https://ntrs.nasa.gov/search.jsp?R=19660020967 2020-03-24T03:08:36+00:00Z

NASA TECHNICAL NOTE

NASA TN D-3471

NASA TN D-3471

LOAN COPY: R
AFWL (WI
KIRTLAND AFI

TECH LIBRARY KAFB, NM
0130406

POTENTIAL OF LIQUID-METHANE FUEL FOR MACH 3 COMMERCIAL SUPERSONIC TRANSPORTS

by John B. Whitlow, Jr., Joseph D. Eisenberg, and Michael D. Shovlin

Lewis Research Center
Cleveland, Ohio

RECEIVED
25 JUL 1966

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • JULY 1966

TECH LIBRARY KAFB, NM
0130406

POTENTIAL OF LIQUID-METHANE FUEL FOR MACH 3 COMMERCIAL SUPERSONIC TRANSPORTS

By John B. Whitlow, Jr., Joseph D. Eisenberg, and Michael D. Shovlin

Lewis Research Center
Cleveland, Ohio

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

For sale by the Clearinghouse for Federal Scientific and Technical Information
Springfield, Virginia 22151 – Price $2.00

CONTENTS

	Page
SUMMARY	1
INTRODUCTION	1
SYMBOLS	2
METHOD OF ANALYSIS	3
Mission	4
Engines	6
Aerodynamics	8
Structural Weight	9
Fuel System	10
Cost Estimation	12
RESULTS AND DISCUSSION	13
Engine Performance	13
Engine and Wing Sizing	15
Methane Fuel System	17
Boiloff recovery	19
Subcooling and pressurization	20
Insulation	21
Ice accumulation	23
Combustion Efficiency	23
Methane Heat-Sink Capacity	23
Direct Operating Cost	26
CONCLUDING REMARKS	31
APPENDIX - PAYLOAD AND PERFORMANCE TRADE-OFFS INVOLVED IN ENGINE AND WING SIZING	33
REFERENCES	40

POTENTIAL OF LIQUID-METHANE FUEL FOR MACH 3 COMMERCIAL SUPERSONIC TRANSPORTS

by John B. Whitlow, Jr., Joseph D. Eisenberg, and Michael D. Shovlin
Lewis Research Center

SUMMARY

Methane has certain qualities that indicate it has potential for use as a fuel for a commercial supersonic transport aircraft. Compared with conventional kerosene-type JP fuels, it has a higher heat of combustion that would tend to lower specific fuel consumption. Its higher heat sink could provide better cooling of certain critical engine parts and enhance the use of higher cycle temperatures that promote higher cycle efficiencies. In addition, its availability (as natural gas) is good in many areas of the world. Its low density, however, dictates that configurations with available wing-fuel-storage volume will most fully utilize the advantages of methane.

Analysis showed that a payload improvement up to 31 percent and a direct operating cost reduction up to 36 percent might be realized with methane. These improvements in payload and direct operating cost are dependent on the control of certain problems associated with the use of cryogenic liquid fuels - namely, boiloff loss of the fuel and the weight penalty of a more complicated fuel system.

INTRODUCTION

To date, most discussion of the commercial supersonic transport (SST) airplane has been based on the assumption that conventional kerosene-type JP fuels would be used. It is possible that the use of a different type of fuel could improve the performance and/or lower the direct operating cost of such aircraft. A better fuel should have a greater heat of combustion, a greater heat-sink capacity, a reasonably high density, and be low in cost. Liquid methane, from a consideration of these factors, appears to be one of the possibilities available for the replacement of conventional fuels. It has a 13-percent higher heat of combustion than the better current JP fuels and up to seven times their heat sink capacity, but, unfortunately, in the liquid phase, it is only half as dense. Therefore, if liquid methane is to replace JP fuels for this application, it will be necessary that the advantage of its higher energy content and heat-sink capacity overshadow any volume

penalties that may exist because of its lower density. Methane, as natural gas, also appears attractive from an economic standpoint. Its actual cost per pound when delivered to an aircraft, however, will be dependent on the proximity of existing natural gas facilities and whether the traffic volume will justify methane liquefaction plants at the airport. Since liquid methane is a cryogenic, handling and storage also present problems.

A previous analysis (ref. 1) has shown that lighter hydrocarbons such as methane could improve the performance of supersonic jet fighters and bombers, especially when high-performance engines that impose a high turbine cooling load are used. Of the hydrocarbons investigated in reference 1, methane appears to be one of the most suitable for a commercial application such as the SST.

This investigation was conducted to assess the potential of liquid methane for a Mach 3 commercial supersonic transport. The approach used was to determine the improvement that might be obtained in an overall airplane figure of merit by the use of methane instead of JP fuel in an arbitrarily selected, fixed, arrow-wing configuration of high aerodynamic efficiency. The two airplane figures of merit to be examined in this comparison are payload, or number of passengers, and direct operating cost in cents per seat mile. Engine and wing sizes were varied in an effort to optimize the payload for each fuel, with consideration given to such operational constraints as takeoff velocity, lift-off distance, transonic acceleration, and sonic boom at the optimum cruise altitude.

SYMBOLS

C_L	lift coefficient
D	drag, lb
F	net thrust, lb
I_s	specific impulse, sec
k	thermal conductivity, (Btu)(in.)/(hr)(sq ft)(^oF)
L	lift, lb
S_w	wing planform area, sq ft
T	temperature, ^oF
V	velocity, ft/sec or knots
W	weight, lb
$\dot{W}_a$	sea-level-static design engine corrected airflow, lb/sec
ρ	density, lb/cu ft

Subscripts:

g	airplane gross ramp
it	turbine inlet
LO	lift-off
m	metal
sℓs	sea level static
TO	takeoff
tr	transonic

METHOD OF ANALYSIS

The comparison of liquid methane with JP fuel will be made by analysis of the results obtained from "flying" fixed-wing airplanes similar to the one shown in figure 1 over a standard mission profile. A comparison of some of the characteristics of each of the

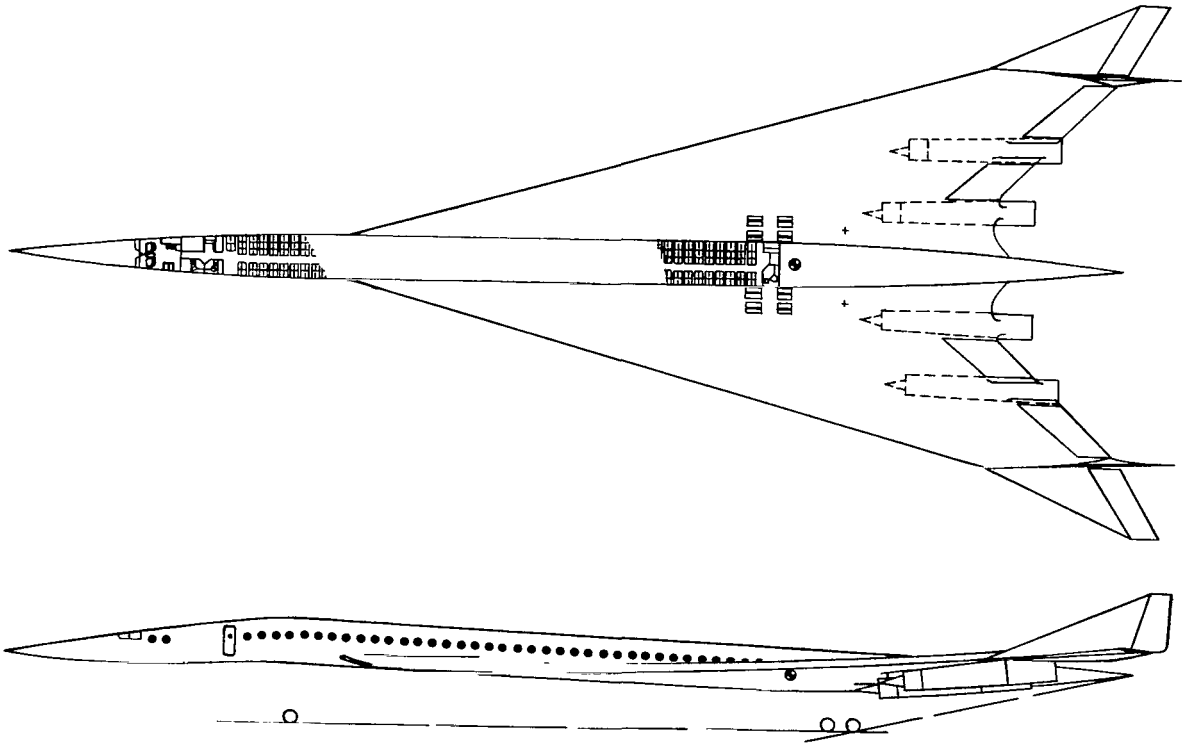


Figure 1. - Layout of 229-passenger methane supersonic transport.

TABLE I. - FUEL CHARACTERISTICS

Characteristic	JP	Methane
Lower heating value, Btu/lb	18 750	21 200
Heat sink, Btu/lb	165 to 365	1100
Density (liquid), lb/cu ft	50	26
Specific heat (liquid), Btu/(lb)(^{o}F)	0.47	0.82
Heat of vaporization (1 atm), Btu/lb	120	219
Freezing point (1 atm), ^{o}F	-85	-297
Boiling point (1 atm), ^{o}F	350	-259
Stoichiometric fuel-air ratio	0.0676	0.0581
Estimated price:		
Cents/gal	12	4.2 to 7.0
Cents/lb	1.8	1.2 to 2.0
Cents/10^{6} Btu	96	57 to 94

two fuels used is presented in table I. The airplane gross ramp weight was fixed at 460 000 pounds for the entire study. Wing and engine sizes were varied in order to optimize the airplane design. The fuselage, fixed in cross section throughout the entire study, was allowed to vary in length in order to accommodate different numbers of passengers. In some cases where very small wings were considered with methane fuel, it was necessary to extend the fuselage to contain fuel that might otherwise have been stored in the wings if they had been larger. The effects of variations in wing, fuselage, and engine size on structural weight are discussed. Also, since liquid methane is a cryogenic, it will be necessary to take into account the penalties associated with the boiloff of fuel, the weight of insulation required to keep the boiloff at an acceptable level, and any increase in the weight of tankage, pumps, and piping that might be encountered with methane. The afterburning turbojet was the only engine type considered. The results of the fuel comparison will be essentially unchanged regardless of which of the two currently competitive types of engines (i.e., the afterburning turbojet and the duct-burning turbofan) are used for the Mach 3 SST application.

Mission

The mission requirements observed in this study are outlined as follows:

Range, n. mi.	3500
Cruise Mach number	3.0
Sonic-boom overpressure limit, lb/sq ft	
Climb	2.0
Cruise	1.5

All airplanes considered were designed for Mach 3 cruise with a total range of 3500 nautical miles (4028 statute miles). The flight path in Mach number and altitude coordinates in all cases was fixed up to Mach 1, but at higher Mach numbers, the altitude was allowed to increase as necessary so that sonic-boom overpressures on the ground would not exceed 2.0 pounds per square foot. The sonic boom was calculated by the method outlined in reference 2. Consequently, all aircraft considered in the study did not necessarily fly exactly the same flight path since, even though they all had the same 460 000-pound ramp weight, the instantaneous weight on reaching the sonic threshold, the wing size, and the fuselage size, all of which affect the sonic-boom overpressure, varied from one case to another. On completion of the climb phase, cruise was begun at that altitude which produced the highest Breguet cruise factor (cruise efficiency), where

$$\text{Breguet factor} = \left(\frac{L}{D} I_s V\right)_{\text{cruise}}$$

The altitude at the start of cruise in each case, therefore, varied as a result of the Breguet factor maximization. To simplify the calculations, it was assumed that the descent time and range remained constant for all cases at 25 minutes and 400 nautical miles, respectively, with fuel consumption calculated with engines idling. A typical flight plan

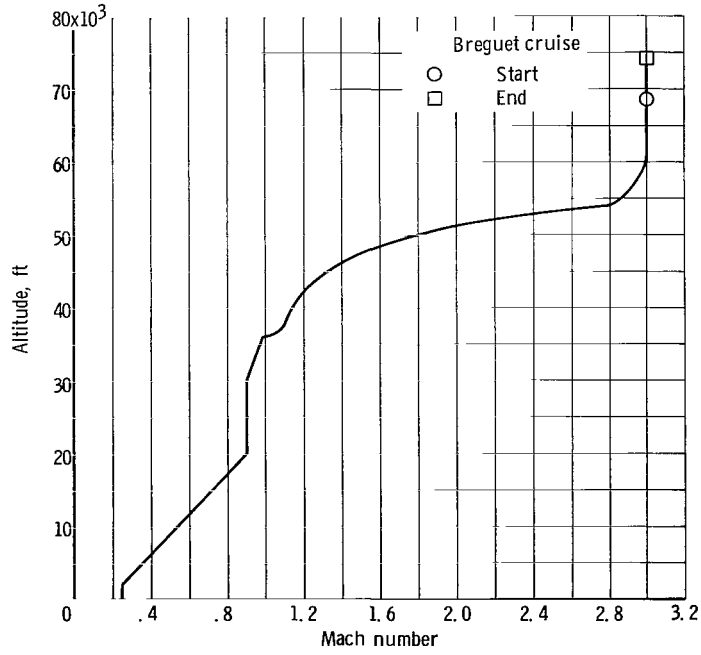


Figure 2. - Typical flight plan. Airplane gross ramp weight, 460 000 pounds; sonic-boom overpressure limit, 2.0 pounds per square foot; takeoff wing loading, 50 pounds per square foot.

for the fixed arrow-wing airplane under study in Mach number and altitude coordinates is presented in figure 2.

