ANNUAL INSPECTION -- OCTOBER 8-9-10, 1947

TUNNEL FACILITIES AND PROPELLER RESEARCH

D.R.Mulholland

Icing conditions in the atmosphere are the source of serious menace to safe aircraft operation and require careful study to provide methods for adequate and reliable protection. The icing research tunnel, shown in the first chart, Fig. 61 is a unique type of wind tunnel in which simulated atmospheric icing conditions are produced and imposed upon various aircraft components to determine their vulnerability to ice formation. The tunnel is a return flow type of rectangular cross section. The air passes from a large drive fan through refrigeration coils, past the water spray system to the test section, and back through the fan. The test section of the tunnel is 9 feet wide and 6 feet high, where air velocities may reach 400 miles per hour and air temperatures can be reduced to minus 30° F. Within the test section, icing research has been carried out on jet engine inlets, frontal engine and cowling surfaces, and many similar units. At the present time, research on wing ice protection is being conducted which will be more fully explained and demonstrated shortly.

Research using various methods of thermal ice protection for propellers was conducted some time ago in the diffuser of the tunnel, where cross-sectional dimensions allowed operation of a full-scale propeller. An airplane fuselage together with the engine was utilized to support and drive the propeller. The first studies were made with electrically heated propeller blades Fig. 62 ~~(chart 2)~~ using special heaters cemented to the external blade surfaces, while later studies involved the use of hot gas which was introduced at the blade shanks, passed through hollow steel blades, and exhausted at the blade tips. Electrical heating is a highly efficient system and lends itself to intermittent application of electrical power to produce controlled shedding of ice formations. This method cyclical de-icing thus allows a significant reduction of the electrical power required for propeller protection.

For some installations where ice throwoff is particularly undesirable, continuous heating must be utilized, either electrically or by some other means, such as heating the propeller blades with hot gas. Protection of propeller by means of hot gas has been investigated using fully hollow blades and also blades partitioned radially to confine the gas near the leading edge where ice formations are most prevalent. Using the hot-gas system, part of the available heat is lost as the gas is exhausted from the blade tip. Partitioning the blade, however, reduces the flow requirements and more fully utilizes the available heat so that the heating efficiency of the blade is increased. The values shown on this chart represent heat input in Btu per hour for icing protection of a singe blade, 2800 Btu per hour are required for cycled electrical heating using a "power-on" period of 20 out of 80 seconds, and 6800 Btu per hour per blade are necessary for steady electrical heating. For these cases, the heater covered the forward 30 percent of the blade and extended to 70 percent of radius. Using the hot-gas system which heats the blade throughout, 19,000 Btu per hour per blade are required for protection with a partition installed at 30 percent of chord, and [illegible],000 Btu per hour per blade for an unpartitioned blade. Although the hot-gas system has many advantages and can be adapted to a great number of installations, greater efficiency of such a system must be realized to justify its use. Analytical studies have indicated that improved internal partitions, plus the addition of more surface for heat transfer to the leading edge, will increase the efficiency of the hot-gas method of propeller blade heating and reduce the heating requirements to approximately 9000 Btu per hour blade using a partition at 30 percent of chord.

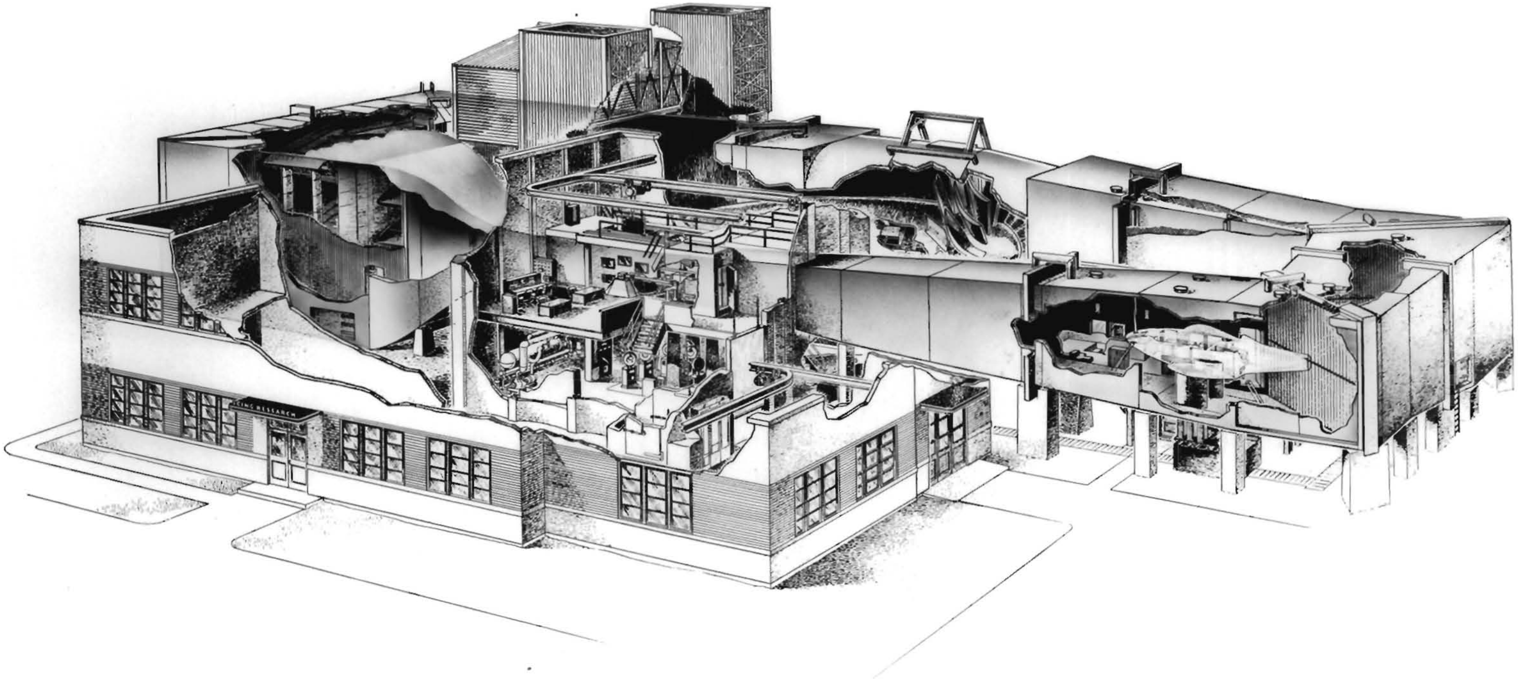
Further information concerning icing research with specific regard to jet engines will be discussed by Mr. Callaghan.

