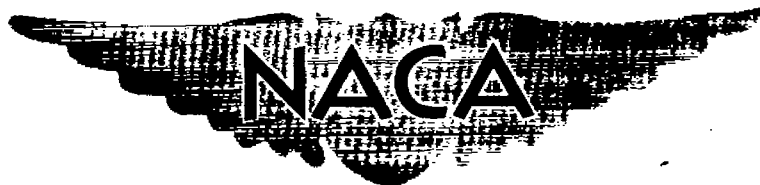


~~CONFIDENTIAL~~

Copy  
RM E55K09

C. 2

NACA RM E55K09



# RESEARCH MEMORANDUM

AFTERBURNER PERFORMANCE WITH COMBUSTION-CHAMBER  
LENGTHS FROM 10 TO 62 INCHES AT SEVERAL  
AFTERBURNER-INLET TEMPERATURES

By Carl C. Ciepluch, Wallace W. Velie, and Richard R. Burley

Lewis Flight Propulsion Laboratory  
Cleveland, Ohio

CLASSIFICATION CHANGED

~~CONFIDENTIAL~~  
UNCLASSIFIED

By authority of *NASA Class. Change Notice No. 19*

CLASSIFIED DOCUMENT

*Attd May 26, 1965*  
*HR-7-1-65*

This material contains information affecting the National Defense of the United States within the meaning of the espionage laws, Title 18, U.S.C., Secs. 793 and 794, the transmission or revelation of which in any manner to an unauthorized person is prohibited by law.

## NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON  
February 1, 1958

~~CONFIDENTIAL~~

UNCLASSIFIED

UNCLASSIFIED

NASA Technical Library



3 1176 01435 4451

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

AFTERBURNER PERFORMANCE WITH COMBUSTION-CHAMBER LENGTHS FROM

10 TO 62 INCHES AT SEVERAL AFTERBURNER-INLET TEMPERATURES

By Carl C. Ciepluch, Wallace W. Velie, and Richard R. Burley

## SUMMARY

A sea-level static investigation was conducted to determine the effect of variation of afterburner combustion-chamber length and inlet temperature on a low-pressure-loss sea-level afterburner. A 70-percentage-point reduction in combustion efficiency resulted when the combustion-chamber length was reduced from 62 to 10 inches. However, a thrust augmentation ratio of 1.20 was obtained with the shortest (10-inch) combustion chamber at a fuel-air ratio of 0.065. A 36-inch combustion chamber was found adequate with respect to combustion efficiency and thrust augmentation. For combustion-chamber lengths up to 36 inches, the afterburner performance increased rapidly, while above this length only small gains in combustion efficiency and thrust augmentation were available. Reducing the afterburner-inlet temperature from 1700° to 1550° R resulted in a reduction in combustion efficiency of  $11\frac{1}{2}$  and  $6\frac{1}{2}$  percentage points for 38- and 10-inch combustion chambers, respectively.

## INTRODUCTION

For long-range turbojet bombers and transport aircraft, take-off thrust boost or augmentation may be desirable, provided the range of the aircraft is not penalized during cruise conditions. Several investigations have been conducted at the NACA Lewis laboratory to provide design information for a low-pressure-loss, lightweight, compact afterburner augmentor, with a reasonable thrust augmentation suitable for this application. Included in these investigations were studies of the performance of several short annular diffusers (ref. 1) and the over-all performance of several short afterburners (ref. 2). Reference 3 is a continuation of the work of references 1 and 2.

The over-all tail-pipe length of the short afterburner reported in reference 3 was 54 inches, including diffuser and combustion chamber; however, in some applications a shorter tail pipe may be desirable. The

UNCLASSIFIED

minimum practicable diffuser length was established in references 1 and 3. Further variation of the over-all afterburner length can be accomplished by varying the combustion-chamber length. Therefore, an investigation was conducted to determine the effect of variation in combustion-chamber length on the performance of a low-pressure-loss sea-level afterburner. The effect of afterburner-inlet temperature on the performance of the afterburner at several combustion-chamber lengths was also determined to cover the range of interest for engines with different turbine-outlet temperatures.

Afterburner combustion-chamber lengths of 10, 20, 26, 32, 38, and 62 inches were investigated. The afterburner average gas velocity was 410 feet per second. Performance data were obtained over a range of fuel-air ratios from 0.045 to 0.07 with the use of fixed-area exhaust nozzles. An axial-flow turbojet engine installed in a sea-level static test stand was used as a gas generator.

## APPARATUS

### Engine and Installation

The engine used for the afterburner investigation was an axial-flow turbojet engine with a 5-minute military thrust rating of 5970 pounds at 7950 rpm and an exhaust-gas temperature of 1735° R with an engine-inlet temperature of 519° R. The engine was mounted in a sea-level static test stand. The engine air, which entered the test cell through air-measuring orifices, was heated to a fixed temperature with a can-type preheater. The engine was mounted on a thrust stand, and engine thrust was measured with a null-type air-pressure diaphragm.

### Afterburner

A schematic diagram of the afterburner tail pipe is shown in figure 1. The combustion-chamber length was considered as the distance between the flameholder and the exhaust-nozzle exit. The afterburner was similar to the one investigated in reference 3, except that the inside diameter of the tail-pipe liner was 31 inches compared with 28 inches for the configuration in reference 3. The afterburner average velocity at the flameholder was 410 feet per second. The annular diffuser was composed of a cylindrical outer shell and a 11.5°-half-angle conical inner-body with a blunt base which blocked approximately 31 percent of the tail-pipe area. Flow-straightening vanes were installed at the turbine outlet to reduce the turbine-exit whirl. The fuel-injection system was designed to provide uniform radial fuel distribution. The region adjacent to the outer shell was kept lean, however, to prevent burning between the liner and afterburner shell.

A total of 28 spray bars were used. The spray bars were flattened, or faired (fig. 2(a)), to reduce the total-pressure loss. Each spray bar had five radially located orifices that injected fuel upstream. The annular flow passage in the fuel-injection plane was divided into six equal annular areas, and the orifices were positioned in the spray bar at points coinciding with the center of each of the annular-area increments, except the one nearest the afterburner shell. The fuel orifices were 0.041 inch in diameter.

