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RESEARCH MEMORANDUM

LIQUID HYDROGEN AS A JET FUEL FOR HIGH-ALTITUDE AIRCRAFT

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NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS
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INTRODUCTION

The urgent requirement that military aircraft fly ever farther and higher has led to an intensive search for fuels of higher energy as a means for extending performance. Thus far, only casual attention has been given to the possibilities of liquid hydrogen as a fuel for conventional air-breathing engines despite the fact that it is the element with the highest heating value (fig. 1), and has good combustion characteristics over wide ranges of fuel-air mixture ratio.

A deterrent to early and easy use of liquid hydrogen as fuel has stemmed largely from its high specific volume (cu ft/lb), which is about ten times that of the conventional hydrocarbons. Problems of supply and handling also discourage interest in a new fuel unless it is shown that military requirements can be met in no other way. Reference 1 pointed out the desirability of research effort on problems of aircraft structure, and fuel tankage and handling in sufficient detail to determine whether a significant part of the thermodynamic promise of hydrogen can be realized in actual flight. Both current military considerations and major advances in the aeronautical field have now itensified this interest in liquid hydrogen as an aircraft fuel.

Recent research on turbine and ram-jet engines and concurrent research in aerodynamics have provided information for the design of military engines and aircraft that will fly far higher than our present military aircraft can. These technological gains emphasize the need for sound re-evaluation of liquid hydrogen as a fuel, since it is at the high altitudes that its advantages are most apparent. It is now expected that gas-turbine-engine specific weight (lb of engine weight/lb of thrust) may become less than one-half the value for engines in current military use. Unconventional jet-engine configurations such as the ducted-rocket, ducted-fan, and ram-jet engines may have even lower specific weight. Specific engine weight, based on altitude engine performance, is the primary variable that now establishes the ceiling of aircraft. With lighter engines, flight at higher altitudes within the next few years may be confidently predicted.

Aircraft that fly at higher altitudes will have large wings to provide lift in the rarefied upper atmosphere. At 80,000 feet altitude, air density is about one-fourth that at 50,000 feet altitude. An airplane designed to fly at 80,000 feet may require a wing area four times as great as that of a similar airplane of equal weight designed to fly at 50,000 feet altitude. If the aircraft are dimensionally similar, so that aircraft efficiency (lift/drag) is about the same for both designs, the volume of the fuselage for the 80,000-feet-altitude airplane could be about eight times that of the 50,000-feet-design-altitude airplane.

It is apparent, therefore, that as aircraft flight altitude is increased, aircraft of about equal aerodynamic efficiency will have much larger fuel-storage volume available in the fuselage and wings. This increase in relative aircraft storage volume without sacrifice in aerodynamic efficiency provides the key to the successful exploitation of the high heating value per pound of the low-density liquid hydrogen.

This paper will review some of the analytical and experimental studies of the use of liquid hydrogen as a jet-engine fuel that have been conducted at the Lewis Flight Propulsion Laboratory, and show the possible extension of aircraft performance that will follow adequate research and development effort on the problems of its use.

Assumptions made in analytical studies of this kind regarding performance and weight of components and the complete aircraft investigated are always to be questioned prior to the manufacture of an aircraft that accomplishes the mission intended. This fact neither vitiates the analysis nor reduces the need for it. No other course is open but to use assumptions consistent with the state of the art and the progress anticipated. It is fortunate that in the present analysis many of the gains possible are large enough so that gross errors in assumptions are tolerable.

FUEL CHARACTERISTICS

The physical properties of liquid hydrogen that have been used in the present analysis are summarized in table I and in figures 2 and 3. The heating value of the fuel is 51,571 Btu per pound, which is about 2.75 times the heating value of the average hydrocarbon fuel (JP-4) in current military use.

Thermodynamic calculations show that the thrust specific fuel consumptions of like engines burning hydrogen and hydrocarbon fuel at about 2000° R will be about in the ratio of the heating values of their fuels. That is, the thrust specific fuel consumption ((lb fuel/hr)/lb thrust) of the hydrogen-fueled engine will be about 1/2.75 or 0.363 times that of the engine burning an average JP-4 fuel. At cycle temperatures of 3500° R, as are used in afterburning engines, the ratio of hydrogen to JP-4 specific fuel consumption may increase to about 0.375. The assumption was made in the calculations that combustion efficiency was the same for both fuels. Actually, as will be shown later, under marginal burning conditions in high-altitude flight the combustion efficiency of the hydrogen fuel will be greater.

The cycle calculations also show that the thrust per pound of air may be 3 to 5 percent higher with hydrogen as a fuel than is obtained with JP-4 fuel when the maximum cycle temperature is the same for both. This increase in air specific thrust occurs because the water vapor in the exhaust of the hydrogen-fueled engine is of lower molecular weight $(m = 18)$ than the carbon dioxide exhaust of the hydrocarbon-fueled engine $(m = 44)$.

With a density of 4.42 pounds per cubic foot at 1 atmosphere and 37° R, liquid hydrogen has a heating value of 228,600 Btu per cubic foot, which is about one-fourth of the value for JP-4 fuel. Fuel storage is obviously a problem with the hydrogen fuel when airplane volume is limited.

The low temperature of liquid hydrogen and the high value of specific heat of hydrogen vapor (3.40 Btu/(lb)($^{\circ}$F)) are properties of particular interest. In supersonic flight, when cooling of the crew and equipment compartments becomes necessary and cooling of the engine turbine becomes desirable, liquid hydrogen would be available as a refrigerant before injection into the engine. An enthalpy change of about 1600 Btu per pound occurs between liquid hydrogen at 37° R and hydrogen vapor at room temperature (fig. 3). If, as in a sample flight at a Mach number of 2, fuel is burned at a rate of about 15,000 pounds per hour, the total refrigeration capacity is about 24 million Btu per hour or the equivalent of about 2000 tons of refrigeration. A compressor drive of about 2500 horsepower would be required in a conventional refrigeration plant to provide this tonnage. The availability of the hydrogen as a refrigerant before it is burned in the engine will provide extreme simplification of the cooling systems required for aircraft and engines designed for supersonic flight.

Of further interest are the combustion characteristics of the fuel relative to those of JP-4 or similar hydrocarbons. The combustion limits and efficiency are seriously reduced in turbojet engines operating with JP-4 fuel at altitudes of 70,000 and 80,000 feet at speeds for which maximum range can be attained. In order to provide pressures in the engine combustion chamber high enough to sustain efficient combustion at these altitudes and speeds, heavy high-pressure-ratio engines are required. As will be shown later, engine weight is the single most important variable determining the height to which an airplane can fly; if heavy engines are required to obtain good combustion efficiency, the altitude performance is curtailed drastically. In supersonic flight at Mach 2 and 80,000 feet altitude, the pressures in afterburners drop to about 1/2 atmosphere; for these conditions, the efficiency of good JP-4-fueled afterburners is generally about 85 percent. In the ducted-fan engine at subsonic and transonic speeds, at altitudes much above 50,000 feet, pressures and temperatures in the duct passage are low and in the range of values for which efficient combustion has not yet been attained with conventional hydrocarbon fuels. Although applied combustion data for hydrogen are as yet scant, there are excellent reasons to believe that the combustion characteristics of hydrogen will greatly excel those of JP-4 fuel in the low-pressure conditions of high-altitude flight.

Curves showing the minimum pressure for which combustion can be sustained in a standard 2-inch-diameter combustion tube are shown in figure 4. These curves were estimated from experimental data obtained at the Lewis laboratory under similar test conditions. The minimum combustion pressures are plotted against equivalence ratio, which is unity for a stoichiometric mixture of fuel and air. Minimum pressure for combustion at stoichiometric mixture ratio is 8 millimeters of mercury for hydrogen as compared with 32 millimeters of mercury for JP-4 fuel measured under the same test conditions. Just as significant as the low pressure at which combustion is supported, is the extremely wide range of mixture ratios for which the combustion is sustained.

Measurements of laminar flame velocity for hydrogen and for hydrocarbon fuels (fig. 5) are also of interest. These laminar flame velocities were obtained in Bunsen burner and flame-tube experiments (ref. 2). Results show that the laminar flame velocity of hydrogen is about 7.6 times that of JP-4 fuel. These data support expectations that both the combustion limits and combustion efficiencies of hydrogen will be greatly superior to those of JP-4 at marginal altitude burning conditions.

Of even greater significance are results obtained in recent tests at the Lewis laboratory on a J33 turbojet-engine combustor (ref. 3). Tests in this combustor were made using hydrogen vapor as a fuel. The combustor was modified only by adapting the fuel-injector nozzles for the use of a gaseous fuel. Investigations were conducted over a range of pressures in the combustor down to almost 1/10 atmosphere. Despite the fact that the combustor liner and fuel-injector system were not properly adapted to the characteristics of the low-density vapor fuel, excellent combustion efficiencies were measured over wide ranges of combustor pressure and velocity. No combustion instability or flame blow-outs were observed over the entire range of fuel and air flows investigated.

For comparison, a gaseous hydrocarbon fuel, propane, was burned in the same combustor over limited ranges of temperature rise. At the low-pressure test conditions, combustion efficiencies were low and were adversely affected by increases in combustor velocities and decreases in combustor-inlet pressure. Since the combustion characteristics of gaseous propane are superior to those of liquid JP-4, a comparison of hydrogen to JP-4 fuel would reveal an even greater advantage for hydrogen.

From the results of reference 3, the curve of figure 6 has been constructed. Combustor efficiencies are shown for a range of flight altitudes for an engine with a compressor pressure ratio of 5 installed in an airplane flying at a Mach number of 0.75. A combustion efficiency of about 94 percent is shown for an altitude of 80,000 feet. Since these data were obtained in a combustor designed for liquid hydrocarbon fuel, and since it is known that the mixture ratio in the region of the fuel injector was too rich for most efficient burning, it is expected that efficiencies approaching 100 percent can be realized in combustors designed for hydrogen fuel and operated at these flight conditions.

Knowledge regarding the manufacturing, storage, and handling of liquid hydrogen has been advanced in recent years by efforts of the Atomic Energy Commission and the military services. Liquid hydrogen is chemically stable. After converting from the ortho to the para structure, it may be stored for long periods of time in appropriate storage vessels. This conversion evolves 220 Btu per pound converted. Normally, gaseous hydrogen is 75 percent ortho and 25 percent para; at its boiling point it is substantially all in the para form when it is in equilibrium. No large facilities for production of hydrogen now exist. Its cost in limited quantities is about the same as that of other chemical products purchased in small quantities.

Difficulties in handling of the fuel will be aggravated because of its excellent combustion characteristics. Safe handling techniques have been developed among small groups now working with liquid hydrogen.

FUELS SYSTEMS AND TANKS

The properties of liquid hydrogen provide the possibility for the design of an aircraft fuel system without fuel pumps. Pressure to pump the fuel may be provided by tank pressure. For cruising flight at a Mach number of 0.75 at 80,000 feet altitude, pressure in the combustion chamber of a turbojet engine designed to burn hydrogen is likely to be about 0.3 atmosphere. Allowing for pressure losses in fuel lines and regulators, which would be small because of the low density and viscosity of the liquid fuel, a pressure of from 1 to 1.5 atmospheres (15 to 22 lb/sq in.) in the tank should be ample to pump the fuel to the engine combustion chambers.

At a flight Mach number of 2 at 80,000 feet altitude, pressure in the primary combustion chamber of the turbojet engine will be about 0.8 atmosphere. This value is based on an engine with a sea-level static compressor pressure ratio of 6.25, which calculation shows to be a good compromise design value for this Mach number. A tank designed for an internal pressure of about 2 atmospheres will provide more than adequate pumping pressure for the cruising flight condition.

Auxiliary tanks of smaller size with higher internal pressures are required for take-off, climb to altitude, and let-down; however, calculations indicate that for long-range missions only about 10 percent of the fuel must be carried in the high-pressure tanks. The tank pressure requirements will differ for each engine-aircraft configuration, and a separate study will be required for each design.

It is contemplated in a liquid-hydrogen, self-pumping fuel system that most of the fuel will be delivered to the vicinity of the engine as a liquid, and will be carried in vacuum-insulated fuel lines such as are conventional for handling of the fuel. Some fuel will vaporize in the tank at a rate determined by the heat flow into the tank through the tank insulation. This vaporized fuel will also be pumped to the engine combustion chamber by the tank pressure and burned with the remainder of the fuel in the engine. It is expected in any event that the fuel delivered as a liquid will be heated and vaporized before injection into the engine combustion chamber in order to provide the aforementioned cooling.

Liquid hydrogen may be stored at pressures near one atmosphere in liquid nitrogen cooled Dewar vessels with a loss from evaporation of about 1 percent per day. It may be stored indefinitely with no evaporative loss in Dewar vessels equipped with mechanical refrigeration. Aircraft tanks must necessarily be lighter in weight than the standard hydrogen Dewar vessels and new ideas for aircraft tank design are required.

Studies of the tank problem have revealed interesting possibilities for the construction of light-weight insulated tanks that utilize some of the technology developed for the construction of fuel tanks for long-range rocket missiles. It is suggested that liquid-hydrogen tanks may be constructed as a cylindrical balloon of light-gage metal, that depends on internal pressure to maintain its shape. The hydrogen will be in direct contact with the metal tank walls, so that the wall temperature will then be about the same as the temperature of the hydrogen. In this way, advantage can be taken of the favorable increase in the physical properties of the metal at the low storage temperature of liquid hydrogen (40° R). Yield strength of aluminum and of some steels is increased 40 to 70 percent above the room temperature value by reducing the temperature to 40° R (fig. 7). Ductility, as measured in elongation tests, also remains adequate for aluminum and the nickel steels at the lower temperatures (fig. 8). Figures 7 and 8 were obtained from reference 4.

Calculations show that about 25,000 pounds of liquid hydrogen may be contained in a cylindrical tank about 10 feet in diameter and 81 feet long, if 10 percent volume is allowed for fuel expansion in the tank (fig. 9). Such a tank has a volume of 6153 cubic feet, and a surface area of 2564 square feet. If stainless steel is used for the tank and methods of welded tank construction that have been developed for large rocket tanks are applied, it is calculated that a tank of this size, weighing about 2600 pounds, will resist an internal pressure of 4 atmospheres (60 lb/sq in.) before yielding. If the pressure in the tank is limited to 2 atmospheres by blow-off valves, the design factor of safety is 2 based on the yield strength of hard type 301 stainless steel (not shown) at about 40° R.

Studies of tank-insulating material showed that a foam plastic with a weight of 1.3 pounds per cubic foot combines satisfactory characteristics of low thermal conductivity, good structural properties, and effectiveness as a vapor barrier. Foam plastics, available commercially in sizes appropriate for construction, are relatively inexpensive. Calculations show that a 2.4-inch layer of this insulation will provide adequate protection for the tank when it is housed in the fuselage or wing structure with only nominal ventilating flows over the tank insulation surface. If the tank is precooled with refrigerated helium gas before initial filling, calculations indicate that the tank may be filled over 2 hours before a scheduled flight and not require topping off before the flight. If the tank is not precooled, 2 to 3 percent of the liquid hydrogen will be evaporated to cool the tank and insulation. Thus, fuel may be added in the expansion volume of the tank and the tank vent left open to the atmosphere before beginning the flight so as to avoid the necessity for topping the tank.

In subsonic long-range flight at high altitude, fuel will vaporize at a rate less than one-third the rate at which fuel is being used by the engines. In supersonic flight, when higher fuel-flow rates to the engine are used, the vaporization rate in the tank will be a much smaller percentage of the fuel rate to the engines. In either case, as mentioned previously, the fuel vapor will be ducted to the engine and burned.

The foam insulation for the tank is estimated to weigh about 700 pounds, and a layer of aluminum foil for radiation shielding will weigh an additional 64 pounds. The weight of the stainless-steel tank shell, 2600 pounds, and the insulation weight, 764 pounds, add to a tank weight of 3364 pounds to store 25,000 pounds of liquid hydrogen. Thus, the estimated tank weight is 0.134 of the weight of the hydrogen contained.

In the subsequent analysis, a slightly higher value of tank weight of 0.15 times the fuel weight has been used in order to include the heavier specific weight of the small high-pressure tanks used in the take-off, climb, and let-down.

ENGINES AND AIRCRAFT

Extended flight at altitudes of 70,000 and 80,000 feet and above, using air-breathing engines, requires development of aircraft engines and airframes especially compromised for the altitude mission. The weighting of the elements in the usual design compromises change with design altitude, and performance factors that are of first-order importance for attaining long-range flight at 50,000 feet altitude may need to be rated of secondary importance for a similar mission to be accomplished at 80,000 feet altitude. The weighting of the design compromises is also vitally dependent on the heating value of the fuel used and is different for hydrogen and for hydrocarbon fuels.

This shift with altitude in the relative compromise value of the various design variables of the aircraft, such as engine weight, structural weight, aerodynamic efficiency, and specific fuel consumption, occurs because specific weight of air-breathing engines increases with altitude. Since the thrust of these engines decreases approximately as air density decreases, a logarithmic increase in specific engine weight (lb of engine weight/lb of actual thrust) occurs as altitude is increased, if flight speed is unchanged.

If specific engine weight at sea-level static condition is used as a reference, the relative change in specific engine weight with altitude depends on flight speed. Values for a flight speed of Mach 0.75 are given in figure 10, which shows that the specific weight increases 25 fold from sea-level static conditions to flight at 80,000 feet altitude. At a flight speed of Mach 2.5 and 80,000 feet altitude, the change in specific weight referenced to sea-level static specific weight is not as large as at Mach 0.75 because of the increase in engine thrust at high flight speeds due to ram compression. For this flight condition, the specific engine weight increases for a representative case to ten times the sea-level value. It is obvious from these considerations why engine weight is such a powerful and determining variable in aircraft designed for high-altitude flight.

Since thrust is obtained at such a heavy penalty in weight at high altitude, extreme attention must be given to designing an efficient aerodynamic configuration so as to reduce to a minimum the thrust requirement. The compromise here is in the direction of accepting heavier structural weight associated with high wing aspect ratios and thin wing sections in order to increase to a maximum the lift-drag ratio for cruising.

In contrast to engines designed for long-range cruising at altitudes of 40,000 and 50,000 feet, in which engine specific fuel consumption is the most important compromise variable, increases in engine specific fuel consumption may be accepted with less penalty for flight at 80,000 feet altitude if lighter weight engines result. Calculations indicate that engines with sea-level compressor pressure ratios of about 6, although less efficient, will provide a subsonic cruise radius comparable to that with the more efficient but heavier high-pressure-ratio engines. The same engine may then serve effectively for both subsonic and supersonic applications.

Benefits of the trend toward lighter but less efficient engines are accentuated when hydrogen is used as a fuel. Because of its high heating value per pound, a less efficient engine cycle may be accepted even more readily than for the hydrocarbon fuel, if adequate saving in engine weight results. If every pound of weight saved in the aircraft by the use of lighter engines can be replaced by a pound of fuel then each pound of hydrogen added in this way would be over twice as effective in extending range as a pound of hydrocarbon fuel.

A further compromise that must be accepted in high-altitude aircraft using hydrogen for fuel is a high fuselage structural weight to accommodate the large volume of fuel to be carried.

Because of the large engine thrusts available at sea-level and the low wing loadings of aircraft designed for high altitude, take-off, climb, acceleration, and landing present no problems. An exception, of course, is the take-off and landing problems of ram-jet aircraft. Take-off and climb of supersonic turbojet aircraft will normally be accomplished with part-throttle engine operation. High-altitude, design-point engine characteristics need not therefore be compromised for take-off performance. This concept is particularly significant in the case of turbojet engines designed for Mach numbers of 2 and above. Properly applied, it leads to reduction in the weight of the engines designed wholly for supersonic flight.