[illegible]:olh


ICING RESEARCH
ICING RESEARCH TUNNEL
SURFACE THERMOCOUPLES
HOT-GAS BLEEDBACK
INTERNAL WATER SEPARATION
C-19766
10-13-47

NACA

ICING RESEARCH TUNNEL



C- 19888
10- 24- 47

Fig 61

PROPELLER HEAT REQUIREMENTS

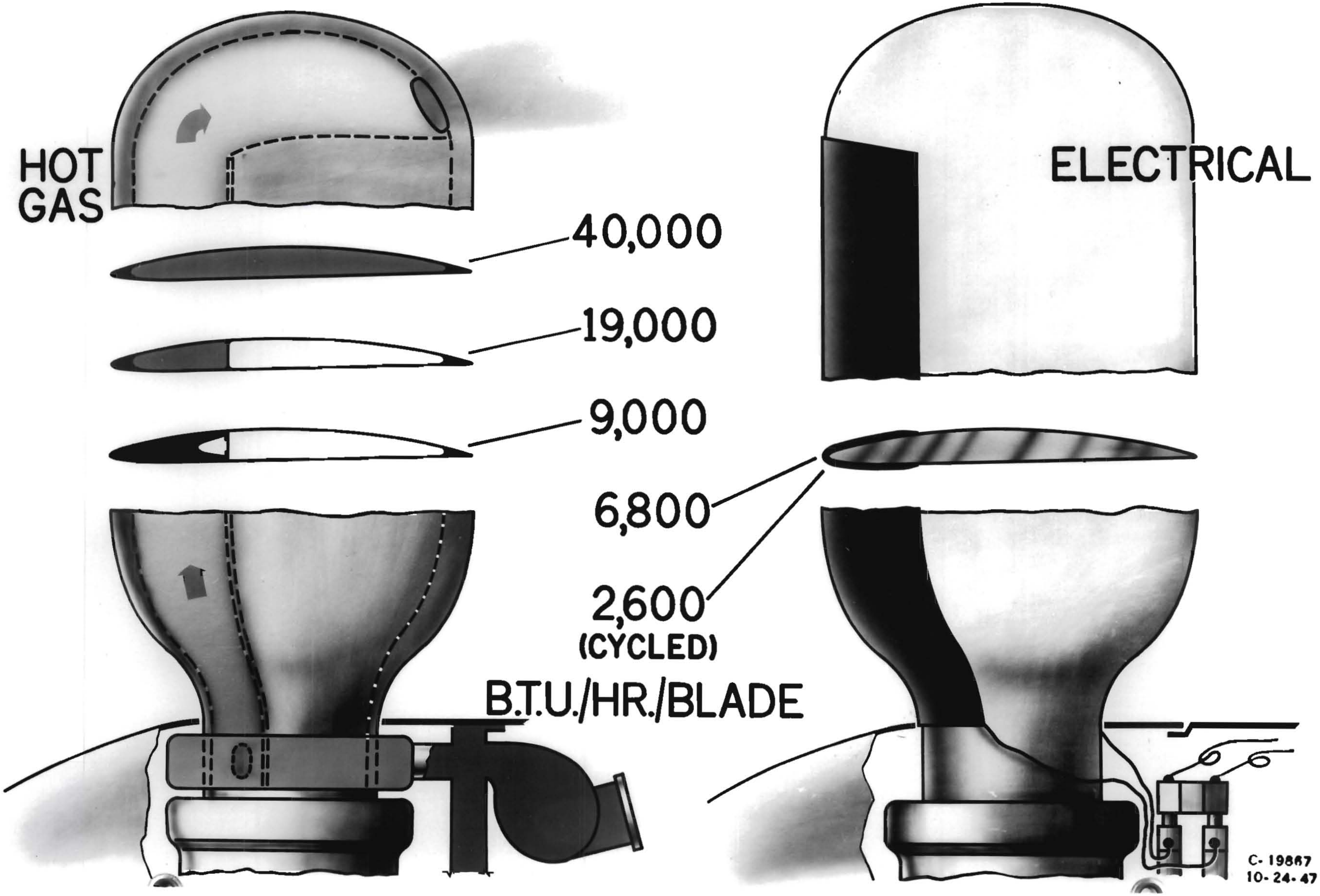
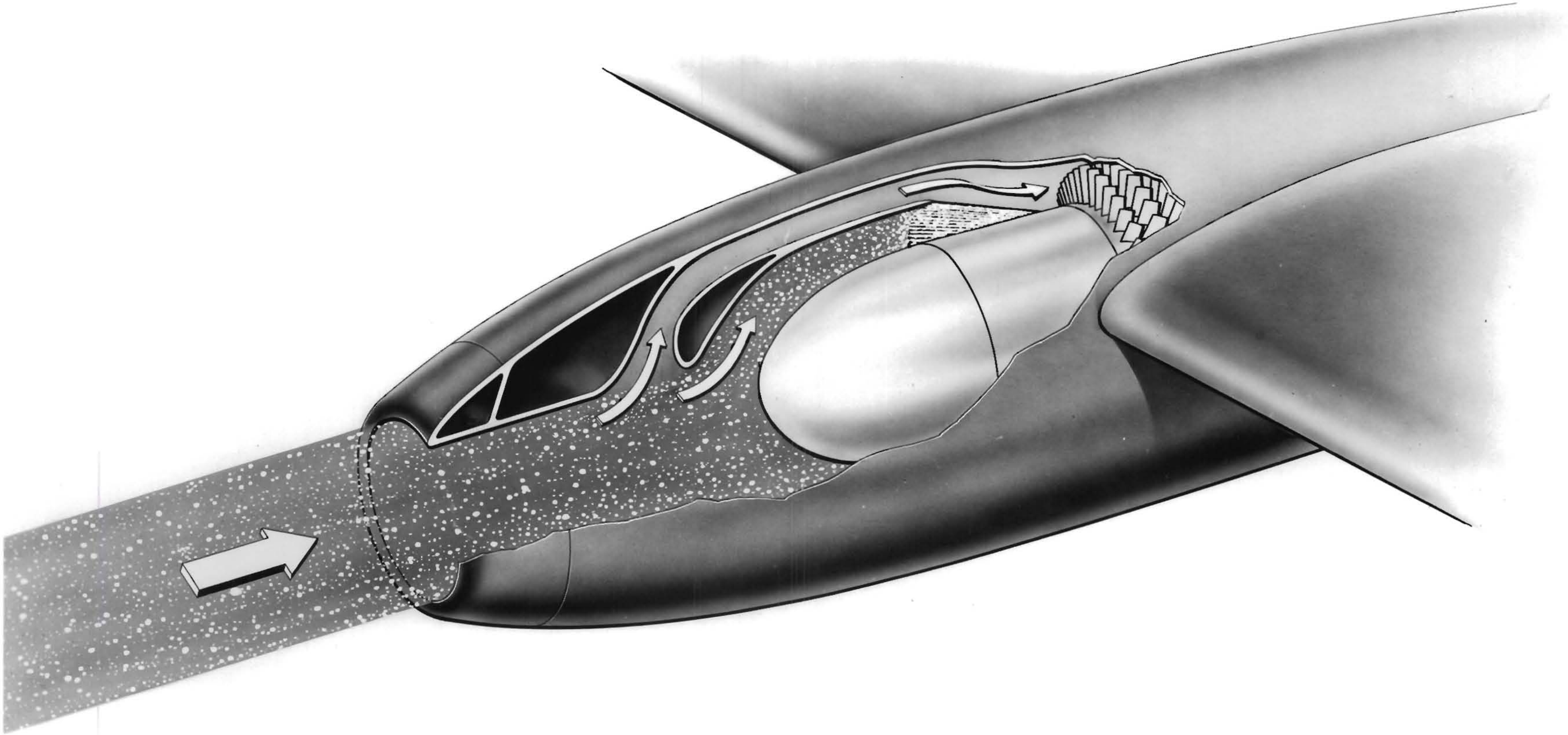


Fig 62

NACA

TURBOJET ICE PREVENTION

BY INTERNAL WATER SEPARATION



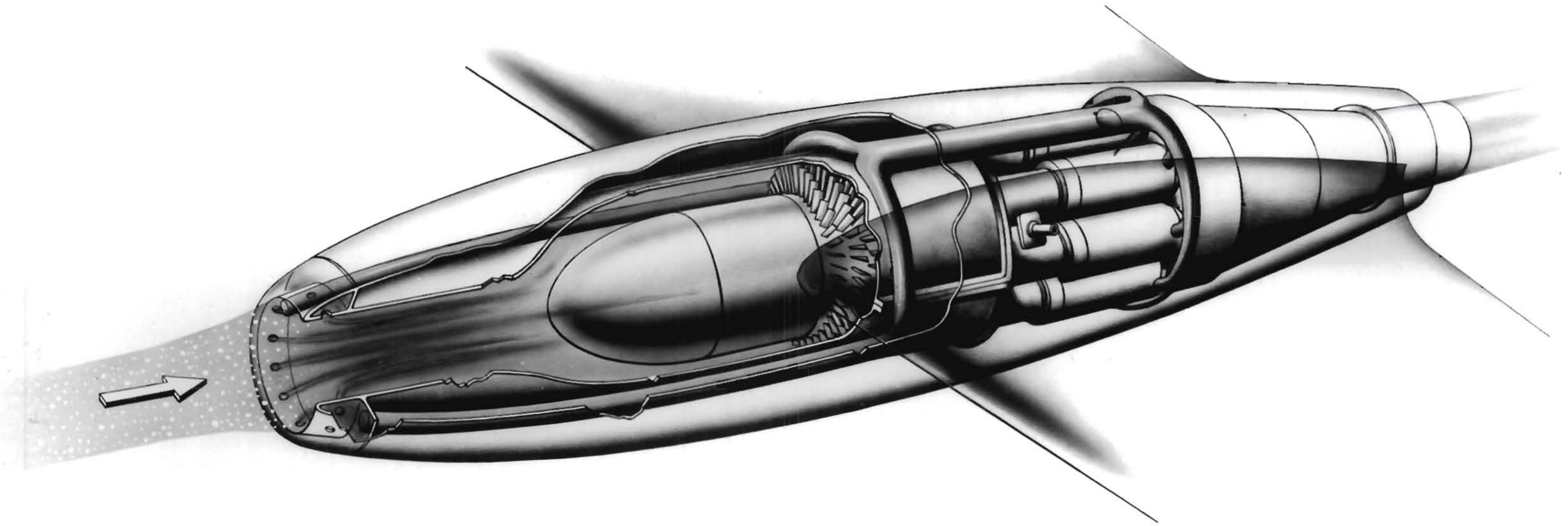
C- 19868
10- 24- 47

Fig. 63

NACA

TURBOJET ICE PREVENTION

WITH HOT GAS BLEEDBACK

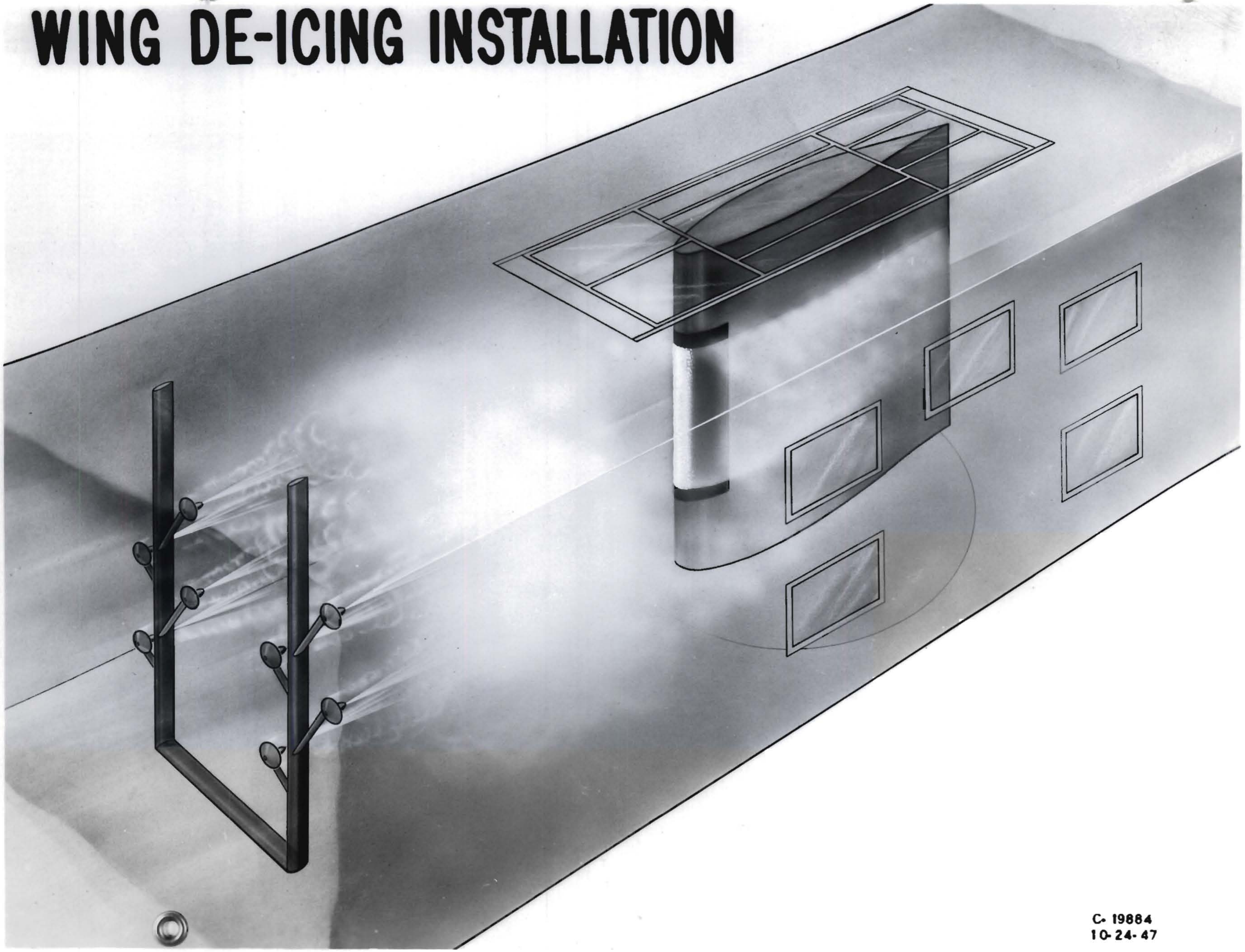


C-19864
10-24-47

Fig. 64

NACA

WING DE-ICING INSTALLATION



C- 19884
10-24-47

Fig. 65

NACA

C- 19873
10-24-47

NACA

Fig. 66


ICING INSTRUMENTS
ICING RATE METER
ANTENNA STRAIN GAGE
ROTATING CYLINDER
DEW POINT METER
BROWN
C-19779
10-13-47

NACA

ANNUAL INSPECTION -- OCTOBER 8-9-10, 1947

TURBOJET RESEARCH

E. E. Callaghan

The turbojet engine is extremely sensitive to impact icing and, even in a mild icing condition, the loss in air flow and the consequent increase in tail pipe temperature is usually prohibitive to continued operation; it is, therefore, essential that some means of automatic ice protection be provided for these engines.