The flameholder (fig. 2(b)), consisting of two annular V-gutter rings 3/4 inch wide, blocked 14 percent of the tail-pipe area. The flameholder gutters were staggered to reduce the pressure loss. The perforated liner was made of 1/16-inch Inconel sheet. Liner perforations were 3/16-inch holes spaced so that the centers were 1/2 inch apart. No liner was needed for operation of the 10-inch combustion-chamber afterburner.

#### Instrumentation

The air flow to the engine was measured by a venturi section located in the atmospheric-air inlet line. The engine fuel flow was measured by a calibrated rotameter, and afterburner fuel flow by a calibrated vane-type electronic flowmeter.

The engine and afterburner instrumentation are listed in the following table:

	Engine inlet	Turbine outlet	Exhaust-nozzle inlet	Exhaust-nozzle outlet
Total-pressure probes	--	16	16	-
Total-temperature probes	20	24	--	-
Static-pressure probes	<sup>a</sup> 2	--	--	3

<sup>a</sup>Engine-inlet total pressure was assumed equal to test-cell static pressure.

## PROCEDURE

The afterburner combustion-chamber length was varied by adding or removing cylindrical sections of the afterburner shell and cooling liner between the flameholder and the exhaust nozzle.

Afterburner performance data were obtained at an engine speed of 7900 rpm and turbine-discharge temperatures of 1700° (approx. rated temperature), 1650°, 1600°, and 1550° R. The variation of turbine-outlet temperature was obtained by varying the afterburner fuel-air ratio with a fixed exhaust-nozzle area while engine speed was held constant at 7900 rpm. The engine-inlet temperature was maintained at 560° R for the tests.

The augmented thrust ratios were based on the thrust of the standard engine operating at the same turbine-outlet conditions at which the afterburning performance was obtained.

## RESULTS AND DISCUSSION

## Effect of Combustion-Chamber Length

Afterburner performance with combustion chambers of 10, 20, 26, 32, 38, and 62 inches is shown in figure 3 for a range of over-all fuel-air ratios from 0.045 to 0.07 with an afterburner-inlet temperature of 1700° R. Afterburner-outlet temperature was calculated from the measured thrust, mass flow, and exhaust-nozzle pressure ratio using a velocity coefficient of 0.98. The combustion efficiency of the afterburner was defined as the ratio of actual temperature rise in the afterburner to the ideal temperature rise. The ideal temperature rise was obtained from the tables of reference 4. A decrease of approximately 70 percentage points in combustion efficiency (fig. 3(a)) is observed for a reduction in combustion-chamber length from 62 to 10 inches. The severe reduction in combustion efficiency as the combustion-chamber length decreased resulted in a reduction in augmented thrust ratio (fig. 3(b)), although a thrust augmentation of 20 percent can be obtained for a 10-inch combustion chamber at a fuel-air ratio of 0.065.

A 36-inch combustion chamber was found adequate with respect to combustion efficiency and thrust augmentation (fig. 4). Only small gains in afterburner performance are available by increasing the combustion-chamber length from 36 to 62 inches. Below a 36-inch length, however, the afterburner combustion efficiency and, consequently, the augmented thrust ratio are approximately proportional to combustion-chamber length. For combustion-chamber lengths less than 36 inches, the time available for combustion (residence time) apparently becomes critical; and, as a result, the afterburner combustion efficiency decreases rapidly with decreasing combustion-chamber length.

The effect of varying the combustion-chamber length on the engine normal thrust (nonafterburning) is shown in figure 5. A 1-percent increase in engine thrust was obtained by decreasing the combustion-chamber length from 38 to 10 inches. The increased thrust resulted from the decrease in afterburner shell friction losses as the length was reduced.

#### Effect of Afterburner-Inlet Temperature

The effect of afterburner-inlet temperature on afterburner combustion efficiency is shown in figure 6 for combustion-chamber lengths of 10, 26, and 38 inches. The afterburner-inlet temperature was varied from 1700° to 1550° R by varying the afterburner fuel-air ratio with a fixed exhaust-nozzle area. Operation of the engine below the rated turbine-discharge temperature of 1700° R resulted in a reduction of both the afterburner average gas velocity and pressure level. The afterburner average gas velocity decreased 2 percent and the pressure level decreased 8 percent for a reduction in turbine-discharge temperature from 1700° to 1550° R. These variations in afterburner-inlet conditions, other than afterburner-inlet temperature, were considered minor, especially since they were compensating effects.

As would be expected, reducing the afterburner-inlet temperature reduced the combustion efficiency (fig. 6). However, the rate of decrease of combustion efficiency with afterburner-inlet temperature diminished as the afterburner-inlet temperature was reduced. For example, at a fuel-air ratio of 0.045 and a combustion-chamber length of 38 inches (fig. 6(a)), the combustion efficiency decreased  $7\frac{1}{2}$ , 6, and  $4\frac{1}{2}$  percentage points as the afterburner-inlet temperature was reduced from 1700° R (in 50° R increments) to 1550° R. Comparison of figure 6(a) (fuel-air ratio, 0.045) with figure 6(b) (fuel-air ratio, 0.055) indicates that the low fuel-air-ratio range was more affected by inlet-temperature reduction than the higher range.

The effect of afterburner-inlet temperature on combustion efficiency also diminished as the combustion-chamber length was reduced. The combustion efficiency of the 38-inch-combustion-chamber afterburner decreased  $11\frac{1}{2}$  points compared with  $6\frac{1}{2}$  points for the 10-inch length for a reduction in afterburner-inlet temperature from 1700° to 1550° R at a fuel-air ratio of 0.055 (fig. 6(b)).

Variation of augmented thrust ratio with afterburner-inlet temperature is shown in figure 7 for the combustion-chamber lengths of 10, 26, and 38 inches at a fuel-air ratio of 0.055. A curve of theoretical augmented thrust ratio is also included in this figure. Although the augmented thrust ratios for the three combustion-chamber lengths increased

slightly as the afterburner-inlet temperature was reduced, the increase was not as great as the theoretical because of the decrease in combustion efficiency with decreased afterburner-inlet temperature.

