ENGINE CONTROL RESEARCH

Speakers: Richard S. Cesaro and David Novik

Gentlemen:

In general, every engine has certain potentialities, and it is the function of the engine-control system to make sure that every bit of this engine potential is realized. It must be possible to operate the engine at minimum fuel consumption, maximum power, maximum acceleration and deceleration, and maximum engine life regardless of the specific operating conditions. For the gas-turbine engine it is difficult to accomplish this general objective, because in order to completely exploit the engine potentialities, it is necessary to operate the engine either at or near the limiting conditions. This means that a control system is required whose response must be critically accurate and the study of required control-response characteristics presents the major problem now being studied in controls research at this Laboratory.

A brief discussion of some of the gas-turbine engine characteristics will serve to illustrate why such critical response is demanded from a control system for this type of power plant. On the first chart, Figure 48 we have plotted the percent maximum horsepower against the percent maximum engine speed for a reciprocating engine and for a gas-turbine engine. You will note that at 75 percent of the maximum engine speed the reciprocating engine can develop up to approximately 90 percent of the maximum engine horsepower. In contrast with this, at the same 75-percent maximum engine speed, the gas-turbine engine can develop only about 30 percent of the maximum engine horsepower. This means, then, that in order for the gas-turbine engine to develop any appreciable power it must be operated in the region of high engine speeds. Operation in the region of high engine speeds necessitates an accurately responsive control system because in this region small changes in engine speed result in large changes in engine horsepower. It is also possible in operating close to the maximum allowable engine speed that this maximum could be exceeded.

The next chart (~~chart 2~~ Figure 49) indicates that the response of a gas-turbine-engine control system is also critical with reference to temperature control. Here we have shown the comparative effect of fuel-air ratio on engine operating temperatures by plotting the engine operating temperature against fuel-air ratio for the gas-turbine engine and the reciprocating engine. The operating temperature of the reciprocating engine, which may be considered as the temperature of some critical part, such as the exhaust valve, peaks at stoichiometric fuel-air ratio; that is, at the chemically correct mixture of fuel and air. This peak temperature is approximately 1400° F. The operating temperature of the gas-turbine engine also peaks at stoichiometric fuel-air ratio but at a temperature of approximately 3600° F. The difference between the two peak temperatures is due to the fact that in the gas-turbine engine there is continuous combustion, whereas in the reciprocating engine charge air is brought into the cylinders during the four-stroke or two-stroke cycle, and this charge air acts as a cooling medium in reducing the temperature. The operating temperature of a reciprocating engine is, therefore, a mean temperature that occurs over the four-stroke or two-stroke cycle. As a result of the continuous increase of operating temperature with increase in fuel-air ratio for the gas-turbine engine, it becomes possible for increase in fuel-air ratios to result in operating temperatures that exceed the maximum allowable operating temperature of the gas-turbine engine. This means, then, that for the gas-turbine engine a temperature-limiting control is an absolute necessity. In contrast with this, if the reciprocating engine is initially designed so as to cool properly at the peak temperature, it will then be inherently temperature-protected because any deviations from stoichiometric will result in reductions of the operating temperature.

It is also required that the control system correct for the effect of altitude on the gas-turbine engine operating characteristics. This effect is shown on the next chart, Figure 50 (~~chart 3~~) by plotting the available engine acceleration against altitude. At sea level a relative value of 5 is used to designate the available engine acceleration. This available engine acceleration decreases as the altitude increases; that is, the maximum rate at which the engine can accelerate decreases, until at 40,000 feet the available engine acceleration has a relative value of only 1. This means that the same power plant operating at 40,000 feet will respond only one-fifth as rapidly to a control setting as it would at sea level. This tremendous change in response characteristics of the engine must be recognized by the control system and in turn it is necessary for the control-system response characteristics to undergo a corresponding change.

In order to obtain a knowledge of the meaning of the word "response," we have found that the response of a controlled gas-turbine engine can be compared with the response of a simple spring and mass system as illustrated on the next chart, Figure 51 (~~chart 4~~). Here, we have a simple spring from which is suspended a mass, and the system is subjected to friction. The response of this simple spring and mass system is well known, and is such that we can say that the spring and mass response is a function of the mass, friction, and spring rate. This relationship may be generalized to the extent that we can say, in general, that response is a function of inertia, damping, and restoring force. In deriving the equations which describe the response of a controlled gas-turbine engine, it has been found that these equations are similar to those obtained for the response of the simple spring and mass system. We can, therefore, compare the two equations and thereby understand the significance of the components contained in the equation for the response of the controlled gas-turbine engine. From this comparison we have found that the response of the controlled gas-turbine engine is a function of the engine inertia plus a control coefficient, a function of the engine damping plus a control coefficient, and a function of the control restoring action. The effect of one of these factors that influences response, the damping of a system, may be seen from this very simple illustration mounted on our demonstration platform. Here we have a simple spring and weight system similar to the one shown on the chart. At present, the only damping to which this system is subjected, is the result of the damping action of the air column contained within this glass tube. Upon disturbance of the equilibrium, you will note that we obtain several oscillations of the type which we normally associate with the response of such a simple spring and weight system. However, by adding water to the tube, we can greatly increase the damping action of the system, and it will be interesting to note the change in the response under such circumstances. You will note that this time, upon disturbance of the equilibrium, we no longer obtain oscillations but merely a restoration of the original equilibrium condition, so that here we have seen the tremendous change in the response of the system resulting from a change in only one of the factors which influence response.

Having some knowledge now of the factors that influence response, the control designer may now attempt at least to obtain a desired response from a controlled gas-turbine engine. The response characteristics of the engine itself cannot be changed inasmuch as they are inherent within the engine. The response characteristics of the control system must, therefore, be adjusted in order that a desired controlled engine response be feasible. This adjustment of the control response characteristics is termed "matching"; that is, matching of the control and engine-response characteristics. The effect of matching may be seen from the next chart Figure 52, where we have plotted engine speed against time. We first assume a low condition of engine speed and then a desired increase of engine speed to this new value. For a case in which the control and engine-response characteristics are improperly matched, we can obtain an increase in engine speed, as indicated by the oscillating curve. The fact that this type of response is undesirable may easily be visualized if we assume that the new value of engine speed happened to be the maximum allowable engine speed.

If this were true, then it can be seen that the maximum allowable engine speed has been greatly exceeded in addition to the fact that a severe fluctuation period follows. On the other hand, when the response characteristics of the control system are more adequately matched with the engine-response characteristics we can obtain an increase in the speed, as shown by this curve, which increases smoothly without oscillations and does not exceed the desired value. We can give you a more visual picture of the effect of matching by referring to this mechanical analog on our demonstration platform. In this mechanical analog the air turbine is used to simulate the action of a gas-turbine engine. The air turbine is operated by air delivered through these two air jets. The pressure at the air jets is determined by the position of a butterfly valve located at this point. The butterfly valve is positioned by a controller located here, and the desired control setting is made with the control lever located here. The controller functions so as to sense and control the pressure immediately downstream of the butterfly valve, which is essentially also the pressure at the air jets. Installed in the line which brings the controlled pressure pulse to the controller is an air accumulator. This air accumulator in effect, imposes a time lag in the control system so that its response characteristics are poorly matched with the response of the simulated engine - the air turbine. During operation of the mechanical analog you will be able to note the actual change in speed as we accelerate from a condition of low speed to high speed by reading the curve which will be drawn on the moving chart. The motion of the pen is proportional to the air-turbine speed, and is actuated by the electrical speed-sensing device at the end of the air-turbine shaft. For this particular condition of operation, it will also be interesting for you to note the movement of the control of the positioning lever as the controller moves the butterfly valve in an attempt to maintain the desired speed setting. You will also be able to hear very audibly the fluctuations in air-turbine speed during operation for this first demonstration of the effect of poorly matched engine and control-response characteristics. (Operation of mechanical analog) We now have a pictorial representation of the oscillations obtained for operation with poorly matched engine and control-response characteristics.