The reserve fuel allowance used in this study includes (1) an additional amount of fuel equal to 7 percent of the mission fuel, (2) fuel for a 261-nautical-mile cruise to an alternate airport at the supersonic cruise altitude and Mach number, and (3) fuel for a 30-minute subsonic hold at Mach 0.6 at an altitude of 15 000 feet. In addition to these reserve fuel requirements, an additional allowance was incorporated in the mission fuel for a 25-minute idle prior to takeoff as well as 1 minute of maximum "dry" (nonafterburning) power application prior to the start of takeoff roll.

Engines

The characteristics of the afterburning turbojet engines considered in the initial part of the study are summarized as follows:

Design compressor pressure ratio	10
Design turbine inlet temperature, ^{o}F	2100
Maximum compressor bleed air for turbine cooling, percent	6.6
Maximum afterburner temperature, ^{o}F	3000
Design compressor efficiency	0.875
Turbine efficiency	0.88
Primary combustor efficiency	0.98
Afterburner combustor efficiency	0.93
Inlet pressure recovery at Mach 3.0	0.850
Exhaust nozzle thrust coefficient at Mach 3.0:	
Minimum afterburning	0.977
Maximum afterburning	0.966

A single-spool afterburning turbojet with a design compressor pressure ratio of 10 and a turbine inlet temperature of 2100^{o} F was the engine used throughout the initial part of the investigation in which the added heat-sink capacity of methane was not considered. Studies outlined in reference 3 indicate that this pressure ratio would be nearly optimum for this turbine inlet temperature. The maximum afterburner temperature in all cases was 3000^{o} F. Takeoff was at the maximum dry power setting that produced a calculated noise level of 119 perceived noise decibels (PNdB) at a distance of 1500 feet from the centerline of the runway at the start of takeoff roll when the effects of engine masking and ground attenuation are considered. Afterburning was not used during takeoff since the noise produced with the maximum dry power setting was deemed to be the maximum acceptable level of airport noise. Afterburning was initiated at an altitude of 36 000 feet

where the transonic threshold is encountered and wave drag increases rapidly. Afterburner gas temperature remained at 3000° F until Mach 2.8, at which point it was gradually reduced to 2540° F at Mach 3 and 60 000 feet prior to entering the cruise mode of operation. Without this afterburner temperature reduction beginning at Mach 2.8, excessive thrust over drag ratios would have resulted, creating computational difficulties when the equations of motion are integrated. The afterburner temperature during cruise was determined as a part of the previously described search for the cruise altitude that yielded the maximum Breguet factor. For most of the engine sizes considered in the study, a small amount of afterburning was required during cruise to maximize the Breguet factor. However, for some of the larger engine sizes, no afterburning was required during cruise.

In calculating the engine design and off-design performance, the engine components were matched to satisfy the relations involving continuity of flow, engine rotational speed, and power balance between the compressor and its driving turbine. The procedures employed were similar to those described in reference 4. The mode of compressor operation was such that constant shaft speed was maintained throughout the entire flight regime except when power was reduced below the maximum dry setting.

In the initial part of the study, no consideration was given to the operating temperatures of critical engine parts such as turbine blades. The design turbine inlet temperature was fixed at 2100° F and the turbine cooling air bled from the compressor exit was assumed (as the result of a preliminary analysis) to be 6.6 percent of the total airflow. In a subsequent part of the study, the benefits that could be derived from the higher heat sink of methane were considered. These benefits might be obtained by utilizing a methane-air heat exchanger to cool the compressor discharge air used for turbine cooling. This lower temperature cooling air would permit either higher turbine inlet temperatures or greater blade life expectancy for the same turbine inlet temperature.

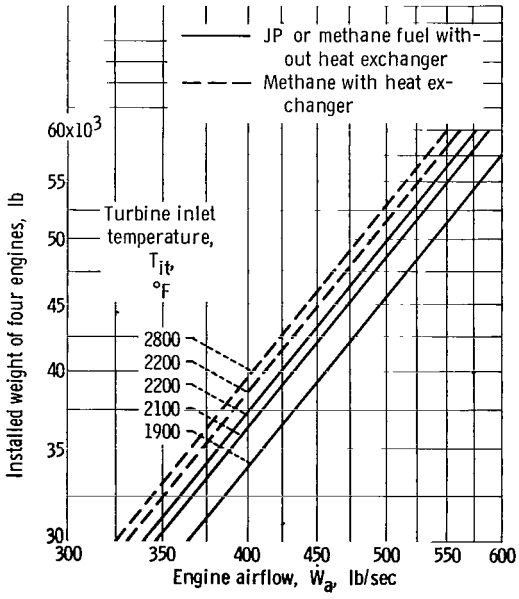


Figure 3. - Effect of engine size on propulsion system weight for afterburning turbojet.

The propulsion system weight as a function of design engine airflow was estimated from empirical data and is presented in figure 3 with turbine inlet temperature as a parameter. This weight includes the weight of the gas generator, nozzle and thrust reverser, accessories, inlet, and nacelle. It can be seen from the curves in this figure that, for engines without the methane heat exchanger, the propulsion system weight

must increase with turbine inlet temperature to increase the tolerance of certain critical engine parts to the higher temperatures encountered. Where heat exchangers are used, the weight curves employ the weight technology of the 2200° F turbine inlet temperature engine. In addition to the heat exchanger used for precooling the turbine cooling air, a methane heat exchanger is used to cool the turbine exit gases used in cooling the afterburner liner when the 2800° F turbine inlet temperature is considered. The addition of a methane-air heat exchanger increases the engine weight about 3 percent for the range of engine sizes of interest when 2200° F weight technology is used; when the afterburner-liner methane heat exchanger is added for operation at a turbine inlet temperature of 2800° F, the engine weight is increased an additional 2 percent. These curves are based on engine characteristics that give an increasing ratio of engine weight to airflow as airflow increases.

Primary combustor efficiency was assumed to remain constant at 98 percent over all operating conditions. The afterburner combustion efficiency was also assumed to remain constant at 93 percent for the basic study. According to reference 1, it is possible that higher combustor efficiencies might be obtained for methane than for JP fuels. A study was made of the sensitivity of payload to improvements in afterburner efficiency. No attempt was made to study the effect of an improvement in primary combustor efficiency, since it is likely that the level already achieved in similar combustor environments with JP fuel (98 percent) is near the maximum that can be justified.

The mixed-compression inlets were sized to capture the entire free-stream tube at the Mach 3 cruise condition. Spillage drag below Mach 3, nacelle wave and friction drag, and the boundary-layer bleed drag associated with dumping the inlet-boundary-layer control bleed overboard were taken into account in calculating installed engine performance. The nozzle considered was a variable geometry convergent-divergent ejector type. The nozzle thrust coefficient, which was adjusted to incorporate nozzle boat-tail drag, was included in the performance calculations as a function of both free-stream Mach number and engine power setting.

Based on the preceding component assumptions, the cycle performance (thrust and specific impulse) was calculated by using the thermodynamic tables of reference 5 with the additional assumption that flow is one dimensional.

Aerodynamics

Aerodynamic performance used in this study was based on data supplied by the NASA Langley Research Center for the SCAT 15F, an advanced fixed-sweep, arrow-wing SST configuration similar to the one depicted in figure 1 (p. 3).

It was necessary to modify the form of the Langley aerodynamic data in order to evaluate the effect of changes in wing, fuselage, and engine size. It was assumed that

the associated change in airplane drag was equal to the net change in the friction and wave drags of these components viewed as isolated elements. The interference lift and drag coefficients were assumed to remain constant throughout the study, even though the relative sizes of the airplane components were varied. The trends obtained with these assumptions should be satisfactory for this preliminary design study.

A representative value of the actual Mach 3 lift-drag ratio for a typical configuration, such as the one shown in figure 1, would be about 9.2. This number would vary somewhat depending on the cruise altitude chosen, the wing area, the engine size, and the fuel used. The drag incorporated in this ratio is the total drag and includes the engine drag.

Structural Weight

The structural weight of the aircraft considered during the course of the study varied, depending primarily on wing size and fuselage length. Wing area was one of the primary variables considered in the airplane design optimization study. Unpublished data from several sources were used for the determination of wing weight as a function of planform area and are presented in figure 4. This curve represents the weight of a wing for the fixed-sweep arrow-wing airplane with a ramp weight of 460 000 pounds, similar to the one shown in figure 1. The wing shown on the representative airplane of figure 1 has a planform area of 9200 square feet, as indicated in figure 4.

The fuselage length was allowed to vary during the aircraft design optimization study. Since a 34-inch seat pitch with five-abreast seating was maintained throughout the study, fuselage length varied only with the number of passengers aboard and, in some cases where the wing volume was small, the fuel volume required for the mission. Throughout the study, the maximum outside diameter of the fuselage was held constant at

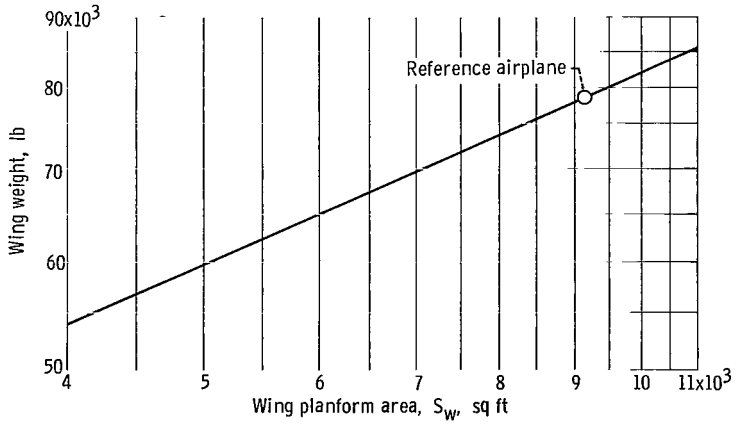


Figure 4. - Variation of wing weight with planform area.

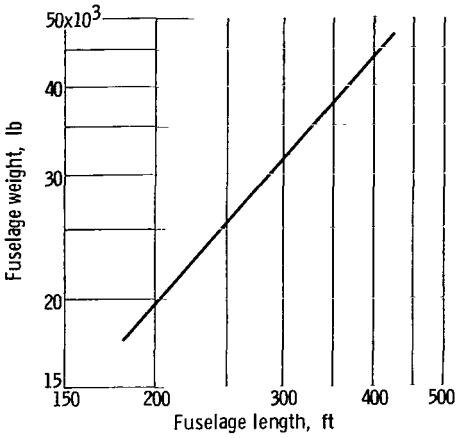


Figure 5. - Variation of fuselage weight with length.

125 inches. The fuselage weight was assumed to vary with length according to the curve presented in figure 5. The fuselage weight term does not include the weight of passenger furnishings and services, emergency equipment, air conditioning, etc., all of which are functions of the number of passengers aboard. In this analysis, each additional passenger was considered to require 116 pounds in furnishings and other equipment. This number does not include a fixed weight of equipment that would be required regardless of the number of passengers the airplane is designed to carry. The payload, a separate item, was assumed to include no cargo - only passengers and their baggage. Each passenger and his baggage was assumed to weigh 200 pounds.

No attempt has been made to discuss the calculations involved in all the component weight estimates. Only those components that were varied in size in the design optimization were considered. Errors in the weight estimates of other components (e. g., landing gear, hydraulic system, etc.,) would apply equally to all the aircraft considered and, therefore, should not influence the comparison of one airplane relative to another, even though on an absolute basis some error in overall airplane figure of merit would be indicated.

Fuel System

In comparing methane with JP-fueled airplanes, it is necessary to take a fuel-tank-insulation weight penalty and a fuel-boiloff weight penalty with methane because of the lower temperatures required to keep it in the liquid phase. The insulation thickness for methane was chosen so that the insulation weight penalty plus the boiloff penalty maximized the payload, since there is a trade-off between insulation weight and boiloff. In the basic study, the methane delivered to the airplane was subcooled 17° F to a temperature of -276° F, the boiling temperature corresponding to an assumed tank pressure of 6.2 pounds per square inch absolute, the methane tank pressurization at altitude. An additional study was conducted to determine how the boiloff rate, and hence payload, would be affected by varying the degree of subcooling received by the methane prior to its delivery to the airplane. This analysis was made for various levels of fuel tank pressurization at altitude. Subcooling was considered all the way down to the triple point of -297° F.

A study was made to estimate the minimum weight penalty associated with a methane fuel system. Factors considered were subcooling the methane loaded onto the airplane

by using a pressurant in the fuel tanks, various types of insulation, and the use of methane vapor pumps to inject boiloff into the engine combustors. The systems and tank weights (not including weight of insulation and methane boiloff recovery pumps) were assumed to be 2.09 percent of the fuel weight for either methane or JP fuel. The higher specific volume of liquid methane would not materially alter the specific tankage weight necessary to contain the two different fuels, since a large part of the tankage was already accounted for as wing or fuselage structural weight.