#### SUMMARY OF RESULTS

The following results were obtained during a sea-level static investigation of the effect of combustion-chamber length and afterburner-inlet temperature on a low-pressure-loss sea-level afterburner:

1. Reducing the combustion-chamber length from 62 to 10 inches resulted in a 70-percentage-point reduction in combustion efficiency. However, an augmented thrust ratio of 1.20 was obtained for the 10-inch combustion chamber at a fuel-air ratio of 0.065.

2. A 36-inch combustion chamber appeared adequate with respect to combustion efficiency and thrust augmentation. Only small gains in combustion efficiency and thrust augmentation were available above a 36-inch combustion-chamber length, whereas below this length a decrease in combustion efficiency and thrust augmentation approximately proportional to the combustion-chamber length was observed.

3. Reducing the afterburner-inlet temperature resulted in decreases in combustion efficiency. The combustion efficiency of the 38- and 10-inch combustion-chamber afterburners decreased  $11\frac{1}{2}$  and  $6\frac{1}{2}$  percentage points, respectively, by reducing the afterburner-inlet temperature from  $1700^{\circ}$  to  $1550^{\circ}$  R at a fuel-air ratio of 0.055.

Lewis Flight Propulsion Laboratory  
National Advisory Committee for Aeronautics  
Cleveland, Ohio, November 10, 1955

#### REFERENCES

1. Mallett, William E., and Harp, James L., Jr.: Performance Characteristics of Several Short Annular Diffusers for Turbojet Engine Afterburners. NACA RM E54B09, 1954.
2. Harp, James L., Jr., Mallett, William E., and Shillito, Thomas B.: Experimental Sea-Level Static Investigation of a Short Afterburner. NACA RM E54B18, 1954.

3. Ciepluch, Carl C., Velie, Wallace W., and Burley, Richard R.: A Low-Pressure-Loss Short Afterburner for Sea-Level Thrust Augmentation. NACA RM E55D26, 1955.
4. Huntley, S. C.: Ideal Temperature Rise Due to Constant-Pressure Combustion of a JP-4 Fuel. NACA RM E55G27a, 1955.



