# SUPERSONIC TURBOJET PROPULSION

The first supersonic flight of a man-carrying airplane took place only ten years ago with the experimental, rocket-powered, Bell X-1. Today we have a number of operational turbojet powered aircraft that are capable of sustained flight well above Mach 1--and even faster airplanes are in the experimental stage. Now one might ask just how fast will turbojet powered airplanes be able to fly. NACA research on turbojet engines has penetrated into what we think may be the ultimate speed capabilities of this type of powerplant and we think this may be around four times the speed of sound. So in this discussion we will indicate the nature of the problems that will be encountered by turbojet powered airplanes flying up to Mach 4. We shall do this in two parts, first treating the problems of applying the engine to a typical flight mission and secondly, discussing the design problems of the basic turbojet engine itself.

As a framework for the first part of our discussion, we shall consider the problems posed by a typical airplane such as that represented by this model. Our hypothetical airplane will be designed for flight at Mach 4, or about 2600 miles an hour, at an altitude of 85,000 feet. We have indicated that the engines will be installed in pods tucked under the wing immediately adjacent to the fuselage. It is this part of the airplane that we shall be concerned with in this discussion.

Now a large number of factors must be considered in the overall selection of the powerplant installation, for if this airplane is to be satisfactory it must combine the advantages from the most advanced concepts in aerodynamics, airplane structures, and powerplants, all of which mutually influence one another. Consequently we are now engaged at this laboratory in a study of the mutual interactions of the powerplant installation and the airframe. In a tour of the 10- by 10-foot tunnel following this discussion, you will see an installation of a model being used for this purpose.

Let us now examine the thrust requirements that will be imposed on the engine installation. To do so we shall look at the airplane drag that will exist along our chosen flight plan. We will plot the drag as a percentage of the airplane gross weight over the Mach number range through which the airplane must fly.

During the ground run of the airplane the drag will be quite low. At liftoff, which will occur at about 190 miles an hour, the drag will increase to about 20 percent of the gross weight, or 20,000 pounds of drag for a 100,000 pound airplane. This is due to the high angle of attack which our small, supersonic wing design will necessitate.

We will begin to climb the airplane as speed picks up, the wing efficiency will improve, and the drag will initially decrease. At an altitude of 20,000 feet we will pass through Mach 1, the resultant supersonic shock waves will then serve to increase the airplane drag. After we pass Mach 2 and 35,000 feet, however, the selected combination of flight speed and altitude will gradually improve the aerodynamic efficiency of the airplane so that the drag will steadily decrease until we reach Mach 4 and 75,000 feet. We will then climb at constant speed until we reach our design altitude of 85,000 feet with a small further reduction in drag.

Now, a satisfactory engine installation will have to develop a thrust that is greater than the airplane drag. Actually, the installed thrust should exceed the drag by about 35 percent throughout the speed range. If it doesn't the airplane will accelerate so slowly as to consume excessive amounts of fuel. Thus with an insufficient thrust margin the Mach 4 range of the airplane will be seriously jeopardized.

Although the basic turbojet engine may be capable of producing more than enough thrust to satisfy our flight requirements when operated under ideal conditions, this thrust can be dissipated by details of the installation so that we end up with an inadequate powerplant. These thrust losses arise primarily from pressure losses in the air intake system, drag due to inlet flow conditions, and shock or overexpansion losses in the exhaust nozzle, so let us look at the inlet and exhaust nozzle problems in greater detail.

Here is an enlarged model of the engine pod shown on our hypothetical airplane. This is the basic turbojet engine--compressor, turbine, and afterburner. Here is the inlet and here is the exhaust nozzle.

The inlet has a number of external ramps which generate oblique shock waves for partial deceleration of the supersonic flow. Considerable research is now underway to determine the best principles for designing inlets having high pressure recovery and low drag at Mach 4 and this one must be viewed as only illustrative of the kinds of solutions which are necessary. You will have an opportunity to see several research models devoted to the study of inlet problems during your later tour of the 10- by 10-foot tunnel.

The exhaust nozzle is characterized by the large expanded area downstream of the primary engine nozzle or sonic throat. At the sonic point the exhaust gas pressure is greatly in excess of the external stream pressure and hence the expanding surface is provided upon which the excess pressure can push in the thrust direction. If we did not provide the expanding passage we would lose about 60 percent of the potential engine thrust at Mach 4.

It would be desirable from the viewpoint of weight savings and mechanical and control simplicity to utilize fixed inlet and exhaust nozzle components wherever possible. Let us therefore examine the likelihood of being able to get by with such an installation. To do so we will refer back to our airplane drag curve that we have just determined and we will superimpose the thrust that could be developed by a fixed geometry installation such as this. This represents the drag curve we just developed. Since we are primarily interested in cruise capabilities at M 4 we will start our example at that point and retrace the flight path. We will select an engine size that will give us the desired thrust margin at Mach 4 and 85,000 feet with full afterburner. As we retrace our flight path we find that the thrust would increase about 60 percent in descending to 75,000 feet, and we would have a substantial thrust margin at this condition. At our Mach 4 design point the inlet shock waves would all focus on the cowl lip and the exhaust flow would completely fill the nozzle.

Although we have sized the engine for Mach 4 requirements, the airplane would never be able to accelerate to this speed because the thrust would be less than the airplane drag at the intermediate speeds. The low thrust around Mach 2 is the result of two factors: the inlet shock structure would create a very high drag which we will consider as an effective thrust loss and the exhaust nozzle would have too large an exit area so that it would be inefficient as indicated by the strong shock waves located inside the nozzle. At lower speeds the thrust would again exceed the drag below Mach 1.4 but would be only marginal at liftoff.

The engine with the fixed inlet and exhaust nozzle is obviously unsuited for the required mission because the takeoff is marginal and the airplane would not be able to fly above Mach 1.4 in spite of its potential capabilities at Mach 4. The only way that sufficient thrust can be provided is with geometrical changes in the inlet and exit along the flight path.

First let us consider the exhaust nozzle problem. Here is a chart of nozzle efficiency over the flight speed range. The nozzle efficiency is the ratio between the thrust actually produced by the nozzle and the thrust that an ideal, no-loss nozzle would deliver. This curve shows the efficiency that might be achieved by a good variable geometry nozzle. At takeoff, for example, a well designed exhaust nozzle should produce about 97 percent of the ideally attainable thrust and should still deliver about 95 percent of the possible thrust at Mach 4.

The lower curve shows the efficiencies that we used in obtaining our fixed geometry thrust performance. Because the discharge area is too great at speeds below the design point the thrust steadily decreases until at Mach 1 the nozzle delivers only about 70 percent of its potential thrust. By using a variable nozzle in place of a fixed one, we can increase our thrust about 25 percent at Mach 1 and about 15 percent at Mach 2. Let us see what this does for our installation if we still retain a fixed inlet.