These general observations of the relative importance of aircraft design variables for high-altitude flight were revealed by a detailed analysis of numerous aircraft configurations in which the important design parameters were varied systematically. Intuition and more general analysis (ref. 5) provide broadly the same results. The more extensive analyses of this paper are useful, however, in providing information on how the general principles adapt themselves into actual engine and aircraft configurations. A few of the results of the analysis are presented to show engine and aircraft types and their performance for several high-altitude flight missions with liquid hydrogen used as the fuel.

Comparisons are made in some of the cases with configurations suitably designed for using JP-4 fuel. For these calculations, the same basic assumptions of engine weight, structural weight, aerodynamic efficiency, etc. were made as in the calculations for the hydrogen fuel. The tank weight and volume requirements of the airplane were, of course, different. The JP-4 fuel was credited with the same value of combustion efficiency as the hydrogen fuel although it is expected that the values will be lower.

The aircraft and engines shown are considered to be no more than schematic representations of how aircraft and engines may look when the new weighting of the compromises introduced by high-altitude flight and a new fuel are applied in design. The intent is to present gross results and not detailed designs. The missions selected for the study were the following:

Subsonic bomber
Subsonic reconnaissance
Supersonic bomber
Supersonic reconnaissance
Supersonic fighter

The results of the analysis are summarized in tables II, III, IV, and V in which the major assumptions and calculated characteristics and performances of the engine and aircraft are given. Brief discussions of the engine and aircraft configurations that evolved are given in the subsequent sections of the paper.

Subsonic Bomber

The problem established for the subsonic bomber was to determine the weight and general configuration of an aircraft using liquid hydro- 3725
gen as a fuel that would carry a 10,000-pound bomb and 5000 pounds of fixed equipment to a target at a radius 5500 nautical miles and arrive over the target at 80,000 feet altitude.

The flight plan for the bomber is shown in figure 11. The climb to altitude is made at a constant indicated airspeed of 105 knots, with initial rate of climb of 6000 feet per minute. Maintaining low flight speeds at low altitudes reduced the structural loads on the airplane. Fuel consumption for climb may be reduced, however, if the climb is made at higher indicated airspeeds.

The bomber cruises to within 1000 miles of the target at a Mach number of 0.75 and an altitude of about 70,000 feet then climbs to 80,000 feet. A schematic drawing of the bomber to accomplish this mission is shown in figure 12. Its sea-level take-off weight is 130,000 pounds, and it is powered by four turbojet engines having a sea-level static thrust rating of about 25,000 pounds.

The unconventional appearance of the airplane results from the high aspect ratio (13) of the 31° swept wing. The relative wing weight is high, but the gains in aerodynamic efficiency resulting from the high aspect ratio more than compensate for the high wing weight. Details regarding the airplane dimensions and characteristics are given in table II.

A possible arrangement of the hydrogen tanks in the airplane is shown in figure 13. Fuel is stored in both the fuselage and wings. Drop tanks are effective for extending the radius of the airplane beyond 5500 nautical miles. Alternatively, they may be used in place of the small internal wing tanks to accomplish the 5500 nautical mile radius, with a considerable simplification in the aircraft fuel system.

Aerodynamic investigations of high-aspect-ratio, swept-wing configurations have been conducted at the NACA Ames Aeronautical Laboratory (ref. 6) at Reynolds numbers comparable to those encountered in high-altitude flight. These results and others served as a guide in establishing values for aerodynamic efficiency (L/D) and for determining the nature of control and stability problems. The lift-drag values used in the study did not account for the possibilities of utilizing boundary-layer control to maintain laminar flow over the airplane surfaces. Techniques for control of the boundary layer will probably be first applied in service for flight at low Reynolds number; the high-altitude aircraft of the present study offer opportunity for its application.

The turbojet engines chosen for the mission (engine A in table IV) have a specific weight of 0.2 pound per pound of thrust in sea-level static operation, and a maximum turbine-inlet temperature of 2000° R. The engine weight is about one-half the weight of the engines currently installed in existing lower-altitude bombers. Advanced development engines currently under contract by the military services have brochure weights comparable to the values assumed for this study. These brochure engines are designed for supersonic flight missions and could possibly be made even lighter for the nominal requirements of the present mission. If engines of current specific weight, about 0.4 pound per pound of thrust, were assumed in the bomber calculations for an 80,000 feet target altitude, the flight radius would be reduced to about 40 percent of that possible with a specific engine weight of 0.2 pound per pound of thrust.

The engines for subsonic flight at 80,000 feet should be designed with consideration of the low Reynolds number of the flow at the compressor inlet. Serious reductions in compressor efficiency and engine stall margins would result if short-chord, low-speed compressor blading were used on the initial compressor stages. Wide-chord transonic blading will probably be a "must" on the initial compressor stages of these engines. The heavier compressor weight of wide-chord blading will probably be offset by the relatively low compressor pressure ratio (6.25) required for the engine, by the higher inflow per unit of frontal area made possible with transonic compressor design, and by the possible reductions in engine combustion-chamber length required to burn hydrogen. The use of four large engines instead of additional smaller engines is based on the desire to maintain highest possible Reynolds numbers at the compressor inlet blading.

The effect of target altitude on flight radius for the subsonic bomber is shown in figure 14. Values are shown for the bomber with and without drop tanks. The curves given are envelope curves of a series of aircraft, each designed for a different target altitude. At a target altitude of 80,000 feet, the bomber without drop tanks has a flight radius of about 5400 nautical miles. With drop tanks containing a total of 9,200 pounds of liquid hydrogen, flight radius is increased to about 6300 nautical miles. The gross take-off weight of the bomber with drop tanks is about 143,000 pounds.

If somewhat larger bomb and fixed-equipment weight had been assumed for the bomber mission, the same range and altitude performance could be achieved but with a larger and heavier airplane.

It was of interest to determine how much farther a larger and heavier airplane could carry the 15,000 pounds of fixed and bomb load assumed for the study. Results of this analysis are shown in figure 15. The flight radius is increased only about 550 nautical miles by increasing the airplane gross weight from 130,000 to 200,000 pounds. This difference corresponds to only a 10.3 percent increase in radius for a 54 percent increase in airplane gross weight.

A bomber fueled with JP-4 and of the same gross weight (130,000 lb) as the hydrogen-fueled bomber would have a flight radius only about 38 3725 to 40 percent of that obtained with liquid hydrogen (fig. 16). If the bomber fueled with JP-4 were increased in gross weight to 300,000 pounds, its flight radius would approach about 60 percent of that shown for the 130,000-pound, hydrogen-fueled bomber.

Subsonic Reconnaissance Airplane

The same flight plan (fig. 11) was chosen for the subsonic reconnaissance airplane as was used for the subsonic bomber. Other assumptions regarding aerodynamic characteristics, engine, and structural weights were held the same in both bomber and reconnaissance airplanes. The design of the reconnaissance airplane differs from that of the bomber only because the 10,000-pound bomb load is eliminated. The fixed-equipment weight of 5000 pounds was held the same. The characteristics of the airplane for a target altitude of 80,000 feet are shown in table II.

Omission of the bomb load enabled reduction of the aircraft weight to 75,000 pounds, achieving a flight radius of over 5800 nautical miles at a target altitude of 80,000 feet (fig. 17).

If airplane gross weight at take-off were increased to about 88,000 pounds by the addition of drop tanks, the flight radius with a target altitude of 80,000 feet increases to over 7000 nautical miles.

Flight radius for this airplane may also be increased by increasing normal gross weight. If airplane weight is increased from 75,000 to 130,000 pounds, flight radius increases (from 5800) to 6400 nautical miles (fig. 18). If it is desired that the fixed-equipment weight be 15,000 pounds instead of 5000 pounds, airplane performance and gross weight will be about the same as that of the subsonic bomber.

Supersonic Bomber

The problem established for the supersonic bomber was that of determining gross weight and general configuration of a liquid-hydrogen-fueled airplane that would carry a bomb load of 10,000 pounds and a fixed equipment load of 5000 pounds at supersonic speeds for a distance of 1500 nautical miles. The flight path of the supersonic bomber is shown in figure 19. The airplane climbs at subsonic speed to about 40,000 feet altitude. It accelerates there to the design flight Mach number of 2.0, and then completes the climb, at the design speed, to the initial cruise altitude of 70,000 feet. The airplane climbs steadily during cruise out, at a constant Mach number of 2.0, until it reaches the target at an altitude of 75,000 feet. After dropping the bomb load, the return is also made at Mach 2.0 with steady climb to near 80,000 feet before reaching the base.

The general airplane configuration to fulfill this mission is shown in figure 20. Some of the general assumptions and results of calculations are presented in table II. All of the fuel is contained in tanks in the fuselage. The airplane has a straight wing with aspect ratio of 3 and taper ratio of 2. In order to gain high aerodynamic efficiency, wing thickness ratio is 3 percent, which results in relatively high wing weight. Similarly, fuselage fineness ratio is 14, which results in low fuselage drag but relatively high fuselage weight. The saving in engine thrust requirement and, hence, in engine weight that results from increasing aerodynamic efficiency more than compensates for the increase in wing and fuselage weight.

This airplane is powered by six turbojet engines of type B, which is illustrated in figure 21. The assumed engine characteristics and performance are presented in table IV. The excellent combustion characteristics of liquid hydrogen and high air-flow capacity of the transonic compressor were exploited in this engine to obtain a low over-all engine-nacelle frontal area. The engine is not equipped with an afterburner. Because of the excellent refrigeration capacity of liquid hydrogen, a cooled turbine with an inlet-gas temperature of 2500° R was assumed. Details of a possible turbine cooling system are discussed in a later section.

The schematic arrangement of the components as they would fit within the nacelle is shown in figure 21. The compressor, which has a sea-level static pressure ratio of 6.2, has a pressure ratio of 4.1 and an equivalent air flow of 35 pounds per second per square foot at the design flight Mach number of 2.0. Combustor-inlet velocity is about 200 feet per second at design flight conditions. For these conditions, a two-stage turbine is necessary in order to obtain a turbine that will fit within the nacelle diameter, which has been determined by the other engine components. Sea-level specific weight of the engine was assumed to be 0.16. This relatively low specific weight could be assumed because of the high turbine-inlet temperature (2500° R). Also contributing to the low weight are the relatively high specific air flow and use of a transonic compressor and short combustors. Inasmuch as take-off and climb present no problem for this airplane, the engine can be designed principally for the design flight condition with little regard for off-design operation at take-off.

This supersonic bomber, with a gross weight of 130,000 pounds, has a 1545 nautical mile flight radius at a target altitude of 75,000 feet, when powered with six turbojet engines, each with a compressor tip diameter of about 42 inches. The effect of target altitude on radius is shown in figure 22. If the airplane were designed for a target altitude of 80,000 feet, larger or more engines are, of course, required and the flight radius would be decreased to 1280 miles.

Calculations were also made to determine the radius that could be obtained using JP-4 fuel. The same basic equations and assumptions were used to compute airplane structural weight and aerodynamic efficiency as were used for computing the performance with liquid hydrogen as the fuel. The results of these calculations (fig. 22) also show the effect of target altitude on flight radius. At all target altitudes the radius with JP-4 is less than 55 percent of that with liquid hydrogen.

The effect on flight radius of changing gross weight of the hydrogen-fueled airplane is shown in figure 23 for a target altitude of 75,000 feet. Increasing gross weight 54 percent (from 130,000 to 200,000 lb) increases flight radius only 6 percent (from 1545 to 1630 miles).

Supersonic Reconnaissance Airplane

The problem established for the supersonic reconnaissance airplane was to determine the general configuration and flight radius of a liquid-hydrogen-fueled airplane with a gross weight of 75,000 pounds that has a target altitude of 80,000 feet and a flight Mach number of 2.5. These flight conditions are more stringent than the 75,000 feet target altitude and 2.0 flight Mach number of the supersonic bomber. The airplane climbs at subsonic speed to near 40,000 feet altitude, accelerates to the design flight Mach number of 2.5, and then completes the climb at the design speed to the initial cruise altitude of about 70,000 feet. The airplane climbs steadily during cruise out at a constant Mach number of 2.5, until it reaches the target at an altitude of 80,000 feet. The return is made at a nearly constant altitude of 80,000 feet.

The airplane configuration is similar to that of the supersonic bomber. The airplane is powered by afterburning engines designed for a flight Mach number of 2.5 (engine C in table IV). The general arrangement of this engine is illustrated in figure 24. Like engine B, this engine has a cooled turbine with a turbine-inlet temperature of 2500° R. Also illustrated in figure 24 is a turbine-cooling arrangement. Air that is bled from the compressor exit is cooled by liquid hydrogen in the heat exchanger. The cooled air enters the turbine disk through the turbine inner cone and struts. After cooling the hollow blades, the air is discharged from the blade tips into the gas stream. The stator blades are cooled directly by hydrogen as it flows to the primary combustor after leaving the heat exchanger. The cooling system shown is one of many that may be devised with hydrogen as the coolant.

Engine C is shown with a one-stage turbine. Smaller nacelle diameter could be obtained if a two-stage turbine were used, except that the afterburner-inlet velocity would be prohibitive. Because the frontal area of a two-stage turbine could not be utilized, a one-stage turbine was used in order to reduce the cooling-air flow required. For the nacelle frontal area as set by the diameter of the one-stage turbine, the afterburner-inlet velocity is approximately 525 feet per second. Each of the components of engine C utilize the nacelle frontal area to obtain minimum length and should therefore result in both a short and light-weight engine. The sea-level static engine pressure ratio of engine C is 4.3. At the design flight Mach number of 2.5, the pressure ratio is 2.5. The specific weight of this engine at take-off was assumed to be 0.18 unaugmented but including the afterburner weight.

For a gross weight of 75,000 pounds and target altitude of 80,000 feet, a radius of 1345 miles was calculated (table II). Four engines (type C) each having a compressor tip diameter of 33 inches are required. The effect of target altitude on flight radius is shown in figure 25. Increasing target altitude from 80,000 to 90,000 feet decreases the radius from 1345 to 1050 miles.

The flight radius of the airplane when powered by the nonafterburning engines B and flying at a Mach number of 2.0 is also shown. At target altitudes below 85,000 feet, the airplane was calculated to have a longer flight radius when powered with engine B at a flight Mach number of 2.0 than when powered with the afterburning engine C at a flight Mach number of 2.5. At a target altitude of 80,000 feet and Mach number of 2.0, the radius is more than 1500 nautical miles with engine B. At 90,000 feet, however, the radius is decreased to 700 miles.

The effect of airplane gross weight on the flight radius of the supersonic reconnaissance airplane with engine C is shown in figure 26 for a flight Mach number of 2.5. The weight of fixed equipment in this airplane is only 6.7 percent of the 75,000-pound gross weight, so that increasing gross weight to 200,000 pounds increases the radius from 1350 to only 1500 miles. In fact, the calculations indicate that increase in gross weight above about 180,000 pounds will decrease flight radius, because of reduction in structural efficiency of the airplane.

Supersonic Fighter

The problem established for the supersonic fighter was to determine the weight and configuration of a hydrogen-fueled airplane that would cruise 500 miles at Mach 2.5, combat for 5 minutes, and return to base. It was assumed that the fixed equipment for crew, armament, navigation, and electronics weighed 3000 pounds. The flight plan for the mission is described in figure 27 for the airplane powered with turbojet engines. The airplane climbs at subsonic speeds to 40,000 feet altitude, where it accelerates to Mach 2.5. At Mach 2.5 it then climbs to 70,000 feet altitude and continues at this altitude to the combat zone where it climbs to 80,000 feet and engages in combat. After combat it returns to base at Mach 2.5 and at the altitude selected for maximum radius.

Several propulsion systems for the fighter aircraft were analyzed to determine whether one type showed outstanding advantages over another. The following propulsion-system configurations were studied:

(a) Two turbojet engines
(b) Two ram-jet engines with auxiliary turbojet
(c) Two ram-jet engines with rocket assist
(d) Two air-turbo-rocket engines

Since nacelle installations were used for all the engine systems, the schematic drawing (fig. 28) of the fighter with turbojet engine installed is generally representative of the airplane configuration for all engine installations studied. The general assumptions of the study and the results of the analysis for a cruise radius of 500 miles are shown for the aircraft and engines in tables III, IV, and V.

In the study, greatest emphasis was given to the fighter equipped with turbojet engines. The engine used, except for size, was the same turbojet engine (engine C, fig. 24 and table IV) that was discussed in the section on the Mach 2.5 reconnaissance airplane. The wing planform and thickness were also about the same as were used on the supersonic bomber and reconnaissance airplanes.

Performance of the fighter expressed in terms of gross weight as a function of combat radius is shown in figure 29. At a design combat radius of 500 nautical miles, the gross weight is 22,350 pounds for the fighter fueled with liquid hydrogen. At this same gross weight, the aircraft fueled with JP-4 has a radius of 285 nautical miles. The results show that a radius of 500 nautical miles and a combat ceiling of 80,000 feet cannot be attained with a JP-4-fueled fighter at Mach 2.5 within the assumptions of this study. A radius of 700 nautical miles can be achieved with a hydrogen-fueled fighter weighing slightly more than 40,000 pounds.

In arriving at the weights just presented for both the hydrogen- and JP-4-fueled aircraft, the engines were sized to provide level flight at 80,000 feet altitude with take-off gross weight. If the fuel burned in climb and cruise out to combat is taken into account, the engine thrust is adequate to provide a combat maneuver of only 1.1 g without loss of airspeed or altitude in the maneuver. If it is required that both speed and altitude be maintained in maneuvers exceeding 1.1 g, additional engine thrust is required for the airplane. Since wing loading of the airplane at combat is 59 pounds per square foot, and the combat lift coefficient is only 0.25, the wings are capable of sustaining high combat-maneuver loadings. The effect on aircraft gross weight due to the additional engine weight required to hold speed and altitude with different maneuver loads is shown in figure 30. The curves indicate little hope of a fighter of any weight accomplishing more than a 1.5 g maneuver at 80,000 feet without losing speed. Exchange of speed for altitude, as in the "zoom" technique, eliminates the need for the excess engine weight and may be a practical combat practice.

For the fighter with a combat altitude of 80,000 feet and maneuverability of 1.1 g, the installed turbojet engine weight is more than 25 percent of the airplane gross weight. Other propulsion-system configurations ((b), (c), and (d)) were therefore substituted to determine if these lighter engines would reduce the gross weight of the fighter airplane. The general assumptions of the engines used in these propulsion-system configurations are given in tables IV and V. Configuration (b), designed for a flight Mach number of 2.5, is a combination of turbojet engine C and the ram-jet engine designed for a Mach number of 2.5. The turbojet component is only large enough to provide adequate take-off, climb, and acceleration performance, but it operates at full power throughout the flight. A schematic diagram of the ram-jet engine is given in figure 31. In the ram-jet engine as in the turbojet, use of hydrogen fuel reduces requirements in combustor size. The ram-jet engine weight was assumed to be 150 pounds per square foot of combustor area.

In configuration (c), the turbojet component of configuration (b) is replaced with a rocket engine to provide thrust during climb and acceleration. Because the ram-jet engine is more efficient at the higher flight speeds, the design Mach number was increased to 3.0. The rocket propellant assumed is liquid hydrogen and oxygen, with a specific impulse of 360 pound-seconds per pound of fuel.

The air-turbo-rocket engine configuration (d) is shown diagramatically in figure 32. Operation of the air-turbo-rocket engine can be described simply as follows. A turbine driven by exhaust gases from hydrogen-oxygen rockets drives a one-stage compressor. Turbine-inlet temperature is held to values near 2000° R, by using fuel-rich mixtures in the rocket chamber. The excess of fuel in the turbine exhaust is mixed with the compressor air and burned in an afterburner. The exhaust gases are discharged to provide thrust. Additional hydrogen may be added and burned in the afterburner to provide additional thrust. When maximum thrust is not required, propellant flow to the rocket and compressor pressure ratio are reduced. For maximum engine efficiency at high flight speeds, compressor pressure ratio is reduced to approximately 1 and the engine is operated like a ram jet. The air-turbo-rocket engine therefore provides essentially ram-jet engine performance for cruise in combination with a high thrust capability for airplane take-off, climb, and acceleration. The weight of the air-turbo-rocket was assumed to be 294 pounds per square foot of compressor-tip area.