Since the density of methane is only about half that of JP fuel, the allocation of fuel storage locations on board the airplane is likely to present a problem with methane because of its relatively large volume requirement. The fuel storage locations for both JP and methane fuel are indicated by the shaded areas on the airplane layouts in figure 6. The two airplanes shown have wing planform areas of 9200 square feet (takeoff wing loadings of 50 lb/sq ft). The shaded area on the JP-fueled airplane layout represents the size of a typical fuel storage volume with some excess capacity. The shaded area on the methane airplane layout represents the maximum volume available for fuel storage. For an airplane of this particular wing size, this volume is approximately 15 percent greater than that actually required to accommodate all the methane fuel, including the reserves, with available wing storage volume accounting for about 48 percent of the total wing volume. The fuel storage volume available in the wing was limited because of the volume occupied by structural members, variable geometry control surfaces, fuel tank insulation, and the wheel wells. The minimum wing fuel tank depth was somewhat arbitrarily set at 18 inches. Although the maximum available volume represents about 15 percent more capacity than is needed for methane, smaller wings with the same percentage of

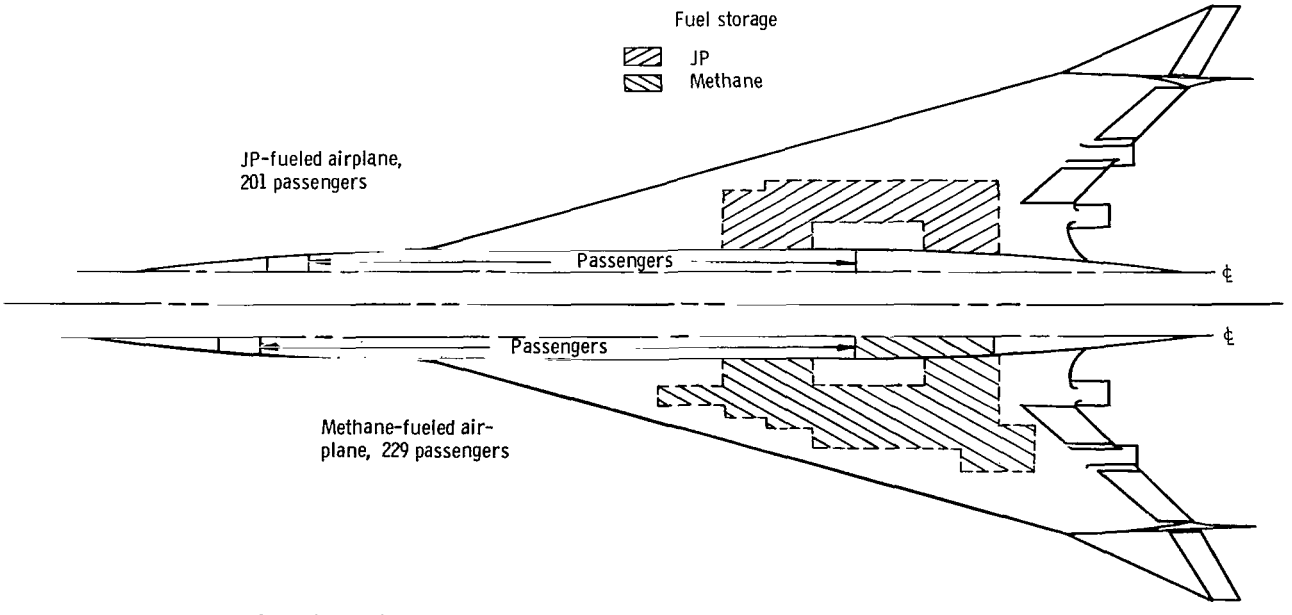


Figure 6. - Fuel-storage locations in JP- and methane-fueled supersonic transport airplanes.

wing volume available for fuel storage might not have enough fuel storage capacity. When still more fuel storage volume is needed, the fuselage is extended and all the fuel that could not be stored in the wing and fuselage tanks shown in figure 6 would be stored in full-diameter tanks in the fuselage extension. Weight and drag penalties would then be suffered. All the methane airplanes considered in the study had some fuel stored aft of the passengers in a full-diameter fuselage tank, even though some excess storage volume in the wings might have been available. An extension of the fuselage was not required because, in each airplane, there is an unused fuselage volume aft of the passengers that could be used for fuel storage if necessary. Since boiloff was a problem with methane and not with JP fuel, however, this fuselage volume was always utilized for methane storage because of its smaller tank-surface-area to volume ratio. A smaller heating load would, therefore, be imposed on the methane contained in the fuselage tank than in one of the outer wing tanks that has a higher surface-area to volume ratio.

For methane in this analysis, the insulation density and thermal conductivity were assumed to be 6 pounds per cubic foot and 0.11 Btu-inch per hour per square foot per ^{o}F, respectively. These assumptions might be satisfied by the use of silica-aerogel powder enclosed within a partially evacuated high-temperature-resistant plastic film (refs. 6 and 7). The effect of using other insulations such as fiberglass blankets and Min K-501 will be shown. Some of the properties of these insulations are presented in table II. The insulation possessing the lowest thermal conductivity - density product $k\rho$ would normally be considered the best. Fiberglass is best from this standpoint, but it cannot take compressive loads and must be sealed from the fuel if its insulating properties are to be retained. A cover to seal the insulation and take the compressive load exerted by the fuel would involve an additional weight penalty that would make the evacuated-powder-insulation concept seem more attractive in spite of its higher $k\rho$ product. The evacuated powder can take the compressive load of the fuel and is sealed from the fuel by the plastic bag. Installation of this material, however, may present problems of such a degree that it may be necessary to use the fiberglass with metal cover in spite of the greater weight penalty involved.

TABLE II. - INSULATION CHARACTERISTICS

Insulation	Density, lb/cu ft	Thermal conductivity, (Btu)(in.)/(hr)(sq ft)(^{o}F)	
		-80^{o} F	120^{o} F
Fiberglass	3	0.18	0.26
Evacuated powder	6	.11	.17
Min K-501	10	.10	.13

Cost Estimation

Direct operating cost is probably a better airplane figure of merit than payload, but the 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

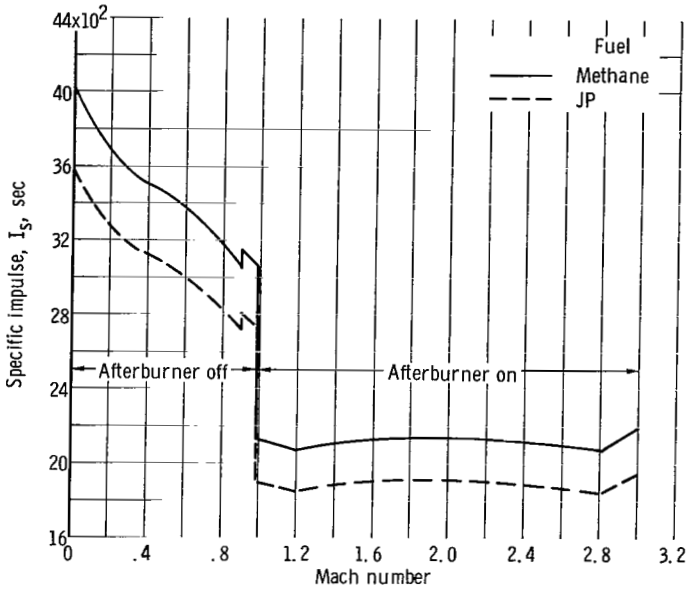
of the airframe weight, and the engine price was assumed to be a function of the engine size (design airflow). Since the methane fuel airframes were in all cases heavier than the corresponding JP airframes, because of longer fuselages and extra fuel tank insulation, a higher airframe price was used in the direct operating cost calculations for methane. Since airplane price is somewhat uncertain at the present time, the influence this parameter has on the direct operating cost of a methane airplane will be shown. The price of liquid methane can vary over a considerable range depending on availability of this fuel at the particular location under consideration. It is expected that a typical price for liquid methane delivered to the airplane in the United States might be about 4.2 cents per gallon (1.2 cents/lb), whereas in England it might be about 7 cents per gallon (2.0 cents/lb). The effect that a variation of methane price has on direct operating cost will be presented. Although a JP fuel price of 12 cents per gallon was used in most of the calculations (as per Federal Aviation Agency SST economic model ground rules), the effect of a price variation in this fuel will also be shown.

Airframe prices for the direct operating cost calculations were estimated to be $118 per pound, development costs included, based on a production of 200 aircraft. In addition, a price of $1 million for electronics was included in the airplane price calculations. 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 selling price per engine, including development costs, was estimated to be $2780 per pound per second of design sea-level static airflow. Thus the selling price of a 470-pound-per-second engine would be about $1.33 million. Unless otherwise indicated, the time between engine overhaul was assumed to be 2000 hours. The direct operating cost calculations were performed in the manner described in reference 8.

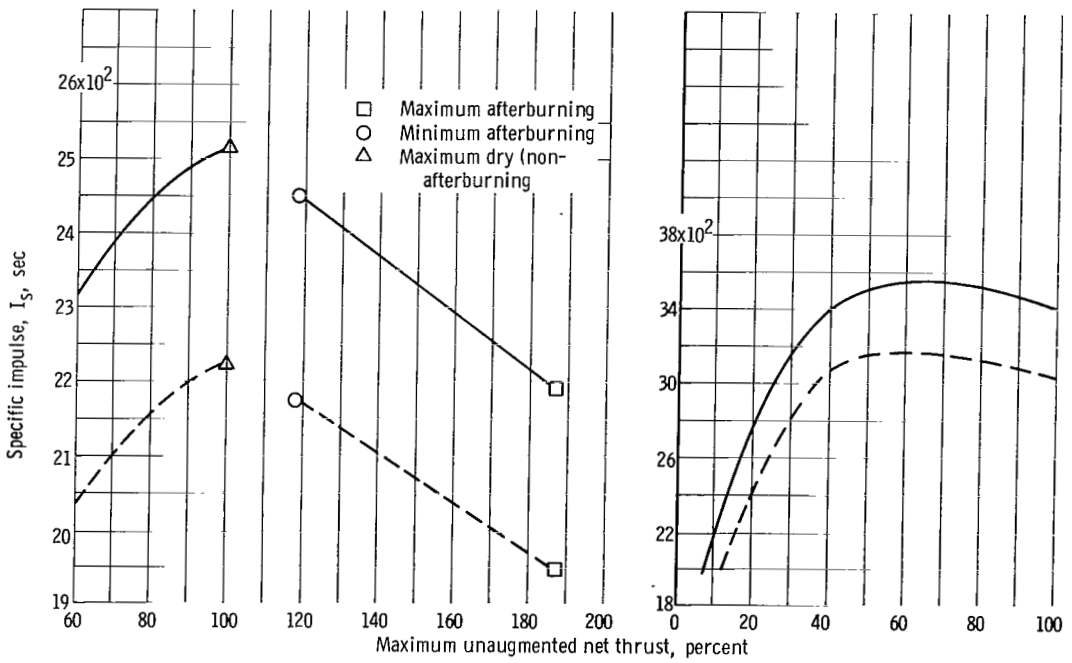
RESULTS AND DISCUSSION

Engine Performance

An important difference between methane and JP fuels is the heating value, 21 200 Btu per pound for liquid methane compared with 18 750 Btu per pound for some of the better current JP fuels. This has a significant effect on the engine specific impulse, as shown in figure 7 for the case where the design turbine inlet temperature is 2100^{o} F and the bleed airflow for turbine cooling is 6.6 percent. The performance shown represents "installed" engine performance which is the performance that has been adjusted to include thrust degradation resulting from inlet spillage drag, nacelle wave and friction drag, inlet boundary-layer bleed drag, internal nozzle performance, and external nozzle boat-tail drag. Figure 7(a) presents the engine specific impulse for the mode of engine



(a) Acceleration and climb (flight plan shown in fig. 2).



(b) Mach 3.0 cruise.

(c) Mach 0.6 hold at 15 000-foot altitude.

Figure 7. - Installed engine performance at 2100° F design turbine inlet temperature.

operation that is used during takeoff, acceleration, and climb. The engines operate without afterburning until Mach 1 at 36 000 feet is reached. Afterburning at 3000° F was begun at Mach 1 and continued until Mach 2.8, where the temperature was gradually reduced to 2540° F at Mach 3 and held constant until the cruise altitude was reached.

At Mach 3 cruise, the afterburner temperature was reduced below the level used for climb until the thrust level required for the altitude that maximized the Breguet factor was reached. Maximization of the Breguet factor, which is a function not only of the specific impulse but also of the lift-drag ratio L/D at any given velocity, minimizes the fuel consumed over a given cruise range. The part-power Mach 3 cruise specific impulse against net thrust is shown in figure 7(b). The peak specific impulse occurs at the maximum dry power setting with the turbine inlet temperature at the design value of 2100° F. Points on the curves to the left of the peaks represent conditions of lower (off-design) turbine inlet temperatures, still without afterburning. Points on the curves to the right of the peaks, going from left to right, represent conditions of design turbine inlet temperature with increasing afterburner temperatures.

Part-power hold specific impulse against net thrust is shown in figure 7(c) for the 30-minute hold condition at Mach 0.6 and an altitude of 15 000 feet. The curves represent conditions of no afterburning with turbine inlet temperature being reduced to obtain lower levels of thrust.

Another difference between methane and JP performance is that, for the same combustion temperature, methane will produce 1 to 3 percent more thrust per unit engine airflow than JP fuel over the range of operating conditions considered. At a given combustion temperature, the specific enthalpy of the products of combustion (as given by the thermodynamic tables of ref. 5) is greater for methane than for JP fuel. This causes a greater jet velocity at the nozzle exit for the methane products, and thus produces a higher specific thrust.