Three fundamentally different methods of obtaining the required protection are being investigated at this laboratory. The first system utilizes the principle of inertia separation; that is, since the water droplets are heavier than the air, it is possible by suitable duct design to separate the water from the air entering the engine compressor due to the greater inertia of the water. The second system involves nothing more complicated than raising the temperature of the inlet air and water droplets above freezing. The third system is to heat each of the separate components subject to icing above the freezing level. Only the first two methods will be discussed at this time. A program to investigate the effectiveness of using each of the separate components has just been started.

This chart Fig. 63 shows a typical installation incorporating one of the designs used in the inertia separation investigation. This design consists of two concentric ducts. Under normal flight conditions, air enters the inlet and passes through the inner duct. When there are sufficient water droplets in the air to cause icing, the screen, which has the least icing tolerance, quickly becomes blocked. As this process progresses, air automatically starts passing through the outer duct in greater quantities until finally all the air is passing through the duct. The air entering the outer duct makes a sharp turn radially and the water droplets, because of their greater inertia, are unable to follow and pass into the chamber formed by the inner duct and screen where they are retained as a harmless ice formation which may be melted after the plane leaves the icing conditions.

The next chart Fig. 64 shows the gas bleedback system by which the temperature of the inlet air and water droplets are raised above the freezing level. Hot gas at about 1200° F is bled back under pressure from a jet engine burner to the nacelle lip and injected into the inlet airstream through small holes located around the periphery of the nacelle inlet. The location of sizes of these holes can be seen in the models shown here. The hot gases enter the inlet airstream at very high velocities due to the high pressures available at the turbine inlet. Because of these high velocities, the gas penetrates into the inlet airstream and mixes rapidly with the inlet air. It has been determined that, if 4 percent of the inlet air is bled back, adequate ice protection can be obtained at ambient air temperatures as low as -20° F.

Both systems discussed involve penalties of an aerodynamic, weight, and structural nature. For the inertia separation type nacelle in icing conditions where the alternate duct is employed, there is a thrust loss of about 10 percent caused by a ram loss of 20 percent. The aerodynamic penalties of the hot-gas bleedback system are a thrust loss of about 19 percent of which 15 percent is caused by the reduction in air density due to heating the air and 4 percent to bleedback. Research is continuing on both the inertia separation and hot-gas bleedback systems to reduce these performance penalties.

EEC:ms

ANNUAL INSPECTION -- OCTOBER 8-9-10, 1947

WING ICING RESEARCH AND TUNNEL DEMONSTRATION

V.G. Rollin

The NACA is currently engaged in a fundamental study of the wing icing problem considering both steady-state heating for ice prevention and cyclical heating for deicing for a series of representative airfoils at medium subsonic speeds. Cyclical deicing with electrical heating elements will reduce the demand on the aircraft generating equipment (if satisfactory ice protection can be demonstrated). This research will include studies to determine local heat-transfer coefficients, optimum heat patterns, heating and cycle time, and the effect of ice formations on the airfoil characteristics. The demonstration of cyclical deicing will be conducted on an NACA laminar flow airfoil, similar to the model on your left, as shown on this chart. Fig. 65 Spray nozzles are located upstream of the model and provide a cloud simulating natural icing conditions. The electrical heater is of the rubber-clad resistance-wire type made up of a series of elements. The heaters are wired to permit a selective heat input to each circuit and chordwise coverage up to 20 percent.

The demonstration can be observed from either the top or side of the test section and will consist of several cycles of ice formation and removal. Progress of the demonstration can be followed on the large chart over the tunnel control panel similar to the small chart. After the tunnel has reached operating conditions of 350 miles per hours and 20° F, the sprays are turned on as indicated by the blue light to the left of the airfoil section. Ice will be permitted to build up on the wing and then the heat is turned on for deicing. This is shown on the chart Fig. 66 by the red light and illuminated portion of the wing leading edge. This cycle of events will be repeated several times.

During the demonstration, ceiling lights will be turned out to permit better visibility of the model. If you will take the stairs on your right, you can now observe the demonstration from the top or side of the tunnel test section. (I will be glad to answer questions when the demonstration is completed.)

ANNUAL INSPECTION — OCTOBER 8-9-10, 1947

SUPERSONIC RESEARCH FACILITIES

D.D.Wyatt

Description of facilities. - Housed in this building are several of the larger continuous-flow supersonic tunnels in the world. The arrangement of the tunnels is shown on the first chart. Fig. 67 The single-pass, continuous-flow, suction tunnels draw atmospheric air from a common source and discharge into a common surge tank, from whence the air is drawn into the altitude wind tunnel exhausting pumps. Before entering the tunnels, the air is passed through an activated-alumina air drier and past electrical heating elements totaling 3000 kilowatts. The drying and heating are necessary in order to avoid tunnel disturbances originating from moisture condensation during the tunnel expansion process. The tunnels have gate valves at each end, since only one tunnel can be operated at a time.

This room will soon house a tunnel having a 2- by 2-foot test section which will operate at speeds up to 1800 miles an hour, corresponding to Mach numbers up to 5.0. On the next floor below is a tunnel with a test section 20 inches in diameter which is operated at a Mach number of 1.90. Two floors down is a tunnel with an 18- by 18-inch test section, currently operated at a Mach number of 1.85, but capable of operation between Mach numbers of 1.3 and 4.0 through the use of interchangeable nozzles. The 20-inch and 18- by 18-inch tunnels were placed in operation in 1945. Below the 18- by 18-inch tunnel is the control room from which all the tunnels are operated and in which pressure and temperature data are recorded.

The 20-inch tunnel is currently being used in the study of supersonic diffusion in the presence of combustion. The 18- by 18-inch tunnel is being used for aerodynamic studies and to investigate the performance of various ram-jet engine inlets.

Discussion of current work. - Improvement of the efficiency of supersonic engine inlets is an important research goal because of the critical effect of total-pressure recovery on the thrust coefficient of the ram-jet engine, which is one of the most promising power plants for supersonic flight. This effect is shown on the next chart, Fig. 68 where thrust coefficient is plotted against flight Mach number for several values of diffuser total-pressure recovery. These thrust coefficients are based on the maximum frontal area of the ram jet, which would be at the combustion chamber for the parts of the curves having positive slope, and at the inlet for the parts of the curve having negative slope. A constant air velocity at the entrance to the combustion corresponding to a Mach number of 0.2 has been assumed on these curves.

In the lower Mach number range where the combustion chamber has the largest area in the engine, pressure recovery in the inlet has a large influence on the thrust coefficient. Part of the increase in thrust coefficient with increased pressure recovery comes from the improved thermodynamic efficiency of the engine, but the major improvement comes from the fact that, as the inlet efficiency is increased, more air can pass through a given sized combustion chamber at the same velocity, and the engine thereby produces more jet thrust.

At high speeds, where it becomes necessary to have the inlet larger than the combustion chamber, an increase in pressure recovery no longer increases the air flow per unit maximum area and the thrust coefficient is only slightly increased.

Because of the great gains in thrust coefficient from increased pressure recovery in the lower speed range, inlet designs which increase both pressure recovery and drag will be permissable, provided of course that the added drag is less than the increase in thrust coefficient. In the higher Mach number range, however, extreme care must be taken that increased pressure recoveries are not accompanied by appreciable increases in inlet drag.

On the next chart Fig. 69 are shown several forms of supersonic ram-jet inlets which have been proposed to maintain efficient pressure recovery at supersonic speeds. The upper left figure shows a convergent-divergent inlet which reduces pressure losses by reduction of the internal air-flow speed through an initial internal contraction of the airstream prior to formation of a normal shock wave. The amount of inlet contraction is limited if the normal shock is to be swallowed, so that great increases in pressure recovery are not possible. At a Mach number of 1.85, we have obtained 84 percent recovery with this type of inlet as compared to 79 percent for a simple diverging, subsonic type inlet.

The lower left figure illustrates a spike type inlet, which achieves increased pressure recovery through initial reduction of inlet airspeed through formation of an external oblique shock. At a Mach number of 1.85, we have achieved 92 percent recovery with a single-shock spike of the type shown, and higher values with a continuously curved spike.

The upper right figure illustrates an NACA perforated convergent-divergent inlet. In this inlet, the contraction can be made great enough to give sonic velocity at the throat. The perforations act to relieve the excess mass flow so that the normal shock can enter, then automatically reduce their capacity once the normal shock is swallowed. Preliminary experiments on this type of inlet at a Mach number of 1.85 have given a recovery of 93 percent.

In the lower right figure is shown a combination of single-shock spike and perforated inlet principles. Internal contraction is added to the spiked inlet, and the perforations are added to permit the shock to enter. With this type of diffuser, recoveries as high as 95 percent have been obtained at a Mach number of 1.85.