The dashed airplane drag curve and the solid thrust curve for fixed inlets and fixed exits are here for reference. At takeoff the variable exhaust nozzle would be closed to about the same area as the primary nozzle. A 12 percent thrust improvement would be obtained by this technique. While the 25 percent thrust improvement looks impressive at Mach 1, the variable geometry exhaust nozzle does not give enough thrust improvement to permit the airplane to fly through the speed range around Mach 2. You'll notice that it is not necessary to expand the nozzle discharge very much in order to get the best Mach 2 performance. Above Mach 2 the secondary nozzle is steadily expanded, and the performance gradually approaches that of the fixed nozzle at the design Mach number 4.

Satisfactory flight will thus not be achieved by maintaining a fixed inlet geometry while making nozzle adjustments. It was indicated earlier that the fixed inlet would incur large drag penalties at the lower supersonic flight speeds. The next figure shows the magnitude of this drag at Mach 2.

The high ramp angles required for efficient compression at Mach 4 cause a detached bow wave to stand ahead of the rearmost ramp at Mach 2. The oblique shock waves from the two forward ramps would now fall well ahead of the cowl lip. At Mach 2 the engine can only use about 34 percent of the air that could come into the inlet as indicated by the region under this dotted line and the remainder is spilled by the shock wave structure, but this spillage is only accomplished with a drag amounting to 61 percent of the potential engine thrust. It is this high drag cost which prevents us from attaining a satisfactory installed thrust in the vicinity of Mach 2.

The drag of the inlet can be reduced by making judicious changes in the spillage pattern. One such possibility is shown on the lower portion of the figure. By reducing the third ramp angle until it becomes an extension of the second ramp, the detached bow wave can be eliminated. But now more air is coming into the inlet as indicated by this dashed line. Since the engine cannot use this extra air, the difference would have to be removed ahead of the engine face, and this can be accomplished by a bypass which returns some of the inlet flow to the external stream.

By substituting flow diversion through the bypass for diversion through a detached bow wave the drag due to spillage can be reduced from 61 to 31 percent of the potential engine thrust. So let's see what this more efficient inlet combined with the variable exhaust nozzle can do for our thrust performance. For comparison the airplane drag curve and the thrust with variable exhaust nozzle but fixed inlet are retained.

The third ramp would be lowered at takeoff and the bypass would be closed. This would result in an improved liftoff pressure recovery that would increase the thrust to about 35 percent more than the liftoff drag, thus assuring good takeoff characteristics. Near Mach 1 the performance would be about the same as the fixed inlet, but above Mach 1 the bypass would be opened at a scheduled rate and the drag savings would result in appreciable thrust increases, so much so that our desired thrust margin could now be attained. Above Mach 2 the third ramp would be gradually raised and the bypass closed so that we would pass through our previous Mach 4 performance point.

Thus we find that through the use of suitable variable geometry inlet and exhaust nozzle components our Mach 4 engine can be made to produce a satisfactory acceleration performance that could not be attained with fixed components. Obviously these components are going to be complex mechanisms. Consequently our research is not confined to just the aerodynamic performance characteristics because the weight, complexity, methods of control, and interaction or interference effects with the rest of the airplane must all be understood and evaluated before a given component design can be said to be satisfactory.

We have said nothing so far about the characteristics of the basic turbojet engine that we have assumed will be used in our Mach 4 airplane. The next speaker will discuss this engine and some of our newest facilities for studying the problems of high speed turbojet propulsion.

For the past 12 years this laboratory has expended a major fraction of its research effort on the study of turbojet engine problems. From our research on the major engine components and from our analyses of desirable engine cycle characteristics, we feel that the turbojet engine can be applied to flight speeds up to Mach 4, though to do so will require a research and development effort of major magnitude.

This model illustrates our goal for a Mach 4 turbojet engine. The installation shown here is a little different from that discussed by the preceding speaker in that the inlet is round rather than rectangular, but the inlet problems are qualitatively the same as those discussed earlier.

The basic engine is much more compact than those to which we are now accustomed, a feature that is made possible from our improved understanding of component performance capabilities.

For example, NACA research has shown the advantages of a new type of transonic compressor which will develop a much greater pressure rise per row than a conventional compressor. Now, cycle studies indicate that a Mach 4 turbojet should not have as great a pressure rise across the whole compressor as current machines. Thus by combining the relaxed pressure rise requirements with the improved transonic performance we are able to conceive of an engine which has only three rotating blade rows, or stages, instead of the 12 to 15 found in current engines.

The compact engine design is also enhanced by the very short primary and afterburner combustion chambers. These are predicated upon recent NACA studies of special aircraft fuels and their combustion properties which have indicated that highly satisfactory combustor performance can be maintained in future engines in much less volume than is current practice. You will hear a discussion of some of these fuels at another stop.

Although the aerodynamic and combustion principles of an engine such as this are, we think, well in hand, the practical construction of such an engine will be difficult, principally because of the temperature environment in which the engine components must operate. Let us therefore examine the temperatures that will exist within this engine and, for comparison, let us initially look at the temperature distribution in a typical Mach 2 engine.

The engine shown on this figure is purely schematic in that the compressor has more stages than our Mach 4 engine but less than a Mach 2 engine. We will look at the temperatures that exist in the compressor, primary burner, turbine, and afterburner. We shall assume that the engine is flying at high altitudes where the temperature is 67 degrees below zero. At Mach 2 the ram effect of the air slowing down in the inlet will bring the temperature up to 240°F. entering the compressor. The pressure rise across the compressor will be accompanied by a temperature rise, so that the air will enter the combustor at 875°F. Enough fuel will be burned in the combustor to bring the turbine inlet temperature up to about 1650°F. but the work extracted by the turbine will drop the temperature down to about a thousand degrees. In order to build up the thrust, more fuel will be burned in the afterburner so that the final discharge temperature will be a little over 3000°F.

Let us now contrast this temperature environment with that in our Mach 4 engine. The ram temperature will now be up to 1230°F. Fortunately, our compressor pressure rise is low, and the temperature only rises about 100 degrees. The primary combustor heat addition is also low so that our turbine will operate with a lower inlet temperature than the Mach 2 engine. The temperature drops about 100 degrees across the turbine due to the low turbine work requirement. To get our desired thrust output, it will be necessary to afterburn to almost 3300 degrees however.

The compressor temperature of our Mach 4 engine is higher than can be handled by present day alloy or stainless steels. If we try to realize the lowest possible weight, then the combination of high compressor stress and high operating temperature will require the use of the strongest super-alloys now used in advanced turbines.