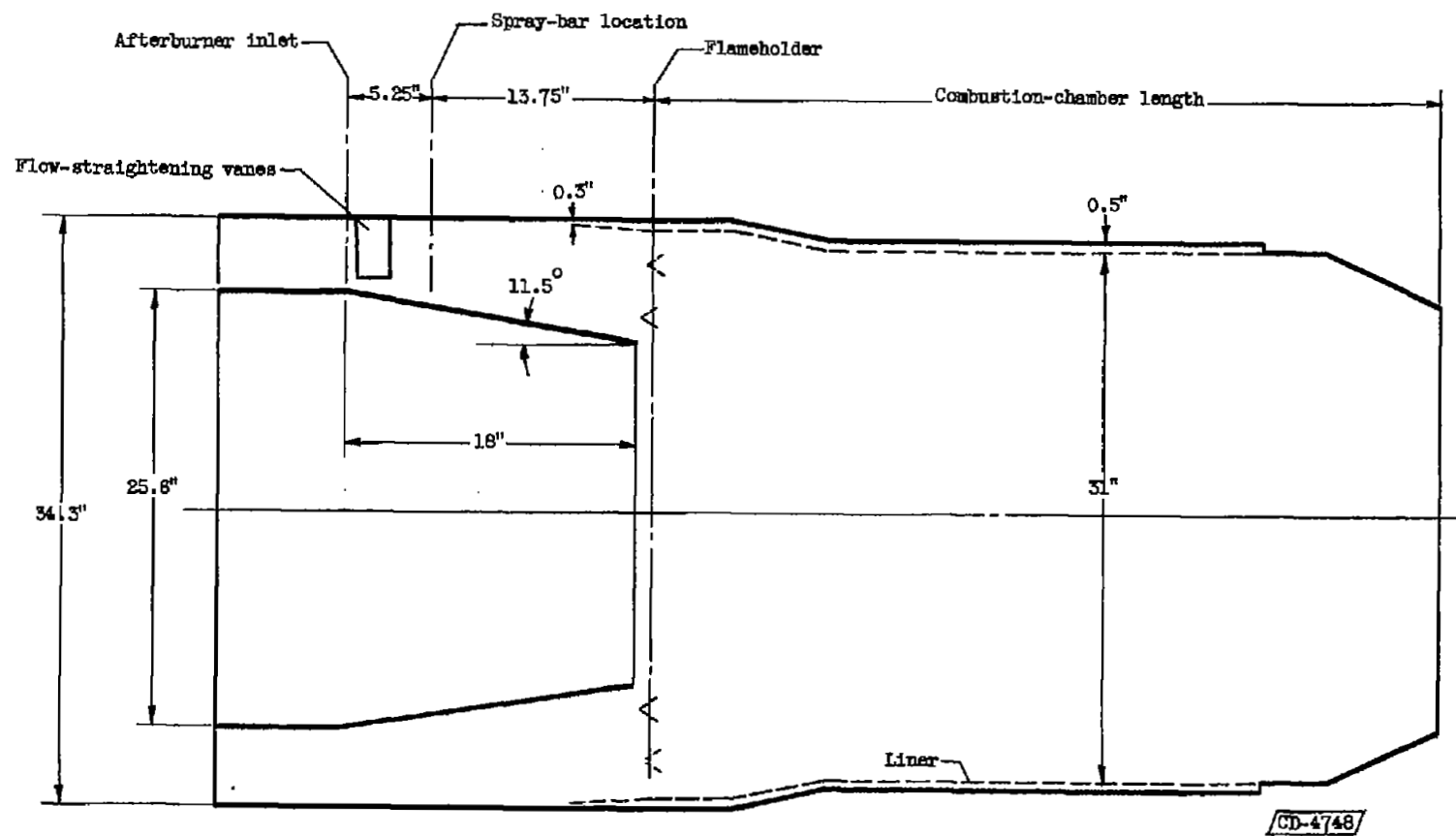
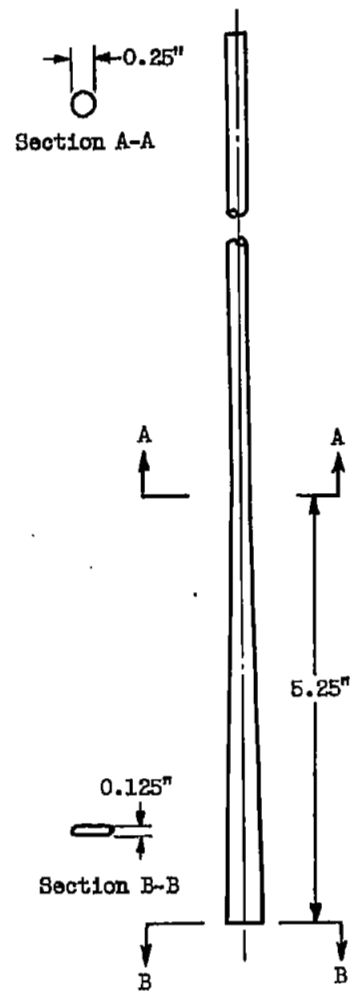
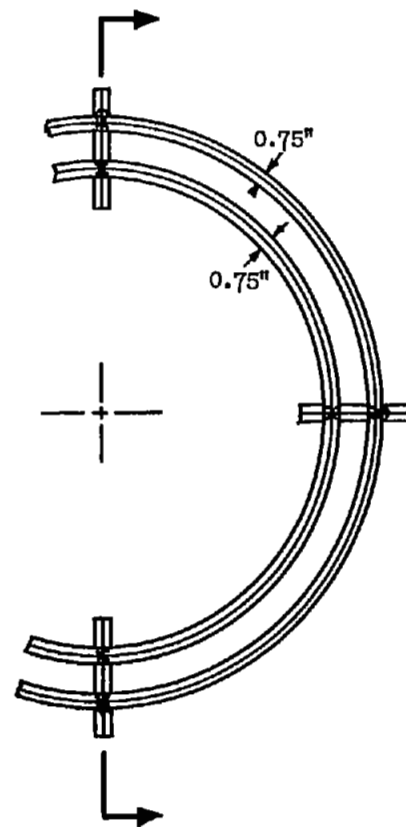
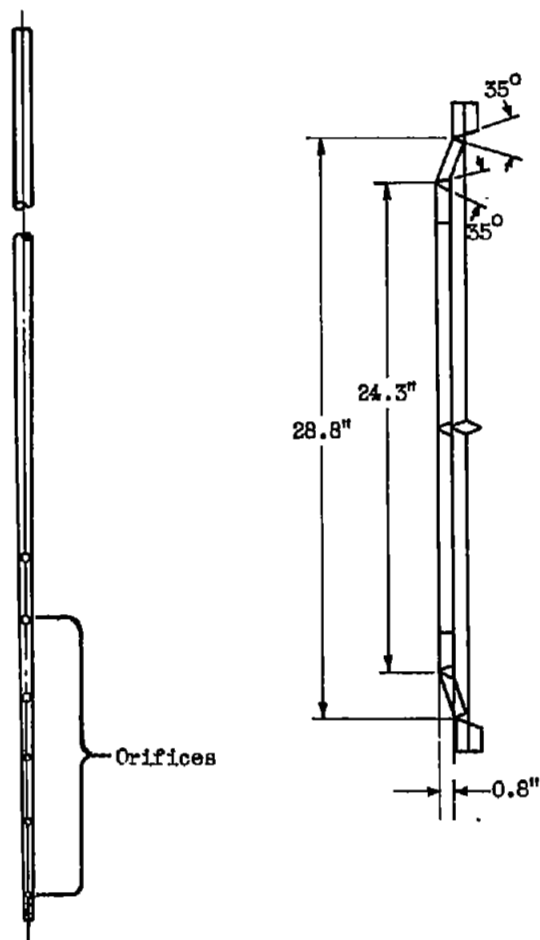


Figure 1. - Afterburner tail pipe.