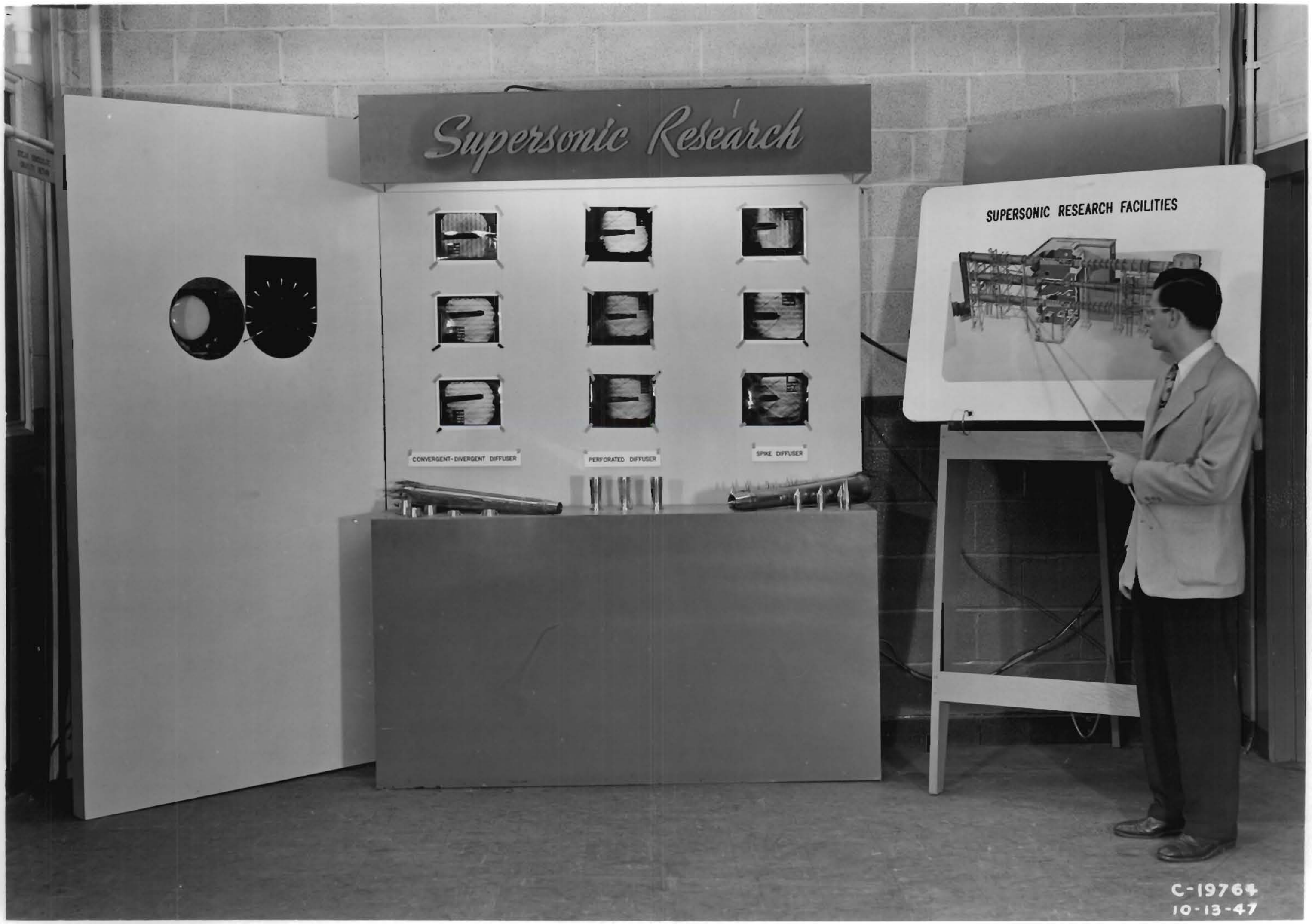
The next chart Fig. 70 summarizes the maximum recovery characteristics of several diffuser types over a range of Mach numbers. The data at a Mach number of 1.85 were obtained at this laboratory, the other data were obtained at the Langley laboratory of the NACA. The lower curve shows the maximum recoveries with unperforated convergent-divergent inlets, the middle curve shows single-shock spike inlets with the normal shock contained in the diffuser, and the upper curve shows single-shock spike inlets with the normal shock ahead of the inlet. Although the latter case gives consistently higher recoveries over the whole range of Mach numbers, drag considerations (which yet remain to be determined) may render these inlets undesirable for high-speed flight. Future research will be devoted to a comprehensive study of the drag as well as pressure-recovery characteristics of supersonic engine inlets.

Discussion of demonstration.- One of our research tools is an optical system (called the Schlieren system) whereby shock waves may be seen and photographed. We have arranged to operate the 18- by 18-inch tunnel and to televise the Schlieren picture to the screen on the left. This next chart Fig. 71 shows some typical photographs of the perforated inlet which is now in the tunnel. After the tunnel is started, you will see a normal shock standing ahead of the inlet as shown in the left photograph. The outlet area will then be enlarged to allow the normal shock to enter as in the second and third photographs. Notice the changed appearance of the shock waves during this procedure. The maximum recovery is obtained with the normal shock completely inside the unit. During the demonstration, the percent pressure recovery of the inlet will be indicated on the calibrated dial next to the televised image.

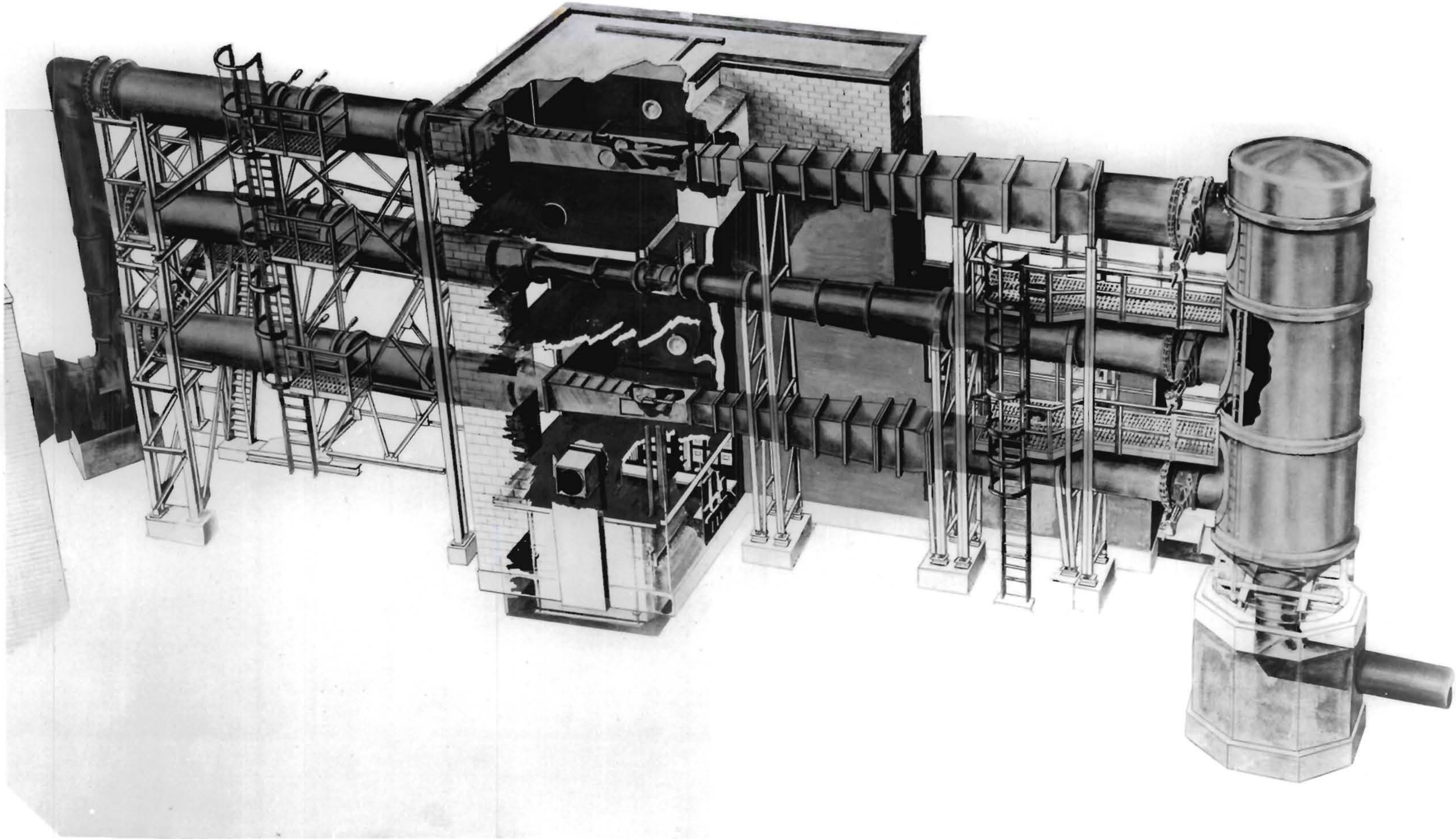
(Demonstration)

DeMDW:olh

9-25-47


Supersonic Research
SUPERSONIC RESEARCH FACILITIES
CONVERGENT-DIVERGENT DIFFUSER
PERFORATED DIFFUSER
SPIKE DIFFUSER
C-19764
10-13-47