The performance of airplanes with the various propulsion-system configurations are given in table III and the airplane gross weights are indicated on figure 29 for a radius of 500 miles. All the airplanes have about the same gross weight for a 500-mile radius and a combat altitude of 80,000 feet. None of the propulsion-system configurations shows large advantages over the others.

Substitution of the ram-jet engine for part of the turbojet engine (engine C) at a flight Mach number of 2.5 reduces the fighter gross weight to about 20,500 pounds. In this combination, the ram-jet combustor area is about two to three times the turbojet compressor area. If the airplane is equipped with even smaller turbojet engines and compensatingly larger ram-jet engines, the lower take-off thrust gives poor climb and acceleration performance of the airplane and results in increased gross weight.

Take-off gross weight of the rocket-boosted ram-jet configuration is about 30,500 pounds. A large part of this weight, however, is rocket propellant and at burn-out of the rocket (Mach number, over 2.0) airplane weight is about 21,500 pounds. Thrust of the rocket engine during boost is about 25,600 pounds. The weight of this combination could be reduced by carrying the rocket propellant for boosting in external drop tanks. In the present configuration, both the rocket engine and propellant tanks are carried throughout the flight, and increase both the weight and fuselage volume.

Gross weight with the air-turbo-rocket engine is about 24,000 pounds. Although this configuration is about 2500 pounds heavier than the rocket-boosted ram-jet configuration at rocket burn-out, it is about 6500 pounds lighter than this configuration at take-off. The heavier weight of the air-turbo-rocket engine is more than compensated for by the lower fuel consumption during climb and acceleration.

CONCLUDING REMARKS

This analysis shows that within the state of the art and the progress anticipated, aircraft designed for liquid-hydrogen fuel may perform several important military missions that comparable aircraft using hydrocarbon (JP-4) fuel cannot accomplish. These include (1) subsonic bomber and reconnaissance flights of over 5500 nautical mile radius without refueling with an altitude over the target of 80,000 feet; (2) supersonic bomber (Mach 2.0) and reconnaissance flights (Mach 2.5) of about 1500 nautical mile radius with altitudes over the target of 75,000 feet for the bomber and 80,000 feet for the reconnaissance aircraft; (3) supersonic fighter aircraft with a combat radius (Mach 2.5) of 700 nautical miles and a combat altitude of 80,000 feet.

For missions of shorter radius, where the desired distance and altitude can be obtained with either liquid hydrogen or JP-4 fuel, the take-off gross weights of the aircraft fueled with hydrogen are one-half or less than those of the JP-4-fueled aircraft. For high-altitude aircraft and missile missions other than those investigated in this analysis, it may be expected that similar gains in radius and reductions in gross weight will be demonstrated when liquid hydrogen is used as fuel.

The performance calculated for the various missions will, of course, not be realized unless the assumptions regarding engine weight, aerodynamic efficiency, tank weight, structural weight, etc. can be realized in the aircraft and its components. Substantial applied research and development effort will be required in many technical fields to achieve the goals outlined.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, April 1, 1955.

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TABLE I. - PHYSICAL PROPERTIES OF HYDROGEN

Heating value, Btu/lb	51,571
Density, liquid at 1 atm, 37° R, lb/cu ft	4.42
Density, vapor at 1 atm, 492° R, lb/cu ft	0.0056
Boiling point at 1 atm, $^{\circ}$R	37
Melting point, $^{\circ}$R	25.2
Critical temperature, $^{\circ}$R	59.6
Critical pressure, lb/sq in. abs	188
atm	12.8
Critical density, lb/cu ft	1.95
Latent heat, melting, Btu/lb	25.2
Latent heat, vaporization at 1 atm, Btu/lb	194
Conversion from ortho to para structure, Btu/lb	220
Viscosity, liquid, centipoises	0.014
Viscosity, vapor, centipoises at T $^{\circ}$K	$0.0084\left(\frac{T}{273.1}\right)^{0.695}$
Specific heat, vapor at 519° R, Btu/(lb)($^{\circ}$R)	3.4
Ratio of specific heats, vapor at 519° R	1.41

TABLE II. - CHARACTERISTICS AND PERFORMANCE OF BOMBER AND RECONNAISSANCE AIRPLANES

	Airplane					
	Subsonic bomber		Subsonic reconnaissance		Supersonic bomber	Supersonic reconnaissance
	Without drop tanks	With drop tanks	Without drop tanks	With drop tanks		
Cruise Mach number	0.75	0.75	0.75	0.75	2.0	2.5
Initial cruise altitude, ft	69,900	68,000	69,600	66,300	71,500	67,500
Target altitude, ft	80,000	79,300	80,000	79,000	75,000	80,000
Gross weight, lb	130,000	142,760	75,000	87,760	130,000	75,000
Payload weight, lb	10,000	10,000	0	0	10,000	0
Fixed weight, lb	5,000	5,000	5,000	5,000	5,000	5,000
Total structural weight, lb	48,200	50,380	26,650	28,830	46,100	29,200
Total installed engine weight, lb	23,450	23,450	13,950	13,950	29,000	13,600
Fuel tank weight, lb	5,650	7,030	3,850	5,230	5,200	3,550
Fuel weight, lb	37,700	46,900	25,550	34,750	34,700	23,650
Engines:	A	A	A	A	B	C
Number	4	4	4	4	6	4
Compressor diameter, each engine, in.	45.7	45.7	34.4	34.4	41.8	33.2
Rated sea-level thrust, each engine, lb	25,400	25,400	14,400	14,400	27,400	16,300
Cruise specific fuel consumption based on net thrust minus nacelle drag, (lb/hr)/lb	0.381	0.381	0.382	0.382	0.571	0.703
Wing:						
Area, sq ft	6,500	6,500	3,750	3,750	2,600	1,150
Sweep angle, deg	31	31	31	31	0	0
Aspect ratio	13	13	13	13	3	3
Average section thickness ratio	0.12	0.12	0.12	0.12	0.03	0.03
Taper ratio	2	2	2	2	2	2
Empennage:						
Area, sq ft	1,625	1,625	937	937	780	345
Fuselage:						
Length, ft	160	160	147	147	194	172
Diameter, ft	12.5	12.5	11.5	11.5	13.8	12.3
Lift coefficient, initial cruise	0.53	0.53	0.54	0.54	0.20	0.14
Lift-drag ratio, airplane less engine nacelles, initial cruise	29.6	27.9	27.8	25.4	5.53	4.33
Radius, nautical miles	5,400	6,280	5,860	7,290	1,545	1,345

TABLE III. - CHARACTERISTICS AND PERFORMANCE OF FIGHTER AIRPLANES

	Engine			
	Turbo-jet C	Ram-jet plus turbo-jet C	Ram-jet plus rocket	Air-turbo-rocket
Cruise:				
Mach number	2.5	2.5	3.0	2.8
Initial altitude, ft	70,600	71,000	74,200	77,600
Combat:				
Mach number	2.5	2.5	3.0	2.8
Altitude, ft	80,000	80,000	80,000	80,000
Time, min	5	5	5	5
Maneuverability, g's	1.1	1.1	1.3	1.2
Gross weight, lb	22,350	20,400	30,700	23,940
Fixed weight, lb	3,000	3,000	3,000	3,000
Total structural weight, lb	9,240	7,100	10,300	10,900
Total installed engine weight, lb	5,730	3,820	2,550	3,060
Fuel tank weight, lb	570	840	[a]1,940	910
Fuel weight, lb	3,810	5,640	[b]12,910	6,070
Engines:				
Turbojet and air-turbo-rocket				
Number	2	1		2
Compressor diameter, each engine, in.	30.5	27.8		30.9
Rated sea-level thrust, each engine, lb	13,650	11,350		8,490
Ram-jet				
Number		2	2	
Combustor diameter, each engine, in.		29.7	34.2	
Rocket				
Rated sea-level thrust, lb			25,600	
Cruise specific fuel consumption based on net thrust minus nacelle drag, (lb/hr)/lb	0.694	0.770	0.863	0.849
Wing:				
Area, sq ft	344	272	253	282
Sweep angle, deg	0	0	0	0
Aspect ratio	3	3	3	3
Average section thickness ratio	0.035	0.035	0.035	0.035
Taper ratio	2	2	2	2
Empennage:				
Area, sq ft	103	82	76	85
Fuselage:				
Length, ft	88	89	98	98
Diameter, ft	7.3	7.4	8.2	8.2
Cruise lift coefficient	0.16	0.19	0.17	0.22
Cruise lift-drag ratio, airplane less engine nacelles	3.7	3.4	3.1	3.3
Combat radius, nautical miles	500	500	500	500

[a]Includes oxidant tank

[b]Includes oxidant.

[redacted]

TABLE IV. - CHARACTERISTICS AND PERFORMANCE OF

TURBOJET ENGINES

	Engine		
	A	B	C
Design flight Mach number	0.75	2.0	2.5
Rated turbine-inlet temperature, $^{\circ}R$	2000	2500	2500
Inlet total-pressure ratio at design Mach number	0.95	0.91	0.82
Compressor:			
Rated pressure ratio at sea-level static conditions	6.2	6.2	4.3
Rated pressure ratio at design Mach number	8.0	4.1	2.5
Rated equivalent air flow at design Mach number, (lb/sec)/sq ft	37.5	35	25
Primary combustor:			
Reference velocity, ft/sec	110	200	180
Pressure at 80,000 ft altitude, atm	0.30	0.82	0.96
Turbine:			
Number of stages	2	2	1
Afterburner:	None	None	
Inlet velocity, ft/sec			525
Pressure at 80,000 ft altitude, atm			0.53
Exit temperature, $^{\circ}R$			3500
Rated performance at design Mach number based on net thrust minus nacelle drag:			
Specific air consumption, (lb/hr)/lb	67.1	70.9	48.7
Specific fuel consumption (JP-4 fuel), (lb/hr)/lb	1.16	1.58	2.30
Sea-level rated specific weight, lb/lb	0.20	0.16	[a]0.18

[a]Unaugmented but including afterburner weight.

TABLE V. - CHARACTERISTICS AND PERFORMANCE OF ROCKET, RAM-JET, AND AIR-TURBO-ROCKET ENGINES

	Engine			
	Rocket	Ram jet		Air-turbo-rocket
Design flight Mach number	-----	2.5	3.0	2.8
Inlet total-pressure ratio at design Mach number	-----	0.75	0.60	0.76
Combustor:				
Inlet Mach number	-----	0.2	0.2	0.15
Pressure at 80,000 ft altitude, atm	-----	0.35	0.58	0.54
Exit temperature, $^{\circ}R$	-----	3900	3950	3500
Performance at design Mach number based on net thrust minus nacelle drag:				
Specific air consumption, (lb/hr)/lb	-----	45.0	46.0	51.7
Specific fuel consumption (JP-4 fuel), (lb/hr)/lb	-----	2.80	2.69	2.43
Sea-level specific impulse (hydrogen-oxygen), lb-sec/lb	360	----	----	----
Specific weight:				
Lb motor/lb thrust at sea level	0.025			
Lb engine/sq ft combustor area		150	150	
Lb engine/sq ft compressor area				294

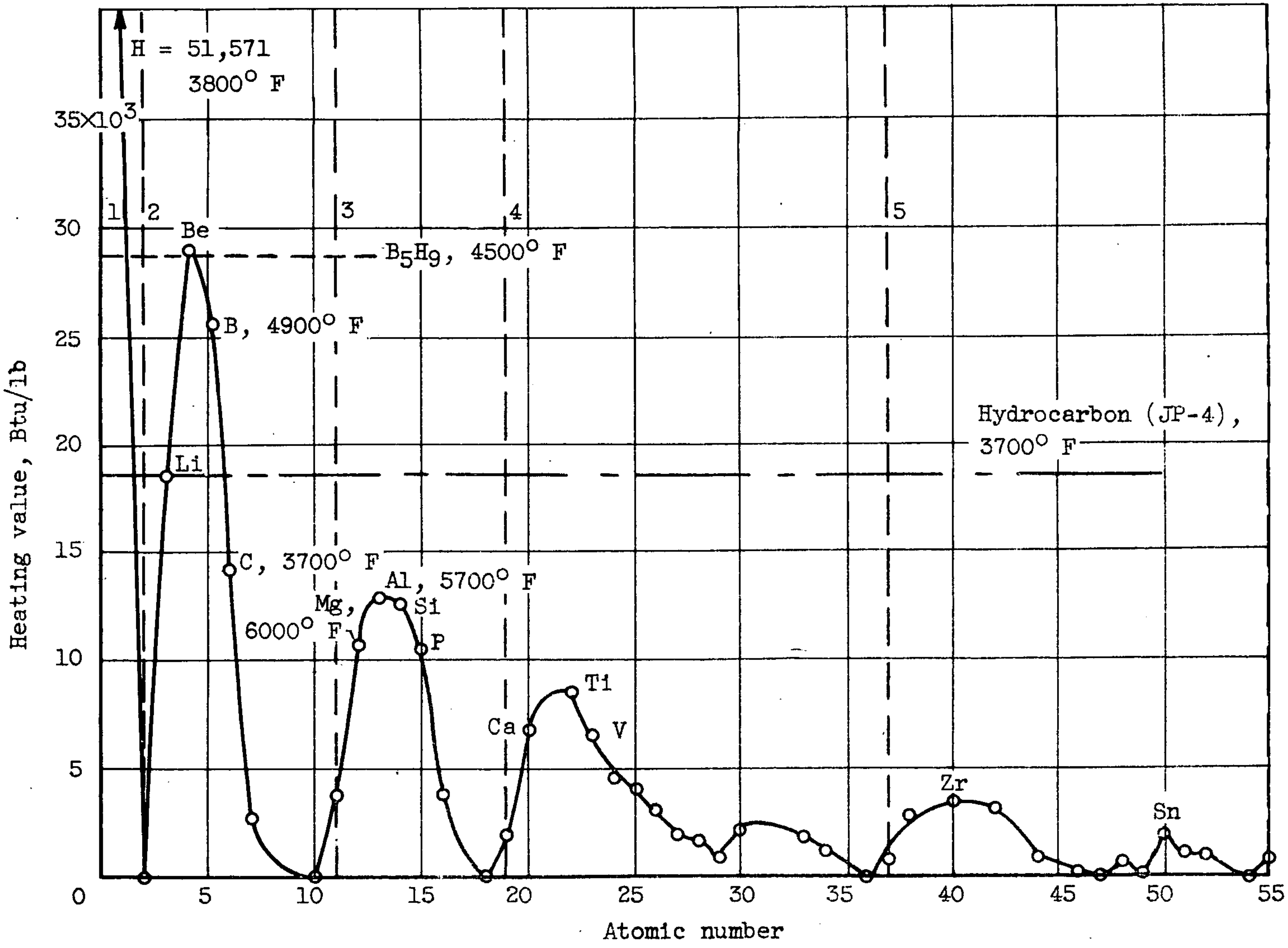

Figure 1. - Heat of combustion of elements classified according to atomic number.

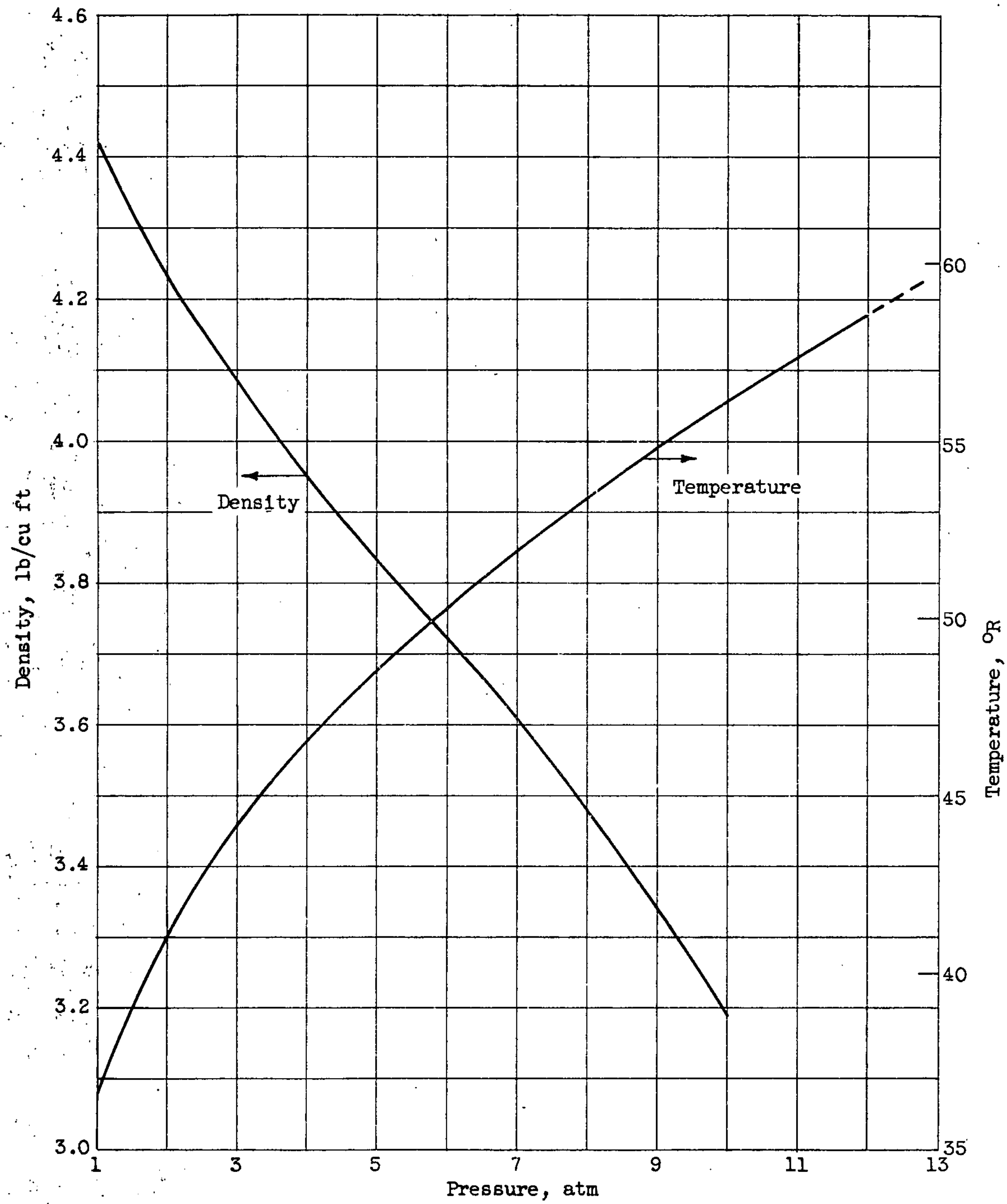

Figure 2. - Temperature-pressure-density relation of saturated liquid hydrogen.