Engine and Wing Sizing

In this analysis, both wing planform area and engine size were varied in order to maximize airplane payload, within the constraint of various operational limitations. As shown in the appendix, for a JP-fueled airplane, a takeoff wing loading of 73 pounds per square foot and a takeoff thrust to gross-weight ratio of 0.26 yield the maximum payload. However, as is also shown, it is takeoff performance that actually sizes the aircraft, requiring that larger wings and engines be used than the combination which would maximize payload. A 50-pound-per-square-foot wing loading and a 0.32 thrust to gross-weight ratio at takeoff were required (at some sacrifice in payload) to reduce takeoff velocity and distance requirements to levels approximating those obtained by current sub-

sonic jet transports. Takeoff performance also sized the methane-fueled airplane, resulting in the same wing loading and thrust to gross-weight ratio (i. e., 50 lb/sq ft and 0. 26, respectively). These parameters correspond to a wing planform area of 9200 square feet and an engine size of 470 pounds per second for JP juel and 463 pounds per second for methane. A payload of 201 passengers was obtained with JP fuel, while the use of methane provided a payload improvement of 14 percent that yielded a 229-passenger total. The airplane layouts for these two design airplanes are compared in figure 6 (p. 11). A comparison of the weights, sizes, and dimensions for the two design

TABLE III. - AIRCRAFT CHARACTERISTICS

Weight summary, lb	Fuel	
	JP	Methane
Fuel including boiloff:		
Takeoff and climb	60 364	56 876
Cruise	101 824	94 696
Letdown	5 223	5 245
Reserves	34 464	31 731
Total fuel	201 875	188 548
Wing and vertical stabilizer	82 879	82 879
Fuselage	24 937	26 780
Installed engines	44 641	43 874
Fuel system, including insulation	4 219	7 466
Landing gear	20 056	20 056
Hydraulic and electrical systems	6 831	6 831
Surface controls	4 011	4 100
Passengers and baggage	40 200	45 800
Furnishings, electronics, passenger services, crew, crew baggage, emergency equipment, air conditioning, etc.	30 351	33 666
Total (ramp weight)	460 000	460 000
Sizes and dimensions	**JP**	**Methane**
Wing planform area, sq ft	9200	9200
Aspect ratio	1. 71	1. 71
Fuselage outside diameter (maximum), in.	125	125
Fuselage length, ft	244	259
Seat pitch, in.	34	34
Number of seats abreast	5	5
Engine design corrected airflow, lb/sec	470	463
Turbine inlet temperature, ^{o}F	2100	2100

airplanes is presented in table III. In all the methane-airplane data presented so far, it was assumed that the methane delivered to the airplane was subcooled 17° to -276° F and was pressurized to 6.2 pounds per square inch absolute at altitude. A total of 3904 pounds of methane boiled off over the entire mission. The fuel system weight penalty incorporated in the design methane airplane was 3526 pounds.

It is possible that significantly different aerodynamic, structural, or engine characteristics might allow the design of an airplane with a more highly loaded wing. In order to determine the benefit of using methane instead of JP fuel for different takeoff thrust and wing loadings, figure 8 is presented. This figure was obtained by subtracting the number of passengers of JP in figure 21(a) (p. 34) from the number of passengers for methane in figure 21(c) (p. 38). Figure 8, therefore, is strictly valid only for fixed-sweep airplanes powered by 2100° F engines. It shows that, with higher wing loadings, less benefit can be obtained from methane than is possible at the lower wing loadings, because of the volume limitation of the low density methane. A sharp cusp occurs in the curves of figure 8 at a wing loading of about 60 pounds per square foot, where the methane volume limitation begins to penalize the airplane payload because of the fuselage extension necessary to accommodate all the methane that could not be stored in the smaller, more highly loaded wings. The figure shows that, as the wing loading increases beyond about 90 pounds per square foot, depending to a certain extent on engine size, JP fuel has the payload advantage over methane. It is important to realize, however, that these particular results are sensitive to the assumptions of the present study. For example, the elimination of methane boiloff losses could shift the curves of figure 8 upward. Different assumptions regarding available wing and fuselage storage volumes or the variation of fuselage weight with length could affect both the slope and level of the curves. Nevertheless, it may be concluded that aircraft with high wing loadings are less likely to benefit from the use of methane than are the low-wing-loading configurations.

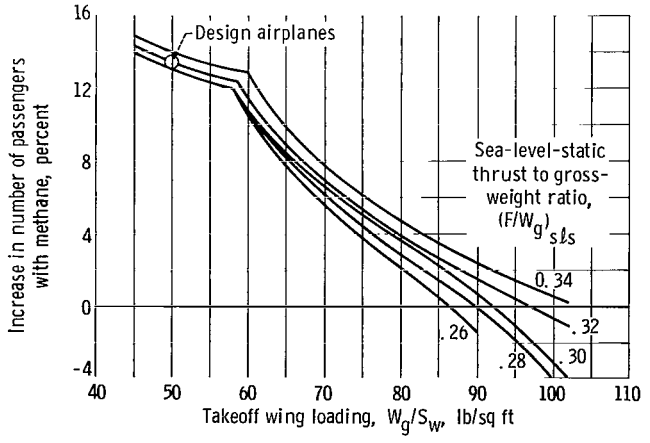


Figure 8. - Comparison of methane with JP payloads. Airplane gross ramp weight, 460 000 pounds; turbine inlet temperature, 2100° F.

Methane Fuel System

Methane is a cryogenic liquid and hence tends to boil off. Use of insulation will restrict but not eliminate boil-

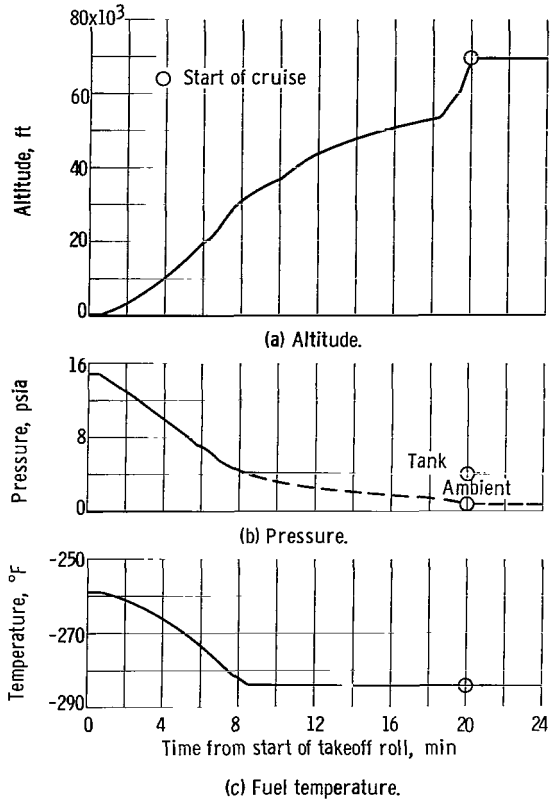


Figure 9. - Climb schedule for methane (saturated liquid).

off. The problem is most acute during parts of the climb and acceleration regime where ambient pressure changes result in the most rapid rate of change in saturated-liquid-fuel temperature. With fuel tanks vented to the atmosphere to avoid excessive bursting pressures, the reduction in pressure with altitude results in a boiloff of the liquid until its temperature is reduced to the corresponding saturation temperature. The problem is not acute in cruise despite high skin temperatures because fuel tank pressure is regulated to be constant, and the boiloff that does occur results only from heating, which is relatively insignificant because of the thermal insulation surrounding the fuel tanks.

A climb schedule showing liquid-methane tank pressure, ambient pressure, and liquid-fuel temperature is presented in figure 9 for a case where there is no initial subcooling of the fuel; that is, it is in the saturated liquid state. The climb rate shown is the result of flying the Mach number against altitude schedule shown in figure 2 (p. 5). It is seen from figure 9(a) that the resultant ambient pressure against time schedule up to about 7 minutes is approximately linear. The corresponding temperatures of the saturated liquid fuel are shown in figure 9(c). Since boiloff rate is proportional to the rate of change of liquid-fuel temperature with respect to time, the boiloff rate will be greatest at 6 to 7. 5 minutes from the start of takeoff roll. Tank pressure was held constant at 4 pounds per square inch absolute at altitudes above 32 000 feet, where boiloff is a result of skin friction heating.

The problem may be more fully appreciated by referring to the boiloff-rate curve presented in figure 10. The total boiloff without subcooling resulting from this schedule amounts to 19 109 pounds, about 9. 6 percent of the total fuel load. If boiloff could be eliminated without any fuel-system weight penalty, the payload could be increased by 34 passengers, or almost 17 percent. For the boiloff rate shown, the use of methane results in only a two-passenger improvement over JP.

Boiloff could be reduced in any of several ways, as follows: (1) recovering and burning in the engines that vapor which would otherwise be vented overboard, (2) subcooling the liquid methane prior to delivery to the airplane, (3) increasing the tank pressurization, and/or (4) changing the initial climb schedule.

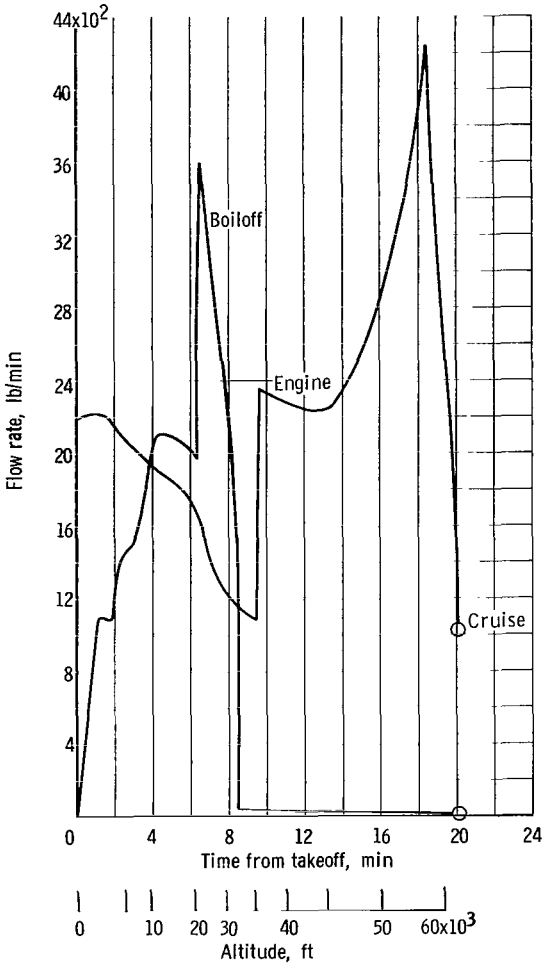


Figure 10. - Engine fuel flow and boiloff rates with no initial subcooling. Turbine inlet temperature, 2100° F; pressurization at altitude, 4 pounds per square inch absolute.

Boiloff recovery. - An attractive alternative to venting the boiloff overboard seems to be to use some of this vapor to supplement the flow of liquid methane to the engines. As seen in figure 10, during most of the flight regime, the fuel-consumption rate of the engines exceeds the boiloff rate of the fuel, even when the fuel is not initially subcooled. The boiloff rate exceeds the fuel consumption rate of the engines only for a period of about 4. 5 minutes during the climb from 10 000 to 32 000 feet altitude, resulting in an excess of about 2000 pounds of boiloff during this interval. It is conceivable that one or more vapor pumps could be added to the methane fuel system for the purpose of compressing the boiloff to the combustor pressure so that it could be burned by the engines; it would be necessary to vent overboard only the 2000-pound excess boiloff. A saving of about 17 000 pounds of fuel would then be effected. Even the 2000-pound excess boiloff might be recoverable if valves to the vents were controlled to allow a buildup in tank gage pressure as the airplane climbs in altitude. This buildup in gage pressure could permit a reduction in the rate at which tank absolute pressure (and hence saturation temperature) would decrease with respect to time. Since the boiloff rate is proportional to the rate of temperature reduction, it could be reduced to a level below the rate of engine fuel consumption during the critical 4. 5-minute time interval. The peak in the boiloff-rate curve might also be eliminated by a rescheduling of the subsonic portion of the climb so that a constant slope of the curve for fuel saturation temperature against time is obtained up to the time when the tank pressure is held constant (8. 5 min and 32 000 ft). This might be accomplished by a more rapid climb to 10 000 feet and a slower rate from 10 000 to 32 000 feet.

Figure 10 indicates that engine fuel consumption more than doubles at 36 000 feet where afterburning is initiated. For the ground rules of this study when boiloff is not a consideration, the optimum time for initiation of afterburning occurs at altitudes between 34 000 and 36 000 feet. If afterburning is begun at altitudes below 34 000 feet, payload

suffers. When boiloff is considered, it might appear that a net increase in payload would be obtained by the earlier initiation of afterburning, which would increase the engine-fuel-consumption rate at the lower altitudes where the peak in boiloff rate occurred. If the boiloff rate schedule did not change, the excess boiloff would be less, but, nevertheless, burning fuel in the afterburner below 34 000 feet would be wasteful. Acceleration would increase, and a small time saving would occur, but the payload would suffer whether the fuel is vented overboard as "unrecoverable" boiloff or is burned in the afterburner at altitudes below 34 000 feet. Furthermore, when afterburning is initiated earlier in the flight, the rate of climb is increased, thereby increasing the rate of boiloff, which is primarily a function of the rate of change of liquid-fuel temperature. When climb path and tank pressurization are rescheduled for a better match of boiloff rate with engine fuel rate, the altitude at which afterburning should be initiated to maximize payload is not expected to be significantly below 34 000 feet.