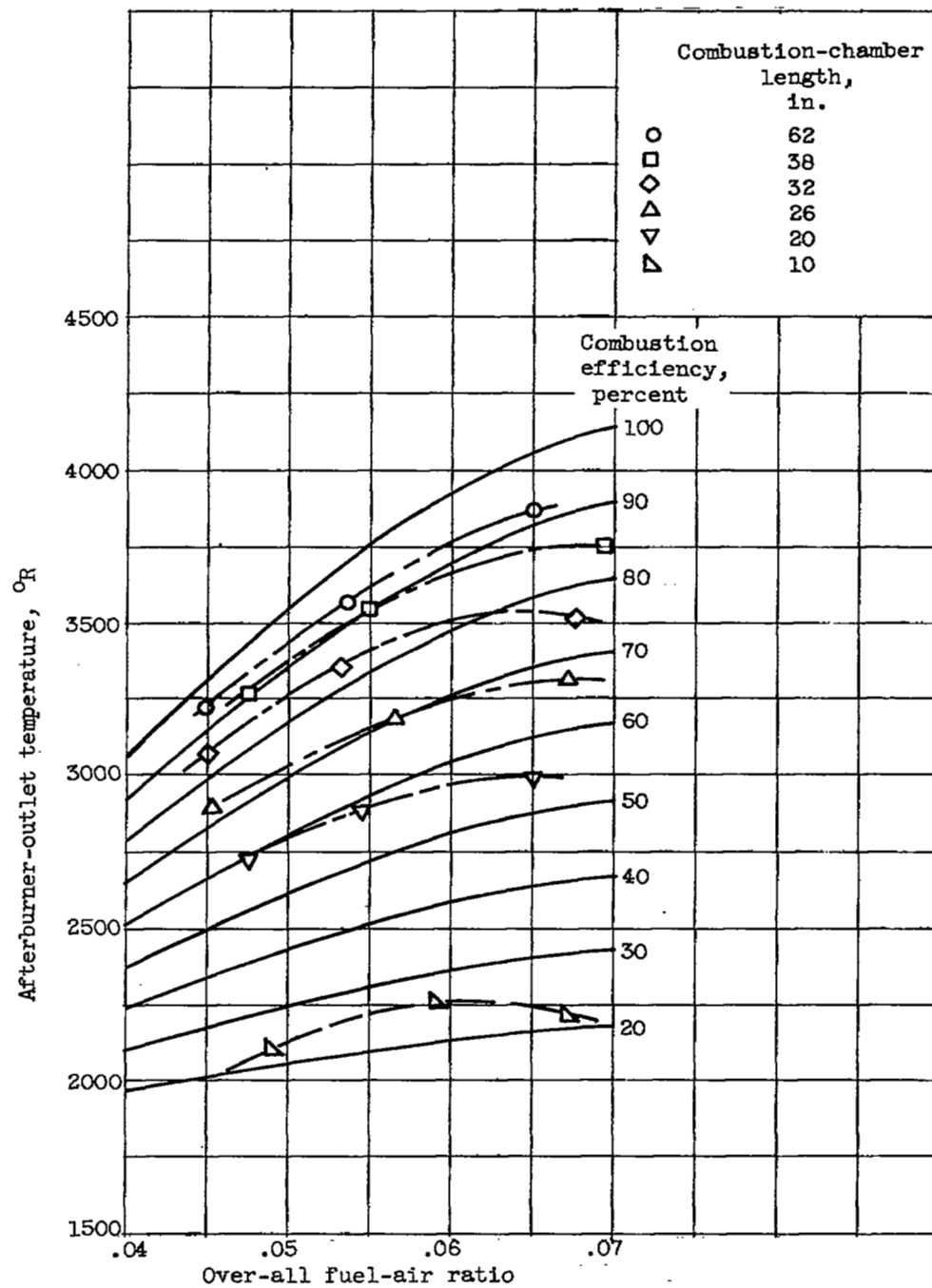
(a) Spray bar.



(b) Flameholder.

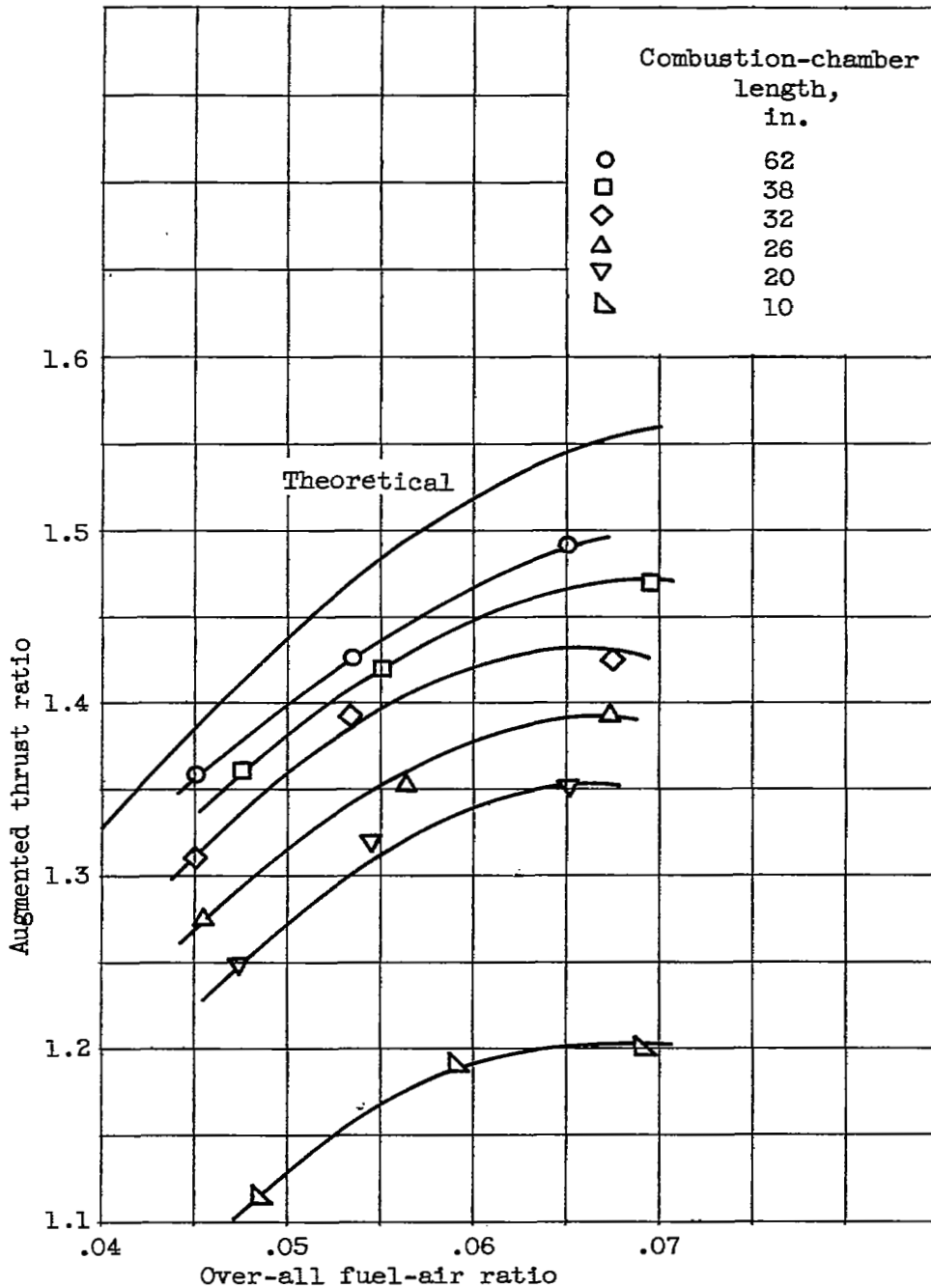
CD-4747

Figure 2. - Spray bar and flameholder.