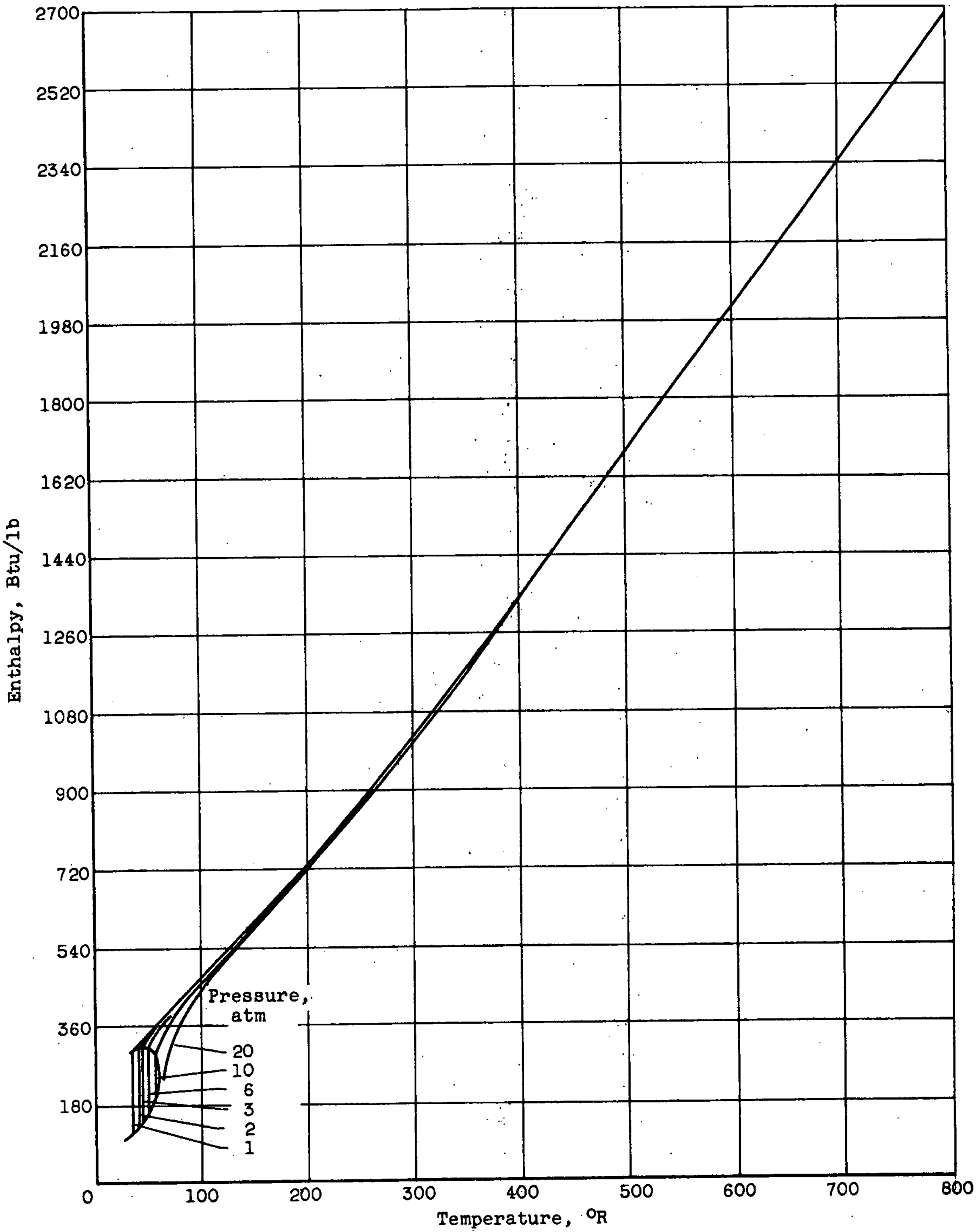

Figure 3. - Enthalpy of hydrogen.

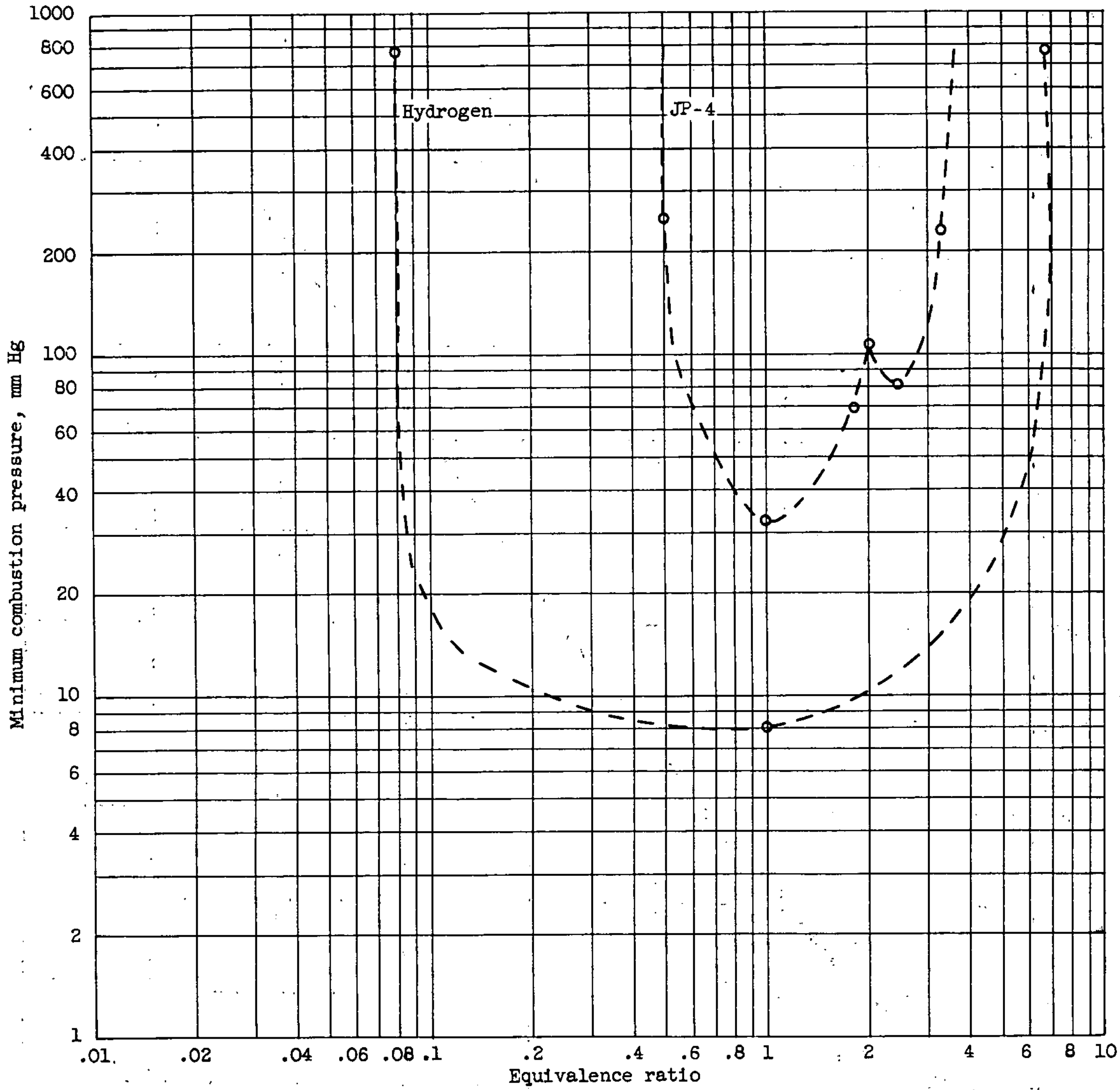

Figure 4. - Estimated minimum combustion pressures for hydrogen and JP-4 fuel.

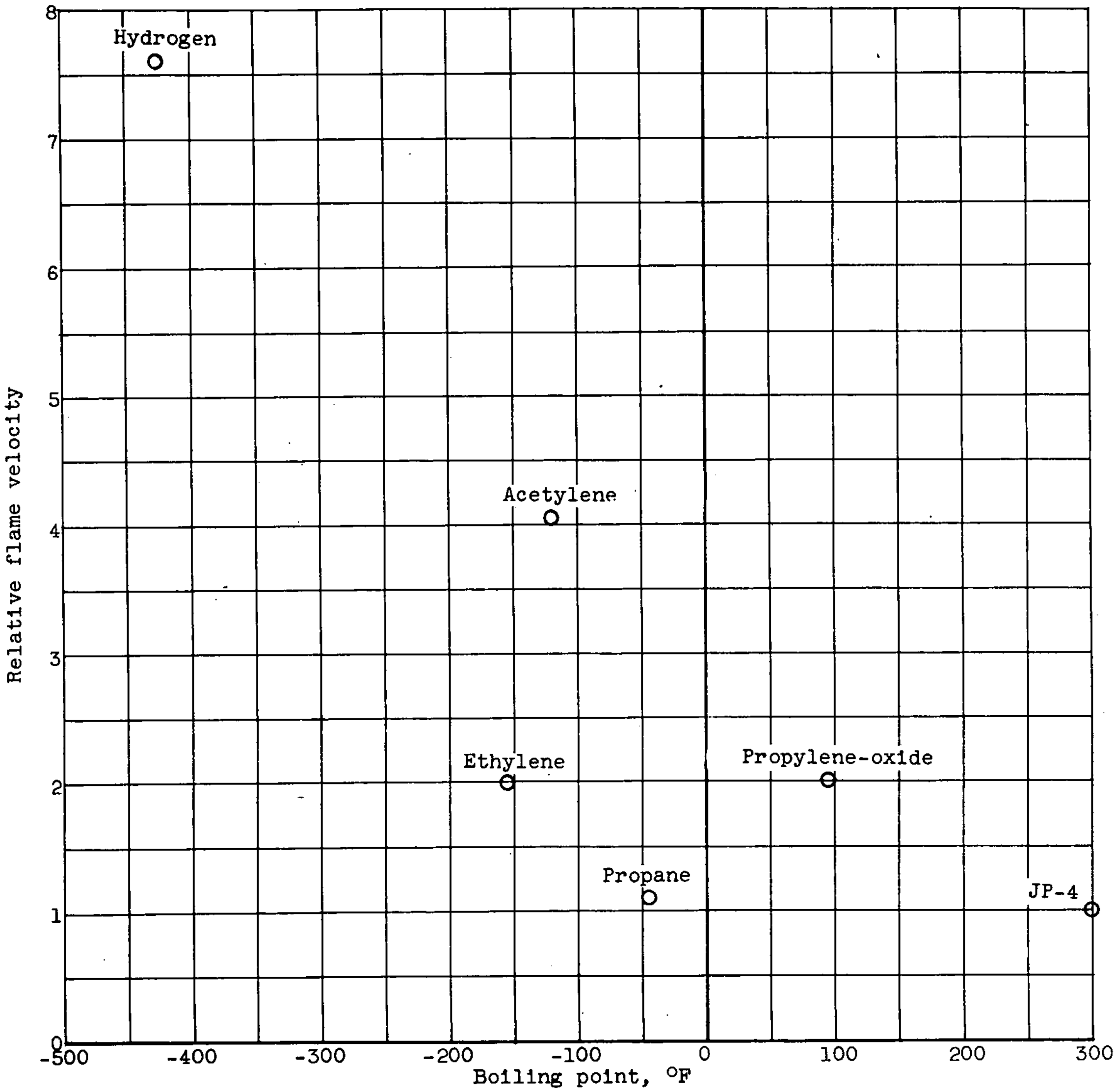

Figure 5. - Laminar flame velocity of fuels relative to JP-4.

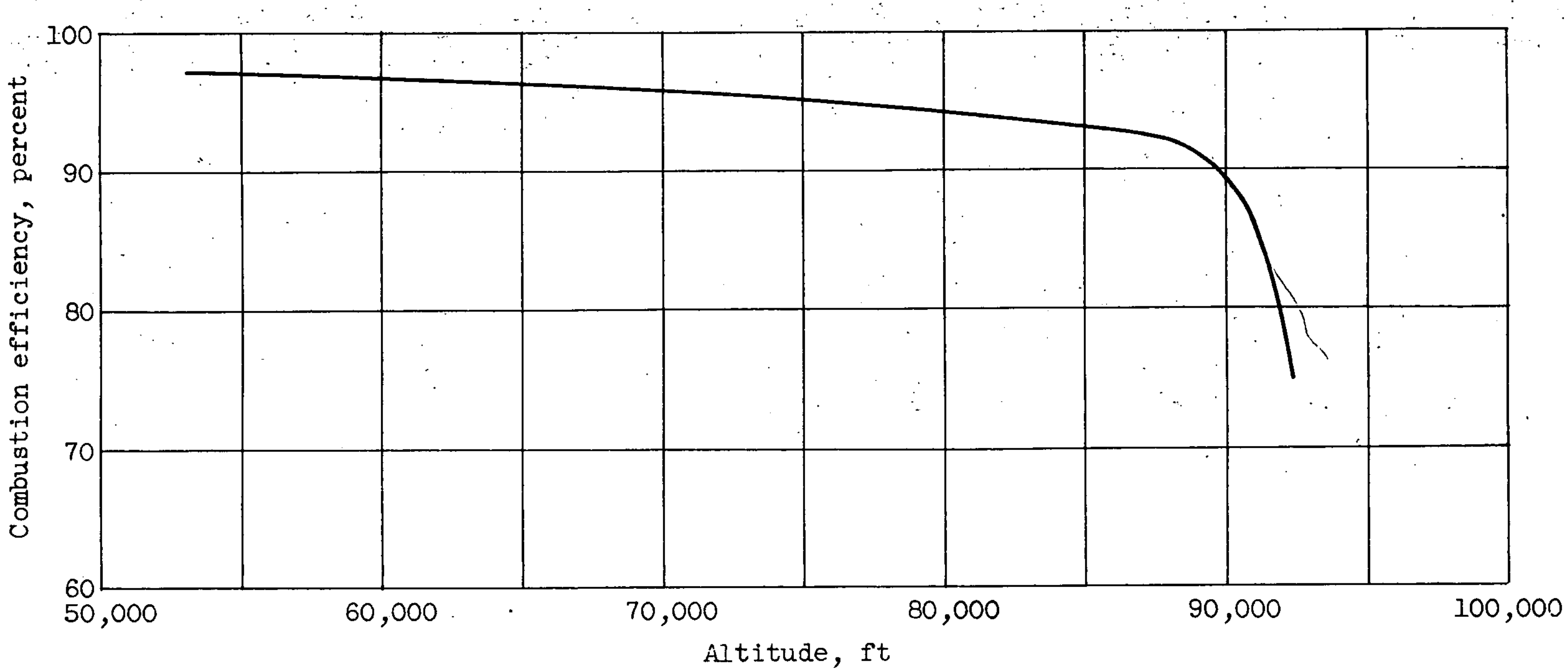

Figure 6. - Effect of altitude on combustion efficiency of hydrogen. Flight Mach number, 0.75; compressor pressure ratio, 5.

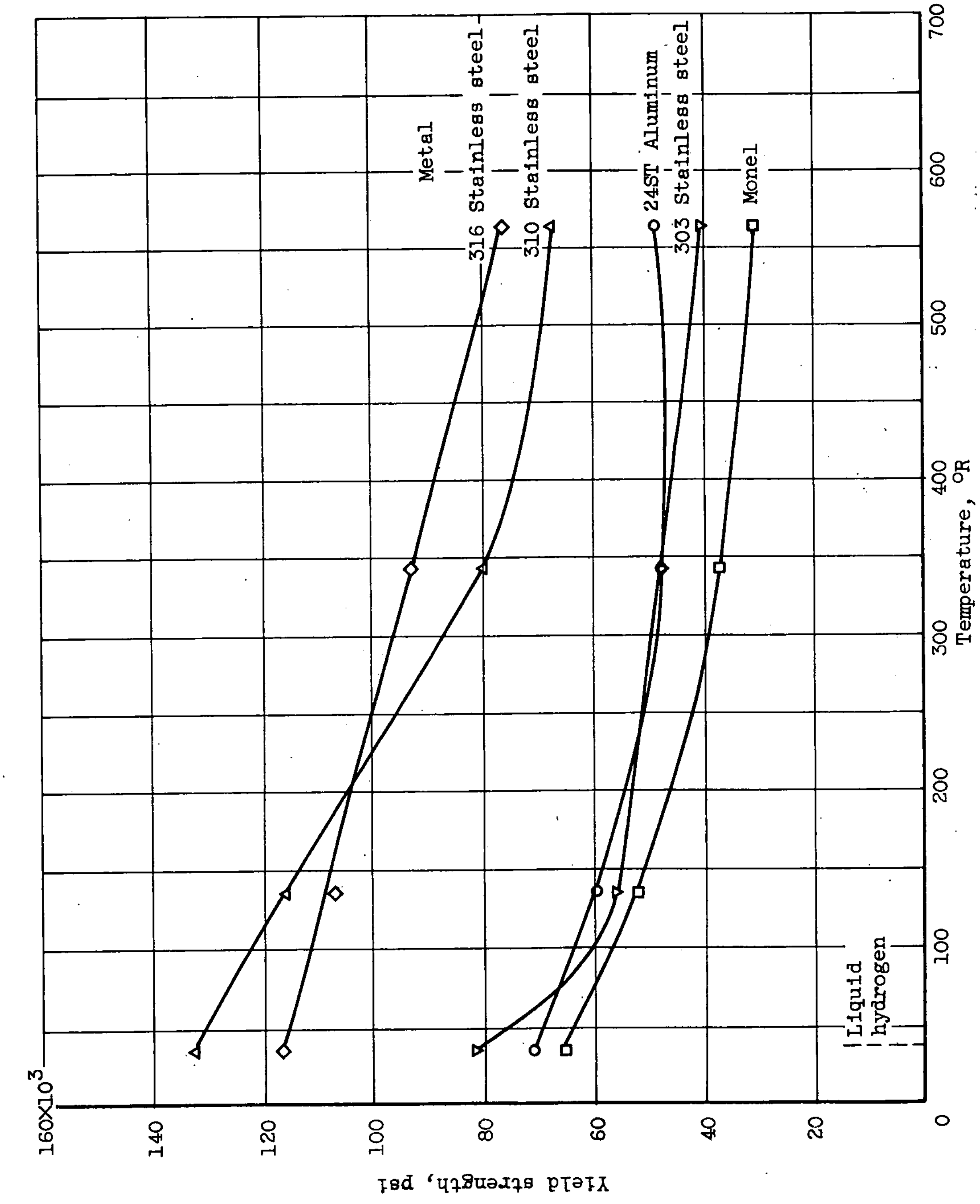

Figure 7. - Effect of temperature on strength of several metals.