Although the turbine temperatures are modest when compared with Mach 2 engines, the necessity for a high stress level to match the aerodynamic capabilities of the compressor will also place heavy demands on the turbine structural materials.

The engine bearings and seals will represent special problems because the environmental temperature is above present bearing capabilities. The entire bearing and seal assemblies will probably require cooling, and here is a special problem, for means will have to be provided to protect these components from heating by the ram air as well as from temperatures generated within the cycle.

Although we are confident that a Mach 4 turbojet can be built, we recognize that these obstacles are formidable. There are other factors which suggest that other powerplants should be employed above Mach 4. For example, we here show the engine pressure ratio developed by the turbojet as a function of flight speed. The engine pressure ratio is the pressure downstream of the turbine divided by the pressure ahead of the compressor.

We want the compressor-turbine combination to produce a pressure ratio greater than unity in order to generate the highest possible thrusts.

The turbojet engine has a high enough engine pressure ratio at takeoff and subsonic flight speeds to permit the engine to develop the necessary large values of thrust under those conditions. When we get into the supersonic speed range, however, the turbine begins to extract more pressure from the cycle than is added by the compressor. When this occurs the engine pressure ratio decreases and becomes less than one above Mach 3 and the compressor-turbine combination then actually hurts the thrust potential.

At these high speeds a more efficient propulsion system could be obtained with a ramjet engine, which doesn't have the compressor and turbine and hence would operate at a pressure ratio of unity in terms of this chart. Because the ramjet doesn't have a compressor it cannot develop any thrust at takeoff. Therefore an aircraft powered by a ramjet engine must be externally boosted up to a speed great enough to permit the attainment of an adequate thrust margin.

If we were to perform our hypothetical mission with a ramjet rather than a turbojet-powered airplane, we could reduce the size considerably, as shown by this model. It is smaller than the turbojet-powered airplane for two reasons. On the one hand, the ramjet would be more efficient at Mach 4 and hence would consume less fuel during the cruise flight. More importantly, however, this airframe would not contain the fuel required to take-off and climb to Mach 4. This energy would have to be supplied by some kind of external boost arrangement.

If we conceive of the booster as a liquid fueled rocket, the boosters might have this appearance. They would be attached to the airplane for launching, but would fall away after final speed conditions are attained. The overall gross weight of the ramjet airplane plus booster would be about the same as the all turbojet airplane. For this and other reasons, neither type of powerplant can be said to have a clear-cut technical superiority for Mach 4 flight and it is probable that each has a place in this speed spectrum for different missions. At speeds above Mach 4, however, our studies indicate that the ramjet engine has a definite superiority. You will hear, or have heard, a discussion of some of the problems of our high-Mach number ramjet research at another stop.

Now, the facilities used to study turbojet engine problems at this laboratory are many and varied. They include test cells for evaluating individual components, and high altitude test chambers in which complete engines can be studied at simulated conditions up to Mach 3 and 90,000 feet. Our newest major facility is the 10- by 10-foot tunnel. In it we study the problems of the engine inlet, exhaust nozzles, the interactions of the engine and the airplane configuration and the performance of full scale engines at Mach numbers from 2 to over 3.5. This tunnel thus complements our 8- by 6-foot tunnel which operates from Mach .9 to Mach 2.1.

You are now seated in a room located in this building. In a few moments you will be conducted on a tour of the main working section of the tunnel, including the test section, the flexible nozzle where the speed settings are controlled, and the second throat where the air is slowed back down to subsonic speed. The tunnel complex contains many other features of interest which you will not be able to see on this brief tour and I should like to point them out at this time.

The supersonic speed in the test section is obtained by expanding air through a nozzle. Two compressors are used to supply the necessary pressure differential, and are located here and here. Each is an axial flow compressor similar to those used in turbojet engines, but on a somewhat larger scale. The primary compressor runs all of the time and sucks the air through the test section. The secondary compressor operates only at Mach numbers above 2.5 and pressurizes the air ahead of the test section.

The compressors are driven by electric motors. Four motors in tandem supply a total of 150,000 hp to the primary compressor, and three motors supply 100,000 hp to the secondary.

The temperature of the tunnel cycle is raised considerably due to the heat of compression as well as from any combustion in the test engine and this heat must be removed continuously by coolers located here and here ahead of each compressor. You will note the size of one of these coolers by comparing it to the man in the lower part of the photograph.

Even with cooling, the temperature of the air entering the nozzle may be on the order of 300°F, but the expansion process by which supersonic velocity is achieved causes the temperature to drop to sub-zero levels in the test-section. Any moisture in the airstream would therefore condense out in the nozzle and would cause flow non-uniformity in the test section. It is therefore necessary to dry the air before it enters the tunnel and this is done in the air dryer, where 1900 tons of activated alumina act as a dessicant. This dryer can adsorb as much as 63 tons of water in a little over an hour on a hot, humid summer's night.

The tunnel can operate on two distinct cycles. When burning engines are being studied the air is brought from the atmosphere, through the dryer, through the tunnel, and is discharged back to the atmosphere through an acoustic muffler. For aerodynamic runs the air is allowed to circulate continuously around the circuit. The cycle path is controlled by the setting of a 24 foot diameter flow transfer valve weighing 34 tons. Here is a photograph of the flow transfer valve taken from the muffler house. The cycle can be changed from one mode to the other during operation of the tunnel.

The entire tunnel operation is conducted from the control room on your right. On the far wall are the controls required to start, change, and maintain conditions in the tunnel. On the left are the controls over the variables of the model under investigation, including the point where the central data taking process is initiated. On the right panel are instruments for automatically recording temperatures, forces, and transient data or data that are changing very rapidly during the reading. Since the control room is remote from the test section, several television screens are located around the room. The pictures from various remote cameras are monitored by the television equipment on the left panel.

Gentlemen, if you will follow your guide through the control room by the door at the rear, you will now be taken on a tour of the working section of the tunnel. This tour will end in the room overhead where the data are received and calculated. Thank you.