SUPERSONIC RESEARCH FACILITIES

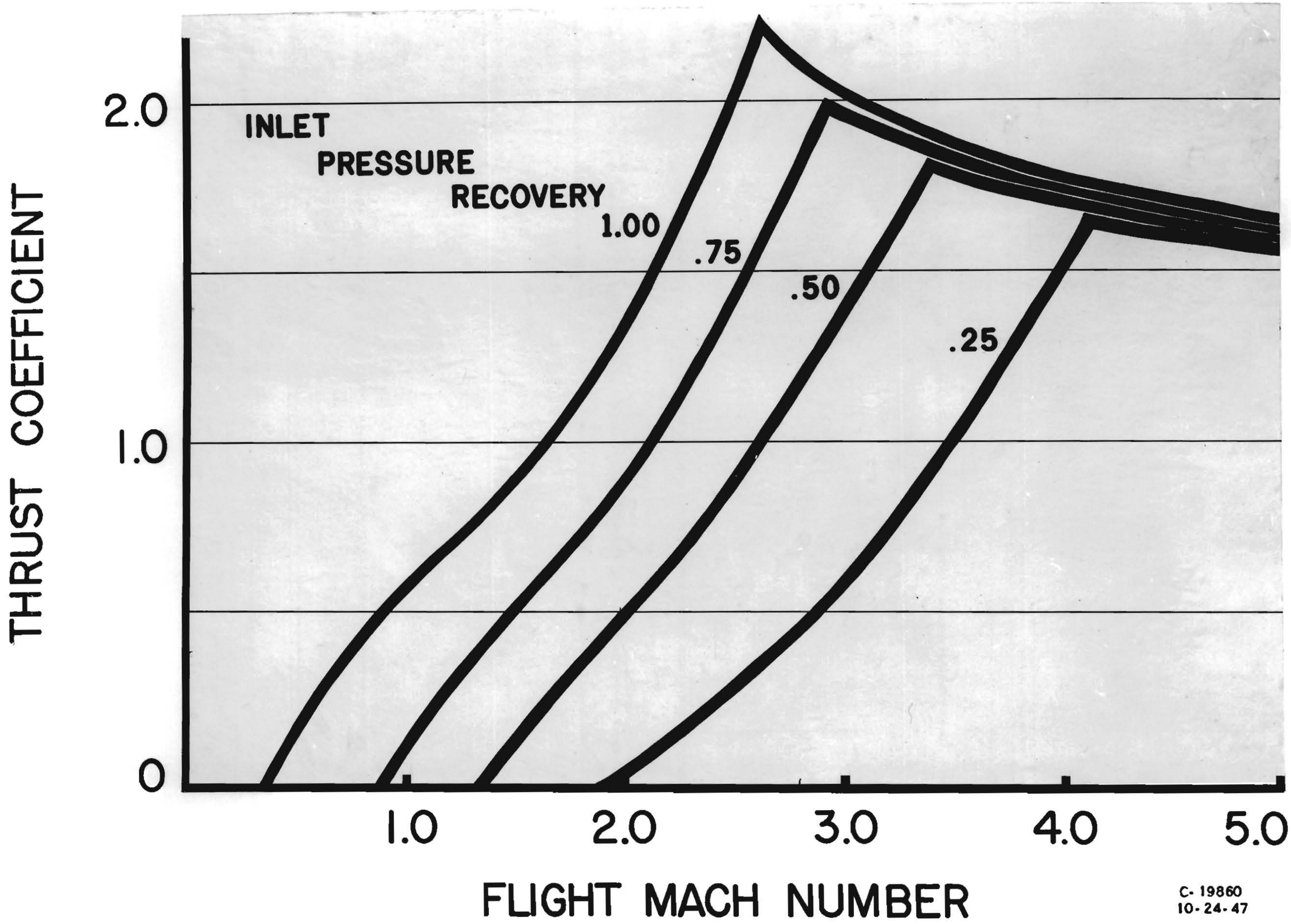


C- 19861
10- 24- 47

Fig 67

NACA

EFFECT OF PRESSURE RECOVERY ON THRUST

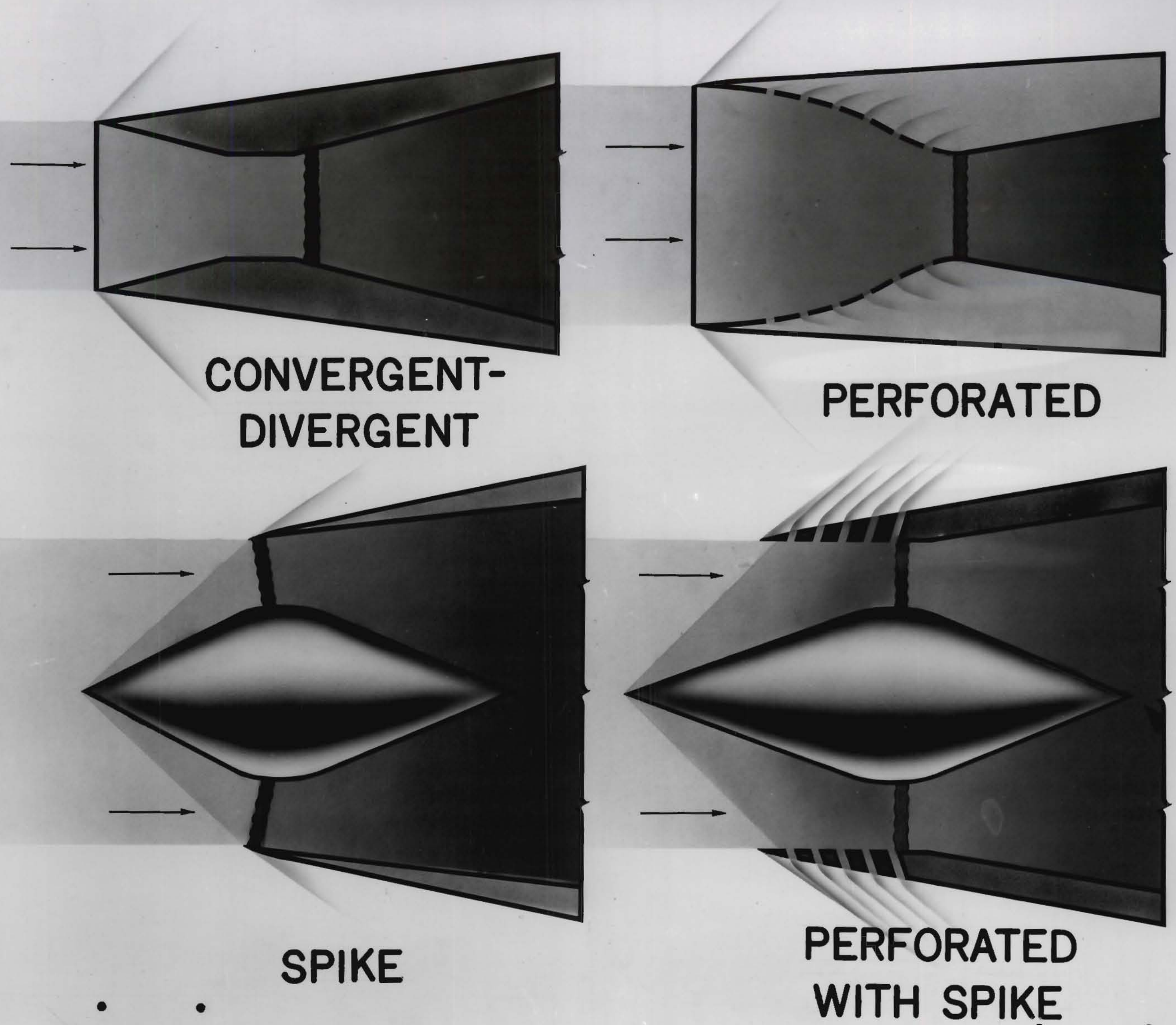


C-19860
10-24-47

NACA

Fig 68

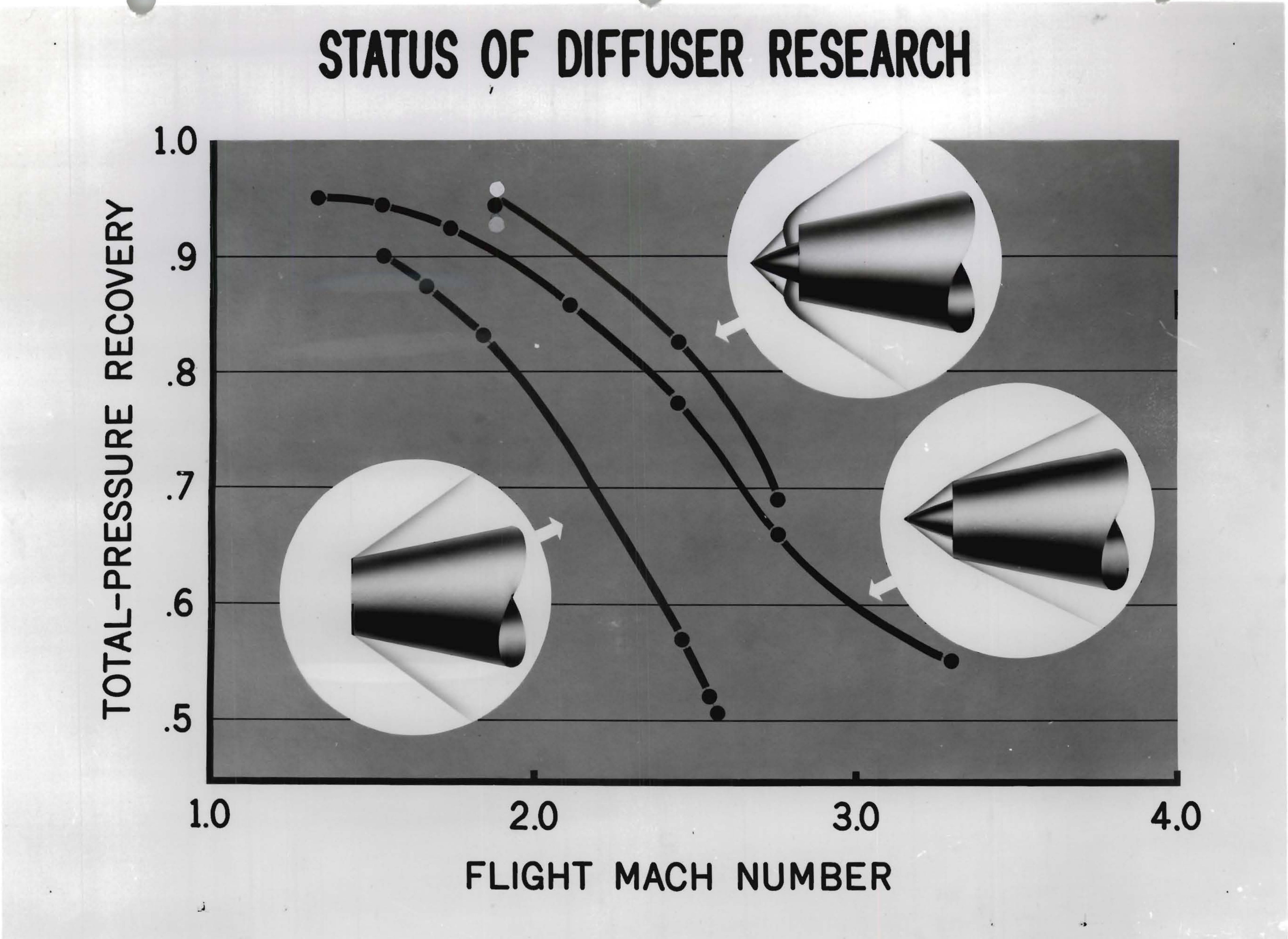
SUPERSONIC ENGINE INLETS



C- 19741
10-9-47

Fig 69

NACA

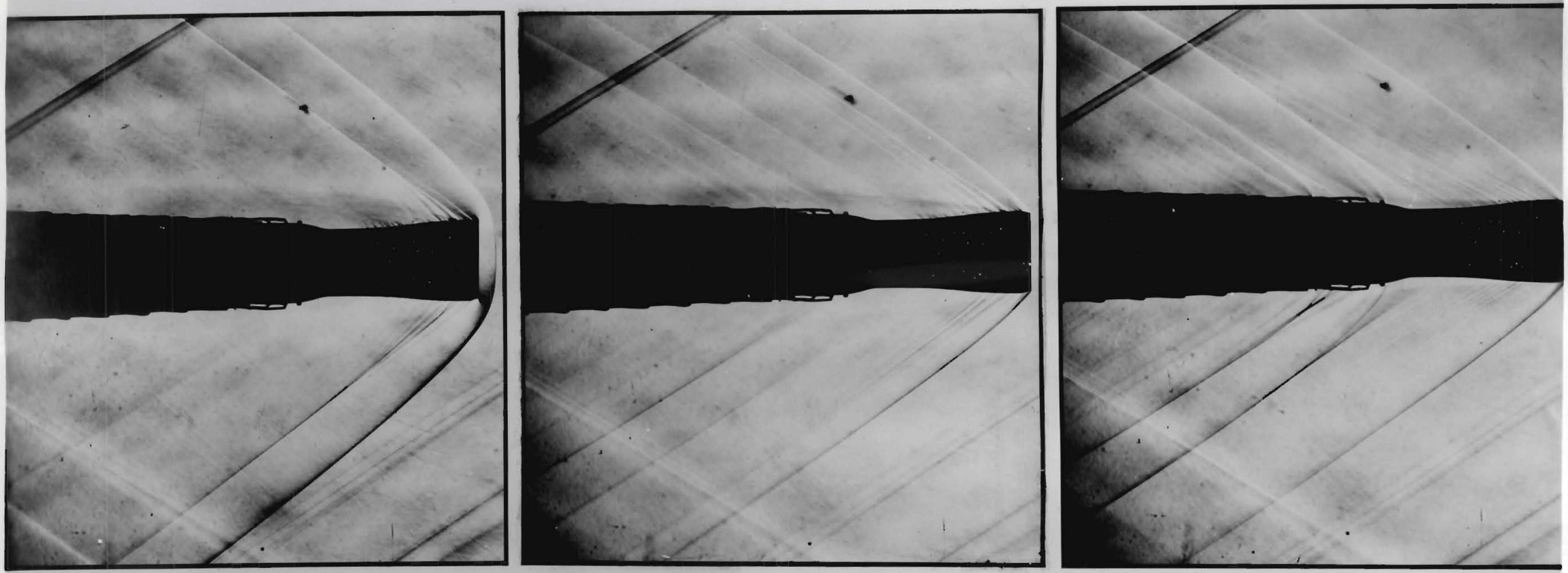


C- 19740
10-9-47

Fig 10

NACA

PERFORATED INLET IN OPERATION



C- 19855
10-24-47

Fig 71


SUPERSONIC WIND TUNNEL INSTRUMENTS
INTERFEROMETER
SCHLIEREN
DANGER
HI-VOLTAGE
MEASUREMENT OF AIR DENSITY
IN
SUPERSONIC TUNNELS
BY
X-RAY ABSORPTION
DANGER
HI-VOLTAGE
C-19777
10-13-47
NACA

ALTITUDE WIND TUNNEL RESEARCH
by W.A. Fleming

The altitude wind tunnel was constructed for research on full-scale aircraft engine installations at altitude flight conditions. A plan view of the wind tunnel and its facilities is shown in the first figure. In order to accommodate full-scale engine nacelles in the altitude wind tunnel, the test section was made 20 feet in diameter and 40 feet long. Since these engine installations were to be investigated at airspeeds between 400 and 500 miles per hour, it was necessary to install an 18,000 horsepower electric motor which drove a 32-foot diameter propeller. To simulate altitude conditions required both a reduced pressure and a reduced temperature within the tunnel. Reduction of the pressures in the tunnel requires four reciprocating-type exhawster units which together absorb 7000 horsepower. Reduction of the temperature in the tunnel, to simulate temperatures encountered at high altitude flight conditions, requires a 21,000 horsepower refrigeration system for cooling the tunnel air. Because the engine exhausted directly into the tunnel test section, this meant that the air within the tunnel had to be constantly changed.. Air at atmospheric pressure is dried, refrigerated, and then bled into the tunnel to replace the burned gas exhausted by the engine. Induction of this additional air requires that the exhauster capacity not only be great enough to maintain the desired test altitude but that the exhausters also have sufficient capacity to handle the fresh air bled into the tunnel. Fig. 72

Work has been done in the altitude wind tunnel on both reciprocating and jet engines. Some of these have been investigated with complete wing nacelles or fuselage installations. Other engines have been installed in special nacelles constructed at this laboratory. Recent work in the altitude wind tunnel has been almost completely devoted to investigations of various types of jet engines.