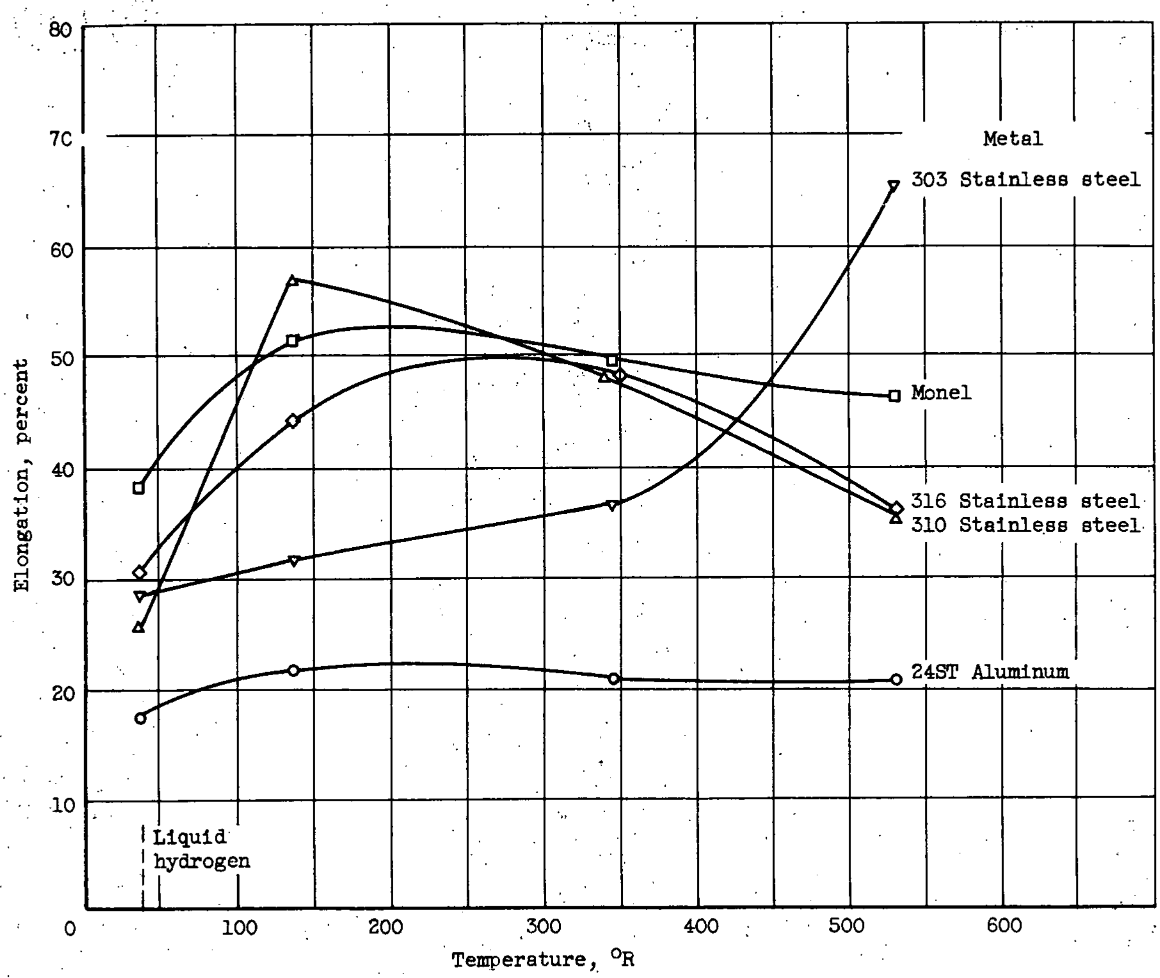

Figure 8. - Effect of temperature on ductility of several metals.

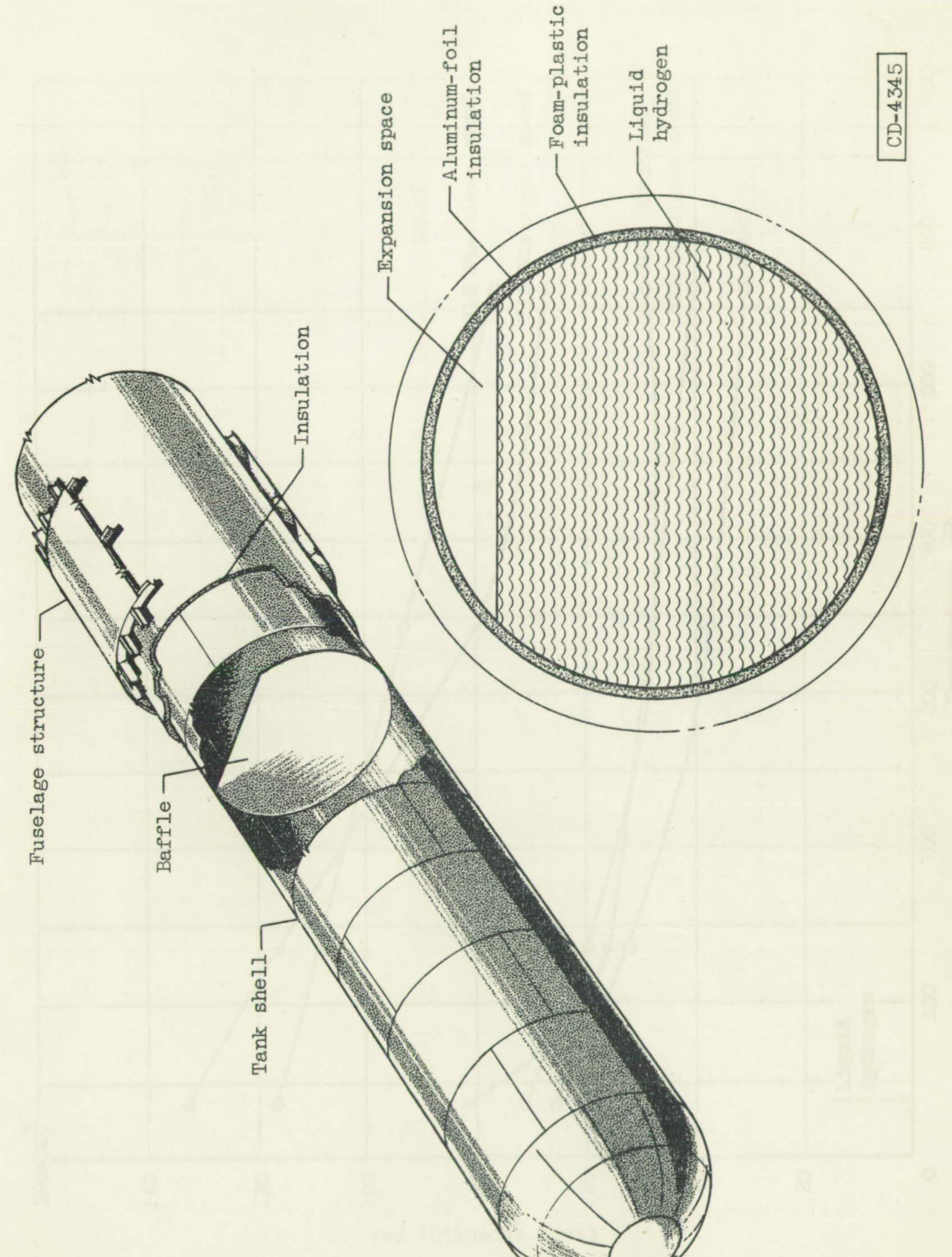

Figure 9. - Liquid hydrogen fuel tank.

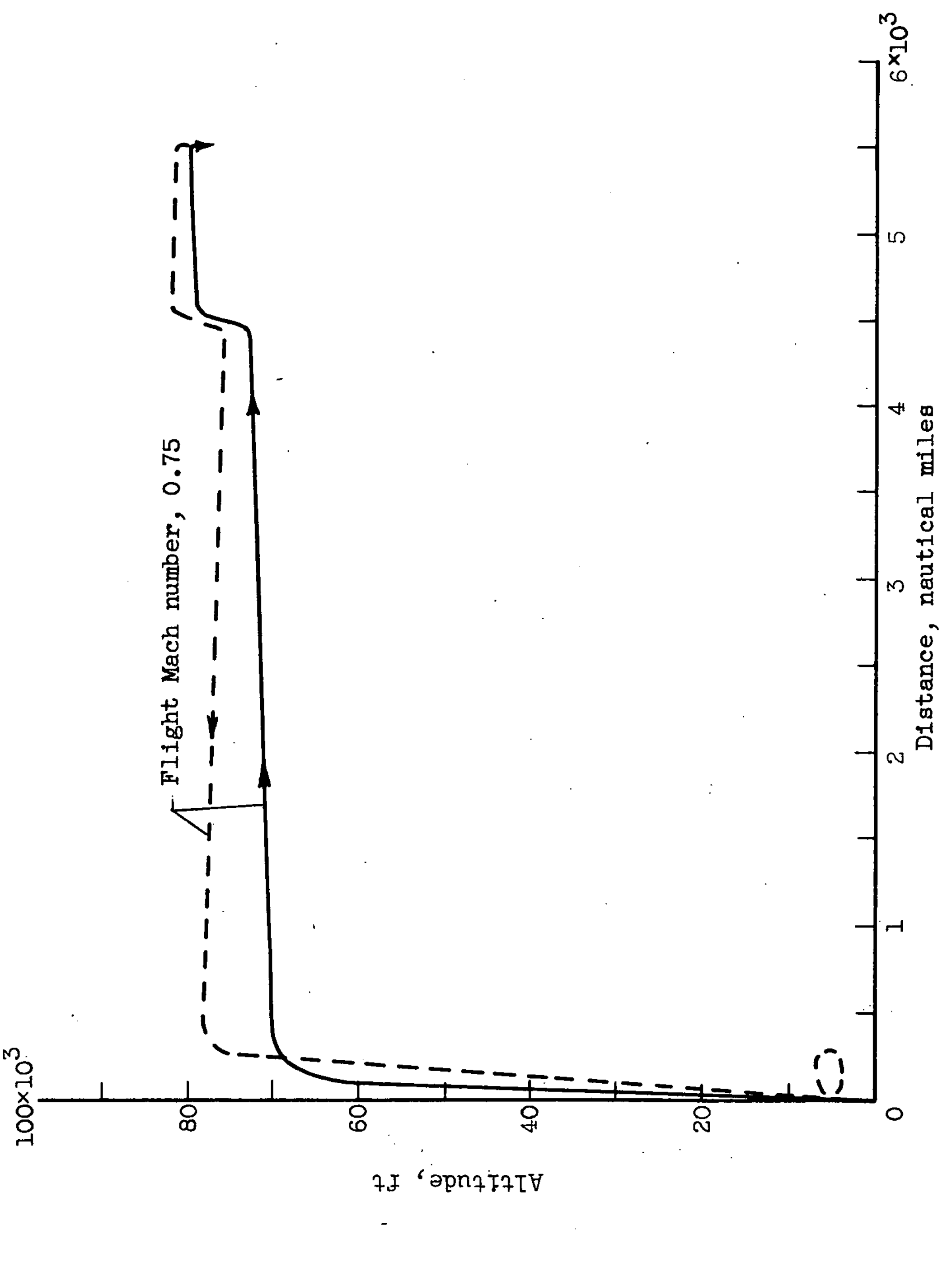

Figure 11. - Flight path of subsonic aircraft.

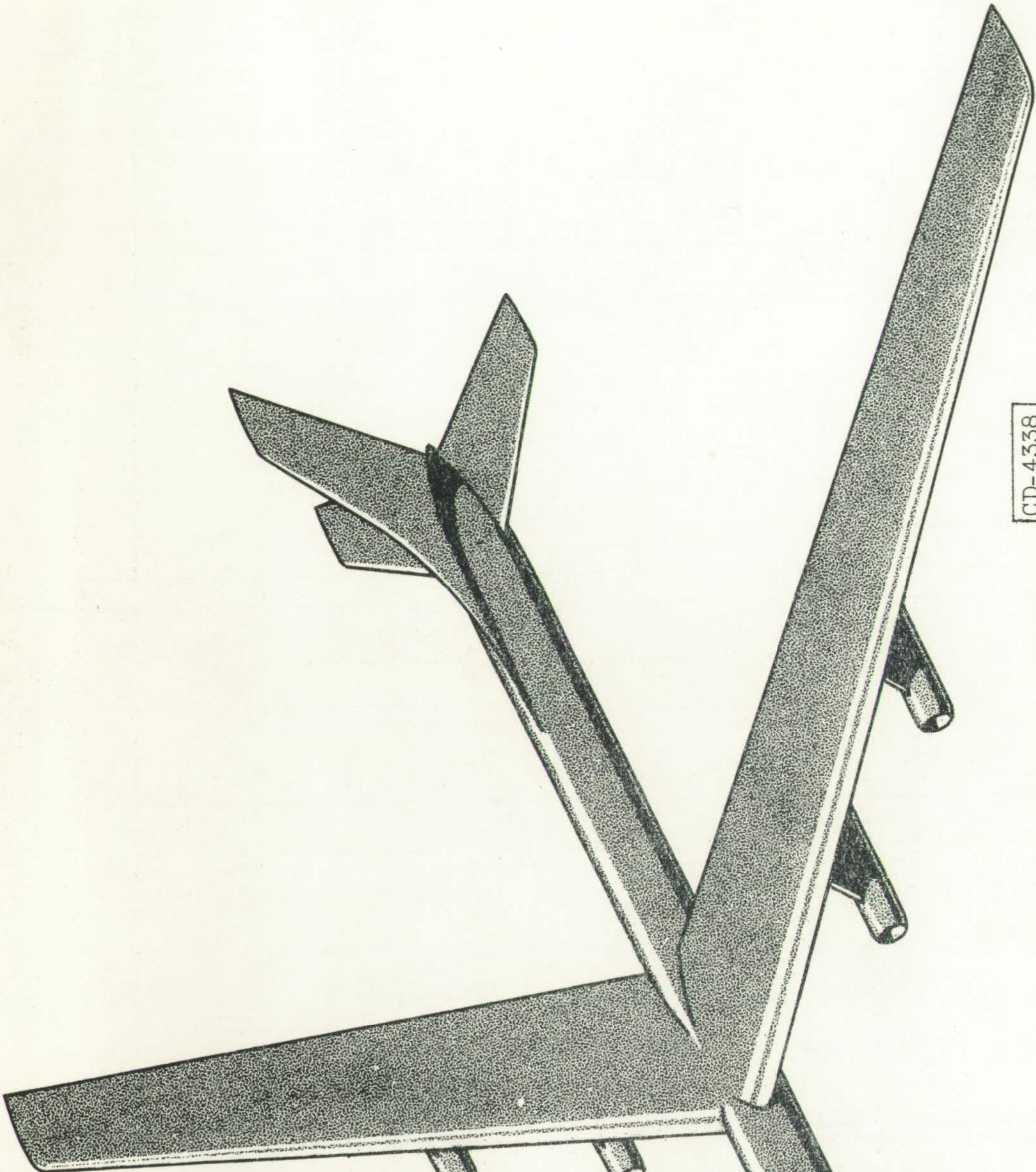

Figure 12. Subsonic bomber. Flight Mach number, 0.75; target altitude, 80,000 feet.

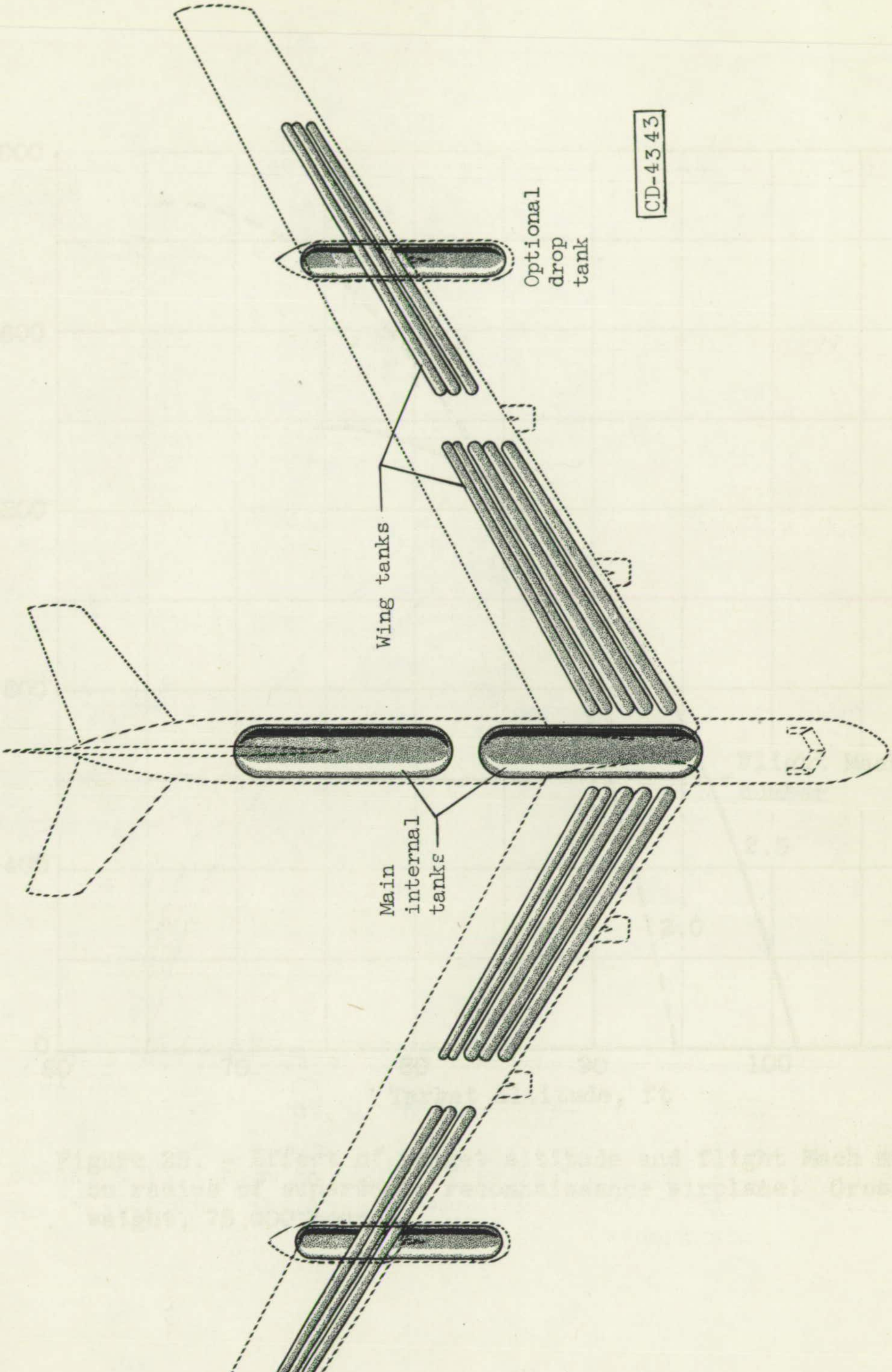

Figure 13. - Aircraft tank installation.

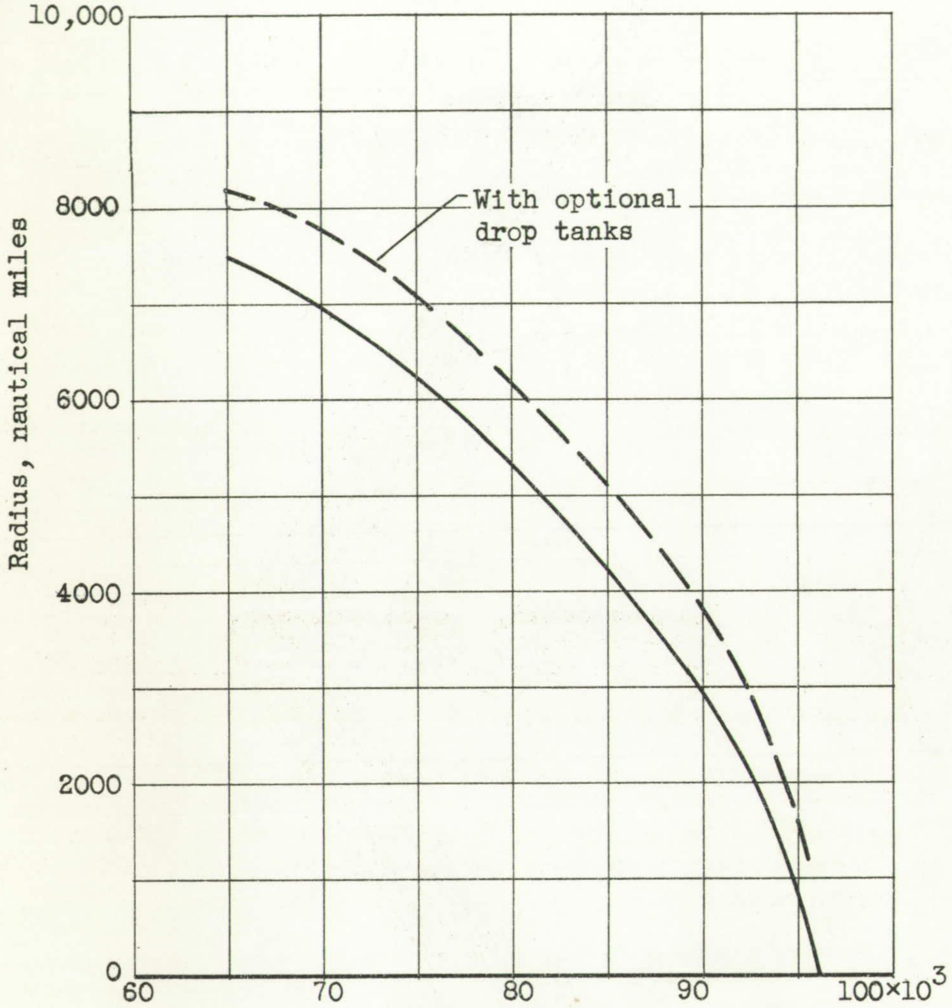

Figure 14. - Effect of altitude on radius of subsonic bomber with and without external drop tanks. Flight Mach number, 0.75; engine A.