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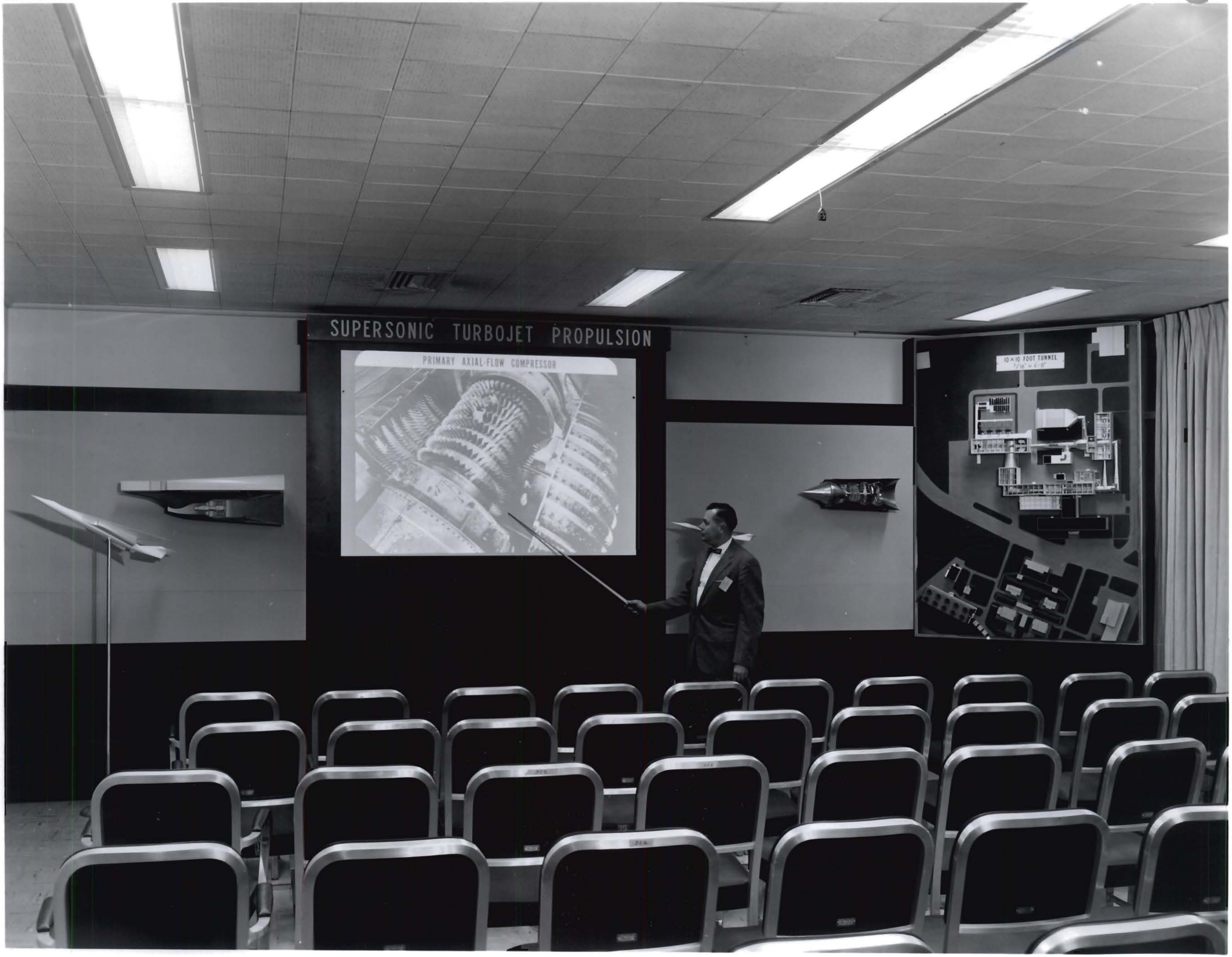
SUPERSONIC TURBOJET PROPULSION
PRIMARY AXIAL-FLOW COMPRESSOR
10 × 10 FOOT TUNNEL

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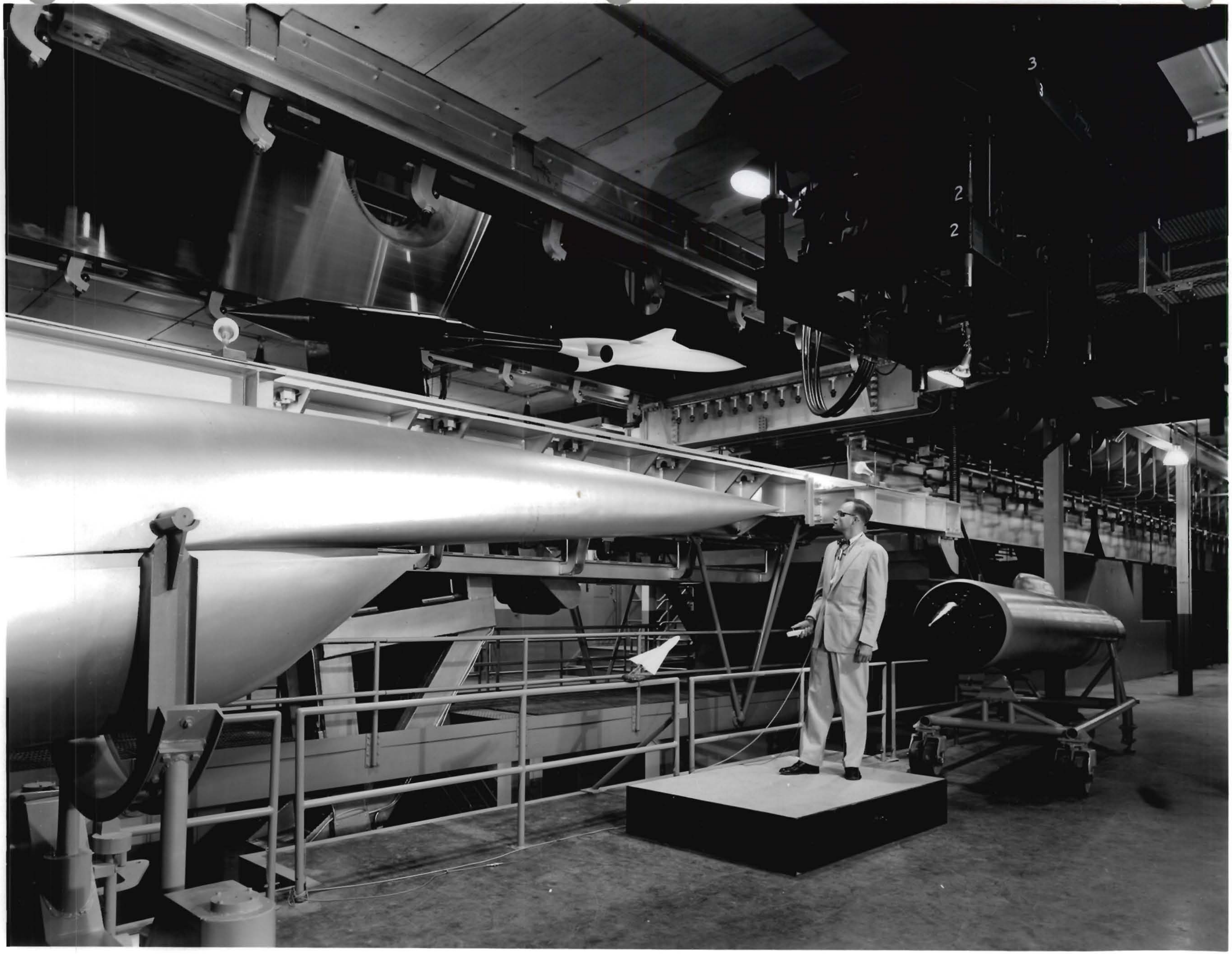