The air accumulator can be bypassed and the time lag removed from the control system by simply turning these valves. Upon elimination of this time lag, the response characteristics of the control system are more suitably matched with the response characteristics of the air turbine; and if we once more accelerate the air turbine, we will be able to note the difference in the type of response obtained. (Operation of mechanical analog) We now have a visual comparison of the two types of response obtained. In the first case, the oscillations resulting from poorly matched engine and control-response characteristics, and in the second case the smooth increase in speed resulting from more properly matched engine and control-response characteristics.

Here, on the end of our demonstration platform, we have an electrical analog constructed for the purpose of investigating the response characteristics of a full-scale experimental electronic control system. I would like to emphasize that although this apparatus has had its face lifted for demonstration purposes, it is in actuality a bench setup from which we have obtained controls research data. The components of the electrical analog are as follows: Here we have the actual housing which contains the motor and gearing for changing the propeller-blade angle. Located here is the electric motor which is simulating the action of the gas-turbine engine. Here, in the proverbial "little black box" is the electronic control, and the electronic control is provided with this adjustment for changing its response characteristics. A desired condition of engine speed is set with the control set knob located here. The control functions to maintain a desired speed setting by increasing the propeller-blade angle should the engine speed tend to increase above the control setting, and by decreasing the propeller-blade angle should the engine speed tend to drop below the control setting. We will first demonstrate the action of this electronic control with the response characteristics poorly matched with the response of our simulated engine, the electric motor. During operation, you will once more note the actual change in speed as it occurs on the moving chart. You will also note the fluctuations in propeller-blade angle as the control moves the propeller in an attempt to maintain the desired speed setting. (Operation of electrical analog) Here, again, we can see the oscillations in speed resulting from poorly matched control and engine-response characteristics.

The response characteristics of the electronic control will now be adjusted so that they are more adequately matched with the response of the electric motor. We will once more accelerate to a new condition of engine speed so that you may obtain a comparison of the response of the system with the response previously indicated. (Operation of the electrical analog) We now have, again, a visual comparison of the types of response which can be obtained through poorly matched and more properly matched engine and control-response characteristics.

These demonstrations have been for the purpose of acquainting you with the nature of the problems now confronting us in controls research. The type of matching which has been illustrated here is obviously much more simple than the matching that is required for full-scale engine operation. Here, on the demonstration platform, we have had to account for the response of only one or two variables. Our research, when extended to ultimate full-scale engine operation, must include the effects of the multitude of variables which influence engine operation; that is, the effects of engine speed, propeller-blade angle, fuel flow, gas temperature, airplane speed, altitude, and so forth. In our research dealing with full-scale engine operation, we have derived mathematical equations to describe the response of the engine and the solution of these equations will provide information leading to the required control-response characteristics. However, these equations are so complicated that they are impossible to solve with conventional higher mathematics and it is necessary for us to resort to the use of a special electronic computer with circuits specifically designed for the solution of these engine equations. In addition to the analytical investigations, we are also instrumenting a full-scale turbo-propeller engine for the purpose of investigating gas-temperature-control response, engine-speed-control response, and propeller-blade-angle control response. Our research engine is located in the adjacent test cell, and although it cannot be operated for you because of the prohibitive noise level, it is available for your inspection. Thank you.

→ See photograph C-22345

David Novik
10-5-48

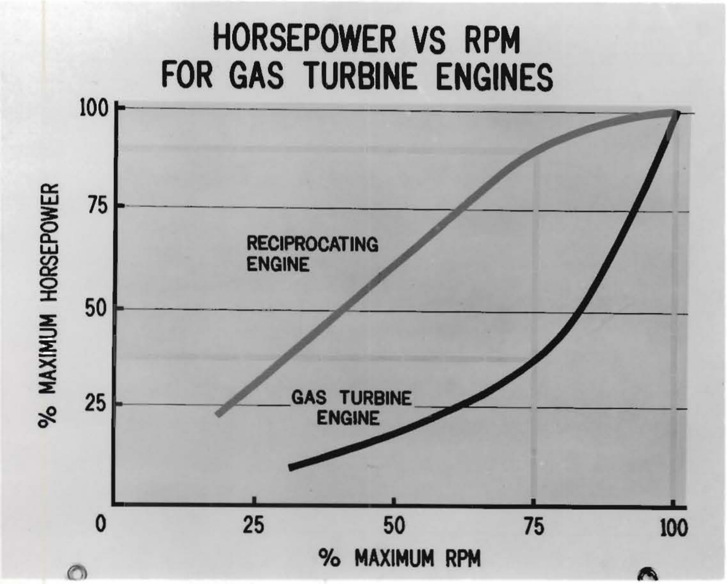


Figure 48.

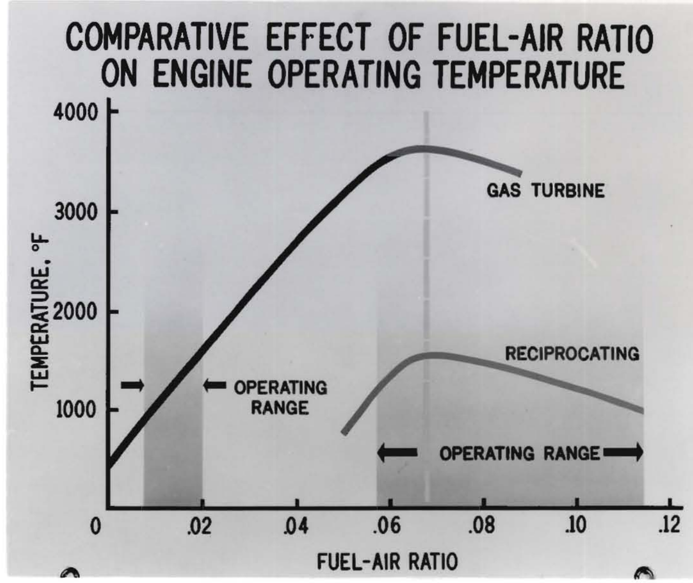


Figure 49.

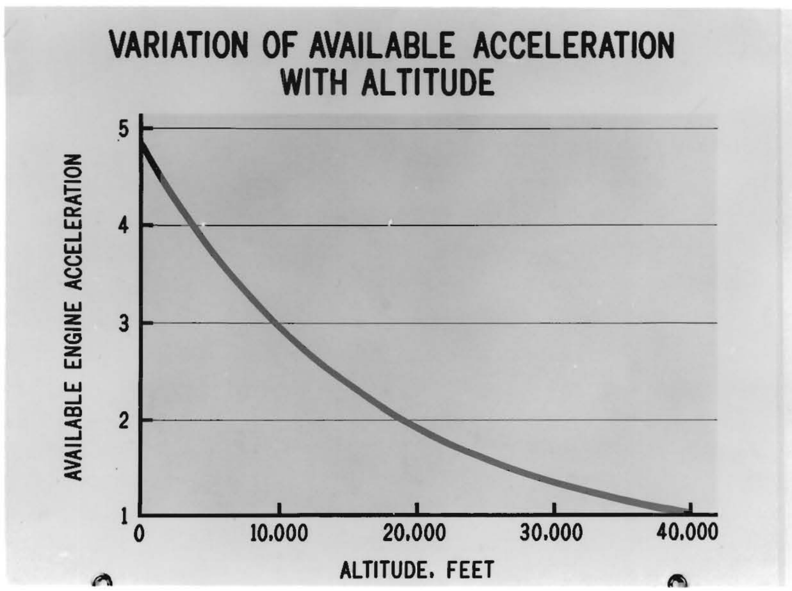


Figure 50.

