

The fan-pressure ratio factor $(\mathrm{FPR})_{fac}$ as a function of design fan-pressure ratio is shown on figure 20(b). As the fan-pressure ratio is increased, the compressor-power requirements are reduced if the design values of overall compressor-pressure ratio and bypass ratio remain constant. Thus, the weight of the compressor spool may decrease, but additional low-pressure turbine stages increase the weight. A split of pressure ratios for the fan and compressor exists that will minimize the fan-pressure ratio weight factor.

An increase in the design turbine-inlet temperature again means additional cooling flow and more rigid construction is required. The heat exchanger weight for four reference duct-burning turbofan engines is 717 pounds (326 kg). The reference engine has a design value of bypass ratio $(\mathrm{BPR})_{des}$ of 2.0, a turbine-inlet temperature of 2200° F (1204° C), and a total airflow of 600 pounds per second (272 kg/sec). The weight of the

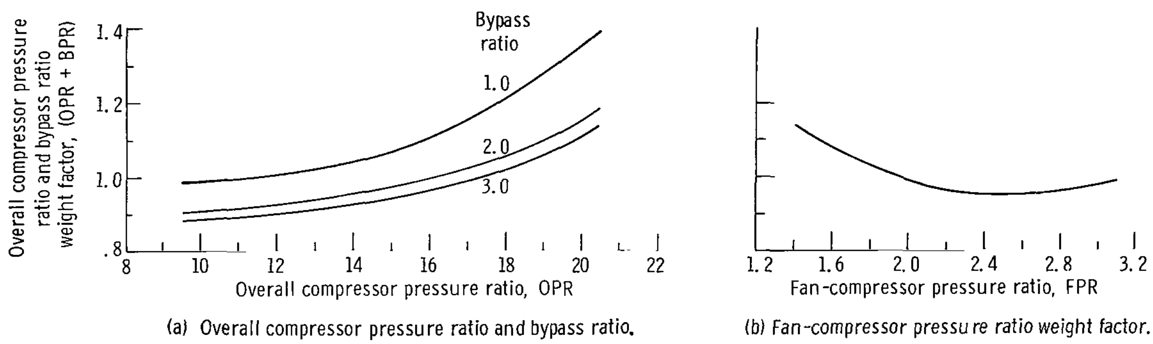


Figure 20. - Duct-burning turbofan weight factors applicable to Mach 3.0 cruise engines only.

heat exchanger is dependent on the amount of air it must cool. Thus, the heat exchanger weight is a function of bypass ratio by the term 1010.5/(BPR_{des} + 1.0) in pounds (or 458/(BPR_{des} + 1.0) in kg). The turbine-inlet temperature affects the heat exchanger and engine structural weight by the expression $0.286\left(T_{4_{des}} - T_{4_{ref}}\right)$ and the constant 380 if temperature is in degrees Fahrenheit and weight is in pounds. The expression becomes $0.233\left(T_{4_{des}} - T_{4_{ref}}\right)$ and the constant 172.0 when temperature is in degrees Centigrade and weight is in kilograms. The turbofan weight is scaled to other sizes by the airflow ratio $W_{a_{des}}/W_{a_{ref}}$ raised to the 1.1 power.

Engine Sizing Details

Without noise restrictions. - Engine size was varied to maximize the number of passengers the SST could carry. However, it was necessary to investigate certain operational limits that might be critical in the selection of the proper engine size. These limits were lift-off distance, second-segment climb, climb-acceleration thrust-to-drag ratio, and sonic-boom overpressure.

Lift-off distance less than or equal to 4450 feet (1460 m) on a standard temperature day was acceptable. To accomplish this, a minimum 0.32 takeoff thrust to ramp gross weight was required. Reference 3 indicates, for the SCAT15F configuration, that this limit is considered to be a reasonable design criterion and is comparable with present-day intercontinental subsonic jet transports.

The second-segment climb limit pertains to the flight of the aircraft immediately after takeoff and after landing gear retraction. It is an FAA requirement that a 1.7-degree climb angle be possible when the aircraft has three out of four engines operating on a nonstandard day (standard day, +15° C). However, maximum thrust can be to meet these requirements.

During climb and acceleration to cruise, a minimum thrust-to-drag ratio of 1.4 was maintained. The lowest thrust-to-drag ratio will normally occur during the transonic Mach numbers (Mach 1.0 to 1.5) where the aircraft drag is the highest. Although this limit is not an FAA regulation, it is an industry guide to limit the airplane range decrement on a hot day.

With noise restrictions. - The jet exhaust noise is not to exceed 116 PNdB at 1500 feet (457 m) from the aircraft sideline at the start of takeoff roll. At the 3-mile (4.8-km) community point, the noise is not to exceed 105 PNdB. With these additional restrictions, the search for an optimum engine size is more involved. First, the engine size that met all the sizing restraints when no noise limits were imposed was considered. With the takeoff thrust set at maximum, a noise level at 1500 feet (457 m) was calculated.

Then the flight path to the 3-mile (4.8-km) point was optimized to determine the Mach number and altitude that would result in the minimum noise after thrust reduction for a 500-foot-per-minute (152-m/min) rate of climb. If the noise limit at either point was exceeded, a takeoff-thrust setting less than maximum was considered, and the noise at both points was calculated again. Takeoff-thrust setting could not be reduced too far, however, or the lift-off distance constraint would be violated. If the noise limits were still exceeded, the calculations were repeated using larger engine sizes capable of producing sufficient thrust at still lower thrust settings. This iteration was done until all takeoff and mission requirements were satisfied. A high-speed digital computer was used to optimize the compromises to ensure the least payload penalty.