(a) Combustion efficiency and outlet temperatures.

Figure 3. - Afterburner performance with various combustion-chamber lengths at an afterburner-inlet temperature of 1700° R.



(b) Augmented thrust ratios.

Figure 3. - Concluded. Afterburner performance with various combustion-chamber lengths at an afterburner-inlet temperature of 1700° R.

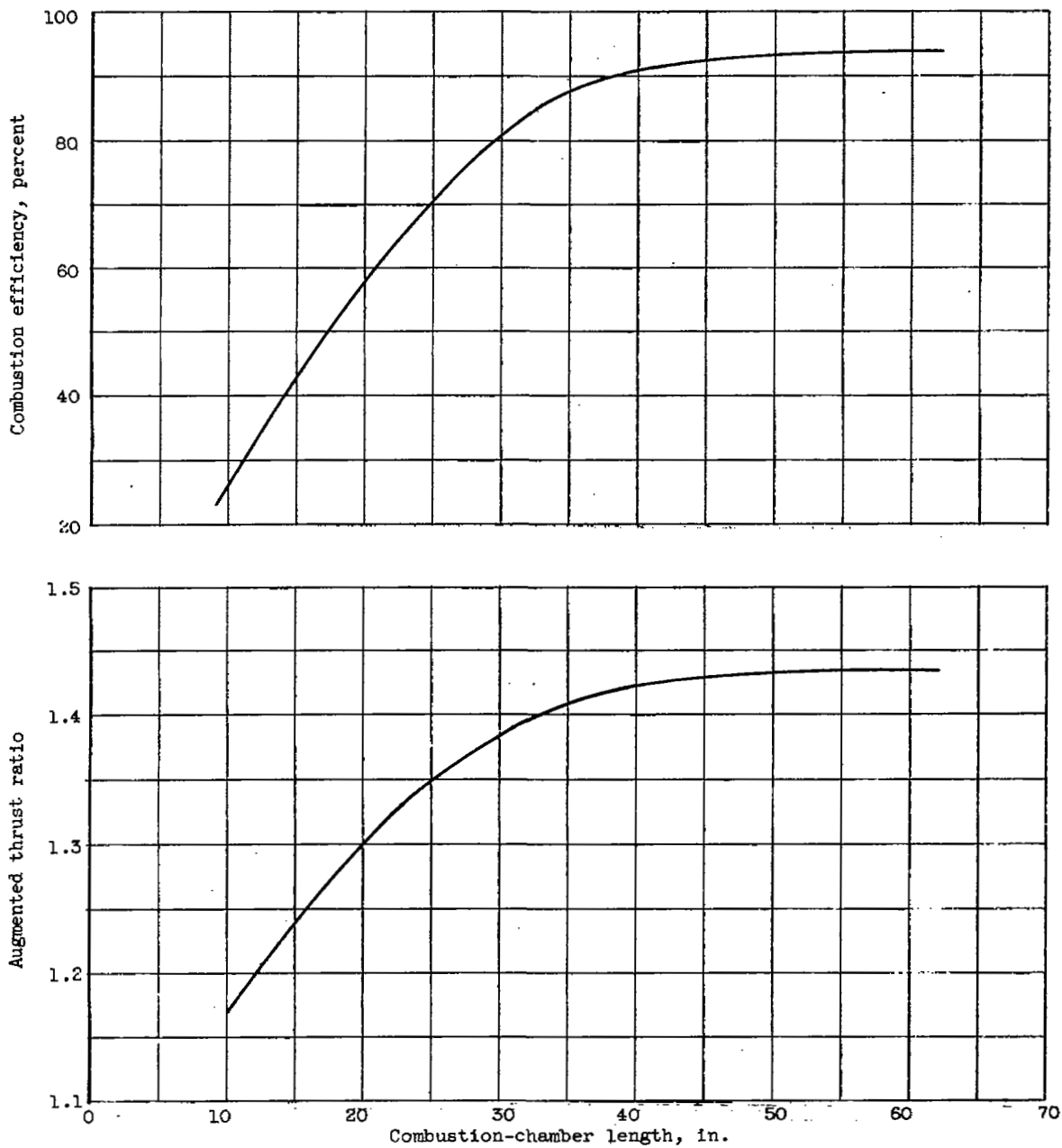


Figure 4. - Effect of combustion-chamber length on combustion efficiency and augmented thrust ratio. Fuel-air ratio, 0.055; afterburner-inlet temperature,  $1700^{\circ}$  R.

3965

Ratio of normal engine  
thrust with afterburner  
tail pipe to thrust of  
standard nonafterburning  
engine