Operational and performance data of the entire engine are determined at various altitude flight conditions, as well as the performance of the various engine components operating in the engine. Engine operational data include a number of phases: The operating range of the engine is established at various altitudes and it is determined whether the operating range is limited by high turbine temperature, faulty combustion, or other reasons. The windmilling drag of the engine is also determined; that is, the drag of an engine which is inoperative and which is allowed to windmill while in flight. Starting and acceleration characteristics of the engine are determined at all altitudes and the maximum altitude at which it is possible to start the engine is established. Investigations are made to test the ability of the engine fuel systems to compensate for changes in altitude and airspeed, in such a manner as to maintain a fixed engine speed for a given throttle setting at all flight conditions.

Fig. 73

The operating range of a typical turbojet engine for a range of altitudes up to 50,000 feet is shown in the second figure. This figure is somewhat similar to one which was shown by the Combustion Research Branch. This figure, however, is the operational limits of the entire engine tested at altitude flight conditions, whereas the curve presented by the Combustion Research Branch was one for an individual combustion chamber operating at conditions simulating those within an engine in flight at high altitudes. It should be noted that at very high altitues the operating range of the engine was greatly reduced from that at lower altitudes.

Fig. 74

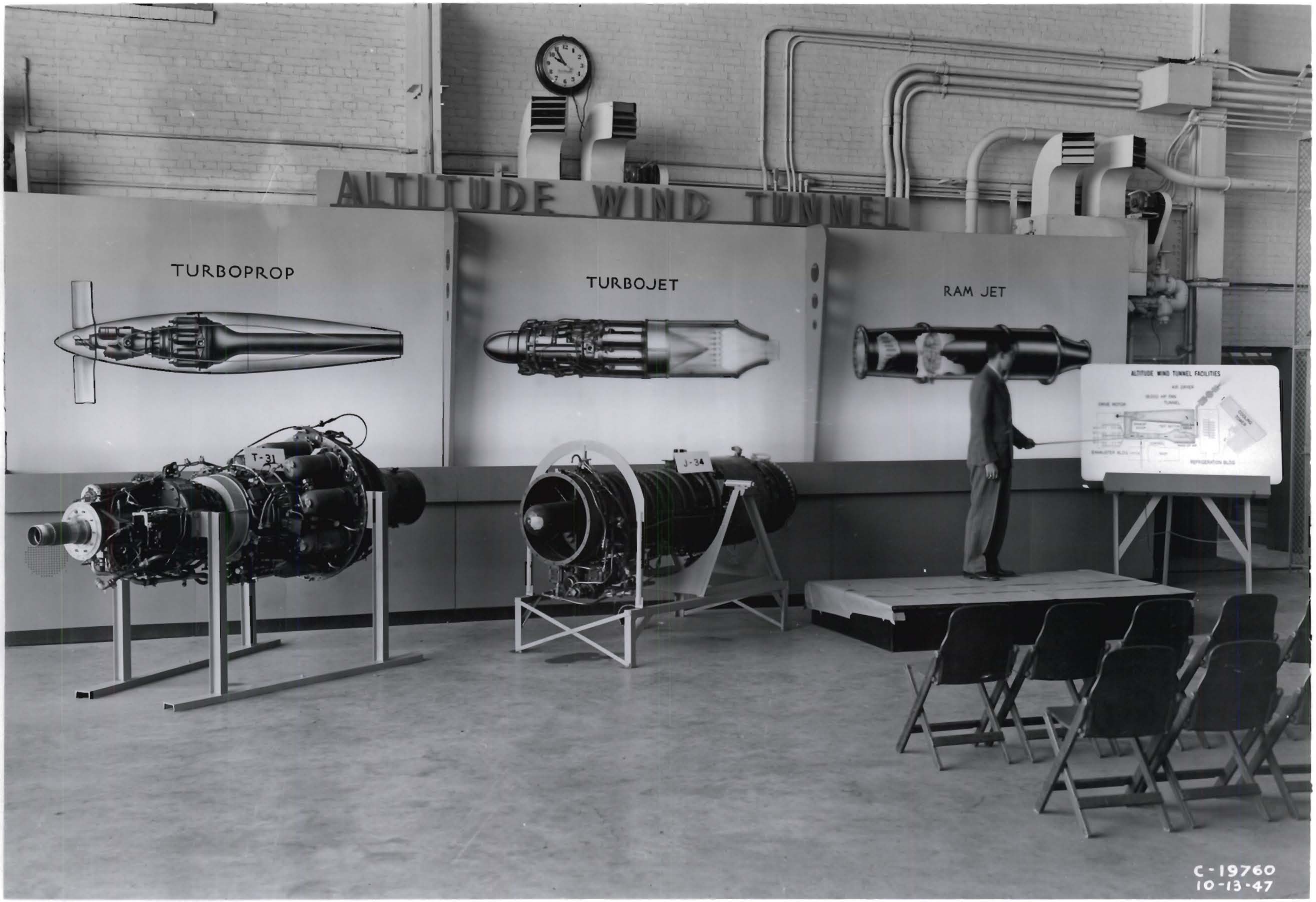
The windmilling drag of a turbojet engine in flight is presented in the third figure. Here the windmilling drag is presented in terms of percent of maximum net thrust of the engine. As the airspeed is raised, the windmilling drag increases quite rapidly. At a speed of 500 miles per hour, the drag of the engine is $12\frac{1}{2}$ percent of the rated engine thrust at that flight condition.

Because the trend in aviation is to fly at ever-increasing speeds, the NACA is conducting a research program to increase the thrust of various types of jet engines. Some typical performance data obtained in the altitude wind tunnel which cover one phase of this program are presented in the next two figures. In the fourth figure is presented a comparison of the net thrust Fig. 75 per unit frontal area obtained with a turbojet engine, a turbojet engine with tail-pipe burning, and a ram jet engine. The curve shows that the thrust of the turbojet engine increases only slightly as the airspeed is raised; however, the thrust of the turbojet engine with tail-pipe burning increases quite rapidly, and at an airspeed of 600 miler per hour the thrust with tail-pipe burning is twice that obtained with the same turbojet engine and no tail-pipe burning. Thrust of the ram jet engine is considerably lower than that of the turbojet engine or that of the turbojet engine with tail-pipe burning at low airspeeds. However, at airspeeds above 500 miles per hour the thrust of the ram jet increases quite rapidly and the ram jet thrust equals that of the turbojet engine at about 630 miles per hour, and equals that of the turbojet engine with tail-pipe burning at a flight speed of about 920 miles per hour.