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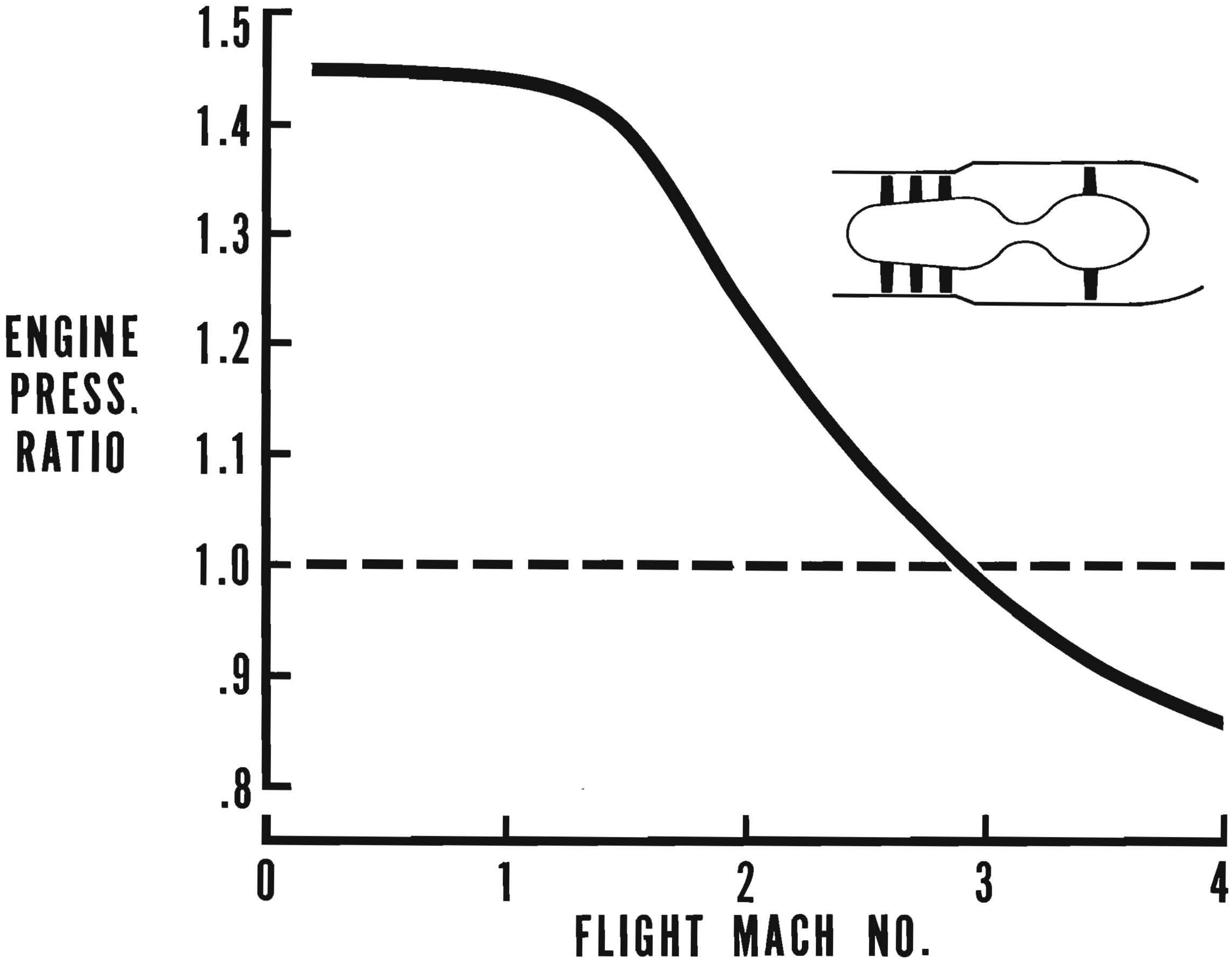
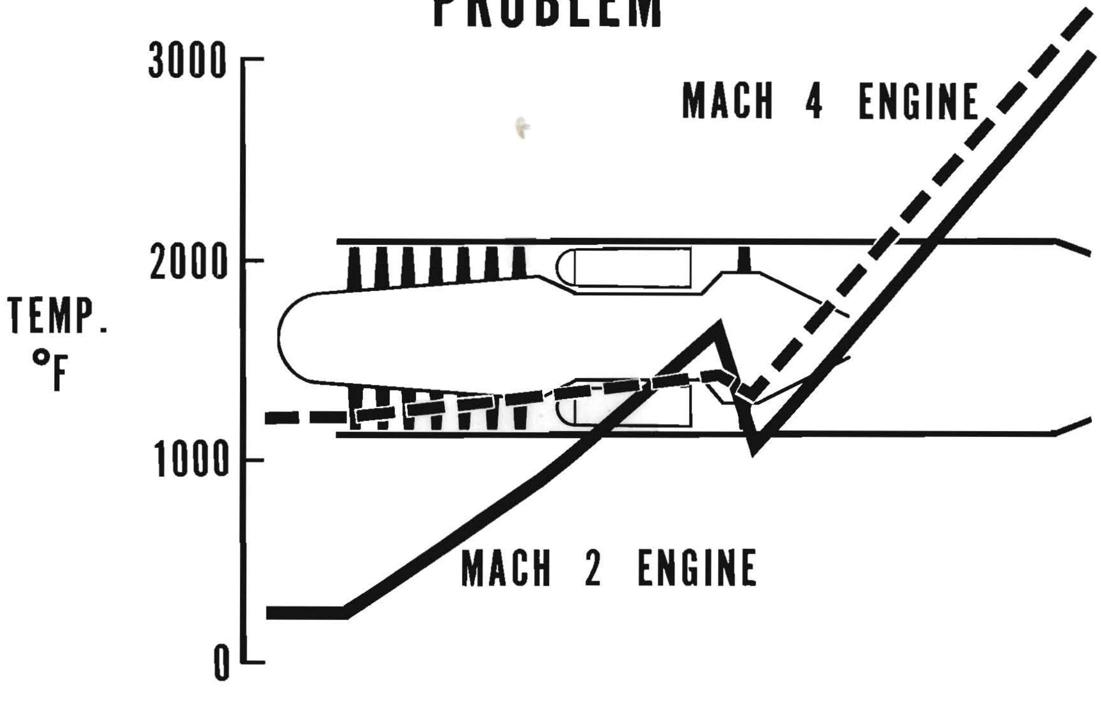
# HIGH SPEED REDUCES ENGINE PERFORMANCE