Direct Operating Cost Airframe and Engine Cost Details

The equation used to calculate the airframe price is as follows:

$$\text{Cost (dollars)} = 19\times10^{6} + (\text{WAF} - 150\,000\ \text{lb})66.7$$

$$\text{Cost (dollars)} = 19\times10^{6} + (\text{WAF} - 68\,000\ \text{kg})147$$

The airframe prices were based on weight of the airframe without engines, fuel, and passengers (WAF in lb or kg). The price of development was included. It was assumed that airframe price for weights ranging from 150 000 to 200 000 pounds (68 000 to 88 600 kg) could be calculated from the equation. A 1 million dollar cost for electronics was included in the airframe price.

The selling price per engine, including development cost was estimated by the following equations. For the afterburning turbojet, the equations used were

$$\text{Cost (dollars)} = 0.00156(W_a - 300\ \text{lb/sec}) + 1.08\times10^{6}$$

$$\text{Cost (dollars)} = 0.00344(W_a - 136\ \text{kg/sec}) + 1.08\times10^{6}$$

For the nonafterburning turbojet, the equations used were

$$\text{Cost (dollars)} = 0.00156(W_a - 300\ \text{lb/sec}) + 1.04\times10^{6}$$

$$\text{Cost (dollars)} = 0.00344(W_a - 136\ \text{kg/sec}) + 1.04\times10^6$$

For the duct-burning turbofans, the equations used were

$$\text{Cost (dollars)} = 0.00127(W_a - 300\ \text{lb/sec}) + 1.21\times10^6$$

$$\text{Cost (dollars)} = 0.00280(W_a - 136\ \text{kg/sec}) + 1.21\times10^6$$

The engine price was assumed to be a function of engine size. The equations were used for a range of engine airflows from 300 to 600 pounds per second (136 to 272 kg/sec).

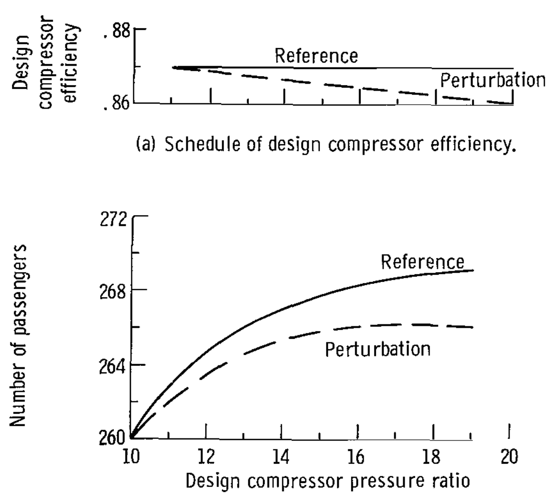
APPENDIX B

EFFECTS OF PERTURBATIONS TO BASIC ENGINE ASSUMPTIONS

Design Compressor and Turbine Efficiency

The calculations presented thus far are for a fixed-design compressor and turbine adiabatic efficiency. Although the compressor adiabatic efficiency varied as the engine operating conditions and power settings were changed, the design value of compressor efficiency was not varied with change of design compressor-pressure ratio. Generally, the design value of overall compressor efficiency will decrease when the design compressor-pressure ratio is increased. To show the effect of changes in compressor efficiency on payload carrying capability, an SST configuration powered by four afterburning turbojet engines was used (fig. 21). The turbojet engines had a design turbine-inlet temperature of 3100° F (1704° C). Airport and community noise restrictions were not considered.

Two adiabatic efficiency schedules are shown as a function of design compressor-pressure ratio on figure 21(a). The schedule showing a constant 87-percent adiabatic efficiency (reference) is the schedule that was used throughout the study. A constant small-stage (polytropic) efficiency of 90 percent was used to determine the other efficiency schedule, which is labeled perturbation. Using this value of small-stage efficiency and



(a) Schedule of design compressor efficiency.

(b) Effect of compressor pressure ratio on number of passengers.

Figure 21. - Effect of compressor efficiency on performance. Ramp gross weight, 460 000 pounds (8652 kg); afterburning turbojet; design turbine-inlet temperature, 3100° F (1704° C); no noise restrictions.

the equations of reference 9, an adiabatic efficiency of 87 percent was calculated with a compressor-pressure ratio of 10. For compressor-pressure ratios above 10, methods described in reference 9 were again used to calculate adiabatic efficiency which decreased approximately 1 percent as the compressor-pressure ratio was increased from 10 to 19.

When the design adiabatic compressor efficiency is changed according to the schedule shown on figure 21(a), the number of passengers and the optimum design compressor-pressure ratio are changed (fig. 21(b)). Instead of optimizing at a compressor-pressure ratio of 19 when the SST was able to carry 269 passengers, a compressor-pressure ratio between 16 and 19 was best with a decrease of three passengers (a 1.1-percent decrease).

The overall payload improvement in going from a turbine-inlet temperature of 2200° F (1204° C) to 3100° F (1704° C) would drop from 12.1 to 10.8 percent.