Subcooling and pressurization. - Figure 11 illustrates the fact that payload can be increased by initially subcooling the liquid-methane fuel or by increasing the level of tank pressurization to minimize boiloff. It was assumed in this figure that tankage weight did not vary with the level of pressurization, but in the actual case it is almost certain that a weight penalty will have to be taken if the pressurization is much above 6 pounds per square inch absolute. Points to the left of the break in each of the three curves, going from left to right, represent decreasing boiloff with constant insulation weight, whereas points to the right of the break represent conditions of no boiloff with insulation weight decreasing. The figure shows that, to eliminate boiloff entirely if an altitude tank pressurization of 4 pounds per square inch absolute is maintained, the fuel would have to

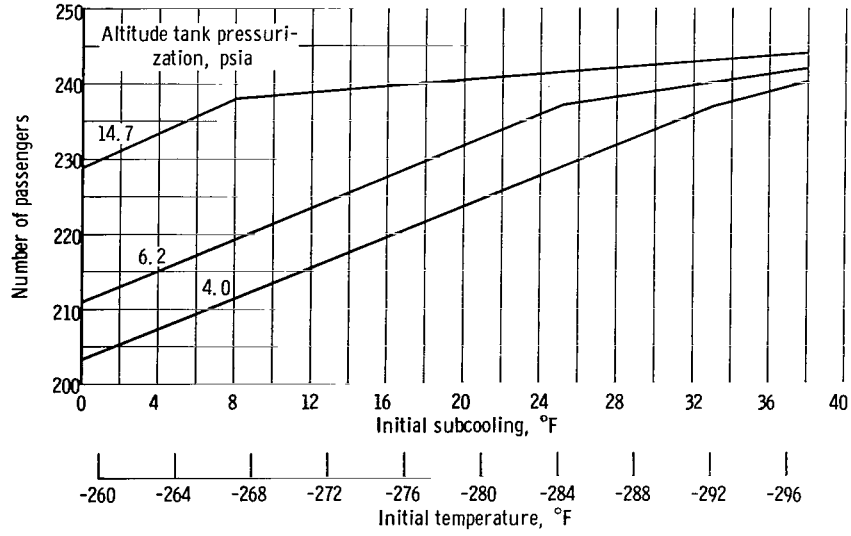


Figure 11. - Effect of altitude pressurization and degree of initial subcooling on payload. Tankage weight assumed not to vary with pressurization. Fuel, methane; takeoff wing loading, 50 pounds per square foot; sea-level-static thrust to gross-weight ratio, 0.32.

be initially subcooled 33° to -292° F. If it were possible to pressurize the tanks at 14.7 pounds per square inch absolute over the entire mission, boiloff could be eliminated with only 8° F subcooling. The boiloff that would occur with no subcooling when the tank pressure is held constant at 14.7 pounds per square inch absolute throughout the entire flight is the result only of atmospheric frictional heating. Raising the minimum tank pressure reduces the total boiloff but does not affect the boiloff rate at altitudes below the pressurization altitude. Subcooling is limited to a maximum of 38° F, corresponding to the temperature where methane solidifies at normal atmospheric pressure. Subcooling to this point could yield a maximum improvement in payload of about 18 percent, for the range of tank pressurization shown in the figure.

Subcooling the liquid-methane fuel presents a handling problem that may be difficult to solve. Since the fuel tanks on board the airplane will not be completely filled with subcooled methane, the unfilled space must be purged of air and refilled with some inert pressurizing gas that is only slightly soluble. The air must be purged from the tanks because its constituents are soluble in subcooled methane. This problem does not exist with the saturated liquid, since the solubility of gases is directly proportional to the partial pressure of the solute gas, and thus would be zero. Nitrogen, the first pressurizing gas that might be considered, is no good because of its high solubility (about 5 percent by weight) in liquid methane subcooled 17° F. With this degree of solubility, it is possible that about 10 000 pounds of nitrogen could be dissolved in the fuel at takeoff, thereby creating an intolerable weight penalty. Helium is a possibility as a pressurant because its solubility is very low, but its availability is currently very low. Its scarcity is at least partly the result of a lack of demand; much helium is wasted each year because no large-scale attempt has been made to recover it from natural gas of which it is sometimes a minor constituent. Warm methane gas is another possibility as a pressurant, but mixing and the subsequent heat transfer from the warm gas to the subcooled liquid fuel might result in the condensation of some of the gas, thereby resulting in a rapid reduction in the tank pressure. Hydrogen gas might also be used as a pressurant, but it is difficult to contain and could present a safety hazard. Its presence might also cause embrittlement of the titanium structure. Of the possibilities mentioned, helium would be the most satisfactory if its scarcity were not a problem. If a flexible membrane could be devised to separate the liquid fuel from the pressurizing gas, air (with its moisture and carbon dioxide removed) should be a satisfactory pressurant.

Insulation. - It has been assumed so far in the discussion that the fuel-tank insulation has a thermal conductivity of 0.11 Btu-inch per hour per square foot per °F and a density of 6 pounds per cubic foot. These assumptions might be satisfied by the use of silica-aerogel powder enclosed within a partly evacuated plastic bag. For the basic methane case with 17° F initial subcooling, the weight of this insulation that maximized payload in the trade-off with boiloff was 3526 pounds. Some difficulty might be encoun-

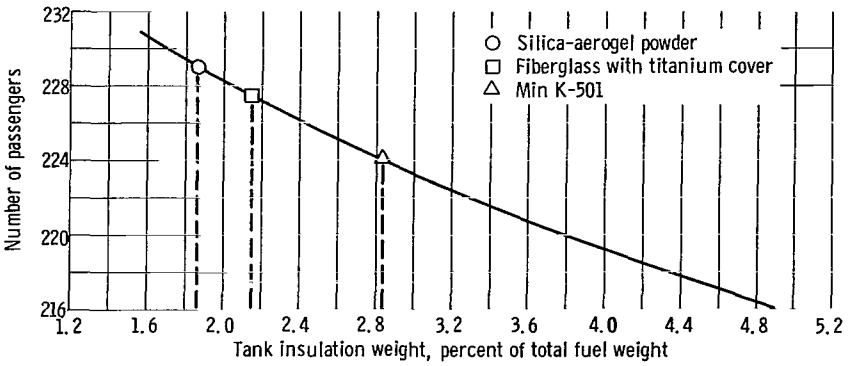


Figure 12. - Effect of methane tankage insulation weight with constant heat flux. Airplane gross ramp weight, 460 000 pounds; takeoff wing loading, 50 pounds per square foot; sea-level-static thrust to gross-weight ratio, 0.32.

tered in the installation of this type of insulation in the airplane. Since a bulk insulation such as silica-aerogel powder would tend to settle within the evacuated bag during usage, it would be necessary to vibrate the bag as it is filled to prevent this occurrence. Bonding the plastic bags to the inside surface of the fuel tanks could also present problems.

Because of these problems with silica-aerogel powder, there is the possibility that alternative insulations might have to be used instead. Figure 12 shows the effect that a change in insulation weight would have on payload when heat flux is held constant. The circle represents the weight of the silica-aerogel powder insulation that produced the 229-passenger payload of the basic study. By considering the insulation properties presented in table II (p. 12), it has been estimated that 2860 pounds of fiberglass insulation (in blanket form) would offer a thermal barrier equivalent to that offered by 3526 pounds of the silica-aerogel powder within the evacuated plastic bags. However, the fiberglass would not be able to take the compressive load of the fuel and, in addition, would have to be sealed from the fuel to protect its insulating qualities. These problems could probably be overcome by the addition of a 0.010-inch-thick titanium cover weighing about 1200 pounds. The resultant fuel system weight penalty would then be 4060 pounds, an increase of 16 percent over the 3526-pound penalty with the evacuated powder insulation. By referring to figure 12, it may be seen that this additional weight penalty causes the payload to drop from 229 to 227 passengers, a reduction of 0.87 percent. Another type of insulation that might be considered for this application is Min K-501. It is estimated that 5340 pounds of this material would provide the same thermal barrier as 3526 pounds of silica-aerogel powder. Figure 12 shows that this additional weight penalty would cause a 2.18-percent decrease in payload (i.e., a reduction of 5 passengers).

The conclusion that should be drawn from the foregoing discussion is that it makes very little difference which of the three types of insulation is selected as far as the effect on payload is concerned. Difficulty of installation and reliability would appear to be the main criteria involved in the selection of the most desirable type.

Ice accumulation. - Ice accumulation on the wings may be a problem during ground hold. One solution to this problem would be to provide nichrome heating wires on the inner surface of the wing skin, but this would create an additional weight penalty of about 300 pounds. An alternative solution might be the use of removable insulating blankets or radiant heat lamps during ground hold. Greatest operational flexibility would be provided by the use of the nichrome heating wires, since they could also be used during the taxi and wait at the end of the runway prior to takeoff.

Combustion Efficiency

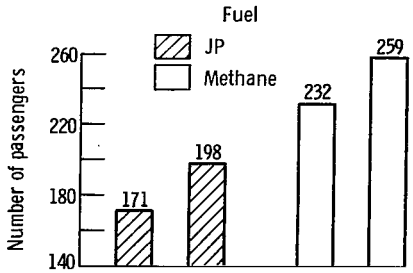
At severe combustion conditions, reference 1 suggests that it might be possible to obtain higher burner efficiencies with methane than with JP, even though this possibility has not been considered in any of the data presented to this point herein. Therefore, a study was made of the sensitivity of the methane airplane payload to perturbations in burner efficiency.

Higher combustion efficiency can usually be obtained at some penalty in total pressure drop, burner weight, or control complexity. If these factors are balanced to result in an afterburner combustion efficiency of 93 percent using JP fuel, it is possible that a change to methane fuel could increase the afterburner efficiency without compromising the other factors. The potential for improvement is less in the main burner where a balancing of all the factors resulted in the higher efficiency level of 98 percent. Therefore, only the afterburner combustion efficiency was increased in the sensitivity study. When this efficiency was increased from 93 to 95 percent, approximately twice the improvement suggested by reference 1, the payload of the methane airplane increased by only one passenger, hardly a significant improvement.

Methane Heat-Sink Capacity

In the preceding results, the advantage of the higher heat-sink capacity of methane has not been considered. In the following discussion, the higher heat-sink capacity of methane will be utilized by providing additional cooling for the turbine blades. The assumption is made that boiloff loss has been eliminated and that the fuel system weight penalty (extra insulation, plumbing, etc.,) is 1.87 percent of the total fuel weight, which is equivalent to a weight penalty of about 3500 pounds, relative to the fuel system weight of a JP-fueled airplane.

Current subsonic jet transports cruise with turbine inlet temperatures of 1450° to 1500° F, which are low enough so that turbine blade cooling is not required. The blades,



Turbine inlet temperature, T_{it}, °F	1900	2200	2200	2800
Blade metal temperature, T_m, °F	1570	1740	1510	1740
Coolant, percent	10	10	10	12
Turbine efficiency, percent	86.7	86.7	86.7	86.0
Weight of heat exchanger, lb	0	0	1300	1700
Engine airflow, $\dot{W}_a$, lb/sec	535	461	455	371

Figure 13. - Benefits from methane cooling capacity with no boiloff loss. Takeoff wing loading, 50 pounds per square foot; sea-level-static thrust to gross-weight ratio, 0.32.

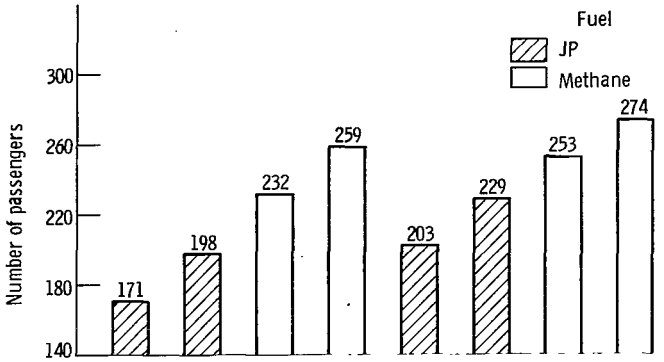
then, reach approximately the same level of temperature as the gas. However, for good SST performance, a much higher level of turbine inlet temperature is desired. Present design goals are of the order of 2100° to 2300° F, which are far beyond permissible metal temperature limits so that cooled blades are required. With a typical bleed of 10-percent compressor-discharge air for cooling, the blade temperature for a 2200° F gas temperature is calculated to be about 1740° F. Since this blade temperature is about 250° to 300° F above current levels in commercial subsonic jets, there is reason to fear that adequate reliability and life expectancy will not be obtained at the higher temperatures. If a more conservative extension of blade technology is envisioned, for example, an approximate 100° increase to 1570° F, the gas temperature at the turbine inlet will be limited to about 1900° F. In this event, the payload of the basic SST will be only 171 passengers (as indicated in fig. 13) instead of 198 passengers, the payload obtained with a turbine inlet temperature of 2200° F. Such a reduction in turbine inlet temperature and payload need not be applied to the corresponding methane airplane, however, if use can be made of its excellent heat-sink capacity. If a methane-air heat exchanger is inserted into the turbine cooling circuit to precool the air bled to the turbine, the blade metal temperature will be reduced to 1510° F, which is about the level currently attained in subsonic jets. As may be seen in figure 13, the payload obtained with this 2200° F methane engine is 232 passengers. This is 61 passengers (36 percent) more than the payload that could be carried with 1900° F JP engines with roughly equivalent blade temperature. If it is possible to achieve the level of turbine-blade technology that would permit a turbine inlet temperature of 2200° F with JP fuel, it is estimated that the turbine inlet temperature of a methane engine (incorporating heat exchangers) could be increased to about 2800° F. (In estimating the performance of this ultra-high-temperature engine, not only were the weight penalties of fig. 3, p. 7, included, but the bleed was increased to 12 percent with an associated drop in turbine efficiency from 86.7 to 86.0 percent.) As shown in figure 13, a payload of 259 passengers is obtained with these engines, 61 passengers (31 percent) more than the 198 passengers that could be obtained with the 2200° F JP engines. In earlier sections where turbine inlet temperature was held constant, the higher heating value of methane provided an 18-percent improvement in payload when boiloff was eliminated. It is now seen that when full advantage is taken of heat-sink capacity and

engines of equal metal temperature are compared, a further benefit of about 13 percent in payload can be realized.

This additional 13-percent improvement in payload obtained by increasing the turbine inlet temperature to 2800° F resulted not from an improvement in specific impulse, but rather from a reduction in engine weight. To obtain the same level of thrust at takeoff, the critical engine sizing criterion, the engine design airflow could be reduced from 455 to 371 pounds per second when the turbine inlet temperature of the methane engines was increased from 2200° to 2800° F. Even though an additional weight penalty of 100 pounds per engine was incorporated in the 2800° F engines for the afterburner-liner heat exchanger, the smaller airflow requirement reduced the engine weight 9944 pounds (21.7 percent) for four installed engines. The total fuel requirement was reduced by only 400 pounds by the increase in turbine inlet temperature. It is possible that a higher design compressor pressure ratio or a different mode of compressor operation would yield higher specific impulses, and thus reduce the fuel requirement, at the 2800° F turbine inlet temperature. The design pressure ratio and mode of operation of the compressor remained unchanged throughout the entire study, regardless of the design turbine inlet temperature.