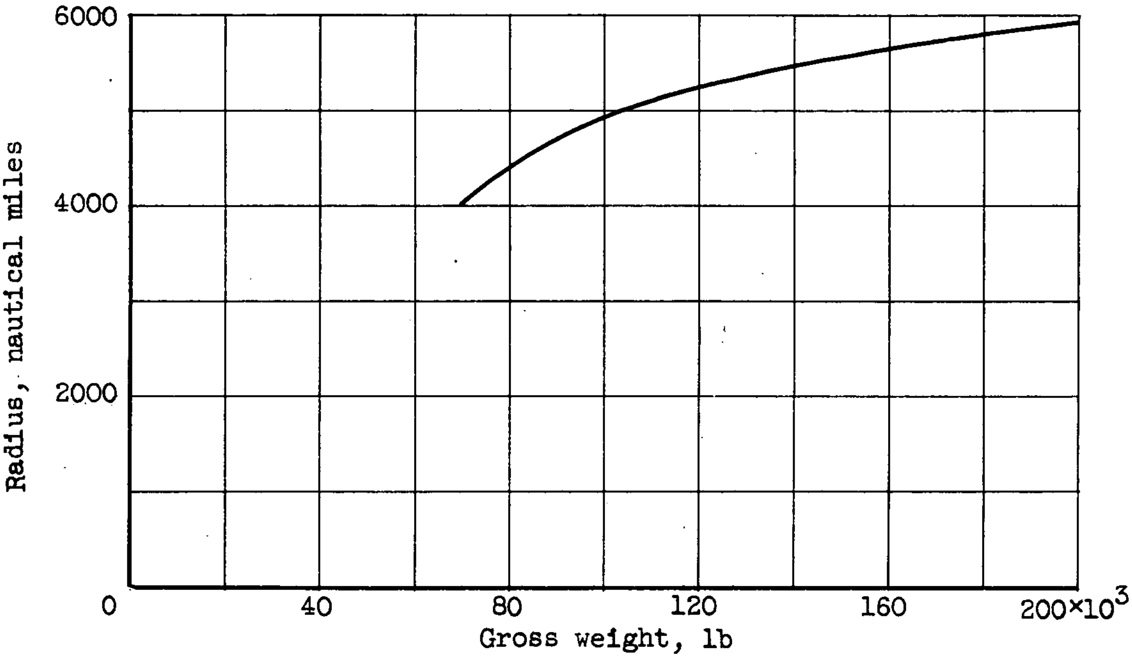

Figure 15. - Effect of gross weight on radius of subsonic bomber. Flight Mach number, 0.75; target altitude, 80,000 feet; engine A.

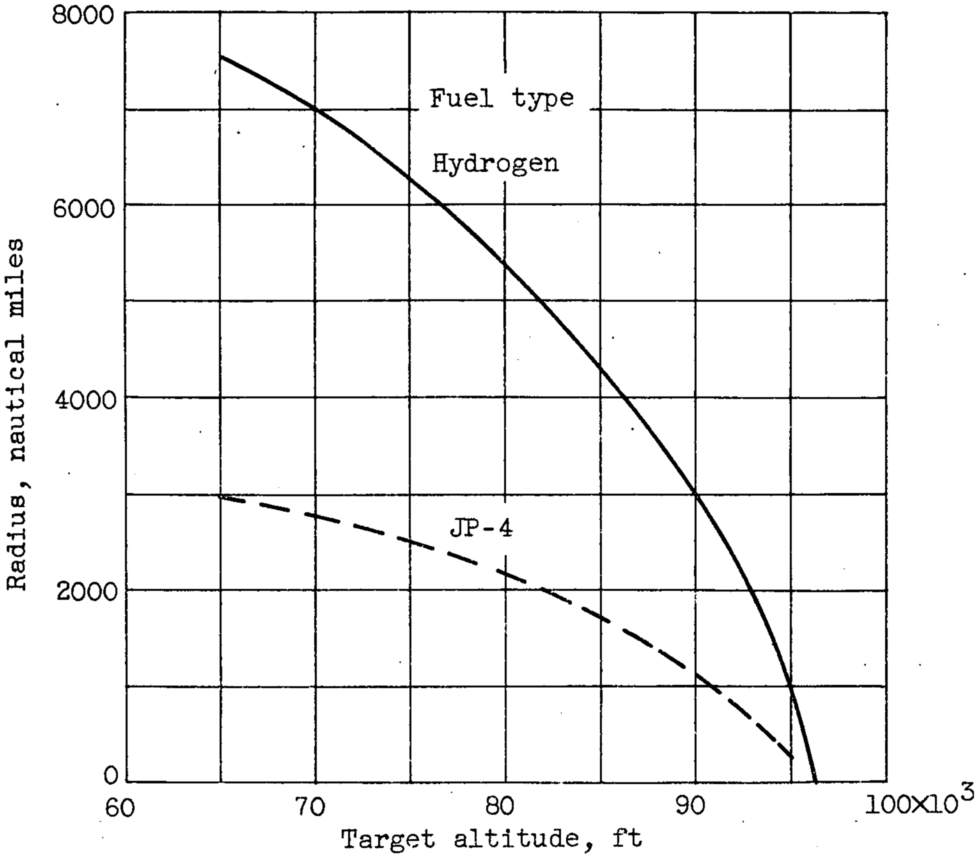

Figure 16. - Effect of target altitude and fuel type on radius of subsonic bomber. Gross weight, 130,000 pounds; flight Mach number, 0.75; engine A.

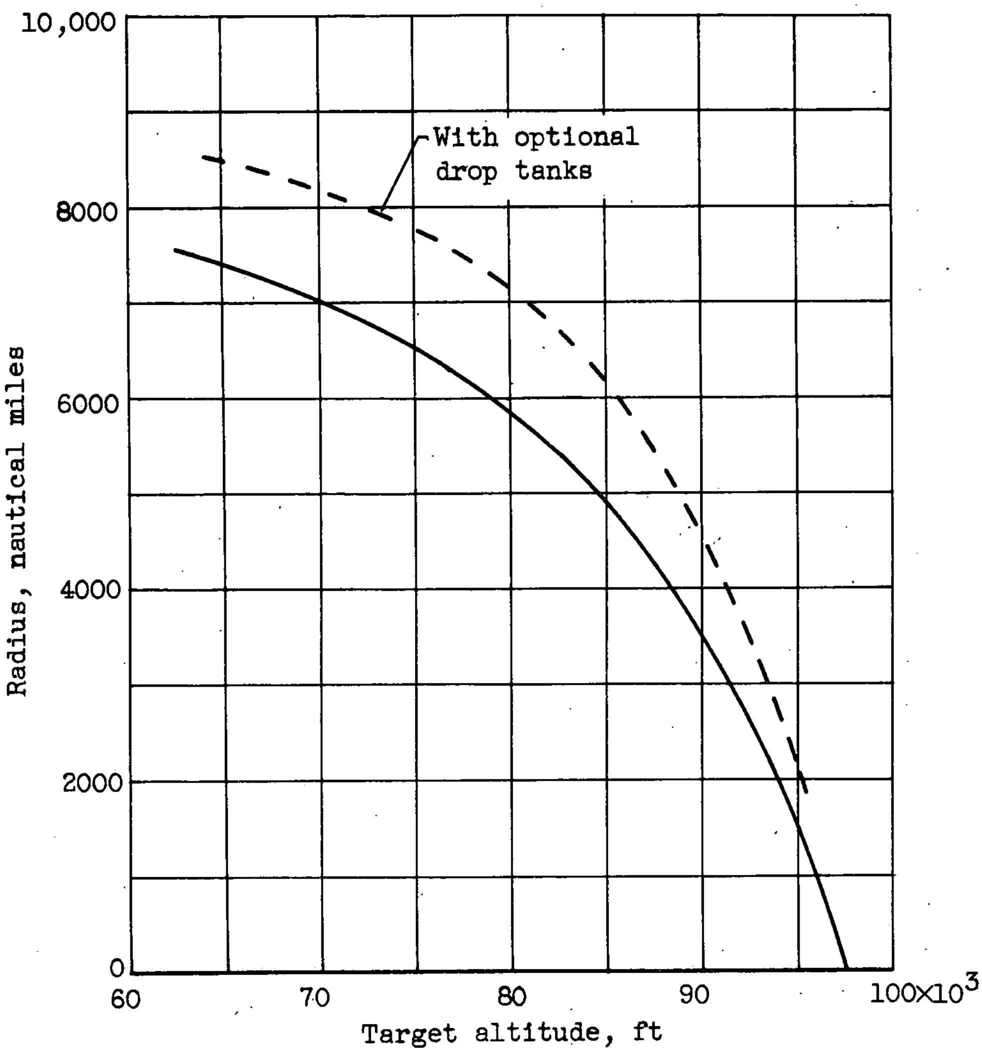

Figure 17. - Effect of target altitude on radius of subsonic reconnaissance airplane with and without external drop tanks. Flight Mach number, 0.75; engine A.

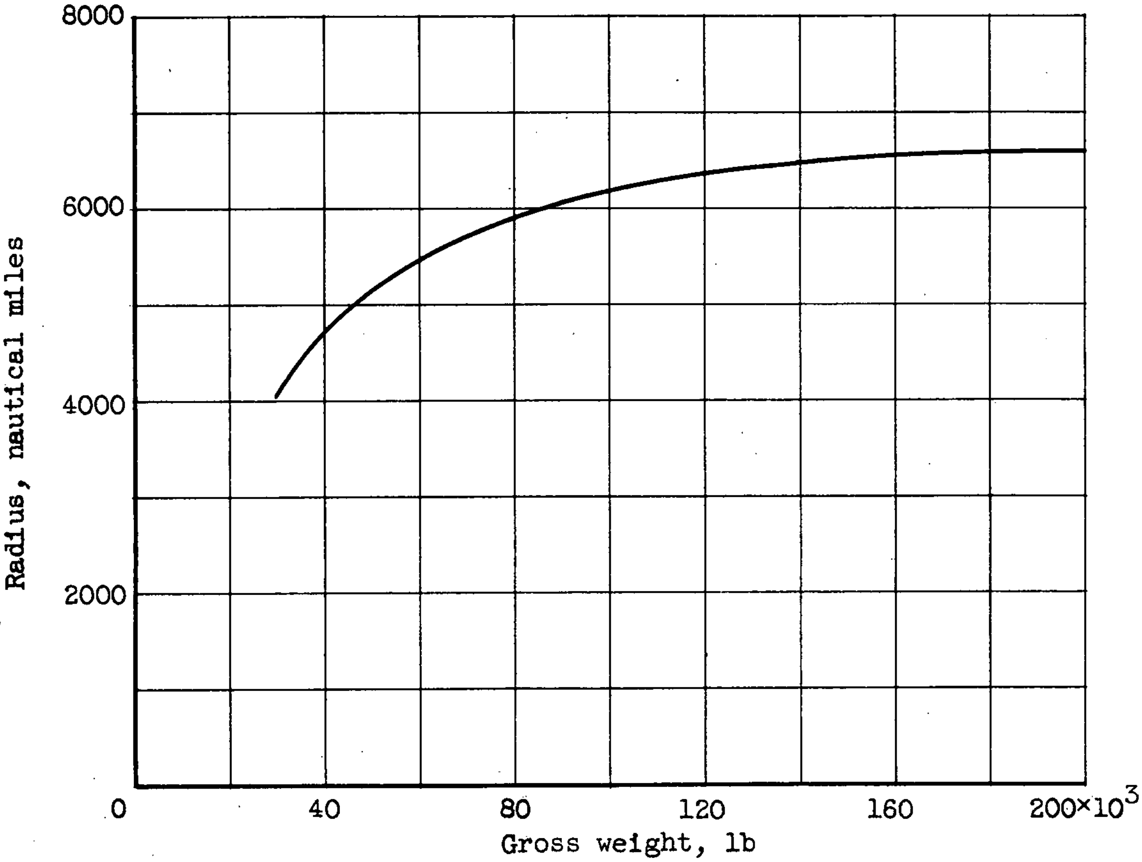

Figure 18. - Effect of gross weight on radius of subsonic reconnaissance airplane. Flight Mach number, 0.75; target altitude, 80,000 feet; engine A.

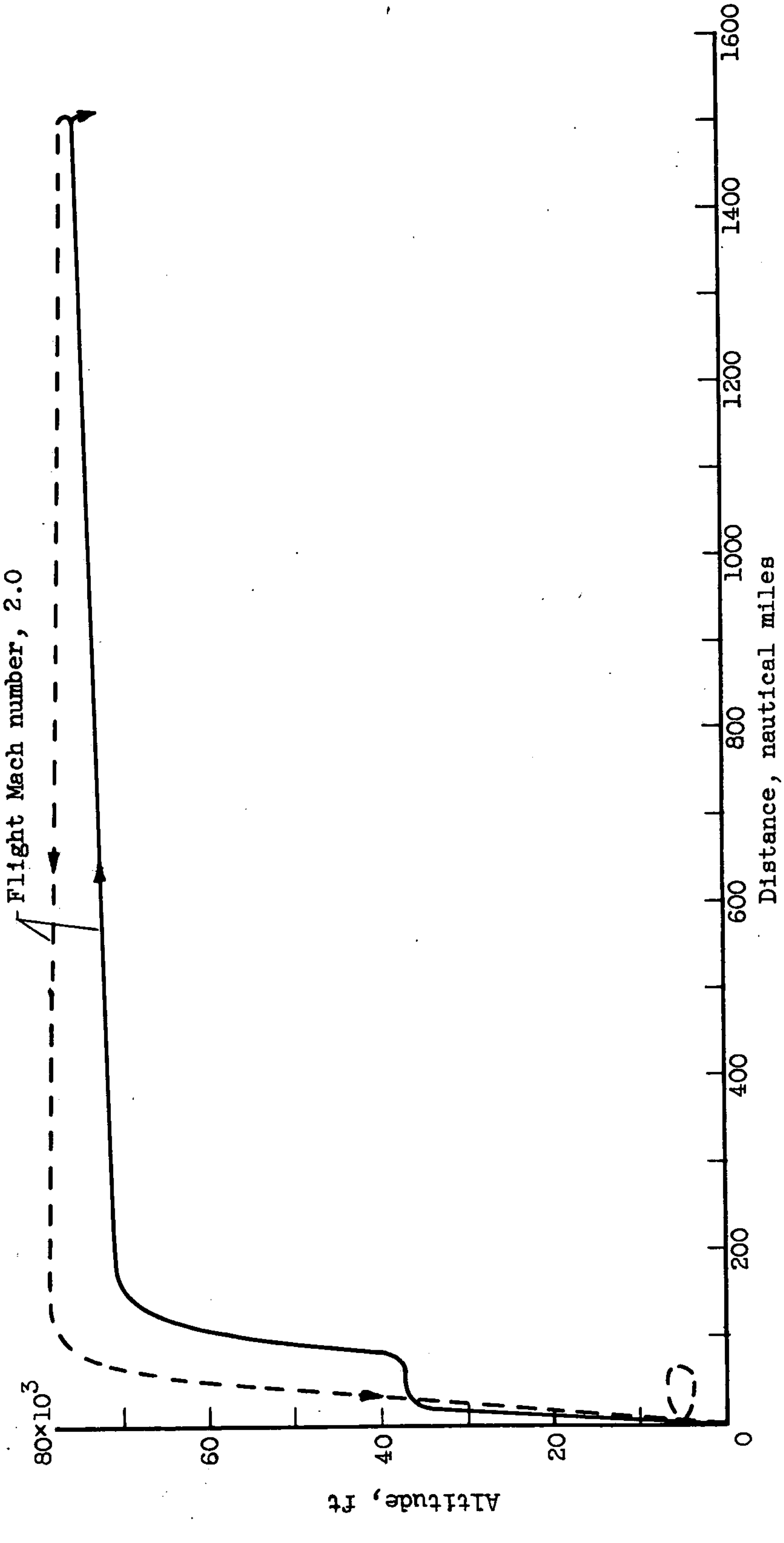

Figure 19. - Flight path of supersonic bomber.

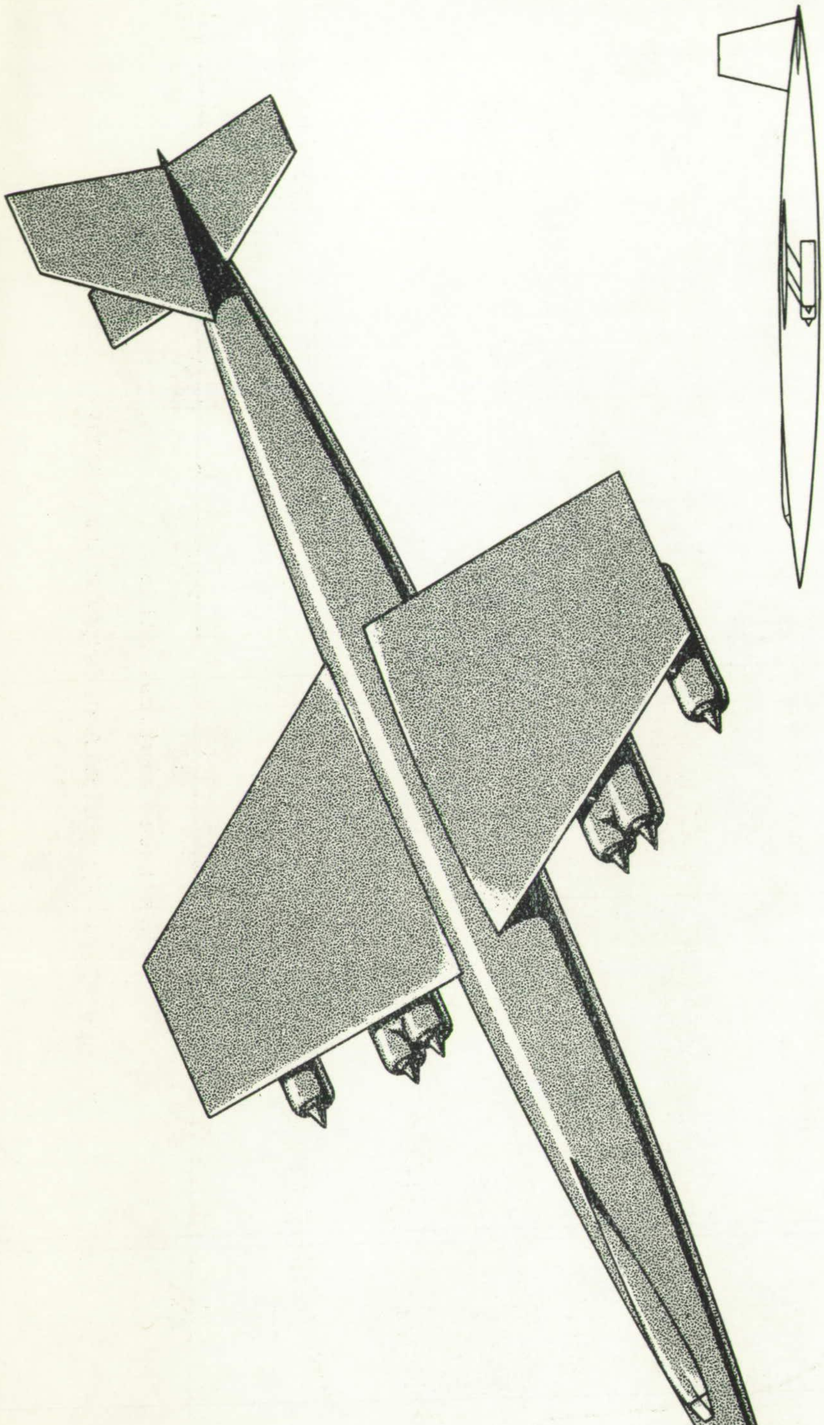

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Figure 20. - Supersonic bomber. Flight Mach number, 2.0; target altitude, 75,000 feet.