The effect of turbine efficiency on the number of passengers of the SST powered by four afterburning turbojet engines is shown in figure 22. The reference design turbine adiabatic efficiency that was used for the study was 88 percent. Many factors affect the level of turbine efficiency. The turbine efficiency will vary somewhat as turbine cooling airflow is changed. Changes in design pressure and bypass ratio will also affect turbine efficiency. The design turbine efficiency was decreased over a range up to 6 points on the airburning turbojet engine. The engine used has a design compressor-pressure ratio of 19 and a turbine-inlet temperature of 3100° F (1704° C). A 6-point decrease in turbine efficiency caused a decrease of number of passengers from 269 to 259.

The decrease in payload that accompanied the drop in turbine and compressor efficiency illustrated that, in order to realize any potential gains of high-turbine-inlet temperature operation, turbine and compressor efficiency must not be degraded appreciably.

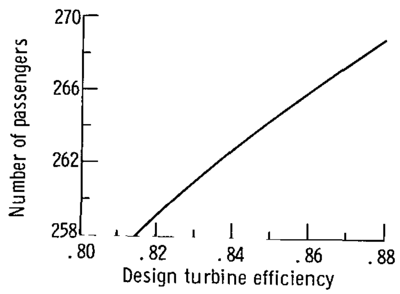


Figure 22. - Effect of design turbine efficiency on number of passengers. Afterburning turbojet; design turbine-inlet temperature, 3100° F (1704° C); design compressor pressure ratio, 19.0; ramp gross weight, 460 000 pounds (208 652 kg); no noise restrictions.

Turbine Cooling Airflow

The turbojet engine with a design turbine-inlet temperature of 3100° F (1704° C) uses 15 percent of the compressor airflow to cool the turbine. Advances in turbine technology may decrease the amount of compressor bleed air that is required. Thus, the cooling airflow requirements shown in figure 16 are subject to change. The effect that changes in the cooling airflow requirements had on the number of passengers the SST can carry is shown on figure 23. The SST that uses four afterburning turbojet engines having a design turbine-inlet temperature of 3100° F (1704° C) and a compressor-pressure ratio of 19 was used to show the effect. A cooling airflow of 15 percent was used as a reference. If

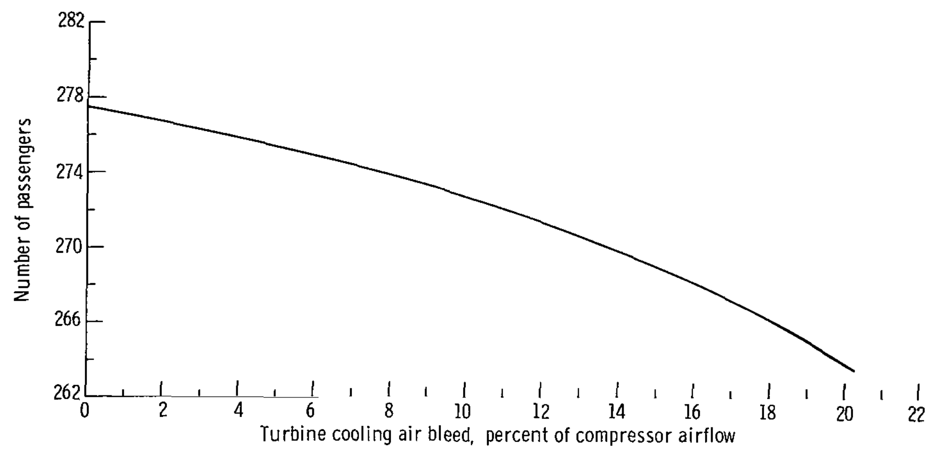


Figure 23. - Effect of compressor bleed air for turbine cooling on number of passengers. Afterburning turbojet; design turbine-inlet temperature, 3100° F (1704° C); design compressor pressure ratio, 19.0; ramp gross weight, 460 000 pounds (208 652 kg); no noise restrictions.

cooling requirements were such that 20 percent of the compressor airflow was needed, a decrease of five passengers would result. If the 15-percent cooling airflow were completely eliminated, approximately nine more passengers could be carried.

The increase in payload would be even more if a decrease in cooling airflow would result in a more efficient turbine operation. Also, the possibility of a lighter engine exists because a smaller heat exchanger would be required.

Engine Weight

Because the engine weight associated with higher temperature levels may be somewhat optimistic, figure 24 shows the effect of changes of engine weight on the number of passengers the SST can carry when four afterburning turbojets are used. The effect

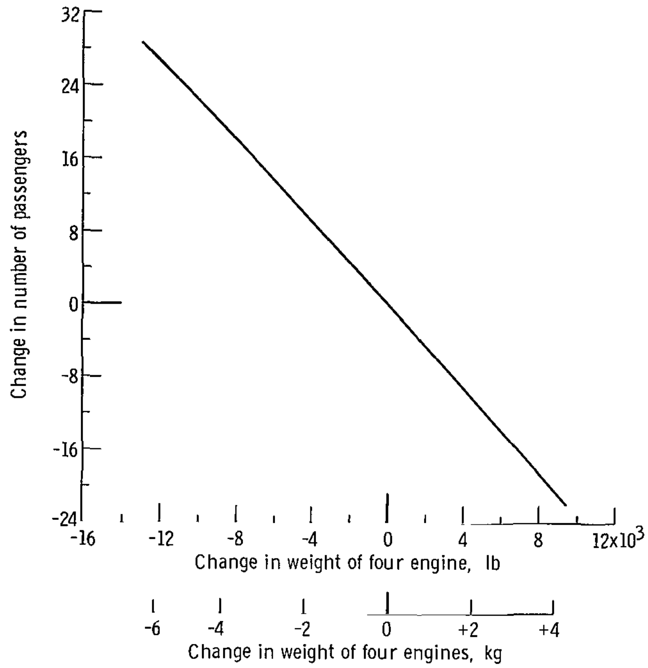


Figure 24. - Effect of change in installed engine weight on number of passengers. Afterburning turbojet; design turbine-inlet temperature, 3100° F (1704° C); design compressor pressure ratio, 19.0; ramp gross weight, 460 000 pounds (208 652 kg); no noise restrictions.