The overall benefit of 31 to 36 percent in payload shown as the result of using methane instead of JP is based on takeoff wing and thrust loadings of 50 pounds per square foot and 0.32, respectively. As discussed in the appendix, however, improvements in aerodynamics or the easing of takeoff performance goals might permit higher wing loadings and lower thrust loadings, which would tend to reduce the advantage of using methane. If it becomes possible to operate the JP- and methane-fueled airplanes at their optimum wing loadings of 80 and 60 pounds per square foot, respectively, (per figs. 21(a) and (c), pp. 34 and 38) and thrust loadings of 0.30, the payloads are changed to those indicated in the right half of figure 14. Overall advantage for equal turbine metal temperature is cut to 20 to 25 percent, still a very substantial improvement.

The utilization of the methane heat sink permits higher turbine inlet temperatures that allow smaller and thus lighter engines. Similar results would have been obtained by afterburning during takeoff. The gain in payload would not have been as much, however, since greater cycle efficiency is achieved when the maximum possible amount of burning is accomplished at the higher level of pressure in the primary combustor. Since it was deemed advisable to takeoff without afterburning to minimize takeoff noise, similar ground rules should be observed when considering higher turbine inlet temperatures. For example, increasing the turbine inlet temperature of the methane engine from 2200° to 2800° F, and at the same time resizing the engine, causes the calculated sideline noise 1500 feet from the airplane to increase from 120 to 122 PNdB. For this 2-PNdB penalty in engine noise, a greater engine weight saving could be effected by afterburning, since the 100-pound-per-engine weight penalty of the afterburner-liner heat exchanger would



Turbine inlet temperature, T_{jt}, °F	1900	2200	2200	2800	1900	2200	2200	2800
Takeoff wing loading, W_g/S_w, lb/sq ft	50	50	50	50	80	80	60	60
Sea-level-static thrust to gross-weight ratio, $(F/W_g)_{s\ell s}$	0.32	0.32	0.32	0.32	0.30	0.30	0.30	0.30
Engine airflow, $\dot{W}_a$, lb/sec	535	461	455	371	502	432	426	348

Figure 14. - Effect of wing and engine size with no boiloff loss.

not be required with the lower turbine inlet temperature. Transonic thrust margin would suffer, however, provided that the same 3000° F maximum afterburner temperature limitation were observed in both cases. Engine fuel consumption would also suffer subsonically with afterburning, since specific impulses would be lower than at the higher turbine inlet temperature without afterburning. Supersonic cruise fuel consumption would also suffer because a greater amount of afterburning would be required to obtain the required thrust.

It is possible, then, that takeoff noise restrictions may restrict turbine inlet temperatures to levels below 2800° F, even though blade technology and/or methane heat exchangers might permit higher temperatures. The higher heat-sink capacity of methane in that event could be used to provide additional cooling, thereby increasing the stress-rupture blade life expectancy by a multiple of approximately 100 relative to a JP-fueled engine of the same turbine inlet temperature, based on a typical blade material like Udimet 700. Hence, although the heat-sink capacity of methane would not be utilized in this case to increase payload, the increased blade life expectancy might permit a greater time between engine overhaul, which would result in a lower direct operating cost.

Direct Operating Cost

Direct operating cost is a better airplane figure of merit than payload, but it is more subject to uncertainties in calculation since fuel, airframe, and engine prices all must be known. Figure 15 shows a comparison of the direct operating cost that might be expected between the 2100° F JP- and methane-fueled aircraft considered in the basic study and previously compared in table III (p. 16). The 201-passenger JP airplane would have a

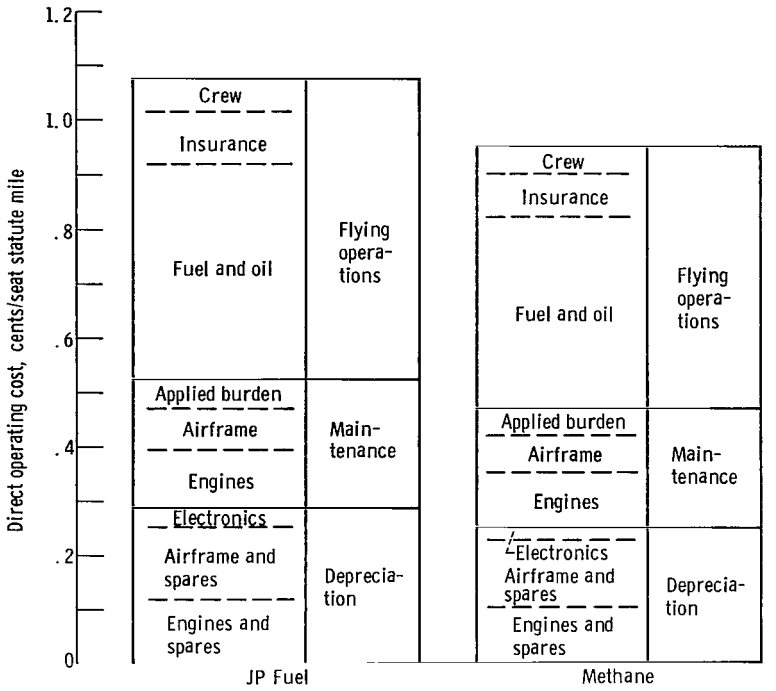


Figure 15. - Direct operating cost comparison. Airplane gross ramp weight, 460 000 pounds; takeoff wing loading, 50 pounds per square foot; sea-level-static thrust to gross-weight ratio, 0.32; turbine inlet temperature, 2100° F; estimated liquid-methane price, 7 cents per gallon.

direct operating cost of 1.07 cents per seat statute mile, based on the 12-cent-per-gallon (1.8 cents/lb) fuel price specified by the FAA for the SST economic model (international version). Based on a fuel price of 7 cents per gallon (2 cents/lb), the estimated price of liquid methane delivered to the airplane in England, the 229-passenger methane airplane would have a direct operating cost of 0.96 cent per seat statute mile. Since a large part of direct operating cost is fuel cost, as may be seen from figure 15, the direct operating cost of the methane airplane will be very sensitive to this presently highly uncertain item. Figure 16 shows the effect of variations in the price of methane. The price of fuel delivered to an airplane in the United States has been estimated (in consultation with the Institute of Gas Technology) to be about 4.2 cents per gallon (1.2 cents/lb). Figure 16 shows that, for this fuel price, the direct operating cost would be only 0.82 cent per seat statute mile. Hence, if only the heating-value advantage is considered and 3904 pounds of boiloff is allowed for, methane gives a direct operating cost of 11 to 24 percent less than JP fuel (at 12 cents/gal). The price of methane would have to increase to about 9.3 cents per gallon before its direct operating cost advantage relative to 12-cent-per-gallon JP fuel is eliminated. The effect of a price variation in JP fuel is also shown in figure 16.

Figure 17 shows the effect airplane price has on the direct operating cost of the

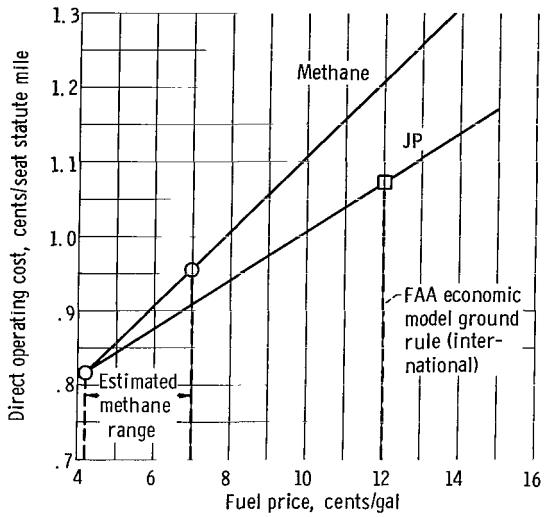


Figure 16. - Effect of fuel price on direct operating cost. Takeoff wing loading, 50 pounds per square foot; seal-level-static thrust to gross-weight ratio, 0.32.

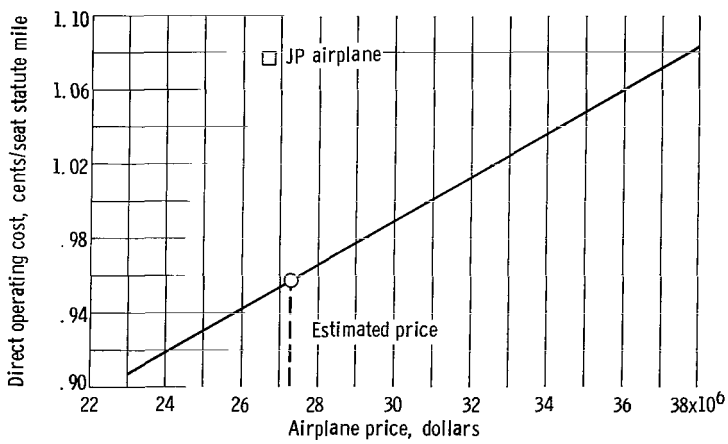
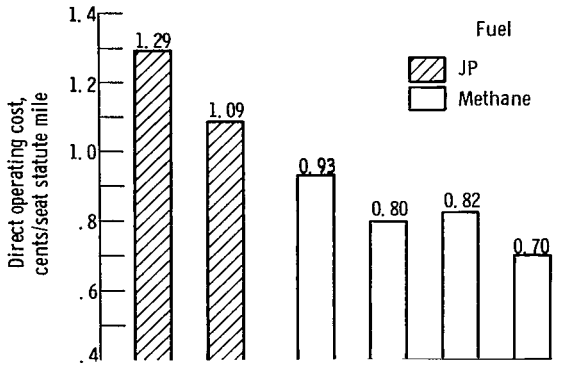


Figure 17. - Effect of methane airplane price on direct operating cost. Airplane gross ramp weight 460 000 pounds; takeoff wing loading, 50 pounds per square foot; sea-level-static thrust to gross-weight ratio, 0.32.



Turbine inlet temperature, T_{it}, °F	1900	2200	2200	2200	2800	2800
Blade metal temperature, T_m, °F	1570	1740	1510	1510	1740	1740
Number of passengers	171	198	232	232	259	259
Fuel cost, cents/gal	12.0	12.0	7.0	4.2	7.0	4.2
Engine airflow, $\dot{W}_a$, lb/sec	535	461	455	455	371	371

Figure 18. - Effect of methane on lowering direct operating cost. Wings and engines sized for takeoff; no boiloff loss. Wing loading, 50 pounds per square foot; sea-level-static thrust to gross-weight ratio, 0.32.

229-passenger methane airplane if a fuel price of 7.0 cents per gallon is assumed. The estimated prices of the methane- and JP-fueled airplanes are seen to be $27.3 million and $26.7 million, respectively. To nullify the direct operating cost advantage of methane, the price of the methane airplane would have to increase $10 million (36.6 percent) over the estimate. Lower methane fuel prices would permit even greater increases in original airplane price before the methane direct operating cost advantage is eliminated.

Figure 18 shows the effect that higher turbine inlet temperatures (made possible by the greater heat-sink capacity of methane) would have on direct operating cost. The cases presented here are the same as those presented in figure 13 (p. 24) with number of passengers as the figure of merit. If it is necessary to decrease the turbine inlet temperature of the JP-fueled engine from 2200° to 1900° F to get adequate turbine blade life, figure 18 shows that the direct operating cost would increase from 1.09 to 1.29 cents per seat statute mile, an increase of 18.3 percent. If differences in time between engine overhaul had been considered, this increase might have been trimmed somewhat. The 2200° F methane engines, which should have about the same blade life as the 1900° F JP engines when the methane heat sink is utilized for turbine cooling, would give a direct operating cost of 0.80 to 0.93 cent per seat statute mile, for fuel prices of 4.2 to 7.0 cents per gallon, respectively. This represents a reduction in direct operating cost of 28 to 38 percent for methane. If takeoff noise restrictions are waived, the 2800° F methane engines would give a direct operating cost of 0.70 to 0.82 cent per seat statute mile for the range of fuel prices considered, a reduction of up to 36 percent below the level obtained with the 2200° F JP engines with the same blade life.

It is interesting to consider once again what would happen if takeoff noise restricted

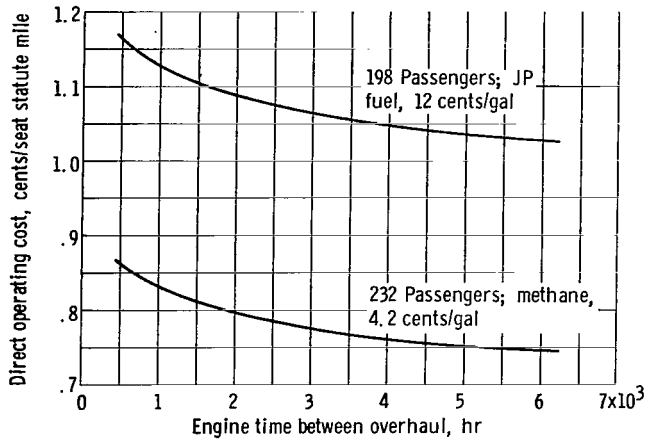
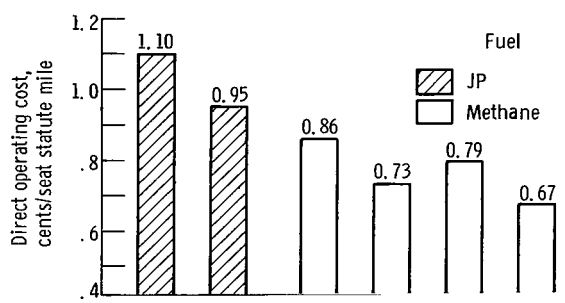


Figure 19. - Effect of time between overhaul on direct operating cost. Airplane gross ramp weight, 460 000 pounds; turbine inlet temperature, 2200° F.