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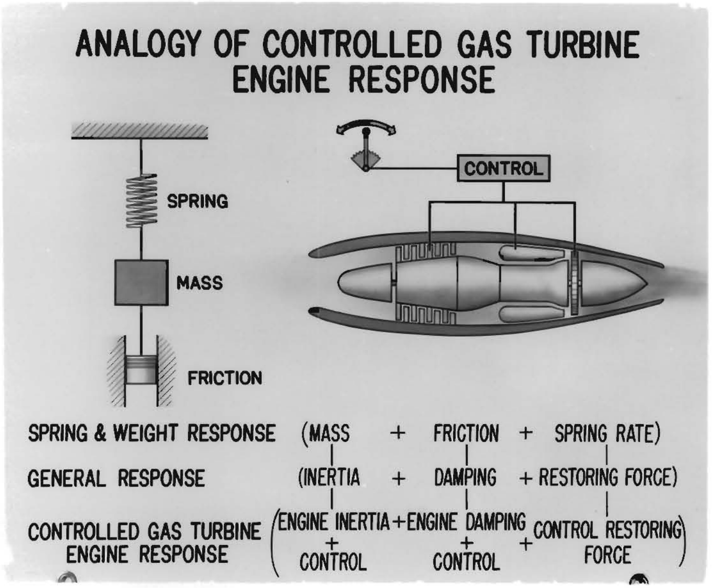


Figure 51.

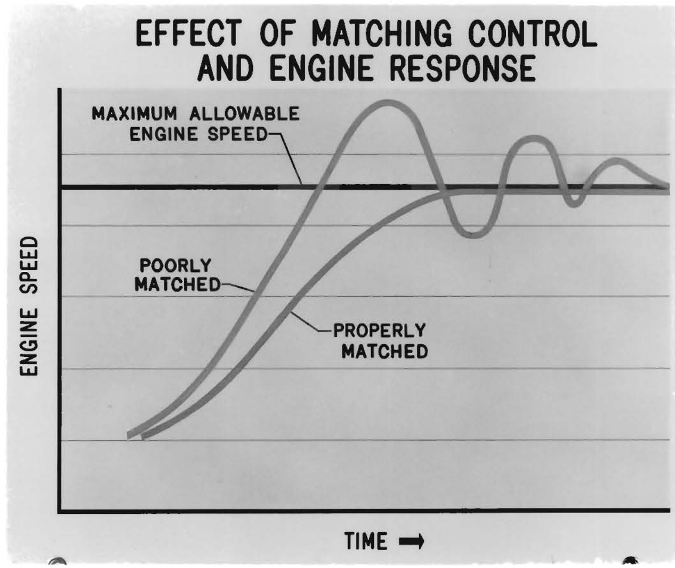


Figure 52.

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CONTROLS RESEARCH
ANALOGY OF CONTROLLED GAS TURBINE ENGINE RESPONSE
SPRING
MASS
FRICTION
CONTROL
SPRING & WEIGHT RESPONSE
GENERAL RESPONSE
CONTROLLED GAS TURBINE ENGINE RESPONSE
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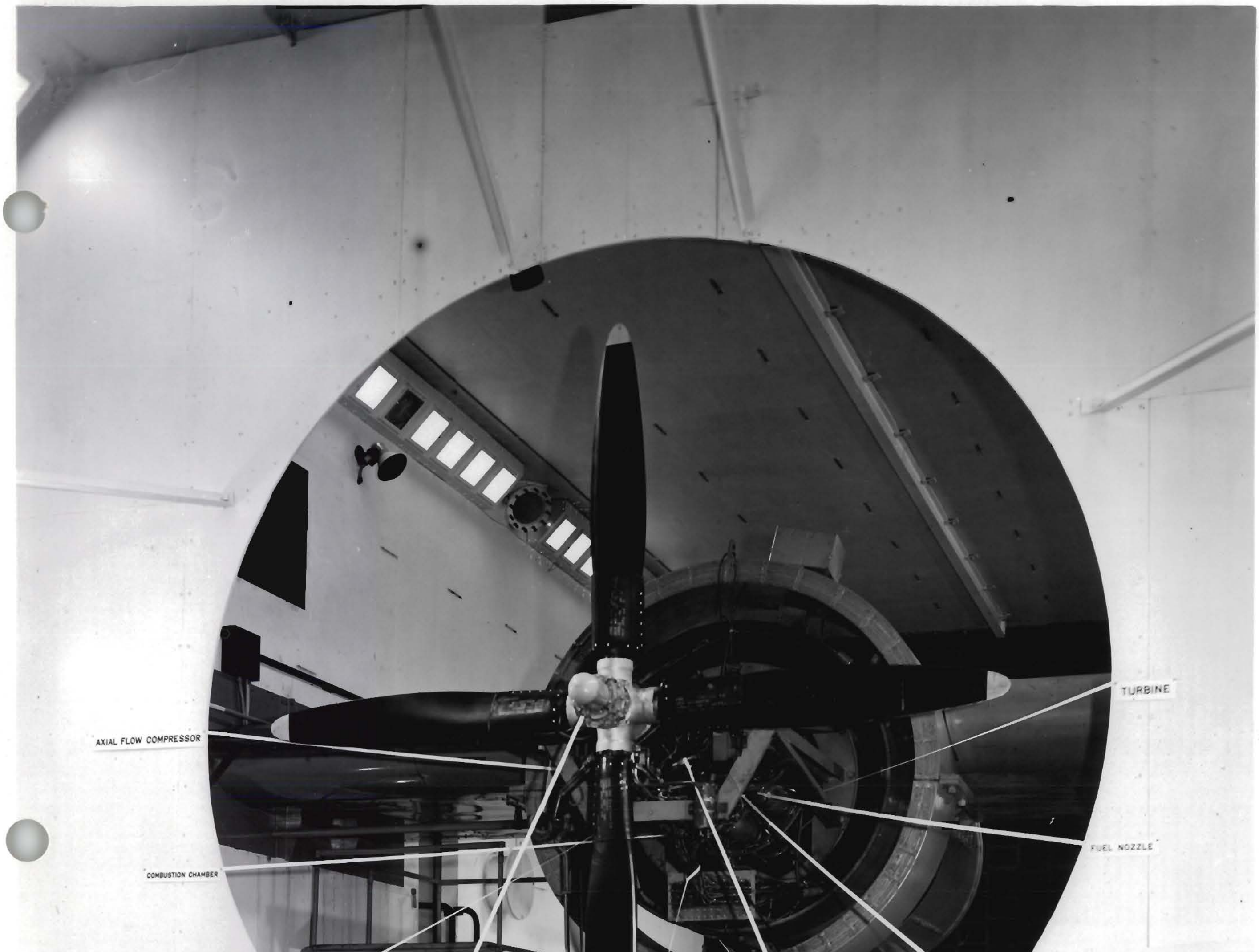
TURBINE
AXIAL FLOW COMPRESSOR
FUEL NOZZLE
COMBUSTION CHAMBER
FUEL MANIFOLD
COMPRESSOR INLET
PROPELLER PITCH
MOTOR HOUSING
REDUCTION GEARS
& ACCESSORY DRIVES
RESEARCH INSTRUMENTATION TUBING
C-22345
9-28-48

NACA

Talk I
AWT

TURBOJET RESEARCH

Theoretical and experimental research has shown that the greatest cruising range of turbojet-powered aircraft is obtained at high altitudes. However, the turbojet engine in its present stage of development has several characteristics which restrict its effectiveness as a high-altitude power plant. Two such characteristics of major importance are combustion blowout and engine acceleration. A study of these characteristics by the NACA has indicated certain design trends which would improve the effectiveness of the turbojet engine at high altitudes.