would be similar if duct-burning turbofan engines were used. For example, a decrease of total engine weight by 1800 pounds (815 kg) will increase the payload by 4 passengers. An engine weight decrease of 1800 pounds (815 kg) is approximately equivalent to removing all the turbine cooling heat exchangers.

APPENDIX C

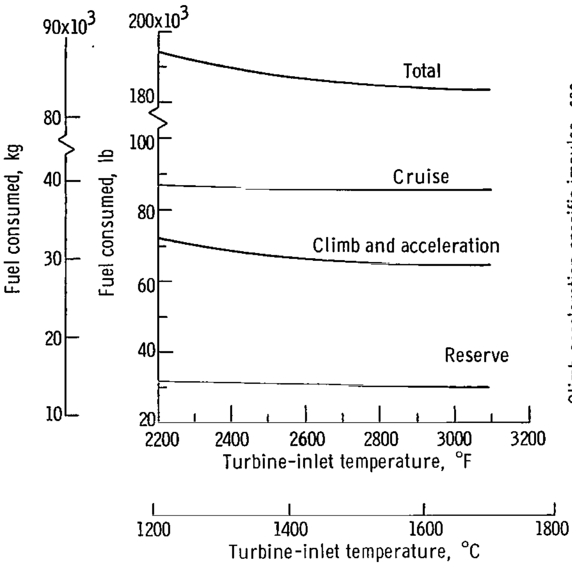
AFTERBURNING TURBOJET SST MISSION FUEL DETAILS

The effect of turbine-inlet temperature on mission fuel weight and engine specific impulse is shown in figure 25 for the afterburning turbojet powered SST without noise restrictions. The total fuel decreased by 10 800 pounds (4900 kg) for the SST mission (fig. 25(a)), when the design turbine-inlet temperature is increased from 2200° to 3100° F (1204° to 1704° C). Approximately 8000 pounds (3628 kg) of this fuel difference can be accounted for during climb-acceleration to cruise altitude and Mach number.

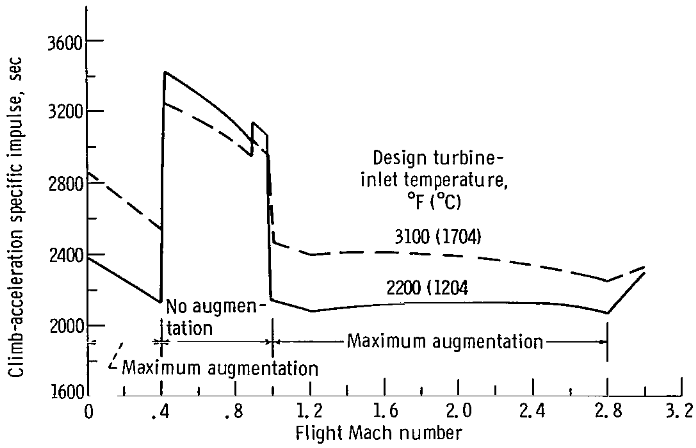
Figure 25(b) shows installed-engine specific impulse for climb and acceleration as a function of flight Mach number. Flight Mach number, power setting, and altitude will affect the value of specific impulse. The figure shows a specific impulse advantage for the turbojet having a 3100° F (1704° C) design turbine-inlet temperature over one using 2200° F (1204° C) turbine-inlet temperature during the period of operation when maximum augmentation is used. Between Mach 0.40 and 1.0, the afterburning turbojet with the higher design turbine-inlet temperature has the lower specific impulse. During this portion of the flight, both engines are operating at maximum nonafterburning power. Increased compressor bleed flow for turbine cooling of the 3100° F (1704° C) engine lessened the specific-impulse advantage this engine has during maximum augmentation operation. During afterburning, a maximum afterburner gas temperature of 3100° F (1704° C) was used. Because of the high turbine-discharge temperature of the engine with the design turbine-inlet temperature of 3100° F (1704° C), less temperature rise was required in its afterburner than for the 2200° F (1204° C) temperature engine. Consequently, the engine with the higher design turbine-inlet temperature will benefit through a better specific impulse from the fixed value of maximum allowable afterburner gas temperature. The reason being that less fuel was required in the afterburner to reheat the turbine discharge gas to the maximum afterburner gas temperature.

The ratio of engine thrust to aircraft ramp gross weight is another factor affecting the amount of fuel required during climb and accleration. The optimum thrust to ramp gross weight ratio used was the one that maximized payload when no noise restrictions were observed. At a 2200° F (1204° C) design turbine-inlet temperature, the optimum thrust to ramp gross weight ratio was 0.338; at a 3100° F (1704° C) design turbine-inlet temperature, it was 0.355. This simply meant that the 3100° F (1704° C) engine had more thrust for acceleration and climb. Thus, the SST reached its cruise speed and altitude in less time and range, which also helped decrease the fuel used during the climb and acceleration.