# HIGH SPEED CREATES TEMPERATURE PROBLEM



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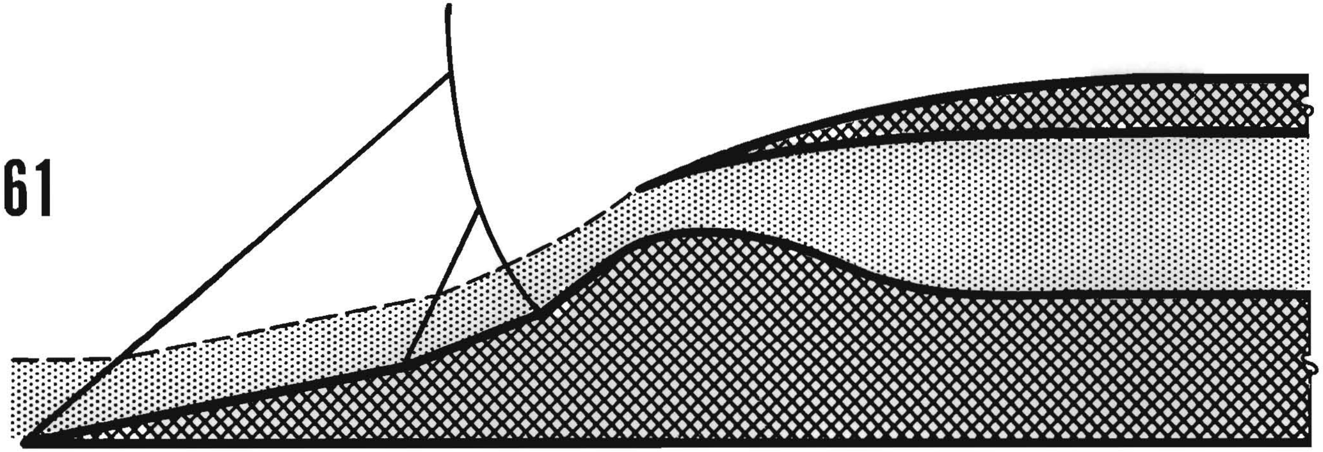
# SPILLAGE PATTERN AFFECTS DRAG

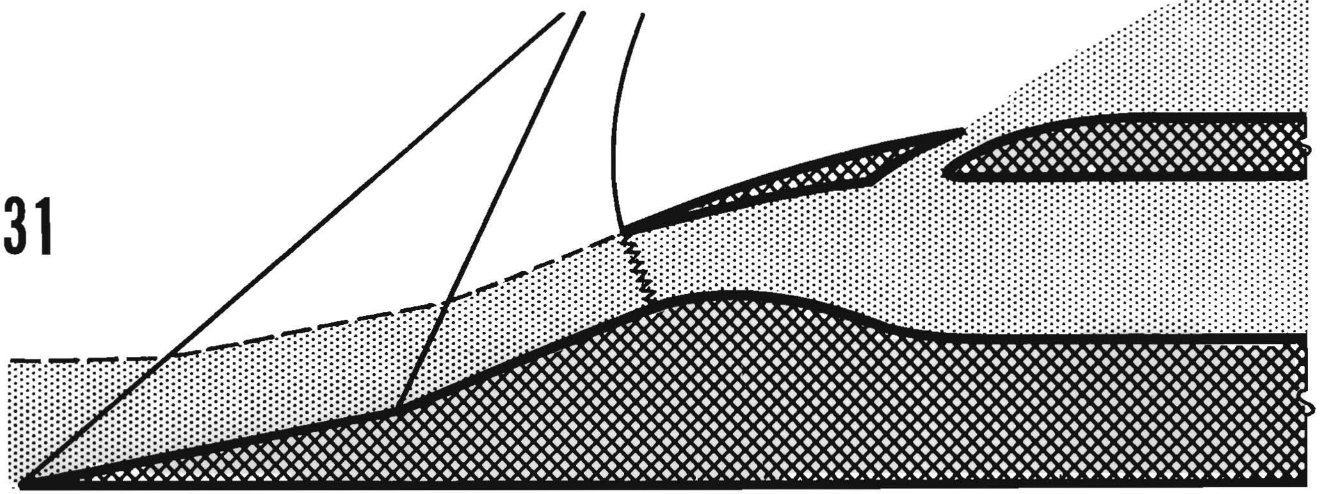
## FIXED INLET AT MACH 2

$$\frac{\text{DRAG}}{\text{THRUST}} = .61$$



## VARIABLE INLET AT MACH 2

$$\frac{\text{DRAG}}{\text{THRUST}} = .31$$



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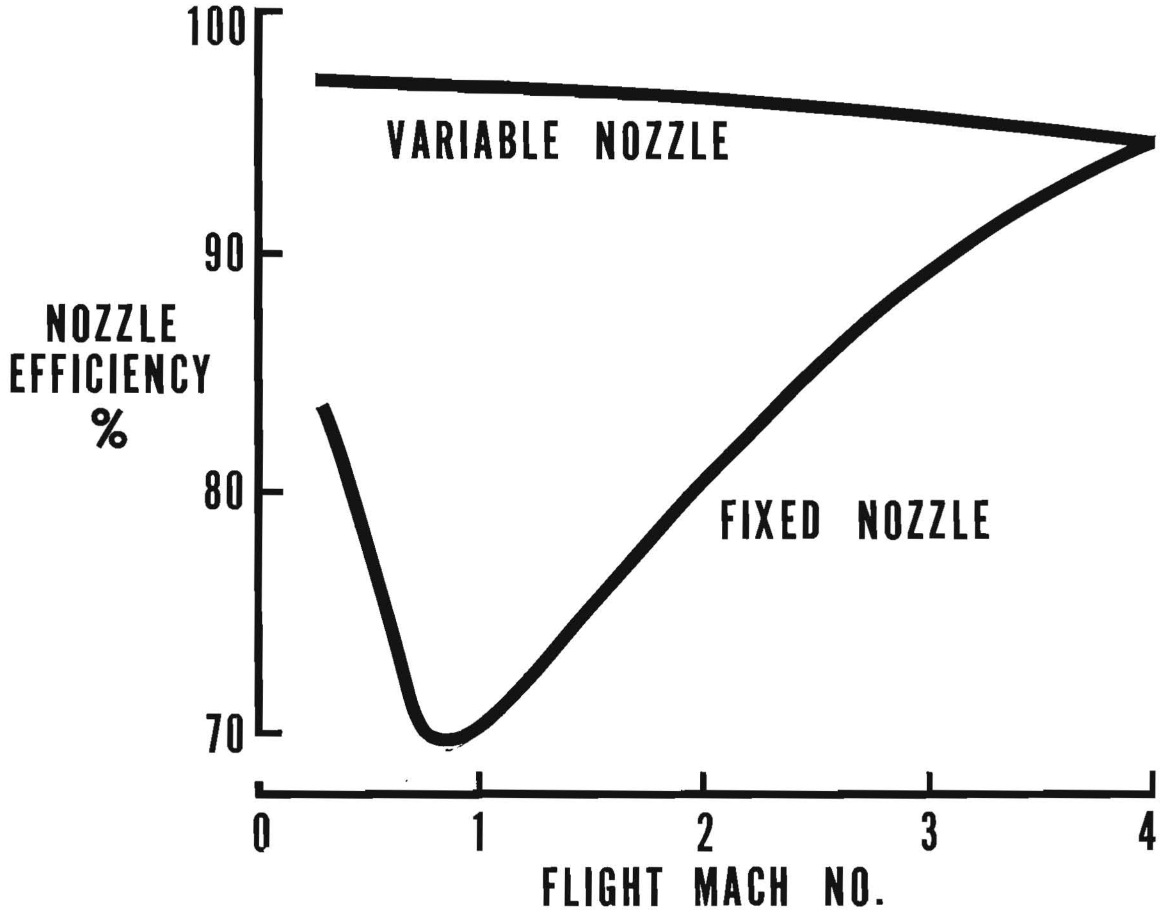
# FIXED EXHAUST NOZZLE IS INEFFICIENT