Research on turbojet engine components, particularly combustion chambers, has considerably increased the altitude at which an engine can be operated over a full range of engine speeds without encountering combustion blowout. The first slide (Figure 53) shows results obtained in the altitude wind tunnel on a particular series of engines. In 1944 the altitude limit of a typical turbojet engine was 19,500 feet. Mainly through progressive improvement of combustion chamber design, the maximum altitude was increased to 30,000 feet in 1945 and to 45,000 feet in 1947. Inasmuch as combustion blowout is associated with low combustion chamber pressures which occur in high-altitude flight at low engine speeds, the maximum altitude could logically be increased by raising the pressure level throughout the engine. Consequently the effect of increasing the design compressor pressure ratio on the maximum altitude and engine performance should, therefore, be considered.

The next slide (Figure 54) shows the maximum altitude at which an engine can be operated from idling to rated speed plotted against compressor pressure ratio. For an engine with a compressor pressure ratio of 4 to 1, which is representative of most existing turbojet engines, the maximum altitude is

45,000 feet. By increasing the pressure ratio to 8 to 1, the maximum altitude is raised to 59,000 feet and a further increase in pressure ratio to 16 to 1 would allow operation at all engine speeds at 74,000 feet.

Further studies indicate that increased compressor pressure ratio will also result in reduced fuel consumption which can be interpreted in terms of increased cruising range. The next slide Figure 55 shows the relation between compressor pressure ratio and relative range for an airplane with a turbo-jet engine cruising at 500 miles per hour at an altitude of 30,000 feet. The range of an engine with a pressure ratio of 4 to 1 is used as a reference. Calculations indicate that by doubling the pressure ratio, the range is increased by 18 percent. An increase in pressure ratio from 4 to 16 results in a 40 percent increase in range. These results, therefore, indicate that increasing the compressor pressure ratio will not only increase the maximum operating altitude but will also give substantial gains in range. Considerable effort is being devoted at this laboratory toward improving compressor performance, as is being demonstrated in other presentations today.

Now let us consider the acceleration problem. The next slide Figure 56 shows the time required to accelerate the engine so as to increase the thrust from 30 percent to 100 percent of rated thrust as a function of altitude. At sea level, the time required to obtain this thrust increase is about 4 seconds. It is characteristic of a turbojet engine that, as the altitude is raised, a longer period of time is required to obtain a given thrust increase by accelerating the engine because the available turbine power decreases with a reduction in air density while the inertia of the rotating parts remains constant. The acceleration time increases with altitude until at 60,000 feet the same thrust increase requires 42 seconds. However, from a tactical viewpoint it is often necessary to increase the thrust more rapidly than this. Because very little can be done to reduce the inertia of the rotating parts or increasing turbine output without exceeding existing allowable temperature limits, the situation demands some means of changing thrust without changing engine speed. At a constant engine speed, the velocity of the exhaust jet and consequently the thrust can be changed by a variation in exhaust nozzle area. By use of a variable-area exhaust nozzle, the engine can be operated continuously at rated speed and the time required to change the thrust from 30 percent to 100 percent of rated thrust is in the order of 2 seconds or less at all altitudes. A typical variable-area exhaust nozzle is shown here (See photograph C-22347) and the rate at which thrust can be increased or decreased is a function only of the speed of the nozzle actuator.

The variable-area exhaust nozzle is a solution to the acceleration problem, but consideration must be given to its effect on operating economy over the range of engine thrusts.

The next slide (Figure 57) shows specific fuel consumption plotted against percent of rated engine thrust for a fixed-area and a variable-area exhaust nozzle. Over the range of thrust values shown, the specific fuel consumption of an engine with a variable-area exhaust nozzle operating constantly at maximum engine speed was about 6 to 8 percent higher than for the fixed-area nozzle where thrust changes are obtained by changing engine speed. However, these results were obtained with an experimental variable-area exhaust nozzle which had a considerably lower efficiency than the fixed-area nozzle. Calculations indicate that for the engine considered here having a variable-area nozzle with the same efficiency as the fixed-area nozzle, the specific fuel consumption will be slightly better than that obtained with the fixed-area nozzle. It should be emphasized that the effect of the variable-area nozzle on specific fuel consumption would vary from one engine to another depending on the characteristics of the engine components. Investigations at this laboratory of variable-area nozzles on other engines have shown as much as 6 percent reduction in specific fuel consumption in comparison with the fixed-area nozzle. Research work is being continued toward improving the efficiency of the variable-area nozzle.

Thus from the theoretical and experimental results presented here, it is evident that the effectiveness of the turbojet engine at high altitudes can be greatly improved by increasing the compressor pressure ratio and by use of a variable-area exhaust nozzle.

Mr. Gray will present the next discussion on icing research on turbojet engines.

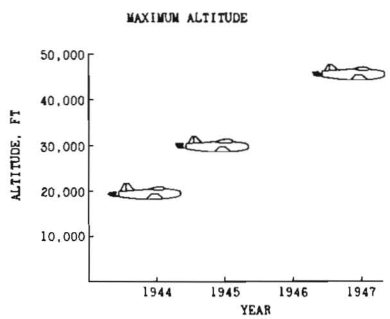


Figure 53.

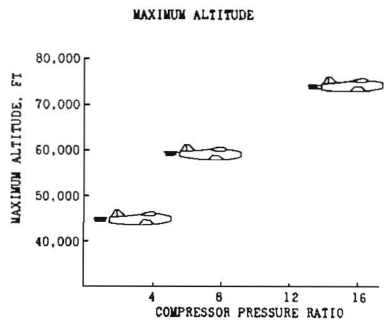


Figure 54.

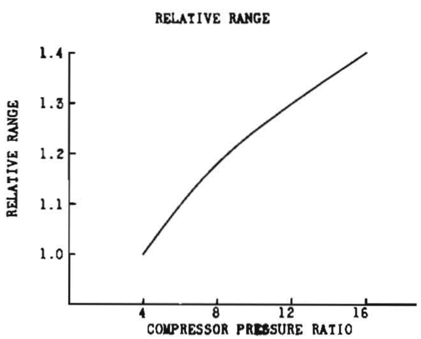


Figure 55.

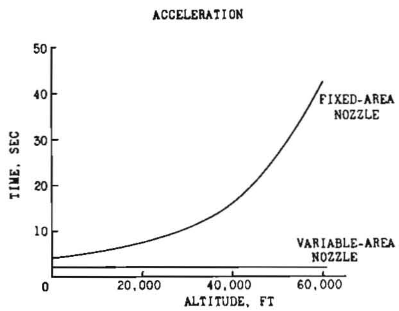


Figure 56.