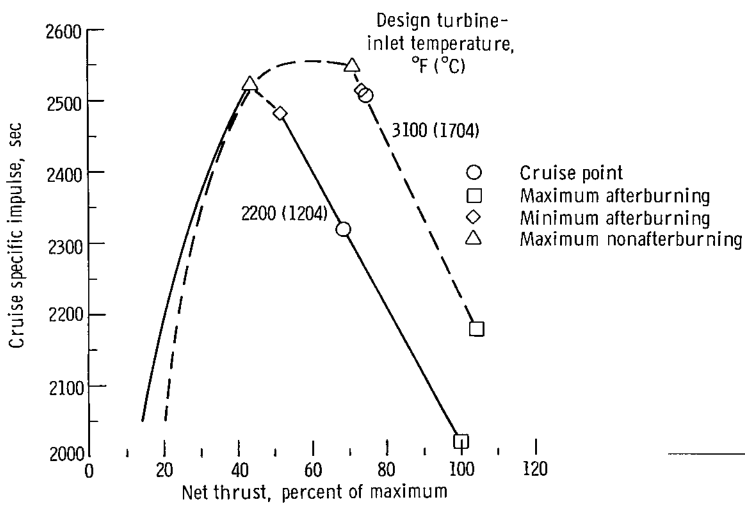
As shown on figure 25(a), a 1400-pound (635-kg) fuel savings occurred during cruise as the engine design turbine-inlet temperature was increased from 2200° to 3100° F



(a) Fuel requirements for 3500-nautical-mile (6482-km) mission.



(b) Installed engine specific impulse during climb-acceleration for afterburning turbojets.



(c) Installed-engine specific impulse during Mach 3.0 cruise for afterburning turbojet.

Figure 25. - Effect of turbine-inlet temperature on aircraft fuel requirements and engine performance; afterburning turbojet engines with optimum design characteristics; no noise limits.

(1204° to 1704° C). During cruise, there was approximately an 8-percent cruise specific-impulse gain (figs. 25(c)) for the higher temperature afterburning turbojet. In addition, for the fixed-range mission of 3500 nautical miles (6482 km), less range during climb and acceleration to cruise was required for the SST with that engine. Thus, a longer cruise range was necessary when this engine was used. Therefore, the savings in cruise fuel was not as much as the cruise specific-impulse improvement would indicate.

APPENDIX D

DUCT-BURNING TURBOFAN POWERED SST WEIGHT DETAILS

As the design turbine-inlet temperature was increased from 2200° to 3100° F (1204° to 1704° C) with noise restrictions, the number of passengers the SST could carry increased by 4.3 percent. The payload increase was the result of weight differences in engine, fuel, payload, fuselage, and associated equipment for an SST with a fixed ramp gross weight of 460 000 pounds (208 652 kg) table III shows these weight differences when comparing the SST using duct-burning turbofans designed for 3100° F (1704° C) turbine-inlet temperature to the SST using 2200° F (1204° C) temperature engines.

Less fuel was required for climb and acceleration to cruise speed and altitude for the high temperature engine-SST combination. This fuel decrease partially came about from an increase of the optimum takeoff thrust to ramp gross weight ratio as the design turbine-inlet temperature was raised. A value of 0.353 takeoff thrust to ramp gross weight was best at 2200° F (1204° C), while 0.374 was best at 3100° F (1704° C) turbine-inlet temperature. Similar to the turbojet, a larger value of takeoff thrust to ramp gross weight ratio meant more thrust was available during climb and acceleration. Thereby, the time and range up to cruise altitude and speed was less. This alone does not necessarily mean less fuel would be used. However, the specific impulse of the engines was nearly the same; consequently, a fuel decrease of 2878 pounds (1306 kg) resulted by using the high-temperature engine.

During cruise, the SST with the 3100° F (1704° C) design turbine-inlet temperature used more fuel than did the SST using the 2200° F (1204° C) engine. For the fixed-range mission of 3500 nautical miles (6482 km), less range during climb and acceleration to cruise was required for the SST with the high-temperature engine. Therefore, a longer cruise range was necessary for the SST with that engine. Only a slight specific-impulse improvement occurred for the high-temperature engine; therefore, it used 1102 pounds (500 kg) more fuel during cruise.

The change in reserve fuel was negligible. As shown in table III, a 1807-pound (820-kg) fuel saving was realized for the SST using the engine designed for a 3100° F (1704° C) turbine-inlet temperature.

The engine weight decreased by 4.3 percent (1900 lb or 861 kg) when the turbine-inlet temperature was raised.

The net decrease of weight (fuel and engines) is 3707 pounds (1681 kg) as shown in table III. This enabled the vehicle to carry 10 additional passengers as shown in the table. The additional weight carried consisted of 2000 pounds (907 kg) for passengers and baggage, 687 pounds (312 kg) for fuselage weight, and 1020 pounds (462 kg) for as-

TABLE III. - SST WEIGHT DIFFERENCES USING

DUCT-BURNING TURBOFAN ENGINES

[With noise restrictions; ramp gross weight, 460 000 lbs (208 652 kg).]