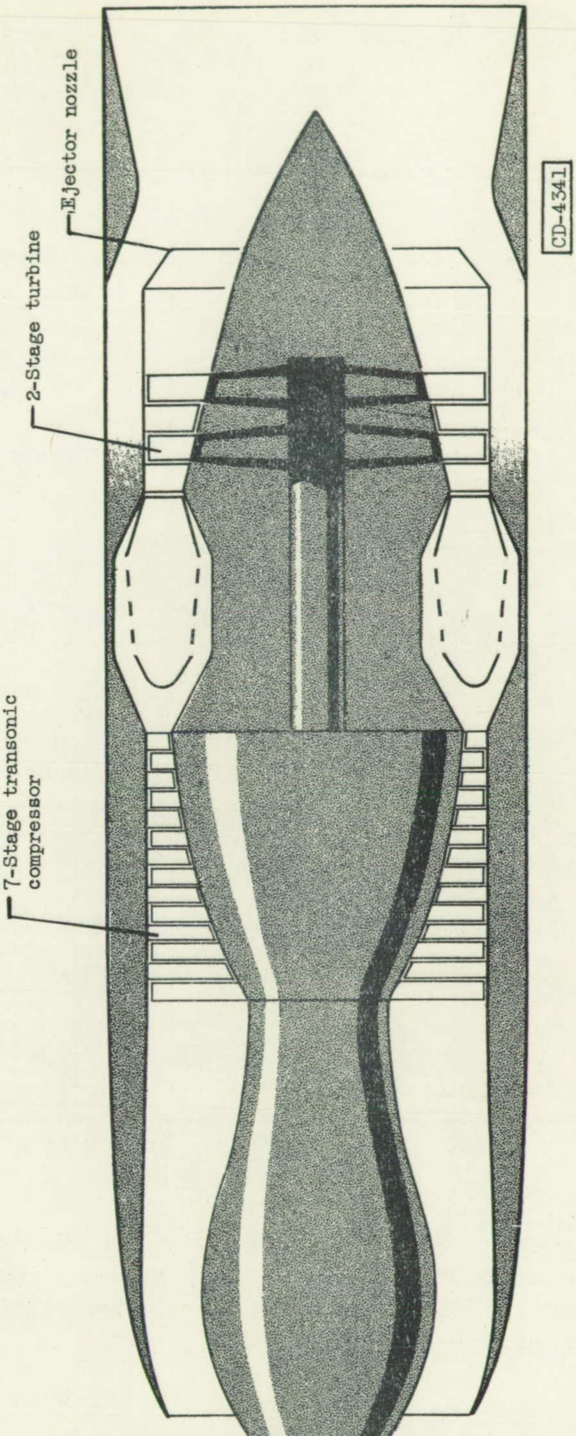

Figure 21. - Schematic diagram of turbojet engine B.

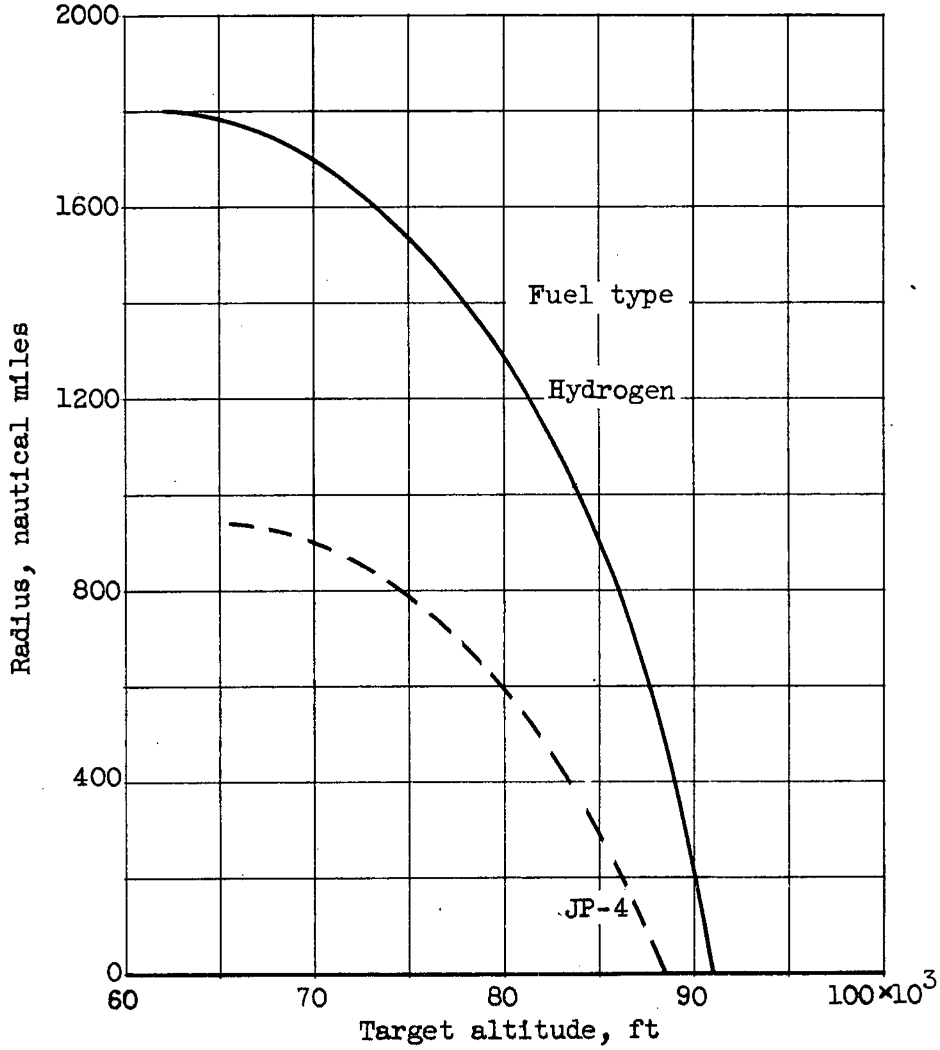

Figure 22. - Effect of target altitude and fuel type on radius of supersonic bomber. Gross weight, 130,000 pounds; flight Mach number, 2.0; engine B.

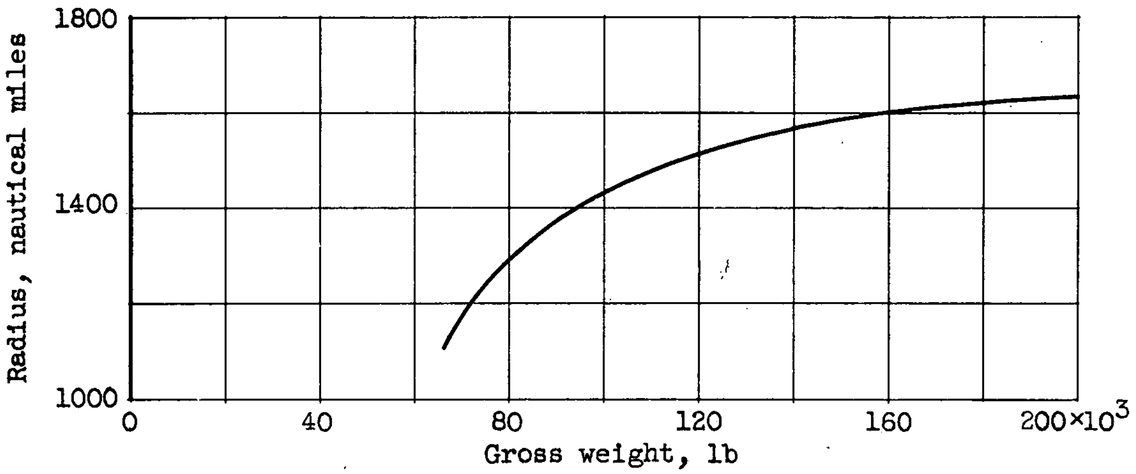

Figure 23. - Effect of gross weight on radius of supersonic bomber. Flight Mach number, 2.0; target altitude, 75,000 feet; engine B.

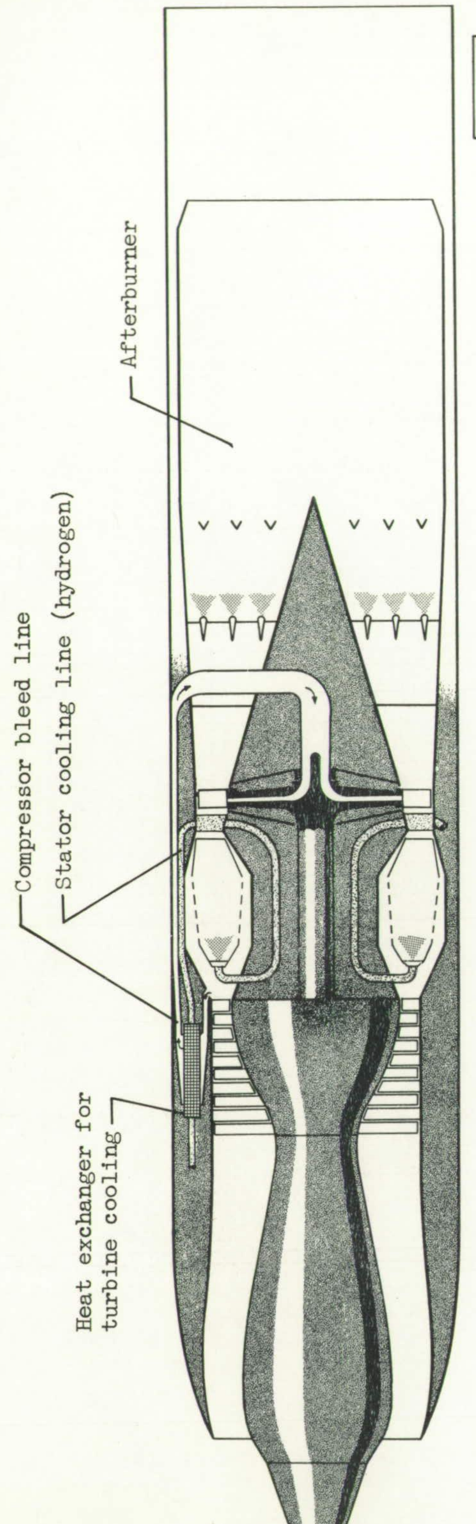

Figure 24. - Schematic diagram of turbojet engine C and turbine-cooling arrangement.

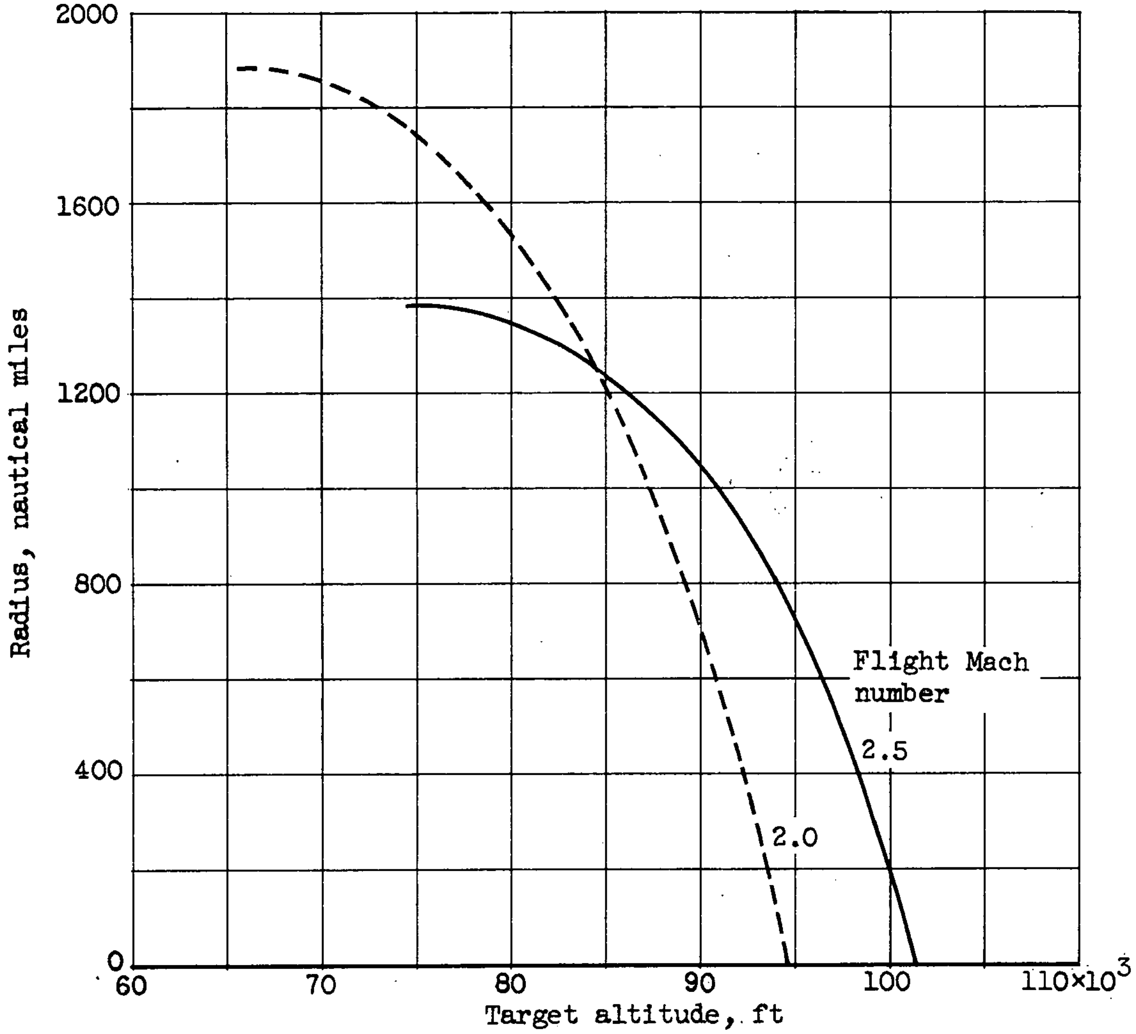

Figure 25. - Effect of target altitude and flight Mach number on radius of supersonic reconnaissance airplane. Gross weight, 75,000 pounds.

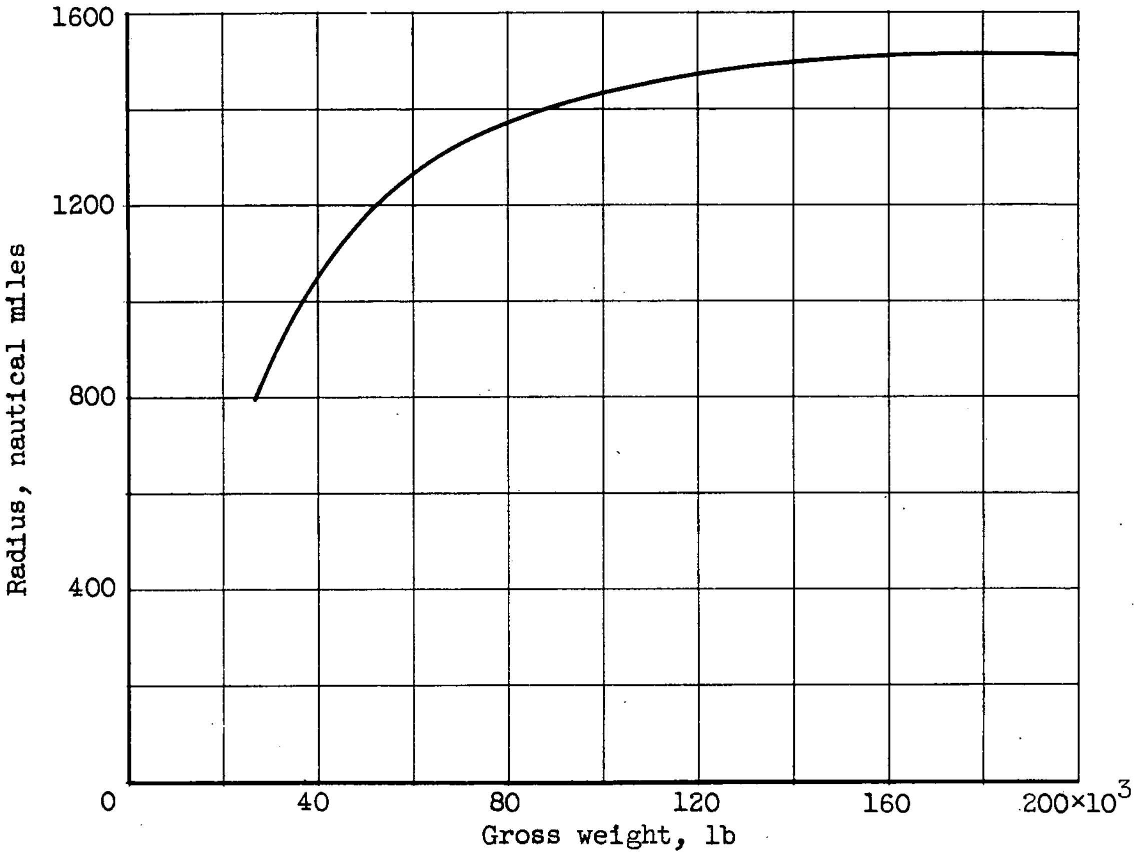

Figure 26. - Effect of gross weight on radius of supersonic reconnaissance airplane. Flight Mach number, 2.5; target altitude, 80,000 feet; engine C.