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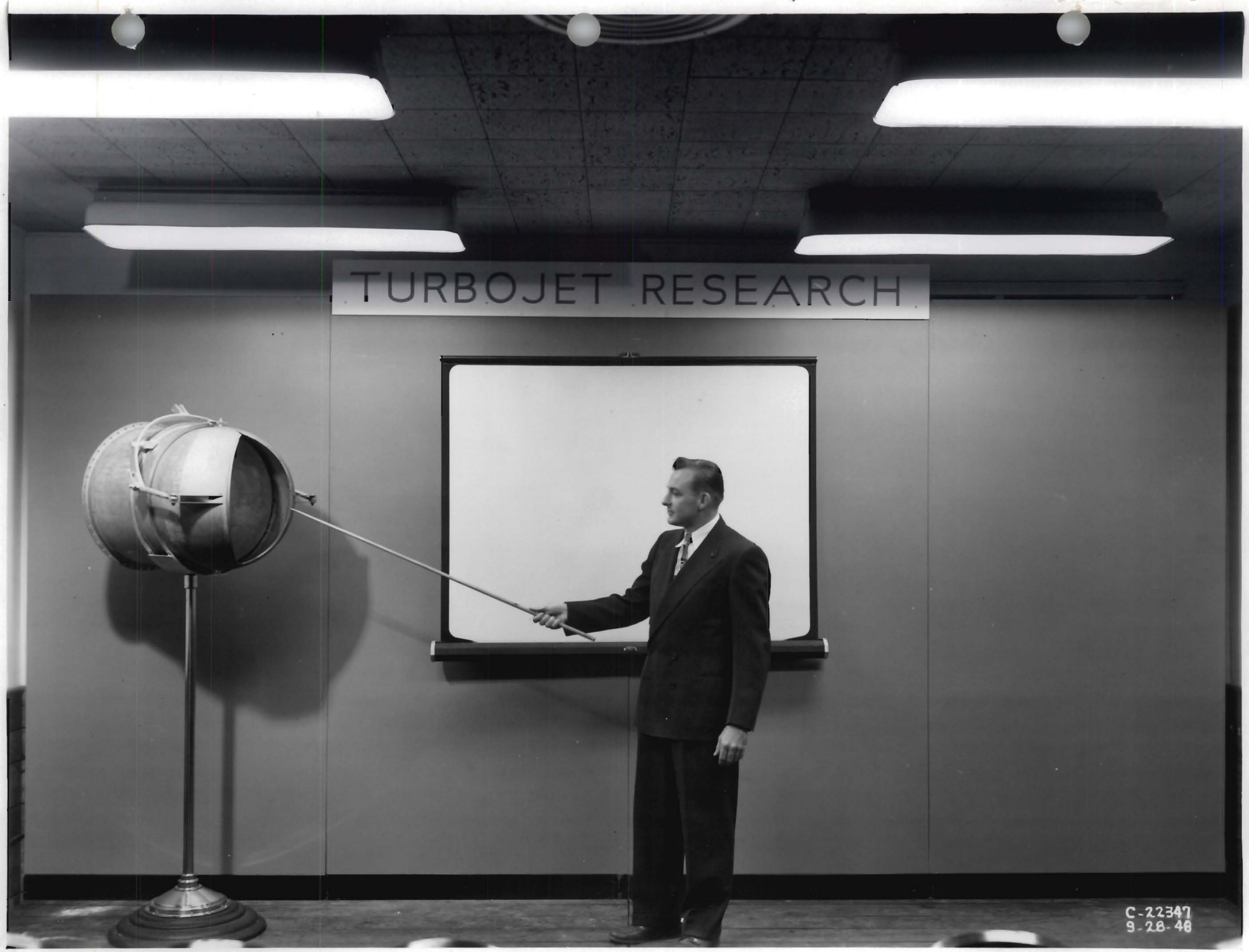


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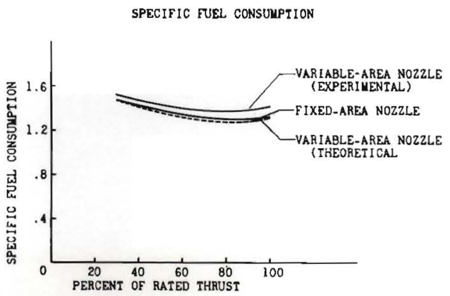


**Figure 57.**

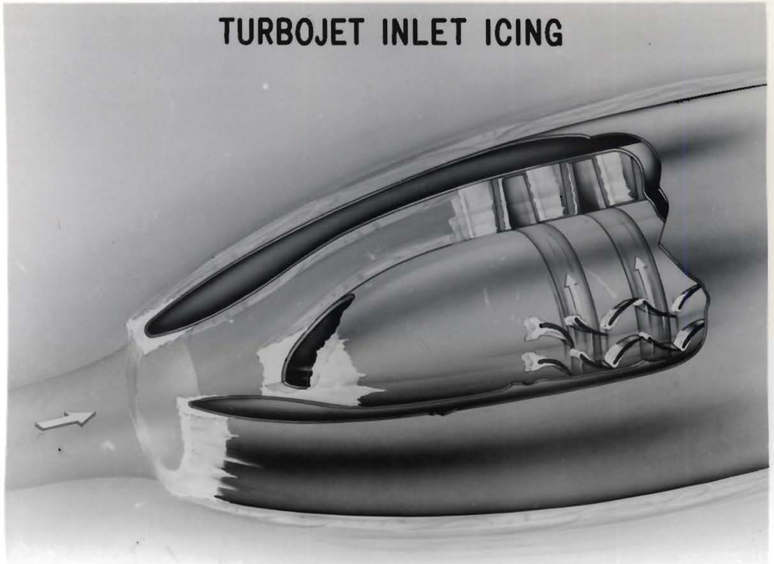


**Figure 58.**

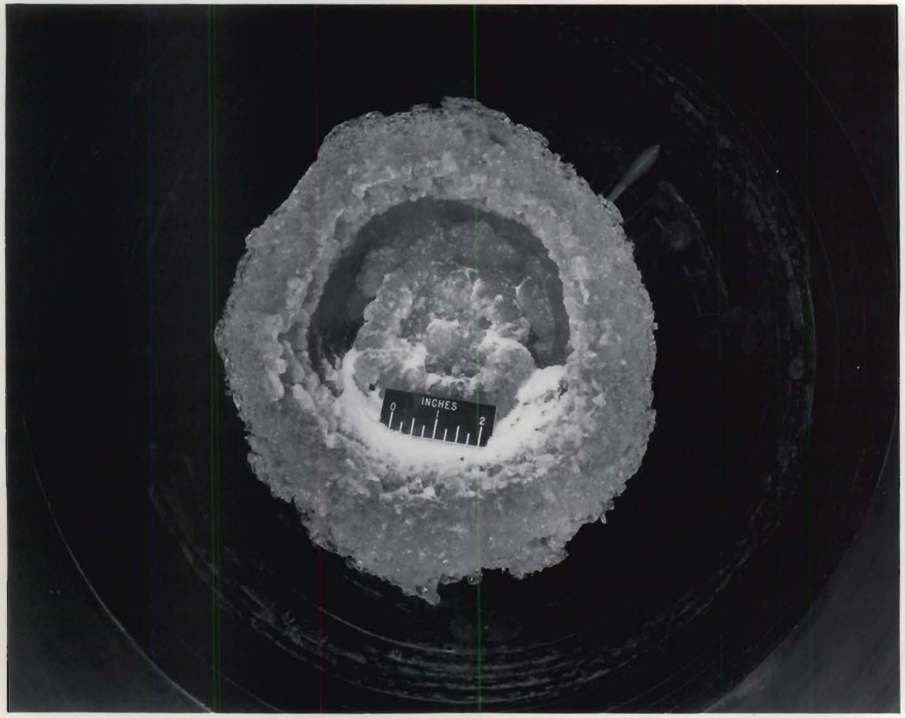


**Figure 59.**

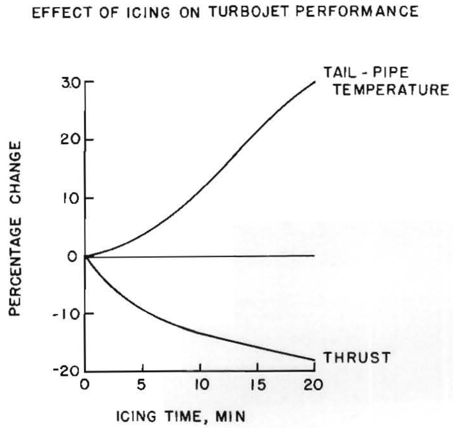


**Figure 60.**

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Talk II
AWT

The Vulnerability of Turbojet Engines to Icing Conditions

Gentlemen:

An axial-flow turbojet engine may be rendered completely inoperative after one to six minutes of flight in severe icing conditions. We shall illustrate some serious turbojet ice formations, explain how these ice deposits handicap a jet engine and describe what methods may be used to prevent the formation of ice. The accretion of ice around a jet engine inlet in flight can be seen in the following short movie which shows several views from last season's icing flights. (start movie) A turbojet engine was suspended beneath the wing of a B-24 which was fully equipped for all-weather flying. The turbojet was operated continuously during the flights and measurements were made of the thrust, airflow, fuel consumption and tail-pipe temperatures. Ice first becomes evident in a thin smooth line around the inlet lips. With passage of time, this formation can be seen to widen, become thicker and more irregular, and take on a variety of shapes. More difficult to see, and not usually visible to the pilot, is the ice which forms on the accessory housing and on the inlet guide vanes.

All of these flights were made in mild icing conditions. From these flight experiences and from icing investigations on a full-scale turbojet in the Altitude Wind Tunnel, it has been established that severe losses in jet engine performance result from icing; the results from one of these flights will be presented later. (end of movie)

The first slide Figure 58 illustrates why such ice formations penalize the performance of an axial-flow turbojet. The reduced inlet opening, the blockage through the inlet guide vanes, and the roughened duct surfaces all contribute towards a marked decrease in the ram pressure recovery. In addition the ice formations on the compressor blades cause a considerable loss in compressor efficiency and a consequent decrease in mass air flow. This reduction in air flow and ram pressure recovery results in a sizeable loss in net thrust.

A close-up photograph of turbojet icing is shown by the next slide. Figure 59 This picture was taken on the ground after a flight of 45 minutes in mild icing conditions at an average air temperature of 22° F with the engine operating at low cruise power. Heavy deposits of ice are seen on the inlet lips, the accessory housing and the inlet guide vanes. There was also measureable amounts of ice on the leading edges of the first stage rotor and stator blades.

The performance losses during the first 20 minutes of this icing flight are shown on the next slide. Figure 60 The loss in thrust after entering the icing condition was rapid, and after 20 minutes the net thrust was reduced 18 percent, while the tail-pipe temperature increased up to 30 percent. Had the engine been operating at rated power no increase in tail pipe temperature would have been permissible and an immediate reduction in engine speed would have occurred. This would render the engine less capable of inducing air flow through the iced passages and progressive reductions in speed would have taken place sufficient to cause complete engine shut-down.

There are two laboratory-developed methods of ice protection which can be applied as modifications to present turbojet engines. The first of these, which protects the engine by the elimination of water from the inlet air stream, is shown on the next slide. Fig. 61 The main passage to the compressor is provided with a screen designed to ice more quickly than the compressor blades. Upon entering an icing condition the screen rapidly becomes blocked with ice, sealing off the main passage and converting it into a trap for subsequent water entrained in the inlet air. The inlet air must then make a sharp bend into the alternate passage while the water particles, due to their greater inertia, continue into the main passage. Upon leaving the icing condition the entrapped ice can be melted to avoid the greater losses associated with the alternate passage. This system can increase the operational range in icing conditions up to several hundred miles.

The other method of ice protection for existing aircraft is illustrated on the next slide (Figure 62) wherein a portion of the hot gas at the turbine inlet is diverted and bled back to the inlet lips. The hot gas is discharged through a ring of orifices in the inlet and mixes with the charge air, which is thereby heated above the freezing level. A thrust loss attends this system of ice prevention, compounded not only from the percentage of weight flow which is bled-back, but from the decrease in density and the loss in ram pressure of the charge air after mixture with bleed-back gas. This system need not be used, however, when icing is not anticipated.

In the basic design of new jet engines, the aerodynamic and thrust losses associated with these two methods of ice protection may be largely eliminated by incorporating surface heating into the design. Vital surfaces may be heated by various means, such as electrical elements imbedded in or cemented to the surface, eddy-current generation, or hot air flowing through passages beneath the surface. The last slide Figure 63 shows schematically the use of hot air for heating exposed surfaces against icing. The inlet lips and ducting and the accessory housing are heated by air ducted forward from either the compressor or the combustion chamber outlet. Heated air is also ducted through the hollow stator and rotor blades and discharges at the blade tips into the inlet air. Calculations have indicated that the losses from such a system will be negligible. It should be emphasized, however, that this system is not an easy modification to existing installations, but should be included in the basic engine design.

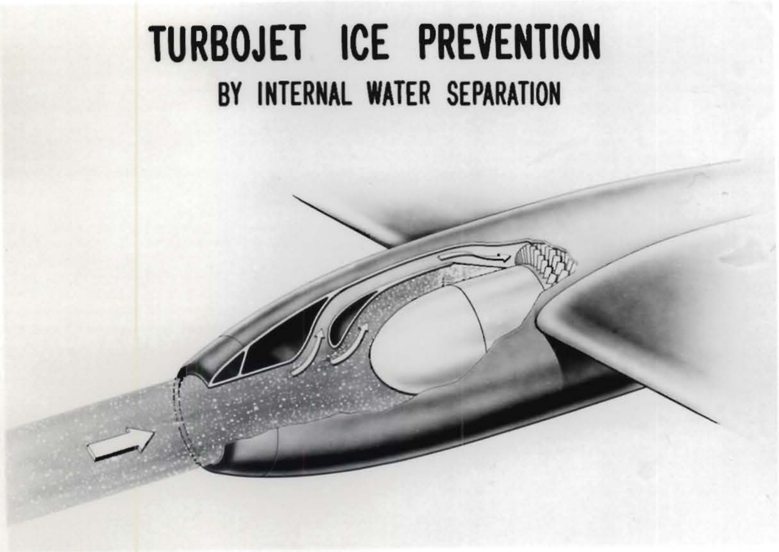


Figure 61.

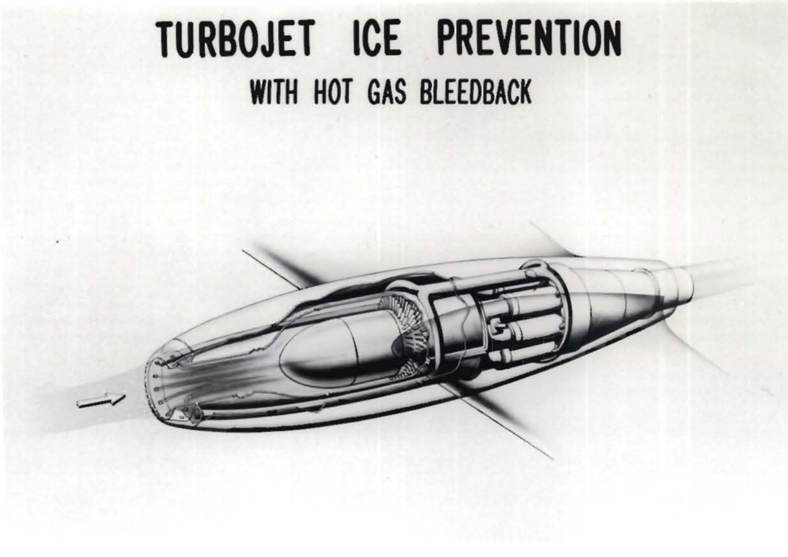


Figure 62.