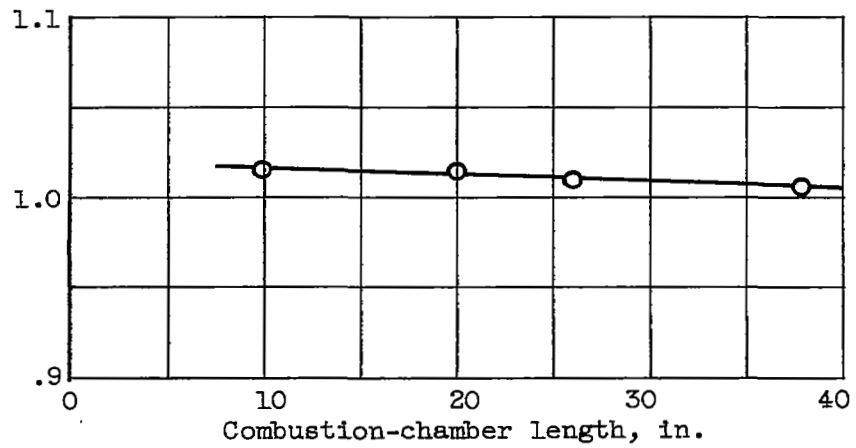
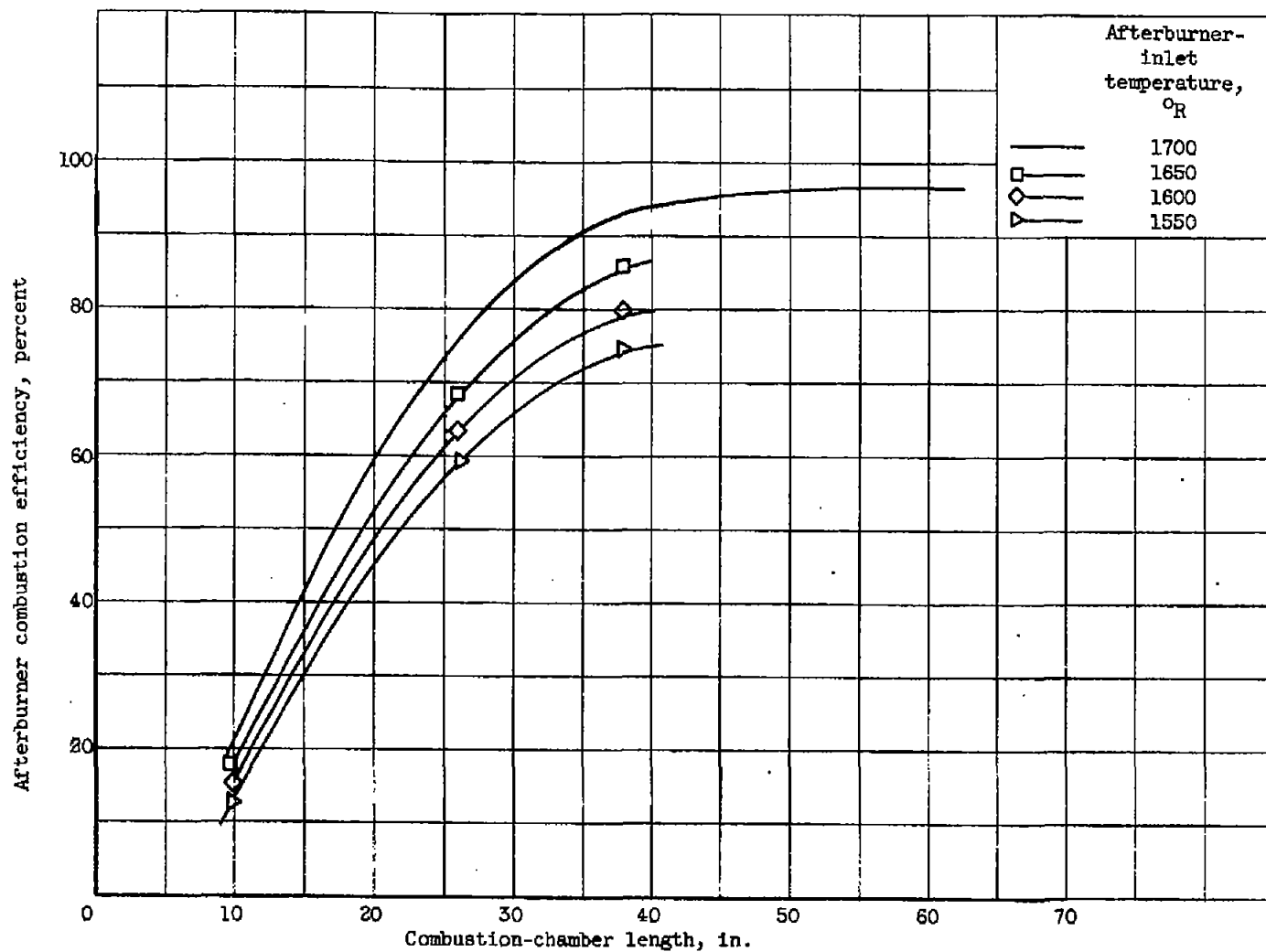
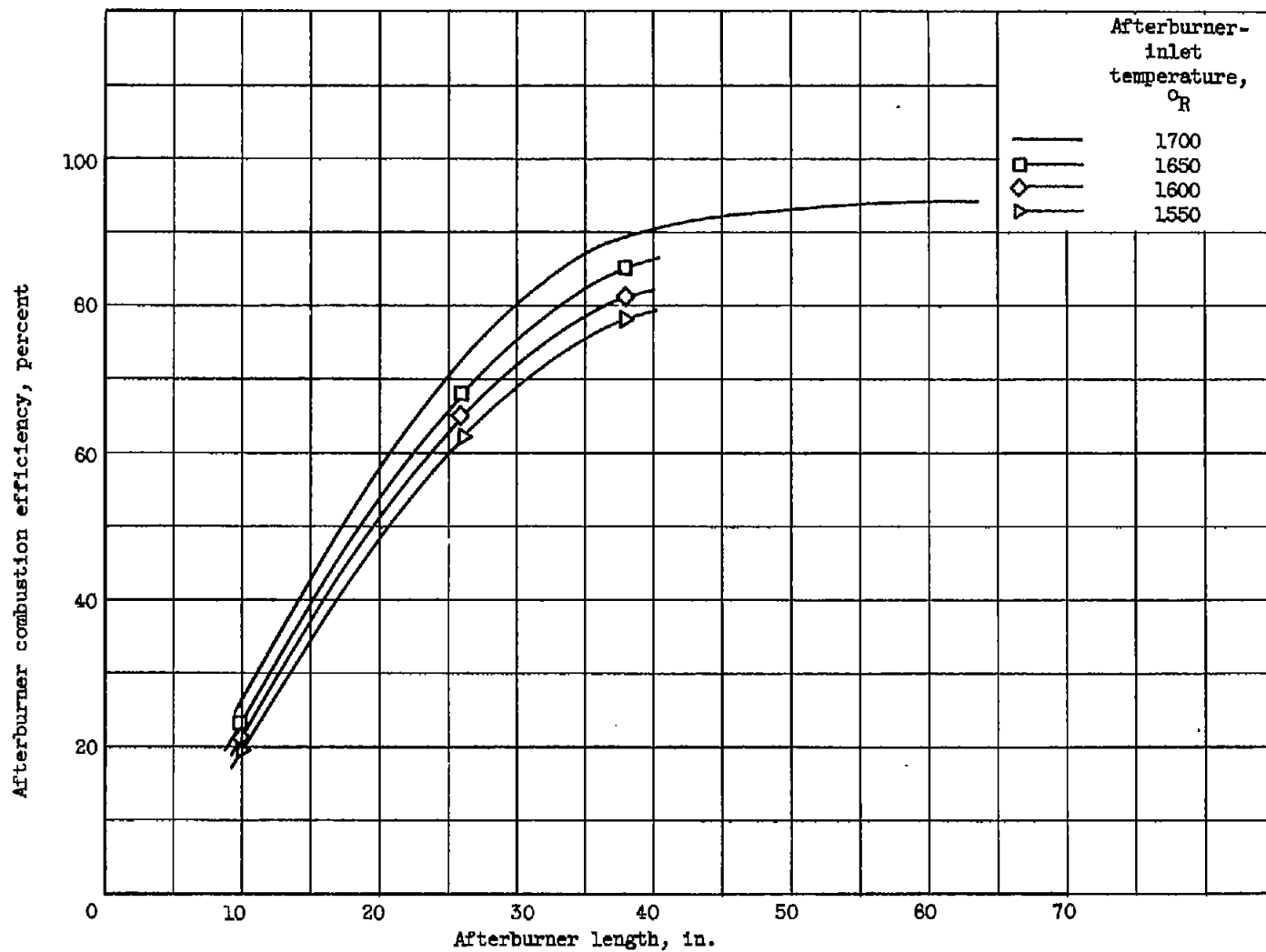