Turbine inlet temperature, T_{it}, °F	1900	2200	2200	2200	2800	2800
Blade metal temperature, T_m, °F	1570	1740	1510	1510	1740	1740
Wing loading, W_g/S_w, lb/sq ft	80	80	60	60	60	60
Sea-level-static thrust to gross-weight ratio, $(F/W_g)_{s\ell s}$	0.30	0.30	0.30	0.30	0.30	0.30
Number of passengers	203	229	253	253	274	274
Fuel cost, cents/gal	12.0	12.0	7.0	4.2	7.0	4.2
Engine airflow, $\dot{W}_a$, lb/sec	502	432	426	426	348	348

Figure 20. - Effect of methane on lowering direct operating cost with wing size selected to maximize number of passengers (no boiloff loss).

the maximum turbine inlet temperature to 2200° F. Figure 18 shows that, when times between engine overhaul are the same (2000 hr), methane gives a 15 to 27 percent lower direct operating cost than JP fuel at this turbine inlet temperature. However, with the methane heat exchanger in the turbine cooling circuit, the blade life expectancy might be many times greater than that attainable with JP fuel. Hence, for a 2200° F methane engine, blade life might not be the critical factor that determines the time between engine overhaul. The most reliable subsonic jets now have attained 6000- to 7000-hour times between overhaul. It is unlikely that any SST engine would be acceptable to the airlines if it could not eventually (after initial debugging) attain a time between overhaul of at least 2000 hours. Figure 19 shows the effect the time between overhaul has on direct operating cost for the 2200° F JP- and methane-fueled airplanes. If it is assumed that the

methane heat exchanger will permit the same engine reliability as obtained by the best subsonic jets (i. e., a 6000-hr time between overhaul), this figure shows that the direct operating cost is reduced to 0.75 cent per seat statute mile (an additional decrease of 4.6 percent) which is 31 percent less than the direct operating cost that could be attained with JP fuel if a 2000-hour time between overhaul is assumed.

The preceding methane direct operating cost benefits have been for airplanes with takeoff wing and thrust loadings of 50 pounds per square foot and 0.32, respectively. If it becomes possible to operate the JP- and methane-fueled airplanes at their optimum wing loadings of 80 and 60 pounds per square foot, respectively, and thrust loadings of 0.30, the direct operating costs are changed to those indicated in figure 20. Overall methane advantage for equal turbine metal temperatures is cut to 22 to 34 percent, still a significant improvement.

CONCLUDING REMARKS

An analytical study was made to determine the advantages that might be obtained by using liquid methane instead of conventional kerosene-type JP fuels in a fixed arrow-wing supersonic transport configuration. Afterburning turbojets were used as the propulsion system.

Among the advantages of methane, relative to JP fuel, are its higher heat of combustion and its higher heat-sink capability. By utilizing only the additional heating value of methane and not its greater heat sink, a payload improvement of about 18 percent could be obtained, relative to a JP-fueled airplane designed to carry about 200 passengers. This payload improvement can be achieved only when the fuel tanks are sufficiently insulated and pressurized to minimize methane boiloff losses. A methane-fuel-system weight penalty (extra insulation, plumbing, etc.,) of approximately 3500 pounds was estimated to be adequate to accomplish this goal. If the higher heat sink of methane is used to provide additional turbine cooling, the methane payload improvement over JP could be about 31 percent. This additional 13-percent improvement would result from the higher turbine inlet temperature it is possible to attain with more turbine cooling.

By utilizing only the additional heating value of methane and not its greater heat sink, the direct operating cost would drop 15 to 27 percent relative to a JP-fueled airplane, depending on the price of the delivered methane. If the additional heat sink of methane could be used to allow higher turbine inlet temperatures, the methane direct operating cost would be as much as 36 percent below that for JP fuel.

The fixed arrow-wing airplane considered in the basic study was sized with large wings (i. e., a wing loading of 50 lb/sq ft) in order to obtain adequate takeoff and landing performance characteristics. Smaller and thus lighter wings would have permitted

greater payloads, but takeoff and landing performance would have suffered. Smaller wings (higher wing loading) decrease the advantage shown by the low-density methane, since a great portion of the fuel would normally be stored in the wings. With the assumptions of the present study, airplanes with wing loadings of the order of 100 pounds per square foot (typical of variable-sweep-wing configurations) suffered so severely from volume limitations that the benefit of methane's greater heat of combustion and/or heat sink was much less than the improvement that could be achieved with the lower wing loadings typical of fixed-sweep configurations. Because the results are sensitive to airframe assumptions, it should not be concluded that variable-sweep designs cannot benefit from the use of methane. However, it does appear that the most striking gains can be obtained with low-wing-loading vehicles.

In addition to the benefits already described for methane, there are other advantages that could accrue from its use. It is expected that coke and smoke formation will be largely eliminated. Since the flame luminosity of methane is much less than that of JP fuel, its radiant heat transfer to the combustor liner is much less, thereby reducing the liner temperature when equal gas temperatures are considered. The use of methane could completely eliminate the present concern about lacquer formation in the oil-fuel heat exchangers and the clogging of fuel injectors. Hence, it is possible that the direct operating cost improvements to be obtained with methane are even greater than those quoted herein, since these factors may affect engine time between overhaul and were not considered in the calculations. Another factor not considered in this analysis is the engine weight reduction that would result from shorter combustors that might be a possibility with methane.

In spite of the problems of handling cryogenic liquid methane, as well as the problems associated with a heavier and more complicated airplane fuel system, methane offers the possibility of significant advantages in the performance and economics of commercial supersonic transports. A more definitive evaluation requires increased knowledge in such areas as fuel system weights, fuel prices, and fuel handling problems.

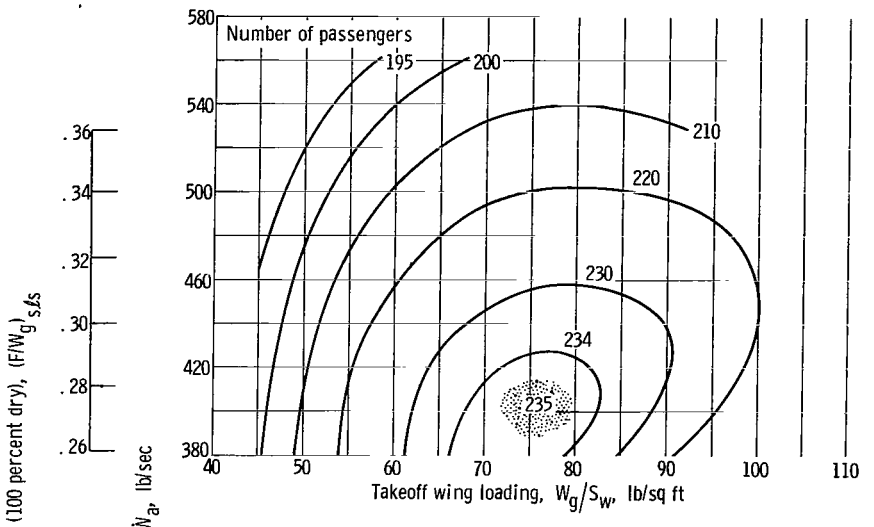
Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, March 28, 1966.

APPENDIX - PAYLOAD AND PERFORMANCE TRADE-OFFS INVOLVED IN ENGINE AND WING SIZING

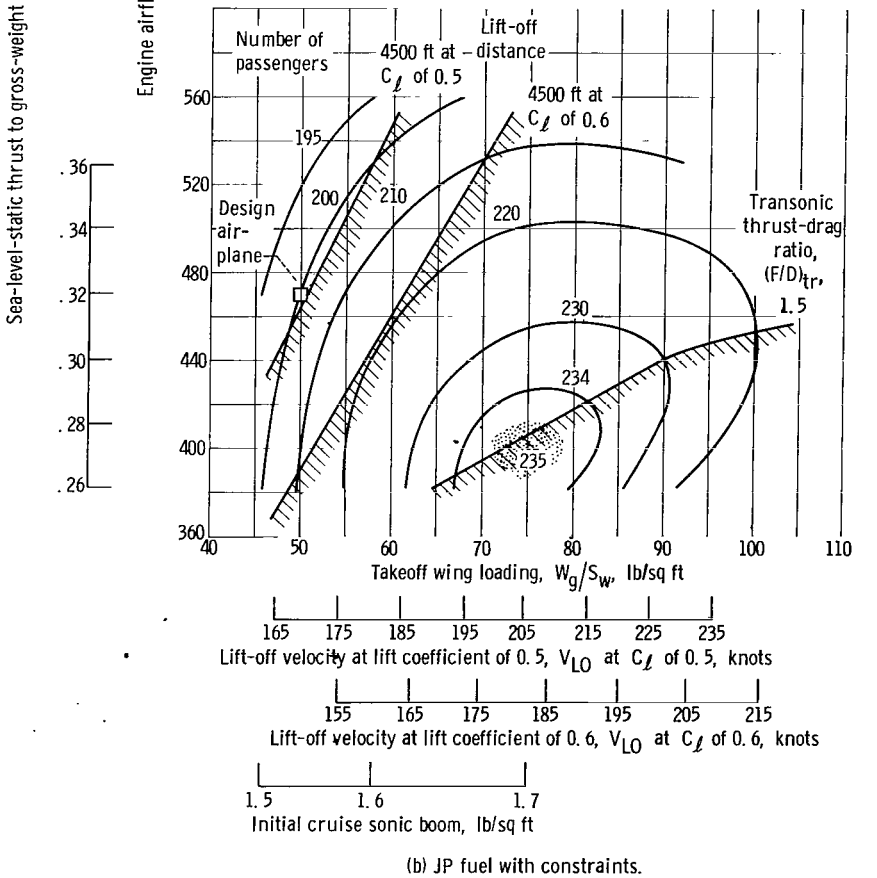
Two of the parameters that were varied in this study were the design engine airflow and wing area because it is possible that the optimum methane-airplane design might differ from the optimum JP fuel airplane design. To ensure that a consistent set of design criteria was used for both airplanes, both the methane- and the JP-fueled airplane engines and wings were sized to yield maximum payload within the constraint of various operational limitations.

Figure 21(a) is a "thumbprint" map for the series of JP-fueled SCAT 15F type airplanes with 2100° F afterburning turbojet engines considered in this study. Each point on a given contour represents a combination of engine and wing size that will permit a particular number of passengers to be carried over the 3500-nautical-mile range when the gross ramp weight is fixed at 460 000 pounds. The map shows that the maximum payload that can be carried is 235 passengers, which is obtained at takeoff wing and thrust loadings of approximately 73 pounds per square foot and 0.26, respectively. These wing and thrust loadings correspond to a wing planform area of 6300 square feet and an engine size of 382 pounds per second. Increases in wing area (i.e., decreases in wing loading) promote higher lift-drag ratios at the expense, unfortunately, of greater wing weight. Because of this trade-off, there is a certain range of wing loading that will permit a maximum number of passengers. This range of wing loading is seen from the thumbprint to lie between 70 and 80 pounds per square foot for the range of engine size considered. When the engine size is increased above 382 pounds per second, which yields the maximum payload in combination with the 73-pound-per-square-foot wing loading, payload suffers at the expense of greater engine weight. With engines smaller than 382 pounds per second, payload decreases in spite of the lighter engine weight because of inadequate performance (excessive fuel consumption). When the engines are too small, acceleration is reduced and the climb to cruise altitude takes longer. Thus, a greater total fuel consumption results over the entire mission, since the cruise range becomes shorter and the least fuel consumption per mile in the supersonic part of the flight occurs during cruise.

Larger engines and wings than the combination that maximizes payload improve performance by increasing the thrust-drag F/D and lift-drag L/D ratios. The thumbprint map of figure 21(a), as it stands, fails to illustrate this point. It is possible, in fact, that the wing and engine combination which maximized payload failed to provide adequate margins of performance. It was necessary, therefore, to investigate certain performance criteria that might be critical in the design of the airplane. Among these performance criteria are lift-off distance and velocity, transonic acceleration, and sonic



(a) JP fuel without constraints.



(b) JP fuel with constraints.

Figure 21. - Wing and engine sizing. Turbine inlet temperature, 2100° F.

boom at the beginning of cruise. One of the mission goals has been to begin cruise with a sonic boom not to exceed 1.5 pounds per square foot, even though the sonic boom in climb to cruise altitude is allowed to reach 2.0 pounds per square foot. Unfortunately, the altitude at which the Breguet cruise factor maximized for airplanes with relatively small wings produced sonic booms in excess of the 1.5-pound-per-square-foot cruise goal, but nevertheless always below the 2.0-pound-per-square-foot-climb sonic-boom limit.

Lower wing loadings and larger engine sizes that produce greater thrust reduce the lift-off distance and velocity. For the engine and wing combination that maximized payload, the required lift-off distance was 9000 feet with a velocity of 203 knots at the point of lift-off on a standard day. The angle of attack at lift-off was not allowed to exceed 11^{o} in order to prevent the tail of the airplane from dragging on the runway. One of the SST design objectives has been to establish a lift-off speed of the order of 160 knots at maximum gross weight (i.e., takeoff speeds comparable with present intercontinental subsonic jet transports). A further objective is for the aircraft to be able to take off fully loaded on a hot day from a 10 000-foot runway. It is obvious that the wing and engine combination that maximized payload did not meet the lift-off speed objective and possibly not the runway distance objective, if the distance had been calculated for a hot day instead of a standard day and if consideration had been given to clearing an obstacle at the end of the runway.