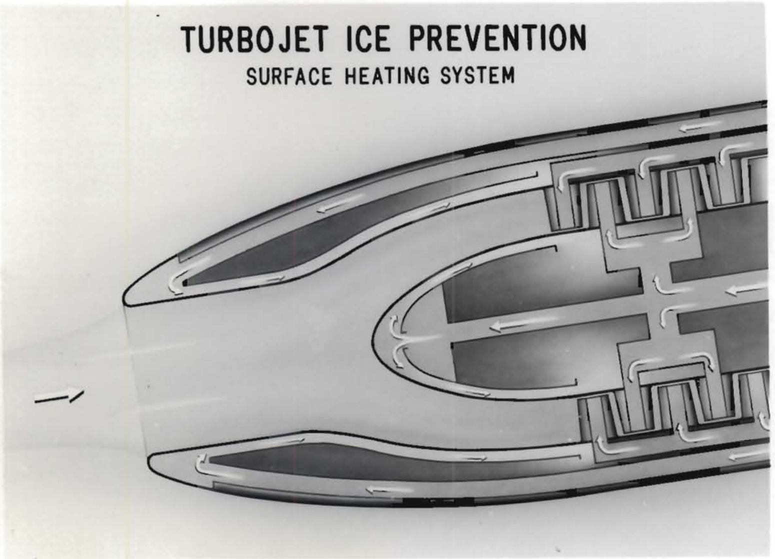


Figure 63.

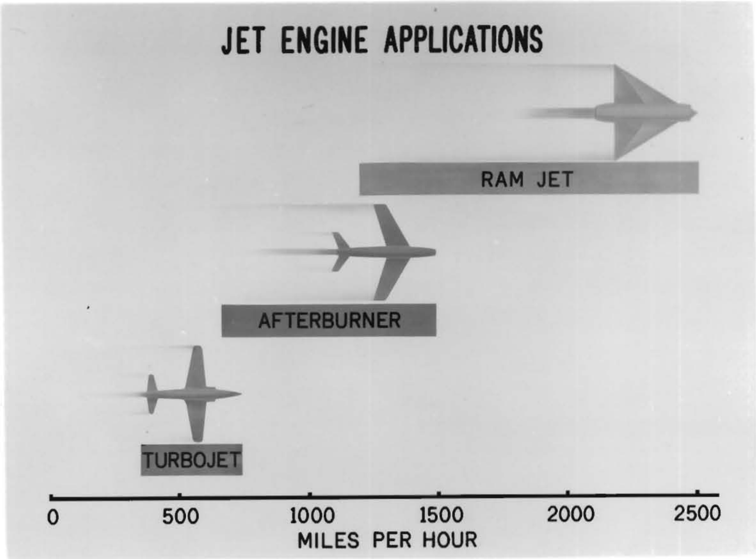


Figure 64.

Talk III
AWT

# TAIL-PIPE BURNING FOR THRUST AUGMENTATION

Increasing the thrust of turbojet engines is of extreme importance, both for sea-level take-off and for high-altitude, high-airspeed operation. One of the most promising methods of increasing the thrust of turbojet engines is the use of tail-pipe burning. An extensive research program is now in progress at the NACA Cleveland laboratory on this method of thrust augmentation. The maximum thrust can be obtained with a turbojet engine when the exhaust gas is heated to the highest possible temperature. However, the temperature ahead of the turbine is limited to about 1500° F because of material limitations of the turbine blades. Because of this temperature limitation, only about one-quarter of the oxygen in the air is burned in the engine combustion chamber. The function of the tail-pipe burner is, therefore, to burn the remaining oxygen left in the air after the turbine and thereby heat the gas in the tail pipe up to a temperature between 3000 and 3500° F.

The first chart (Figure 64) is a speed spectrum showing the range of application of three types of jet-propulsion engines. The turbojet engine without thrust augmentation is most applicable up to an airspeed of about 700 miles per hour. For the speed of 700 to 1500 miles per hour, the turbojet engine with afterburning is the most applicable power plant, and, from 1500 through 2500 miles per hour is the range of application of ram jets.

Drawings of a turbojet engine with both a standard tail pipe and a tail-pipe burner installed are shown in the next chart. (Figure 65) This chart illustrates the physical characteristics of these installations, as well as the relative size. Fuel is injected a short distance downstream from the turbine through a number of streamlined tubes and mixes with the gas going into the tail pipe. A short distance downstream of the fuel injection tubes is installed a flameholder which forms a stagnation or turbulent region upon which the flame seats. Burning occurs over the full length of the combustion chamber and the gases are expanded out of the jet nozzle at a temperature of about 3000° F. Installation of a tail-pipe burner on the engine increases the length of the installation by about 15 percent and the weight by about 20 percent; however, the frontal area remains unchanged. The results, which will be presented in the following charts, were obtained in the altitude wind tunnel with this tail-pipe burner configuration.

Here on Figure 66 is shown the thrust plotted against airspeed for the standard engine and for the engine operating with tail-pipe burning. At static conditions, the thrust is increased 35 percent. Such a thrust increase would reduce the take-off distance of an airplane which might normally require 5000 feet to a distance of only 3200 feet. At an airspeed of 650 miles per hour, the thrust is increased by about 80 percent.

These thrust increases obtained with tail-pipe burning must be paid for with an increase in specific fuel consumption, as is shown in the following chart. Figure 67 At static conditions, the specific fuel consumption with afterburning is 109 percent greater than that of the standard engine with no tail-pipe burning. Increasing the airspeed increases the efficiency of the tail-pipe burning cycle so that, at an airspeed of 650 miles per hour, the specific fuel consumption with tail-pipe burning is 76 percent higher than that of the standard engine without tail-pipe burning. This may appear to be a considerable price to pay for the increased thrust obtained. However, these increases in specific fuel consumption are much less than those obtained with other methods of thrust augmentation.

The use of tail-pipe burning on turbojet engine introduces several difficult problems. Four of these problems are outlined on the following

Figure 68

chart. It is necessary that a tail-pipe burner operate over a wide range of flight conditions. With the tail-pipe burner which has been described, operation has been obtained up to an altitude of 50,000 feet. Another important consideration of a tail-pipe burner installation is that the loss in performance must be kept to a minimum when the tail-pipe burner is inoperative. A loss in thrust of 3 percent was obtained with this tail-pipe burner at all flight conditions.

It has been found that fuel injected into the 1200° gas flowing through the tail pipe will not ignite by itself. It is, therefore, necessary to provide a dependable source of ignition. A considerable number of spark-plug configurations installed in the tail pipe proved inadequate for ignition at some conditions. An ignition system which has been developed in the altitude wind tunnel will operate satisfactorily up to an altitude of 50,000 feet. This system comprises an instantaneous injection of high-pressure fuel into one of the engine combustion chambers which results in a burst of flame back through the turbine into the tail pipe, thereby igniting the tail-pipe fuel. Over 200 starts have been made with this ignition system and no excessive deterioration has been found on either the turbine nozzles or the engine combustion chamber.