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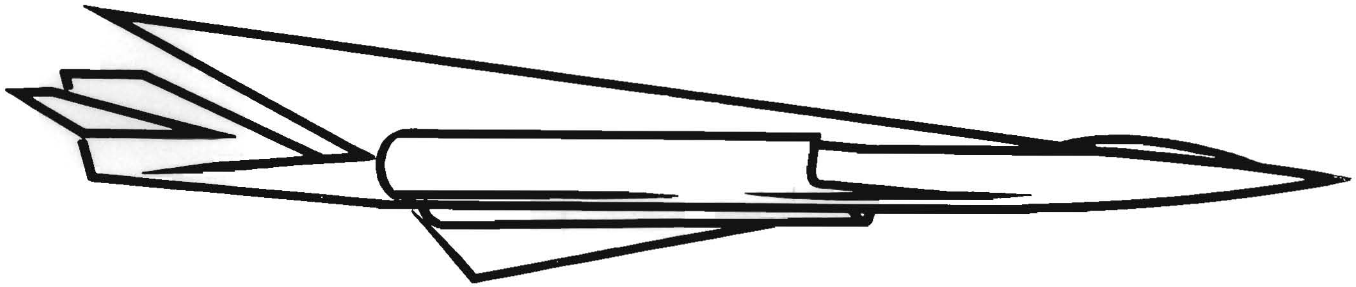
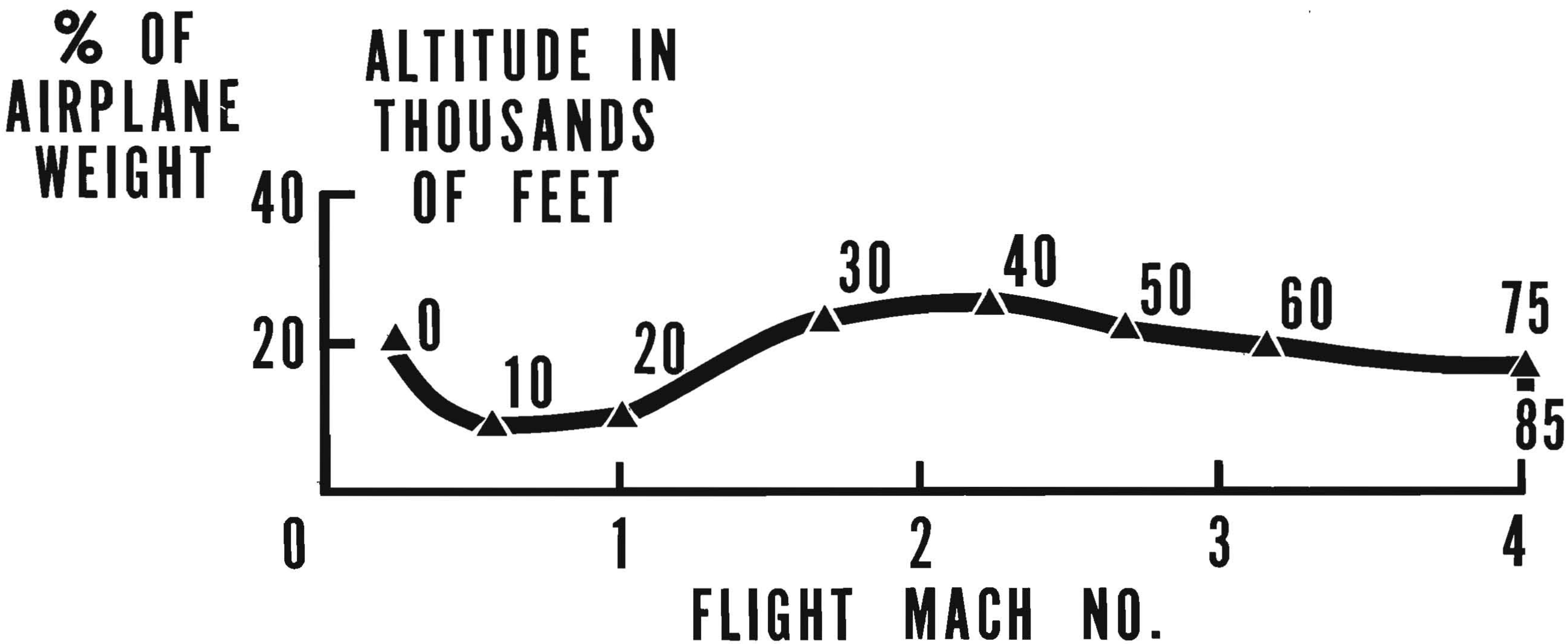
# DRAG OF HYPOTHETICAL AIRPLANE