Fig. 76

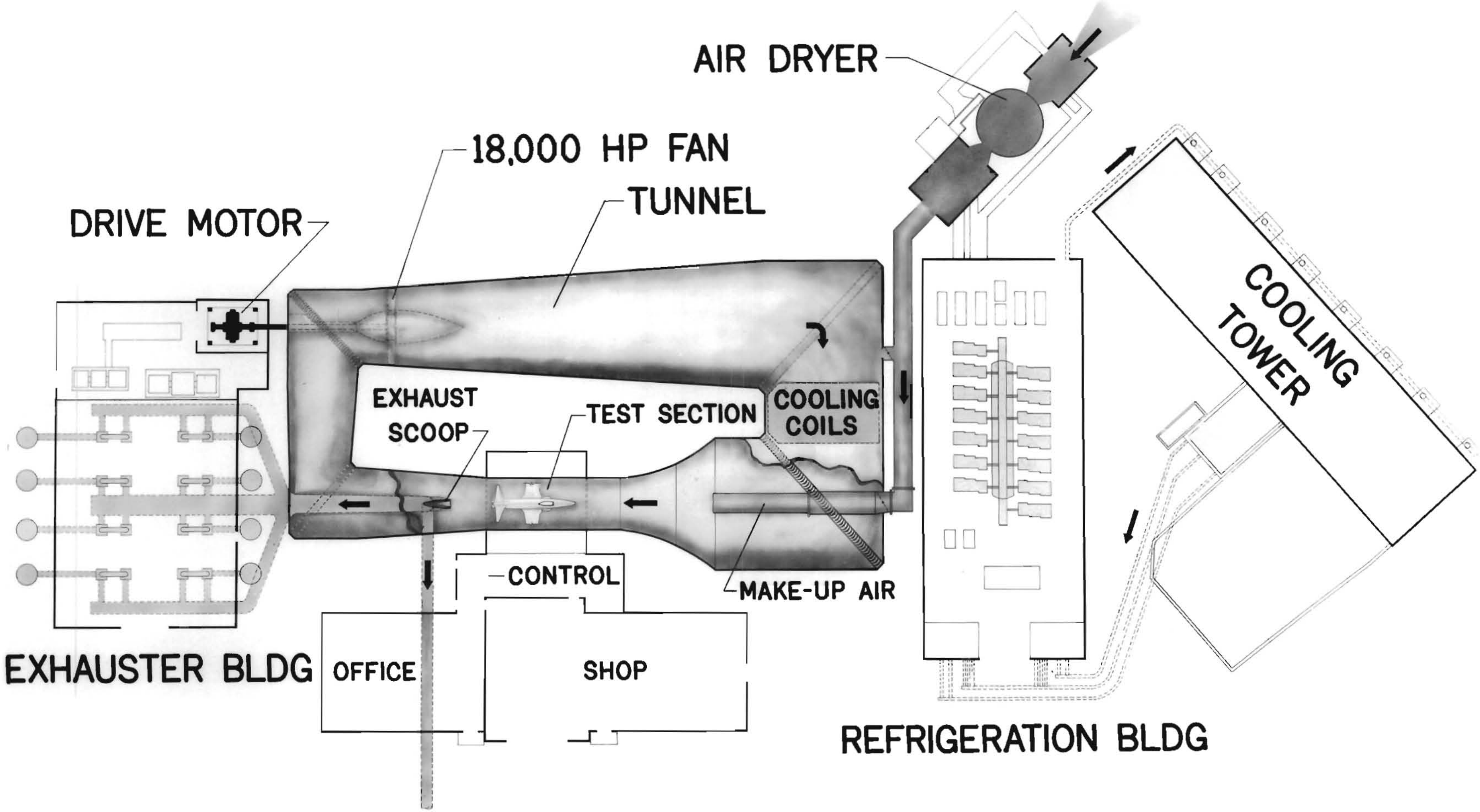
To complete the comparison of the three engines, the specific fuel consumption is presented in the fifth figure. This figure shows that the specific fuel consumption for the turbojet engine increases slightly with airspeed and is lower than that of either the ram jet or the turbojet with tail-pipe burning at all airspeeds. The specific fuel consumption of the turbojet engine with tail-pipe burning is somewhat higher than that of the turbojet engine at all airspeeds, increasing slightly at low airspeeds, and remaining uniform at airspeeds above about 300 miles per hour. The specific fuel consumption for the ram jet is very high at low airspeeds and decreases rapidly as the airspeed is raised. At speeds as high as 1000 miles per hour, however, the specific fuel consumption of the ram jet is considerably higher than that of the turbojet or turbojet with tail-pipe burning. Improvements of combustion efficiency in the ram jet engine and the turbojet engine tail-pipe burner will further lower the specific fuel consumptions of these engines.