In figure 21(b), there is superimposed on the thumbprint contour of figure 21(a) a limiting line representing a standard day lift-off distance of 4500 feet for a takeoff lift coefficient C_L of 0.5, which is representative of the C_L that could be obtained with the SCAT-15F configuration under study if the takeoff angle of attack is limited to 11^{o}. The 4500-foot lift-off distance requirement is considered to be a reasonable design criterion when hot-day conditions, one engine out performance, and clearance of a 30-foot obstacle at the end of a 10 000-foot runway are not considered. The area above and to the left of this limiting line represents conditions where the takeoff distance would be less than 4500 feet. A corresponding lift-off velocity scale to go with the takeoff C_L of 0.5 has been plotted as one of the abscissa scales. If high-lift devices were incorporated into the airplane design without weight or drag penalties, it might be possible to attain a takeoff C_L of 0.6 instead of 0.5, in which case a new 4500-foot lift-off distance limiting line and a corresponding new lift-off velocity scale could be plotted, as shown in figure 21(b). It can be seen from the figure that the use of these hypothetical high-lift devices could allow the design payload to increase considerably because it would then be possible to use smaller and, therefore, lighter wings. However, there are other design criteria that must be investigated before it can be assumed that the airplane wings and engines are sized by the takeoff requirements.

No firm minimum transonic thrust-drag F/D requirement exists today, but many

authorities believe it should be at least 1.4 on a standard day. The engine and wing combination that maximized the number of passengers results in a transonic F/D of 1.43. Larger wings and engines substantially increase the transonic thrust margin. An F/D of 1.5 limiting line has also been superimposed on the thumbprint map. The area above and to the left of this line represents conditions of higher and even more acceptable transonic thrust margins. Apparently, transonic F/D is not a critical sizing criterion for this particular airplane configuration with afterburning turbojet engines.

Airplanes with larger wings tend to create a smaller sonic boom at the start of cruise than do the ones with smaller wings if cruise is begun at the altitude that maximizes the Breguet factor. Aircraft with larger wings have higher optimum cruise altitudes resulting in lower sonic boom overpressures on the ground, since the greater distance between observer and airplane helps attenuate the shock. Larger engine sizes generally result in somewhat higher cruise sonic booms because of the greater airplane weight on entering cruise. This greater airplane weight at cruise with larger engines is the result of less fuel consumption during the faster climb. Over the range of engine sizes considered in this study, the effect of engine size on cruise sonic boom was negligible. For optimum cruise altitudes, it can be seen from the sonic-boom scale at the bottom of figure 21(b) that the nominal goal of 1.5 pounds per square foot requires a wing loading of about 45 pounds per square foot.

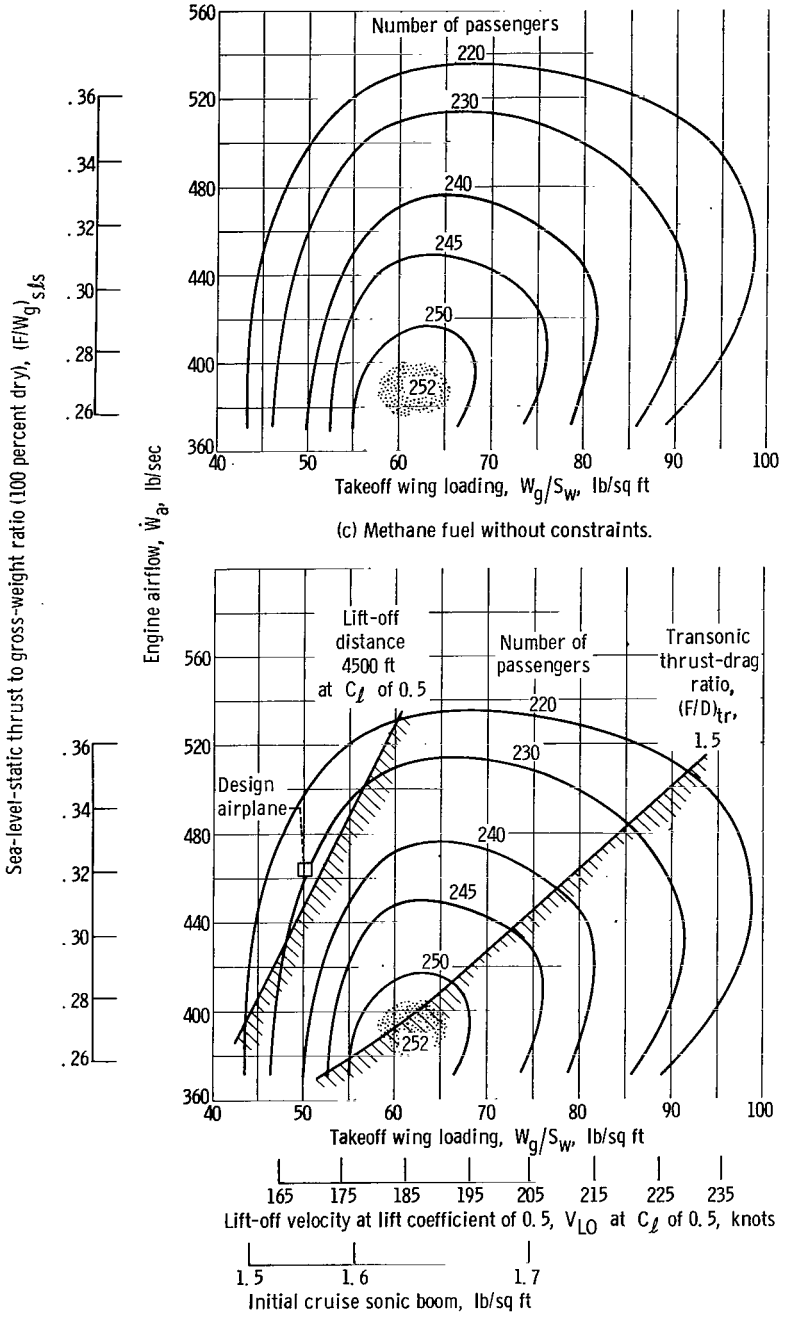
The foregoing discussion, then, shows that the wing and engine combination which maximized payload does not result in a satisfactory airplane because of these other criteria. Thus, the design airplane representative of the configurational class under study was selected to have a takeoff wing loading of 50 pounds per square foot and a takeoff thrust to gross-weight ratio of 0.32, with a resulting payload of 201 passengers. This would be equivalent to a wing area of 9200 square feet and a sea-level static design engine airflow of 470 pounds per second. These design values were selected after considering the performance and safety advantages of larger engines and wings together with the payload penalty involved as the result of their added weight. Still larger engines and wings would have provided even better performance levels, but the additional performance advantage was thought to be outweighed by the further decrease in payload. For this particular design, the lift-off distance on a standard day would be 4450 feet with a corresponding lift-off velocity of approximately 169 knots with the 11^{o} takeoff C_L of 0.5, a transonic F/D of 1.94, and an initial cruise sonic boom of about 1.54 pounds per square foot. These values of the performance criteria are reasonably close to the desired performance goals already outlined. The JP airplane selected, therefore, as the design representative of this configurational class is satisfactory from the standpoint of performance. The same wing area and engine size could provide even better takeoff performance if high-lift devices were available. If it were possible to increase the takeoff C_L from 0.5 to 0.6 without any weight or drag penalties, the lift-off velocity could be re-

duced from 169 to 150 knots with a corresponding reduction in lift-off distance from 4450 to 3550 feet. On the other hand, if it were desired to observe the original takeoff performance limitation of 4500 feet, with high-lift devices, the wings and engines could be resized to 7670 square feet (60 lb/sq ft) and 460 pounds per second, respectively, with the number of passengers increasing from 201 to 219, an increase of almost 9 percent. Lift-off velocity would decrease from 169 to 163 knots. The transonic F/D, however, would decrease from 1.94 to 1.88 and the initial cruise sonic-boom overpressure would increase from 1.54 to 1.62 pounds per square foot. The transonic F/D would thus still be quite adequate although the initial cruise sonic-boom overpressure would be above the desired level of 1.5 pounds per square foot. Although the initial cruise sonic boom would be exceeded, this fact may not be too significant when it is recalled that the climb sonic-boom limit of 2.0 pounds per square foot was considered acceptable and that the sonic-boom overpressure in cruise is continually decreasing as the flight progresses because of the higher altitudes achieved as the fuel load lightens.

A similar thumbprint plot is shown in figure 21(c) for a methane-fueled airplane. This figure takes into account the penalties of boiloff and greater insulation weights for methane, as well as the increased fuel storage volume requirement. It can be seen from the figure that the maximum payload would be obtained with a takeoff wing loading of about 60 pounds per square foot and an engine takeoff thrust to gross-weight ratio of about 0.26. At greater wing loadings (smaller wings), the fuselage must be extended to accommodate fuel that might otherwise be stored in the wings if they were larger. Below wing loadings of 60 pounds per square foot, fuel volume is not a critical factor, since the larger wings provide additional storage volume. Fuel storage volume is also a smaller problem when larger engines are used, since larger engines, in general, result in less fuel consumption because of a faster climb to cruise altitude where the engines operate at a lower rate of fuel consumption than in climb. Also, larger engines that afterburn during cruise will use less fuel per mile of cruise than will smaller engines that must cruise with more afterburning.

The performance criteria referred to in the discussion of JP-fueled airplane sizing require that for methane, also, larger engines and wings be used than the combination which maximizes payload. The constraint curves are shown superimposed on the methane thumbprint map in figure 21(d). These constraints necessitate an engine thrust to gross-weight ratio of 0.32 and a takeoff wing loading of 50 pounds per square foot, as before with JP, when the takeoff C_L of 0.5 is assumed for an 11^o angle of attack. It so happened that the same performance criteria sized both the methane and JP airplanes, resulting in the same wing size and takeoff thrust-weight ratios for both.

It would have been possible, however, if the available methane storage volume had been more limited than was assumed, for the maximum payload to have occurred at a wing loading less than 50 pounds per square foot. All the payload contours would then



(c) Methane fuel without constraints.

(d) Methane fuel with constraints.

Figure 21. - Concluded.

have been shifted to the left on the map, while the performance constraint curves would have remained almost stationary. The maximum payload would then have been the critical methane-airplane sizing factor, since all the performance criteria would be satisfied at the lower wing loading where the payload maximizes. It was necessary, therefore, to determine by means of a plot such as figure 21(d) if the wing and engine combination producing the maximum number of passengers was the critical sizing condition. As it turned out, takeoff performance criteria sized both the methane and JP airplanes. For the engine and wing sizes selected as optimum for the design methane airplane (i. e., 463 lb/sec and 9200 sq ft, respectively), takeoff velocity and distance, transonic thrust margin, and initial cruise sonic boom were essentially the same as the values quoted for the JP-fueled airplane.

REFERENCES

1. Hibbard, Robert R.: Evaluation of Liquefied Hydrocarbon Gases as Turbojet Fuels. NACA RM E56I21, 1956.

2. Hubbard, Harvey H.; Maglieri, Domenic J.; Huckel, Vera; and Hilton, David A. (With appendix by Harry W. Carlson): Ground Measurements of Sonic-Boom Pressures for the Altitude Range of 10,000 to 75,000 Feet. NASA TR R-198, 1964. (Supersedes NACA TM X-633.)

3. Dugan, J. F., Jr.; Koenig, R. W.; Whitlow, J. B., Jr.; and McAuliffe, T. B.: Power for the Mach 3 SST. Astronaut. and Aeron., vol. 2, no. 9, Sept. 1964, pp. 44-53.

4. Dugan, James F., Jr.: Compressor and Turbine Matching. Aerodynamic Design of Axial-Flow Compressors (revised). NASA SP-36, 1965, pp. 469-508. (Supersedes NACA RM E56B03b.)

5. Hall, Eldon W.; and Weber, Richard J.: Tables and Charts for Thermodynamic Calculations Involving Air and Fuels Containing Boron, Carbon, Hydrogen, and Oxygen. NACA RM E56B27, 1956.

6. Anon.: Santocel A - Monsanto's Silica Aerogel for Efficient Thermal Insulation. Tech. Bull. No. I-180, Monsanto Chem. Co., Oct. 1959.

7. Anon.: Dupont H Film - Summary of Properties. Bull. No. H-1, E. I. DuPont de Nemours & Co., 1965.

8. Anon.: Standard Method of Estimating Direct Operating Costs of Transport Airplanes. Air Transport Association of Am., Aug. 1960.

"The aeronautical and space activities of the United States shall be conducted so as to contribute . . . to the expansion of human knowledge of phenomena in the atmosphere and space. The Administration shall provide for the widest practicable and appropriate dissemination of information concerning its activities and the results thereof."

—NATIONAL AERONAUTICS AND SPACE ACT OF 1958

NASA SCIENTIFIC AND TECHNICAL PUBLICATIONS

TECHNICAL REPORTS: Scientific and technical information considered important, complete, and a lasting contribution to existing knowledge.

TECHNICAL NOTES: Information less broad in scope but nevertheless of importance as a contribution to existing knowledge.

TECHNICAL MEMORANDUMS: Information receiving limited distribution because of preliminary data, security classification, or other reasons.

CONTRACTOR REPORTS: Technical information generated in connection with a NASA contract or grant and released under NASA auspices.

TECHNICAL TRANSLATIONS: Information published in a foreign language considered to merit NASA distribution in English.

TECHNICAL REPRINTS: Information derived from NASA activities and initially published in the form of journal articles.

SPECIAL PUBLICATIONS: Information derived from or of value to NASA activities but not necessarily reporting the results of individual NASA-programmed scientific efforts. Publications include conference proceedings, monographs, data compilations, handbooks, sourcebooks, and special bibliographies.

Details on the availability of these publications may be obtained from:

SCIENTIFIC AND TECHNICAL INFORMATION DIVISION

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Washington, D.C. 20546