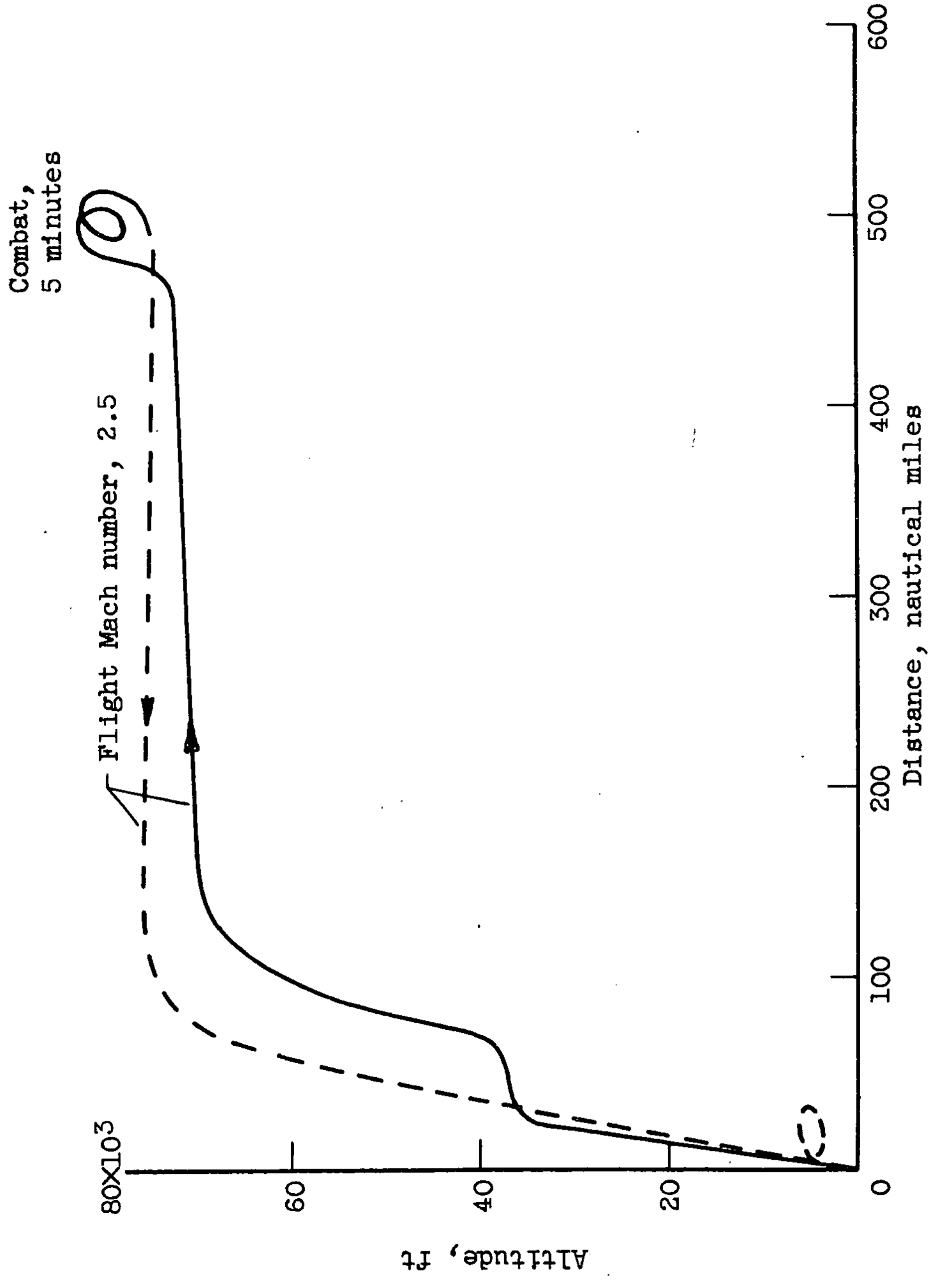

Figure 27. - Flight path of supersonic fighter.

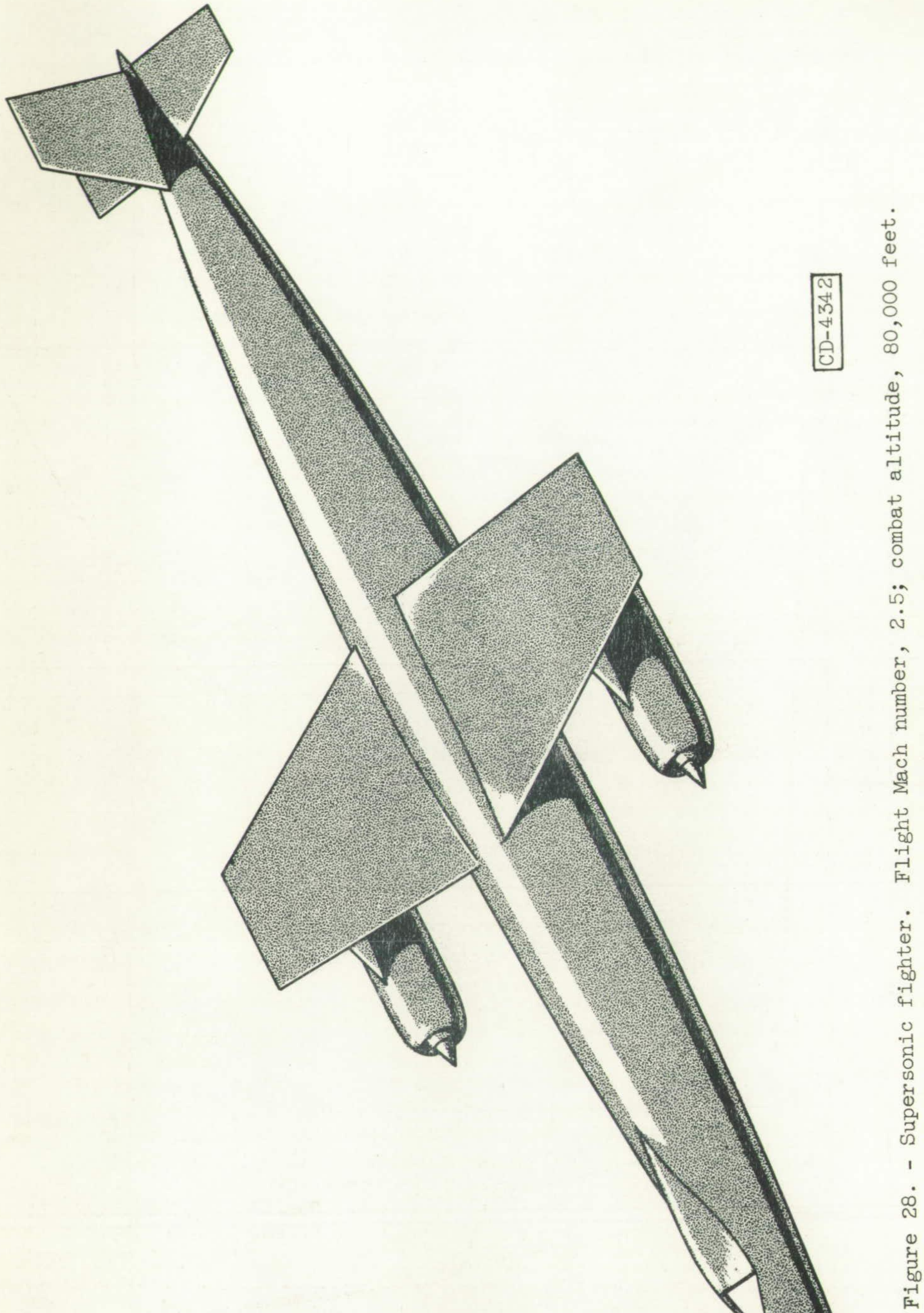

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Figure 28. - Supersonic fighter. Flight Mach number, 2.5; combat altitude, 80,000 feet.

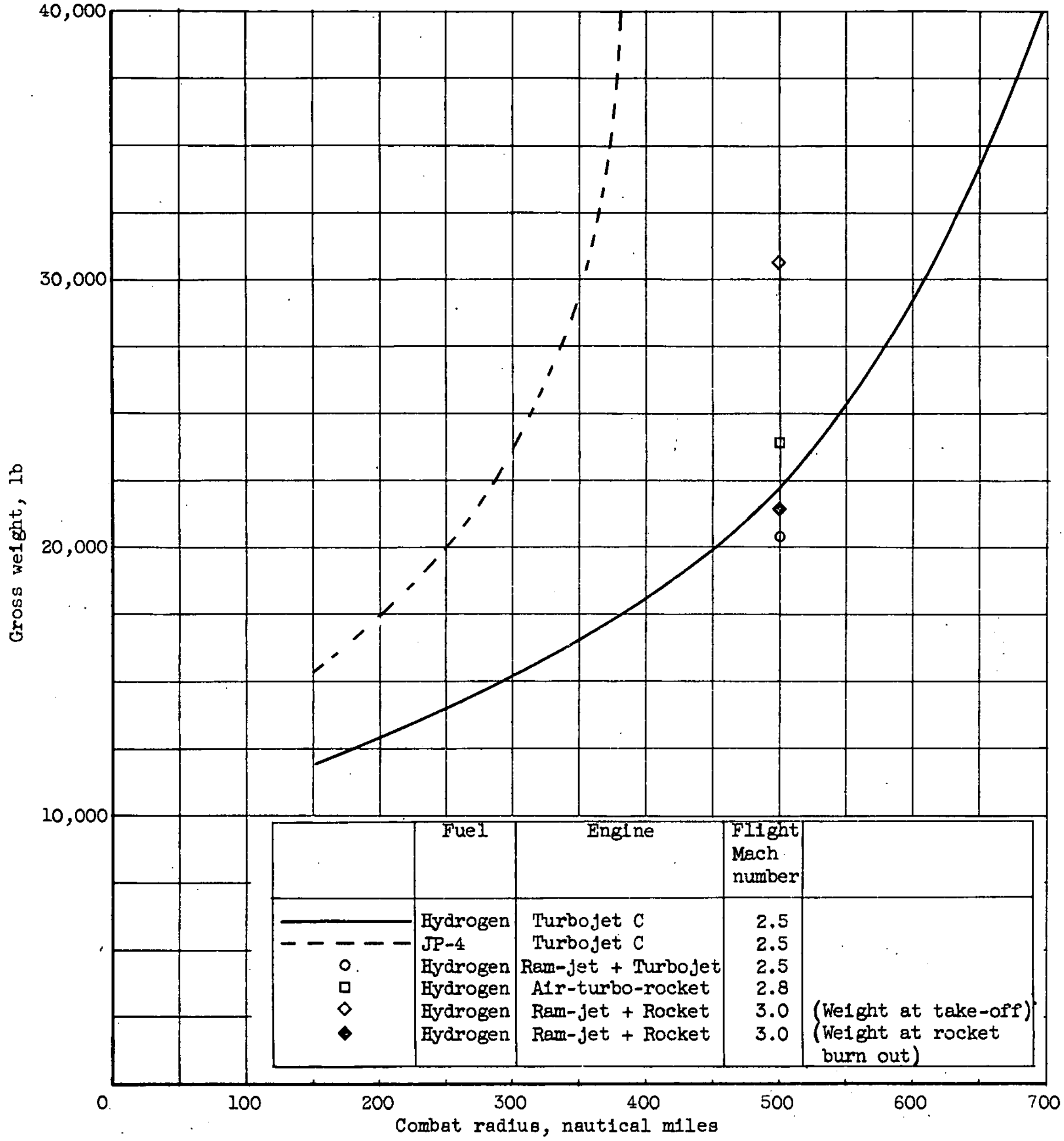

Figure 29. - Effect of combat radius and fuel type on gross weight of supersonic fighter. Combat altitude, 80,000 feet; maneuverability, 1.1 g's.

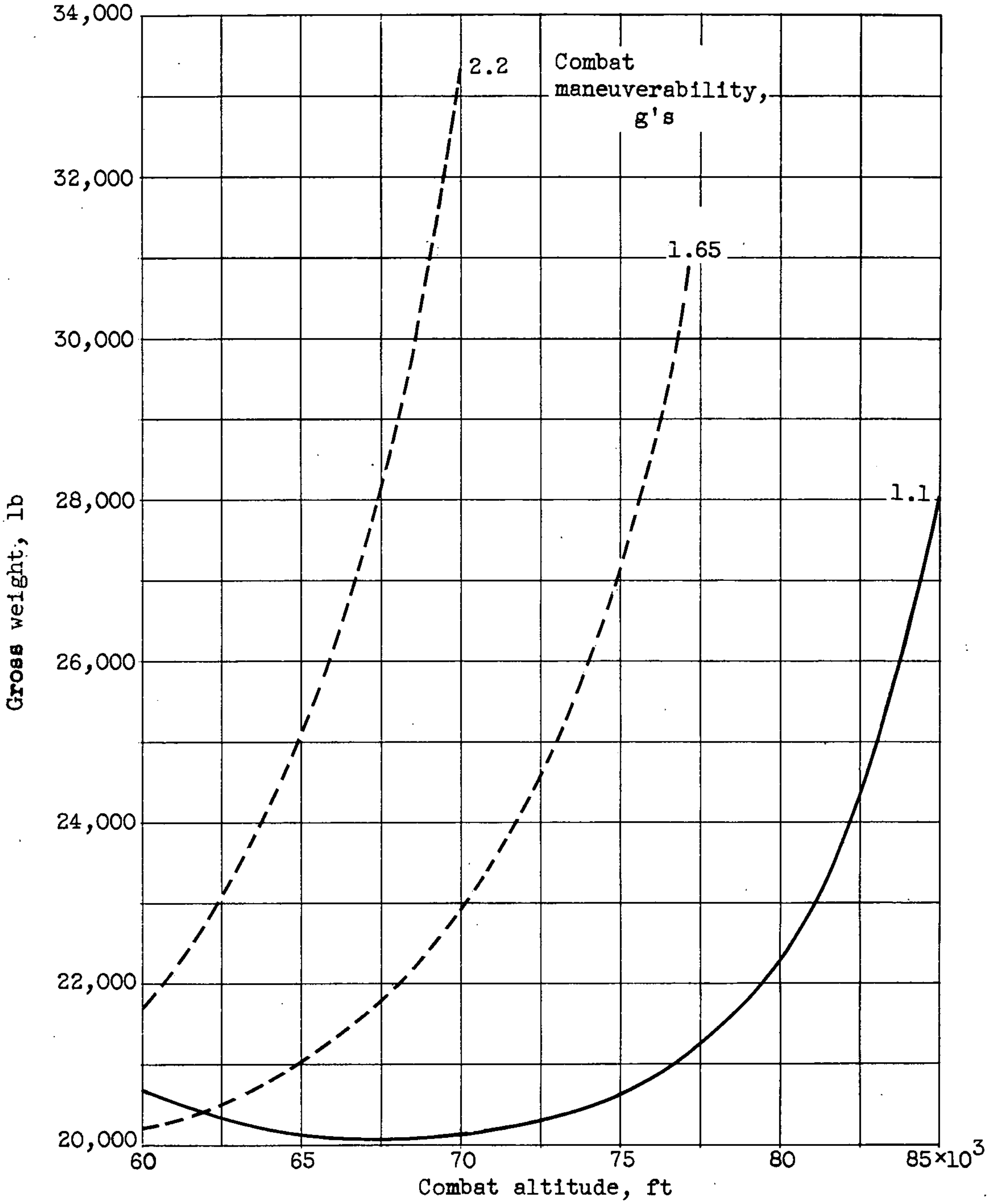

Figure 30. - Effect of combat altitude and maneuverability on gross weight of supersonic fighter. Flight Mach number, 2.5; engine C; combat radius, 500 nautical miles.

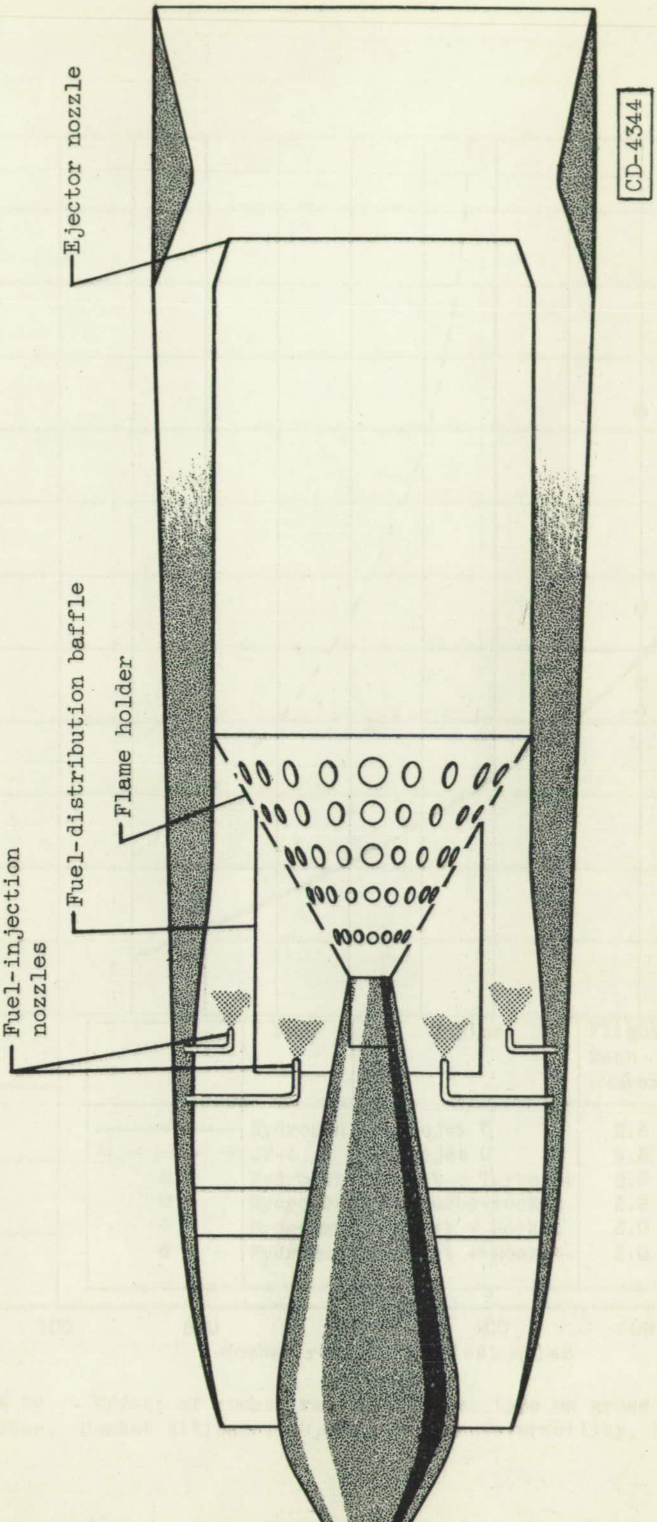

Figure 31. - Schematic diagram of ram-jet engine.

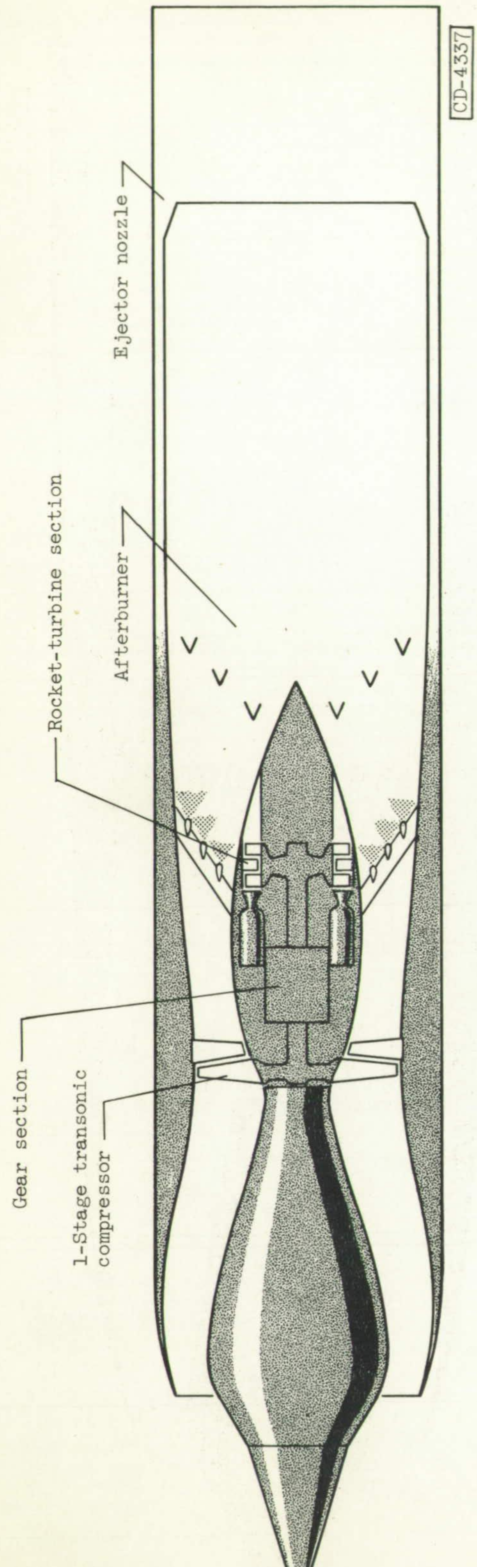

Figure 32. - Schematic diagram of air-turbo-rocket engine.

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