W. A. Fleming:koc
October 8, 9, 10, 1947


ALTITUDE WIND TUNNEL
TURBOPROP
TURBOJET
RAM JET
T-31
J-34
ALTITUDE WIND TUNNEL FACILITIES
C-19760
10-13-47

NACA

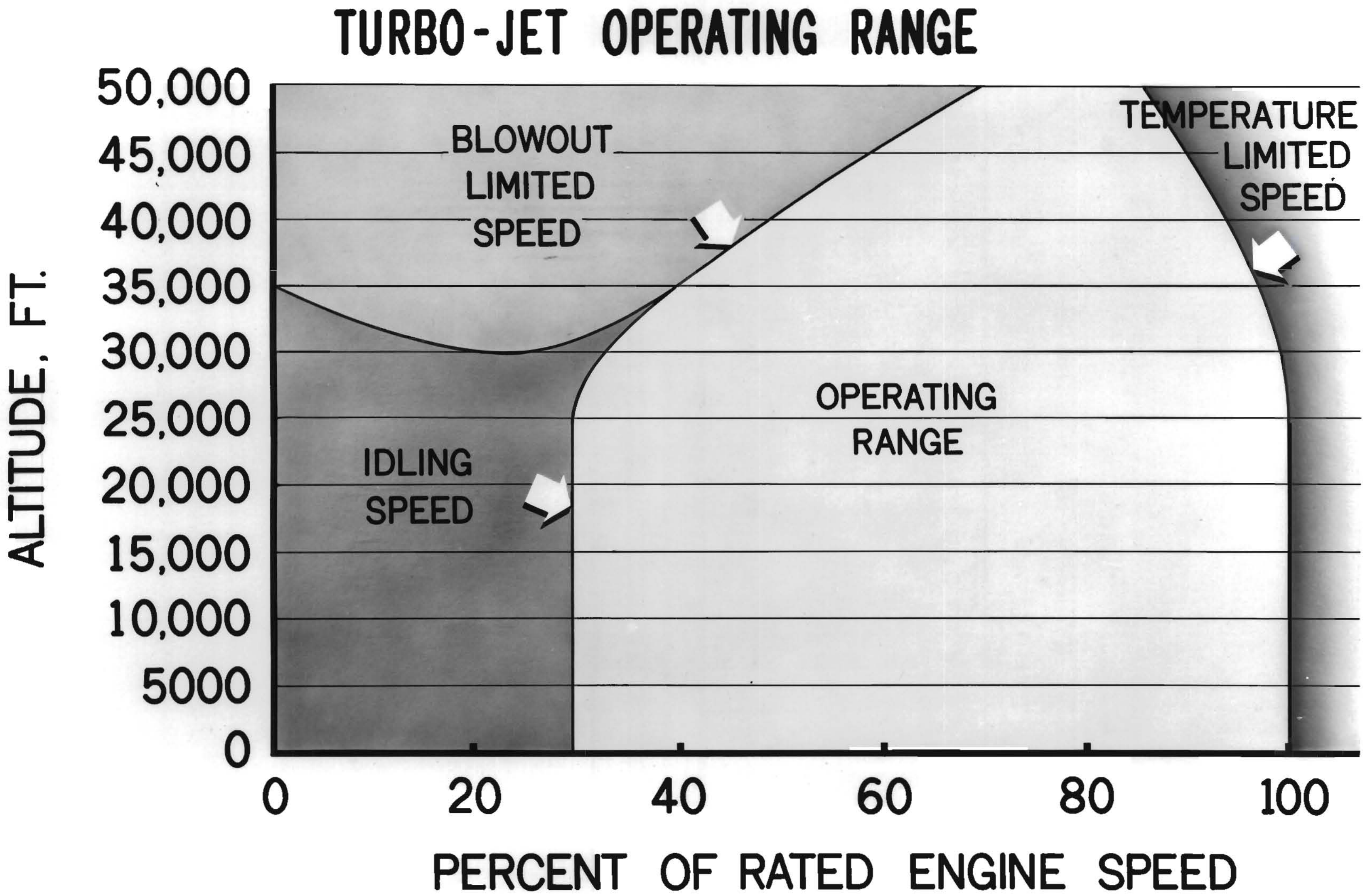
ALTITUDE WIND TUNNEL FACILITIES



C- 19847
10-24-47

Fig 12

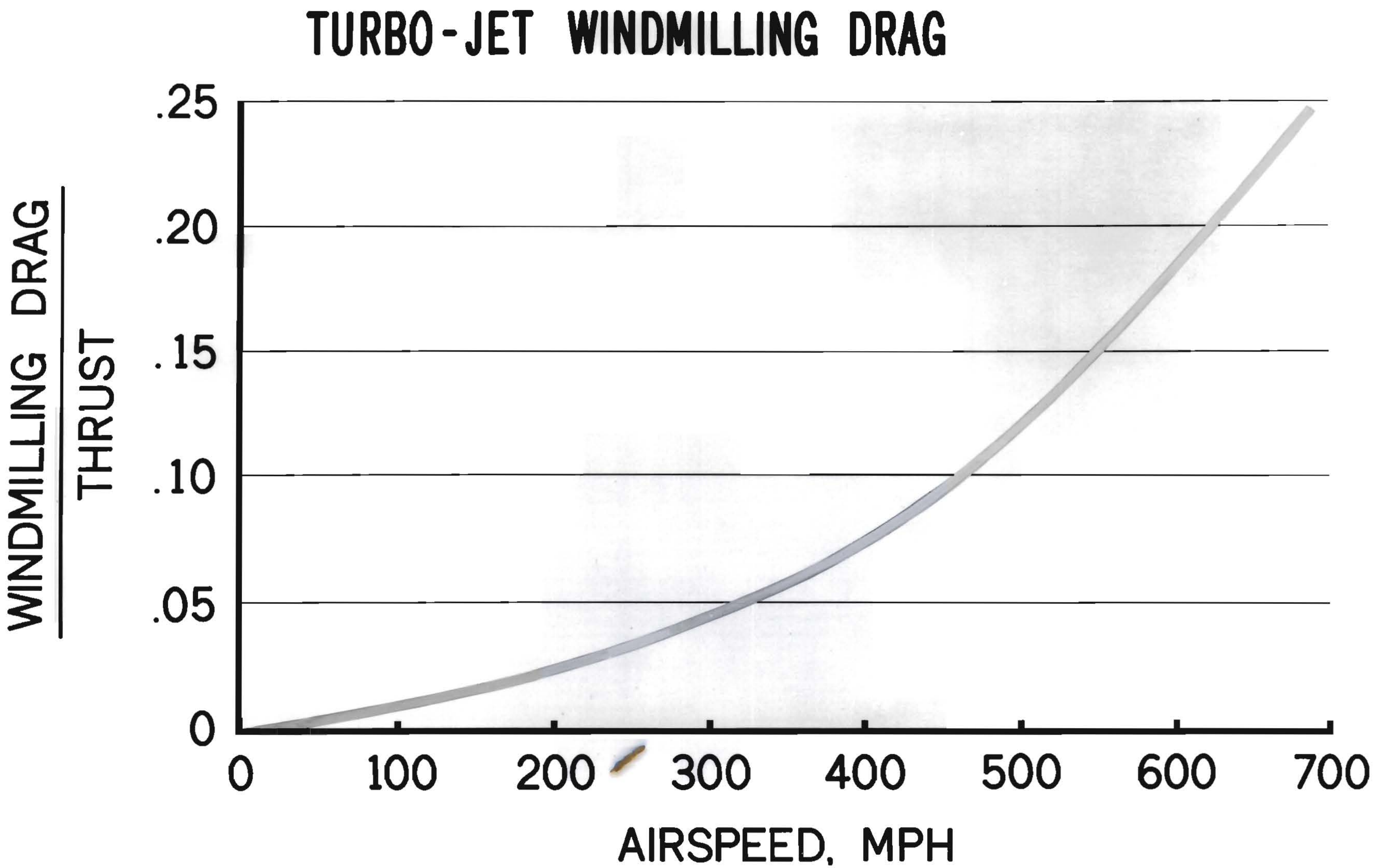
NACA



C-19854
10-24-47

NACA

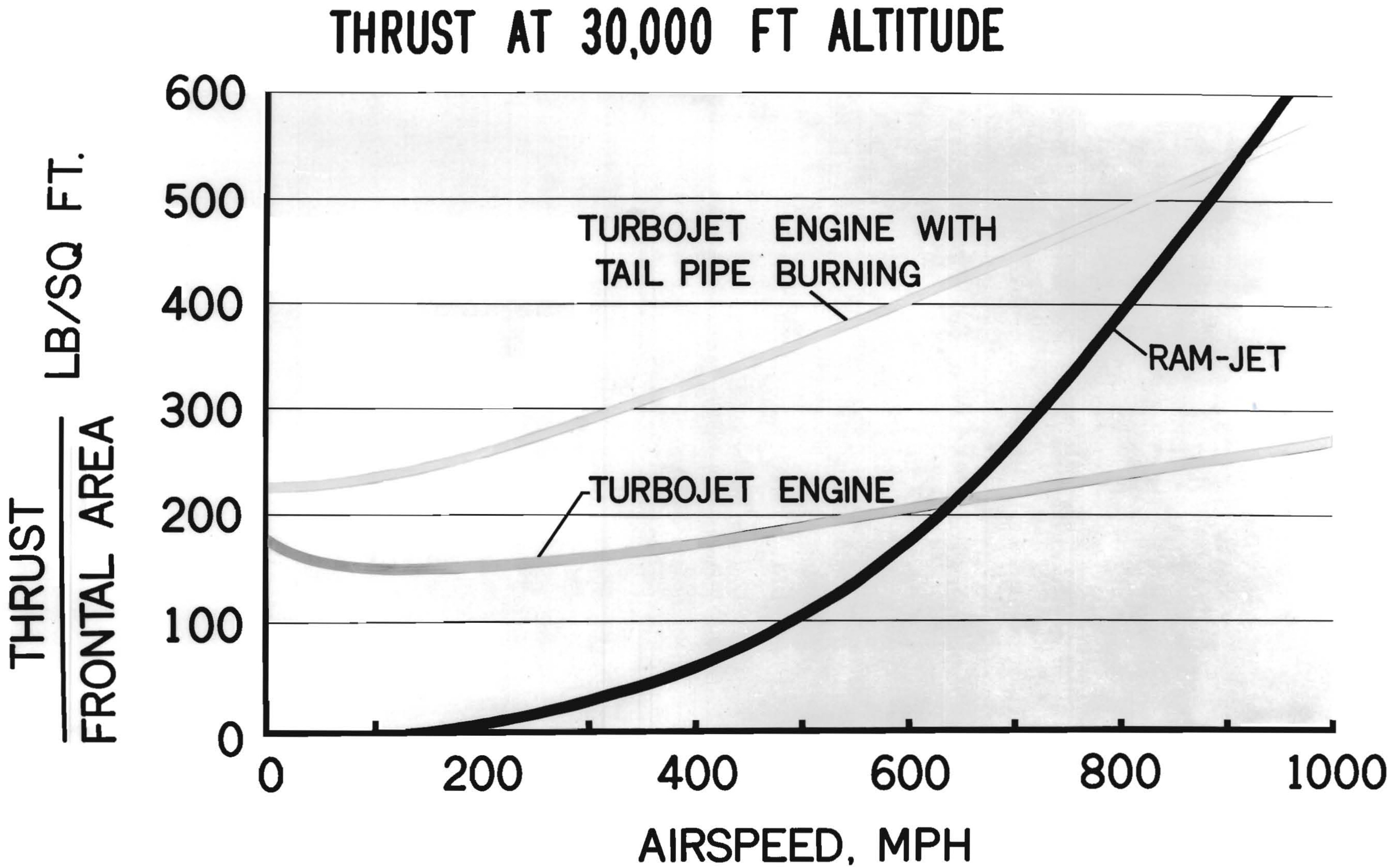
Fig 73



C-19874
10-24-47

Fig 14

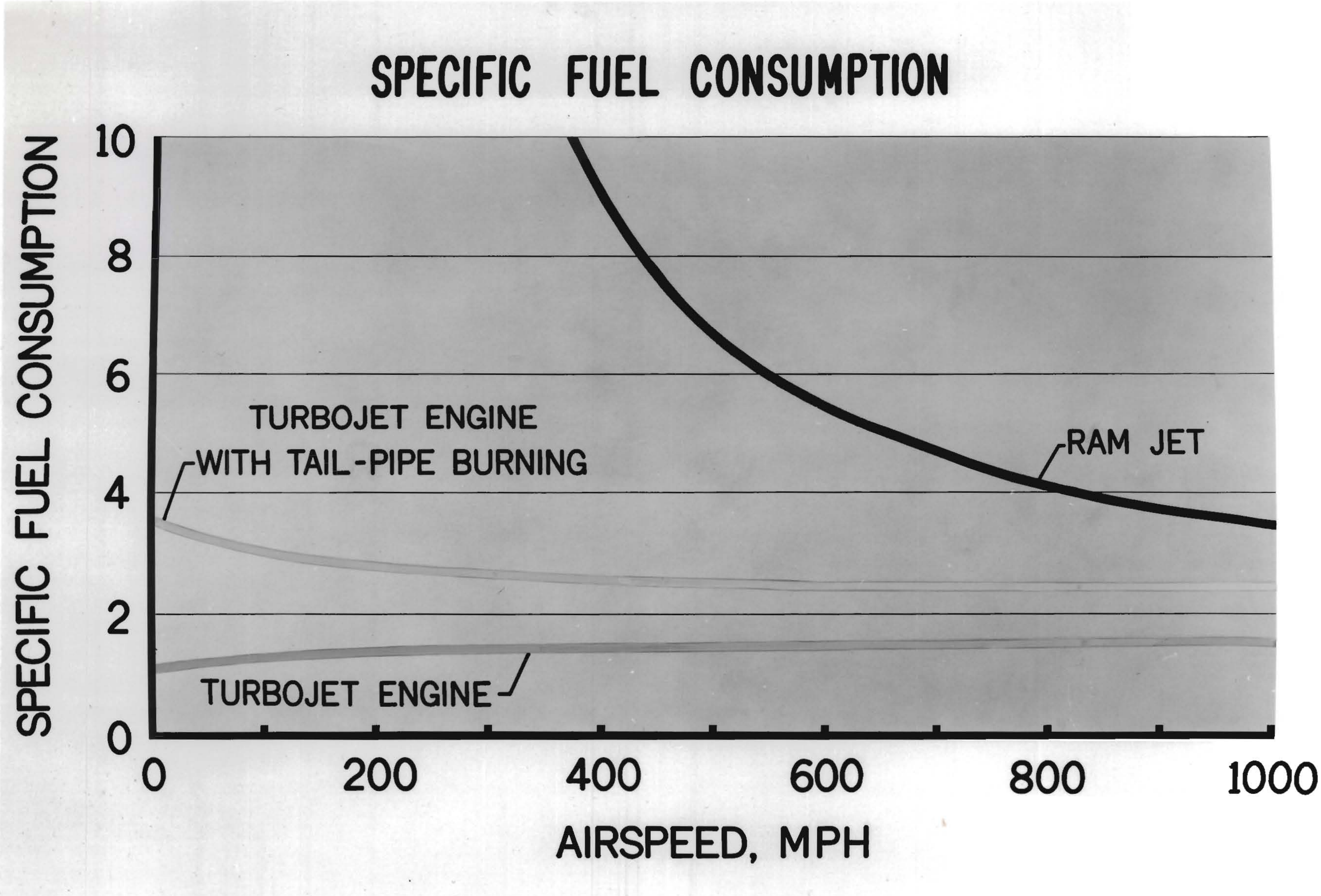
NACA



C- 19863
10-24-47

Fig 75

NACA



C- 19880
10-24-47

Fig 76

NACA