(a) U. S. Customary units

Item	Design turbine-inlet temperature, °F		3100° F engine relative to 2200° F engine
	2200	3100	
	Weight, lb		
Climb, acceleration, and descent fuel	69 079	66 201	-2878
Cruise fuel	88 721	89 823	1102
Reserve fuel	28 922	28 891	-31
Total fuel	186 722	184 915	-1807
Installed engine weight	41 300	39 400	-1900
Engine plus fuel weight difference	------	------	-3707
Passengers and baggage	47 800	49 800	2000
Fuselage	27 464	28 151	687
Associated equipment	34 753	35 773	1020
Net weight difference	------	------	3707
Number of passengers	239	249	10

(b) SI units

Item	Design turbine-inlet temperature, °C		1704° C engine relative to 1204° C engine
	1204	1704	
	Weight, kg		
Climb, acceleration, and descent fuel	31 334	30 028	-1306
Cruise fuel	40 243	40 743	500
Reserve fuel	13 119	13 105	-14
Total fuel	84 696	83 876	-820
Installed engine weight	18 711	17 850	-861
Engine plus fuel weight difference	-----	-----	-1681
Passengers and baggage	21 682	22 589	907
Fuselage	12 457	12 769	312
Associated equipment	15 764	16 226	462
Net weight difference	-----	-----	1681
Number of passengers	239	249	10

sociated equipment. The net result is a 4.3-percent payload increase for the aircraft using the duct-burning turbofan engines when the turbine-inlet temperature is increased from 2200° to 3100° F (1204° to 1704° C).

APPENDIX E

FUEL WEIGHT BY MISSION SEGMENT DETAILS FOR

TAKEOFF, CLIMB, AND ACCLERATION

One type of engine will be better designed to operate at a certain flight condition than another type. Therefore, table IV is used to show fuel-weight differences of the nonafterburning turbojet and duct-burning turbofan relative to the afterburning turbojet engine. (The sum of the differences account for the total fuel difference previously shown on table I in the report.)

TABLE IV. - FUEL WEIGHT DIFFERENCES

[Design turbine-inlet temperature, 2800° F (1538° C); with noise restrictions.]

(a) U.S. Customary units

Item	Weight, lb				
	Afterburning turbojet	Nonafterburning turbojet	Duct-burning turbofan	Nonafterburning turbojet relative to afterburning turbojet	Duct-burning turbofan relative to afterburning turbojet
Takeoff, climb, and acceleration	47 580	49 152	64 830	1572	17 250
Cruise	92 100	89 595	89 470	-2505	-2 630
Letdown	6 200	6 153	1 650	-47	-4 550
Reserves	33 400	33 256	28 850	-144	-4 550
Net weight difference				-1124	5 520

(b) SI units

Item	Weight, kg				
	Afterburning turbojet	Nonafterburning turbojet	Duct-burning turbofan	Nonafterburning turbojet relative to afterburning turbojet	Duct-burning turbofan relative to afterburning turbojet
Takeoff, climb, and acceleration	21 581	22 295	29 406	714	7825
Cruise	41 779	40 640	40 583	-1136	-1193
Letdown	2 812	2 790	748	-22	-2064
Reserves	15 150	15 085	13 086	-65	-2064
Net weight difference				-509	2504

During takeoff, climb, and acceleration, the nonafterburning turbojet used 1572 pounds (715 kg) more fuel than did the afterburning turbojet. During this part of the mission, the fuel-weight difference can be approximated by examining the following equation:

$$dW_f = \frac{\frac{W}{g}}{I \frac{F - D}{F}} dV$$

where

W aircraft weight

g 32.2 ft/sec^2

I engine specific impulse

F engine net thrust

D aircraft drag

dV differential of airplane velocity

dW_f differential of fuel required

High values of specific impulse and thrust-to-drag ratio lower the amount of fuel required to climb and accelerate. Figure 26 shows the thrust-to-drag (F/D) ratio from takeoff to

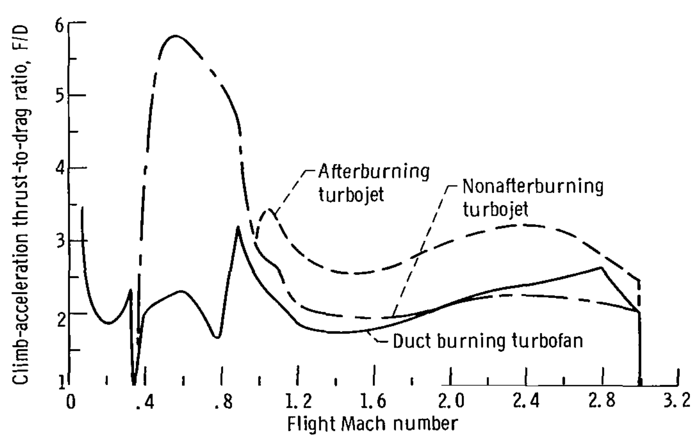


Figure 26. - Climb-acceleration thrust-to-drag ratio as function of engine type and Mach number. Ramp gross weight, 460 000 pounds (208 652 kg); maximum sonic-boom overpressure, 2.0 pounds per square foot (95.76 N/m^2); engine design turbine-inlet temperature, 2800° F (1538° C); with noise restrictions.

cruise as a function of Mach number and engine type. The engines were operating in the manner described in the engine operation section when airport and community noise restrictions were observed. The afterburning turbojet had a minimum transonic thrust-to-drag ratio of 2.55 at Mach 1.5. Using a nonafterburning turbojet, a 1.95 minimum thrust-to-drag ratio results. These minimum thrust-to-drag ratios well exceeded the minimum recommended value of 1.4. The thrust-to-drag ratio of 1.0 at Mach 0.35 represents power cutback at the 3-mile (4.8-km) point for noise abatement purposes. Above Mach 1.0, the nonafterburning turbojet had less thrust than did the afterburning turbojet that was operating with maximum afterburning.