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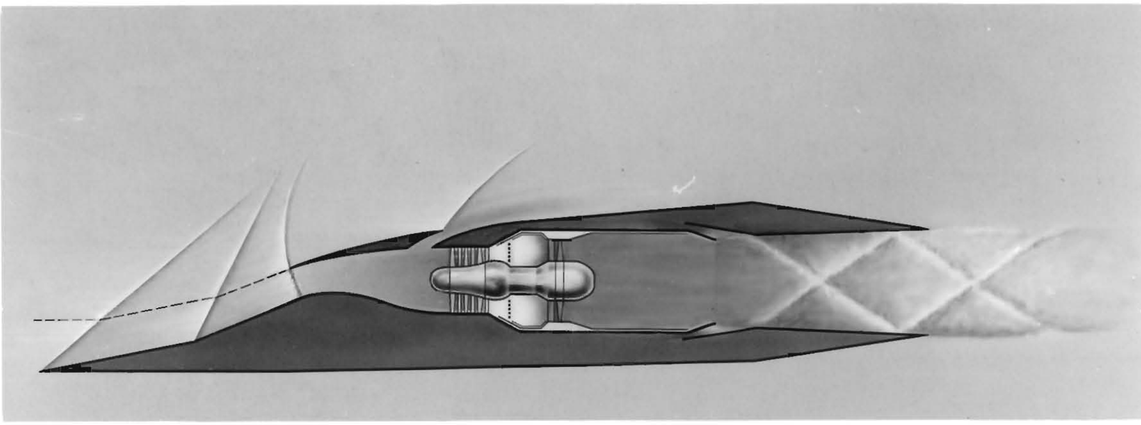
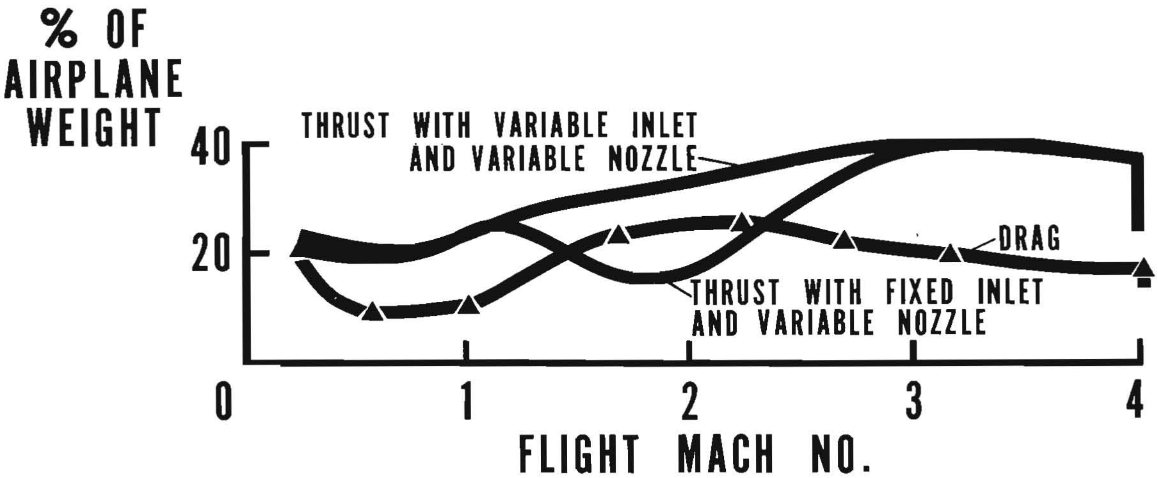
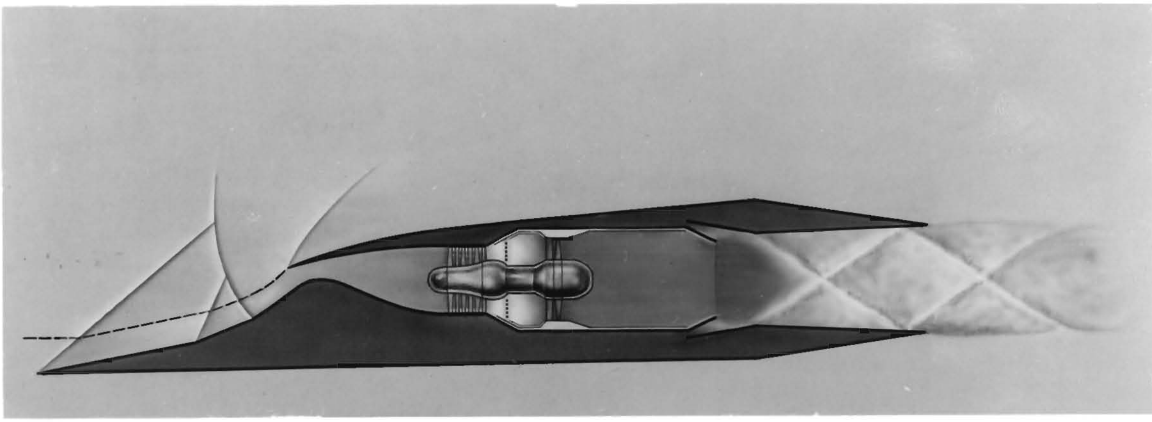
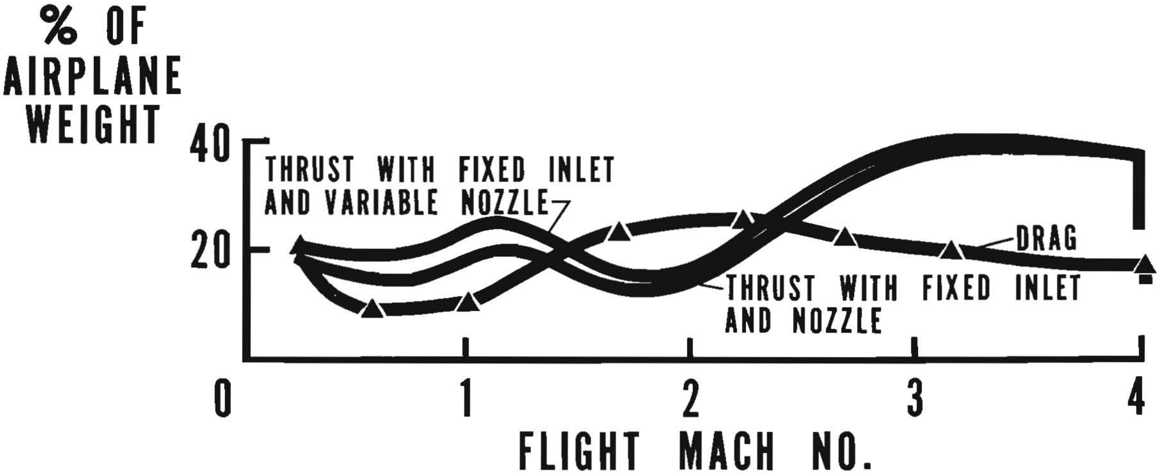
# ADDING VARIABLE INLET SOLVES PROBLEM





# NOZZLE CHANGES ALONE ARE INSUFFICIENT





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# FIXED GEOMETRY THRUST IS INADEQUATE