Because the gas flowing through the tail-pipe burner is at a temperature of about 3000° F, it is obvious that an adequate method of cooling the burner shell must be provided. This cooling should preferably be provided without an appreciable increase in installation weight and without suffering an appreciable loss in airplane or engine performance. Adequate cooling of the tail-pipe burner has been obtained at all flight conditions investigated with a cooling liner installed inside of the burner shell.

The liner extends the full length of the combustion chamber and a ½-inch space is provided between the liner and the burner shell. Approximately 6 percent of the 1200° gas flowing out of the turbine passes between the liner and the outer shell of the tail pipe. This flow through the cooling air passage maintains the temperature of the outer shell between 1200 and 1250° F at all flight conditions. The method of construction and support of the liner in the tail pipe burner is shown by the model displayed.

There are numerous other problems which have been encountered in the investigation of tail-pipe burning on turbojet engines, such as obtaining a satisfactory variable-area exhaust nozzle and an adequate control system. Problems such as these are receiving further attention in the research program on tail-pipe burning at this laboratory.

Motion pictures will following showing the installation and operation of the tail-pipe burner in the altitude wind tunnel.

In conclusion it might be stated that from the knowledge gained of tail-pipe burning here in the altitude wind tunnel, it is now possible to design burners which will operate at altitudes up to 50,000 feet with reasonably good efficiencies.

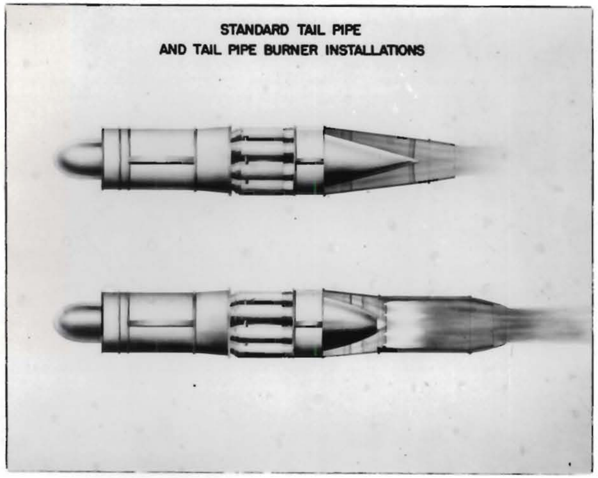


Figure 65.

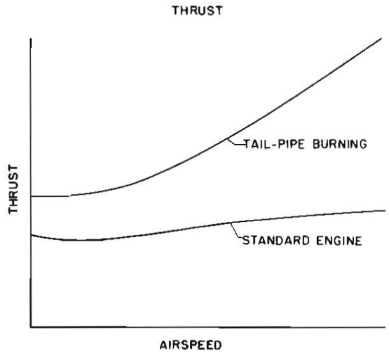


Figure 66.

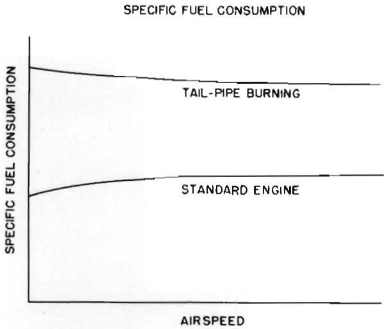


Figure 67.

| TAIL-PIPE BURNER PROBLEMS | RESULTS |
| --- | --- |
| 1-WIDE OPERATING RANGE | 1-50,000 FT ALTITUDE |
| 2-PERFORMANCE LOSS | 2-3-PERCENT THRUST LOSS |
| 3-BURNER IGNITION | 3-50,000 FT ALTITUDE |
| 4-SHELL COOLING | 4-INNER LINER |

Figure 68.

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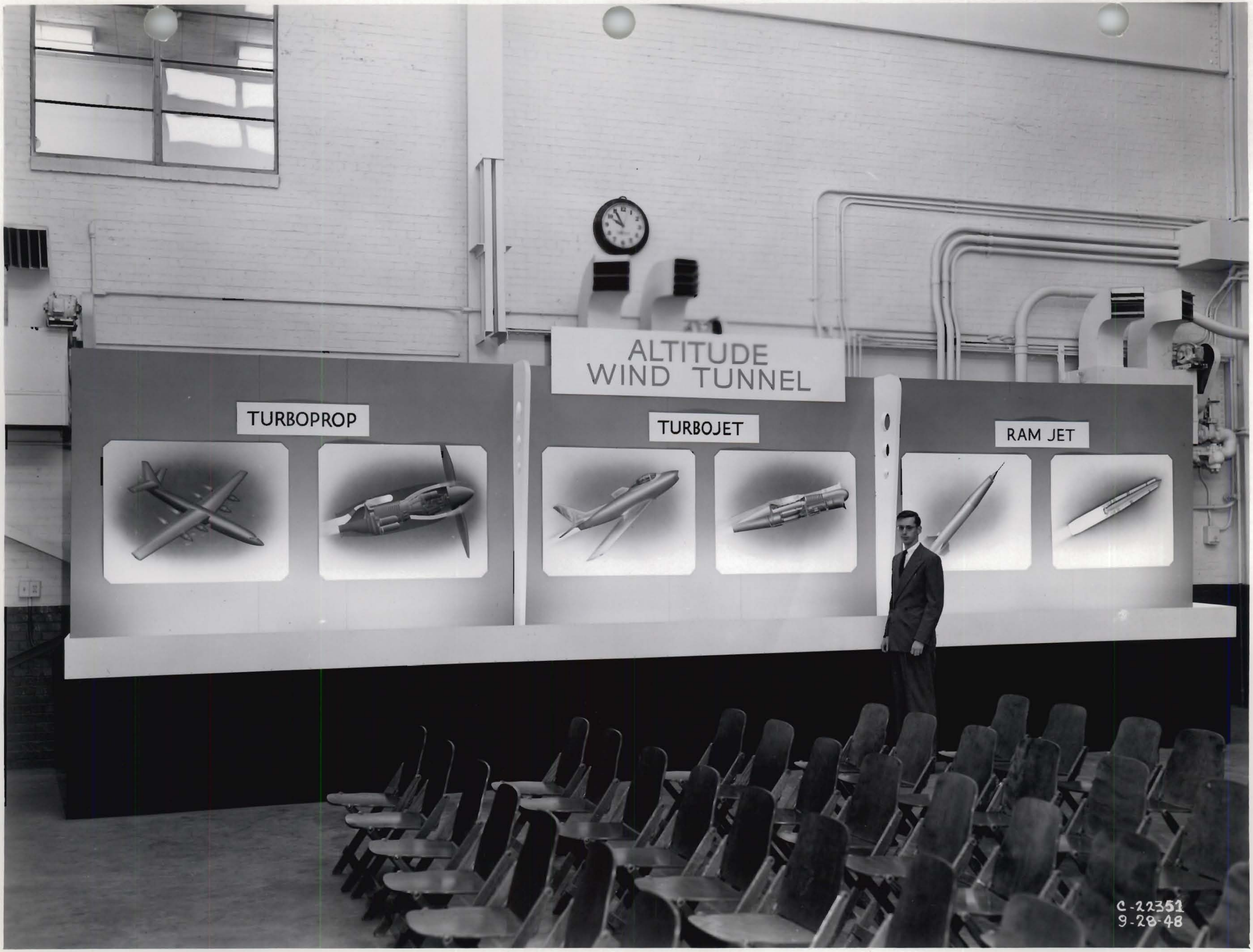
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