The equation shows that specific impulse is inversely proportional to fuel weight. Figure 27 shows specific impulse as a function of Mach number and engine type during climb and acceleration. Above Mach 1.0, a specific impulse advantage exists for the nonafterburning turbojet. Therefore, the product of thrust margin and engine specific impulse nearly equalize the fuel used by both engines during takeoff, climb, and acceleration (table IV).

Cruise. - Cruise fuel was determined by the amount of cruise range and specific impulse during cruise. Table IV shows the nonafterburning turbojet using 2505 pounds (1136 kg) less fuel than the afterburning turbojet. Figure 28 shows cruise specific impulse as a function of percent of maximum thrust for the three engine types studied. The

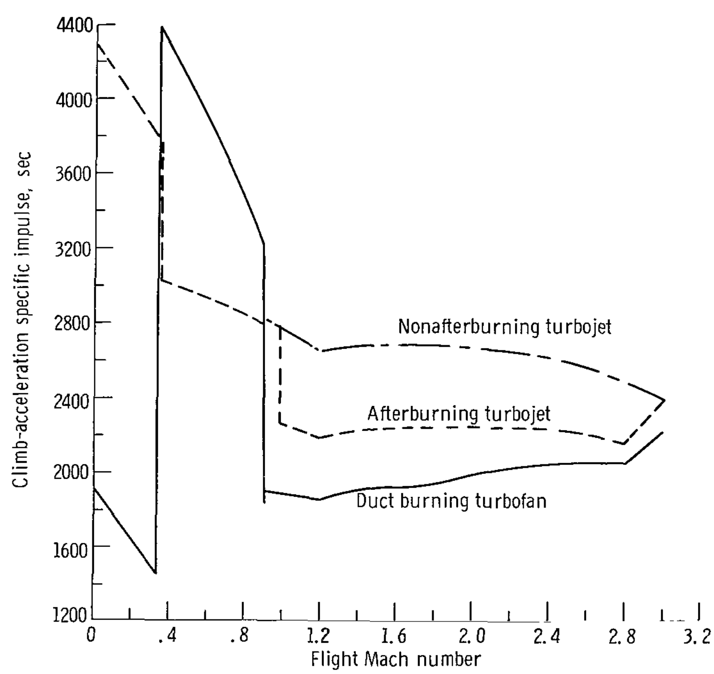


Figure 27. - Installed climb-acceleration engine performance (flight path as shown in fig. 2); design turbine-inlet temperature, 2800° F (1538° C); with noise restrictions.

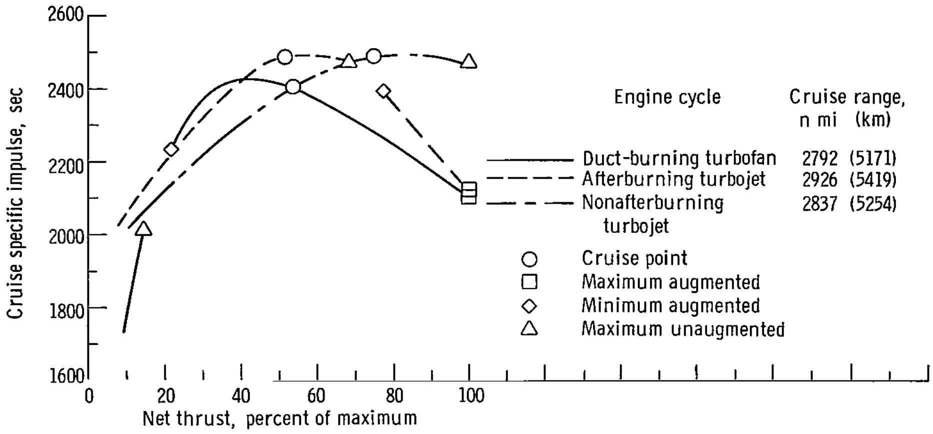


Figure 28. - Installed-engine specific impulse for Mach 3.0 cruise. Design turbine-inlet temperature, 2800° F (1538° C); engines sized with noise restrictions.

cruise point is the point that resulted in the maximum Breguet factor for cruise. It should be noted that the cruise point is located in the region where the engine is operating at less than maximum design turbine-inlet temperature. Thus, in some cases, the high design turbine-inlet temperature capability of the turbojet engines was not used during cruise because the large engine size required to meet the lift-off distance and noise constraints was considerably oversized for cruise.

Also tabulated on figure 28 is the aircraft cruise range for each engine type. The nonafterburning turbojet cruise specific impulse was approximately the same as that of the afterburning turbojet, because the afterburning turbojet did not require afterburning during cruise. However, the aircraft cruise range using the dry turbojet was 89 nautical miles (165 km) less than that of the vehicle which used the afterburning turbojet. The cruise range difference came about from a low climb-acceleration thrust to drag ratio of the nonafterburning turbojet when compared to that of the afterburning turbojet. Therefore, the nonafterburning turbojet used less cruise fuel because of the lower distance required to fly at cruise than the afterburning turbojets.

The duct-burning turbofan used 17 250 pounds (7825 kg) more fuel than did the afterburning turbojet during takeoff climb and acceleration (table IV). This large fuel difference was a result of a combination of a low climb-acceleration thrust to drag ratio (fig. 26) and low specific impulse (fig. 27) for the duct-burning turbofan when compared with that of the afterburning turbojet. During supersonic climb and acceleration, the duct-burning turbofan powered vehicle had a minimum 1.75 thrust-to-drag ratio compared with 2.55 for that with the afterburning turbojets. Consequently, the SST using the duct-burning turbofan required more time and range to reach cruise.

As shown on figure 27, the duct-burning turbofan specific impulse is higher than that of the afterburning turbojet between Mach 0.35 and 0.9. During supersonic operation (between Mach 1.0 and 3.0), the afterburning turbojet had the specific-impulse advantage. This advantage continued for cruise operation as shown on figure 28. Because the duct-burning turbofan powered SST cruised 134 nautical miles (248 km) less than did the one powered by the afterburning turbojet, the specific-impulse advantage of the afterburning turbojet was nullified. Thus, the duct-burning turbofan powered SST used 2630 pounds (1193 kg) less cruise fuel.

Letdown and reserves. - Once cruise was completed, the letdown fuel was calculated as described in the Engine Operation section. The letdown fuel difference between the nonafterburning turbojet and afterburning turbojet was insignificant. The same was true for the reserve requirements, which are listed in the Mission section. Fuel used during the hold condition at Mach 0.6 at 15 000 feet (4570 m) altitude for 30 minutes accounts for nearly half the reserve requirements. A substantial difference could occur in the reserve fuel if the subsonic specific impulse during hold conditions were not nearly the same. Figure 29 shows specific impulse at the hold condition as a function of engine thrust and type. Because the engine design parameters of the nonafterburning and afterburning turbojet with noise restrictions were the same, the same curve applied for both the turbojet engines.

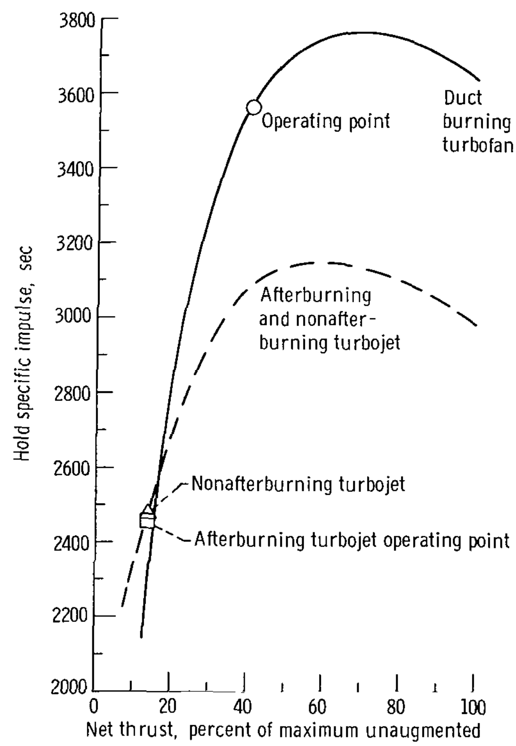


Figure 29. - Installed engine specific impulse for Mach 0.60 hold at 15 000-feet (4570 m) altitude. Design turbine-inlet temperature, 2800° F (1538° C); engines sized with noise restrictions.

The fuel used by the duct-burning turbofans for letdown with engines idling was 4550 pounds less (2064 kg) than that used by the afterburning turbojets. Figure 29 shows that, during the hold condition, the duct-burning turbofan specific impulse was 45 percent better than that of the afterburning turbojet. This accounted for the major part of the reserve fuel reduction required by the SST when using duct-burning turbofan engines.

REFERENCES

1. Whitlow, John B.; Jr.; Eisenburg, Joseph D.; and Shovlin, Michael D.: Potential of Liquid-Methane fuel for Mach 3 Commercial Supersonic Transports. NASA TN D-3471, 1966.
2. Weber, Richard J.; Dugan, James F.; and Luidens, Roger W.: Methane-Fueled Propulsion Systems. Paper No. 66-685, AIAA, June 1966.
3. Whitlow, John B., Jr.; Koenig, Robert W.; and Kraft, Gerald A.: Supersonic Transport Airport and Community Jet Noise During Takeoff and Initial Climb. NASA TM X-1452, 1967.
4. Anon.: Supersonic Transport Economic Model Ground Rules. Rep. SST 65-7 (Rev), Federal Aviation Agency, Sept. 1965.
5. Dugan, James F., Jr.: Compressor and Turbine Matching. Aerodynamic Design of Axial-Flow Compressors (revised). NASA SP-36, 1965, pp. 469-508. (Supercedes NACA TM E56B03b.)
6. Anon.: Jet Noise Prediction. Aerospace Information Rep. 876, SAE, July 10, 1965.
7. Anon.: Definitions and Procedures for Computing The Perceived Noise Level of Aircraft Noise. Aerospace Recommended Practice 865, SAE, Oct. 15, 1964.
8. Anon.: Standard Method of Estimating Direct Operating Costs of Transport Airlines. Air Transport Association of Am., Aug. 1960.
9. Hesse, Walter J.; and Mumford, Nicholas V. S., Jr.: Jet Propulsion for Aerospace Applications. Second ed., Pitman Publ. Corp., 1964, pp. 196-200.

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