Figure 5. - Effect of combustion-chamber length on normal engine thrust.



(a) Fuel-air ratio, 0.045.

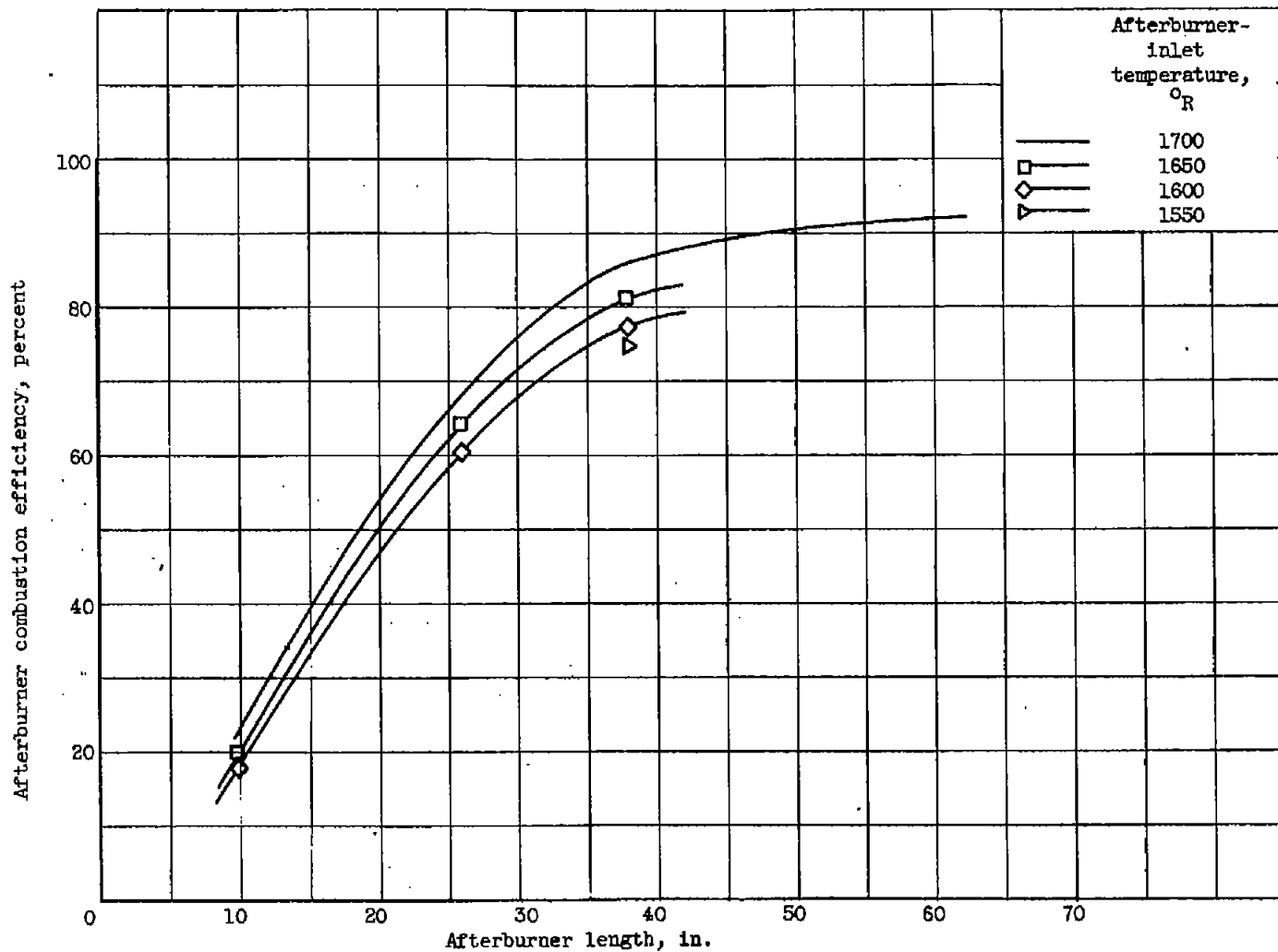
Figure 6. - Effect of afterburner-inlet temperature on combustion efficiency for several combustion-chamber lengths and fuel-air ratios.



(b) Fuel-air ratio, 0.055.

Figure 6. - Continued. Effect of afterburner-inlet temperature on combustion efficiency for several combustion-chamber lengths and fuel-air ratios.





(c) Fuel-air ratio, 0.065.

Figure 6. - Concluded. Effect of afterburner-inlet temperature on combustion efficiency for several combustion-chamber lengths and fuel-air ratios.

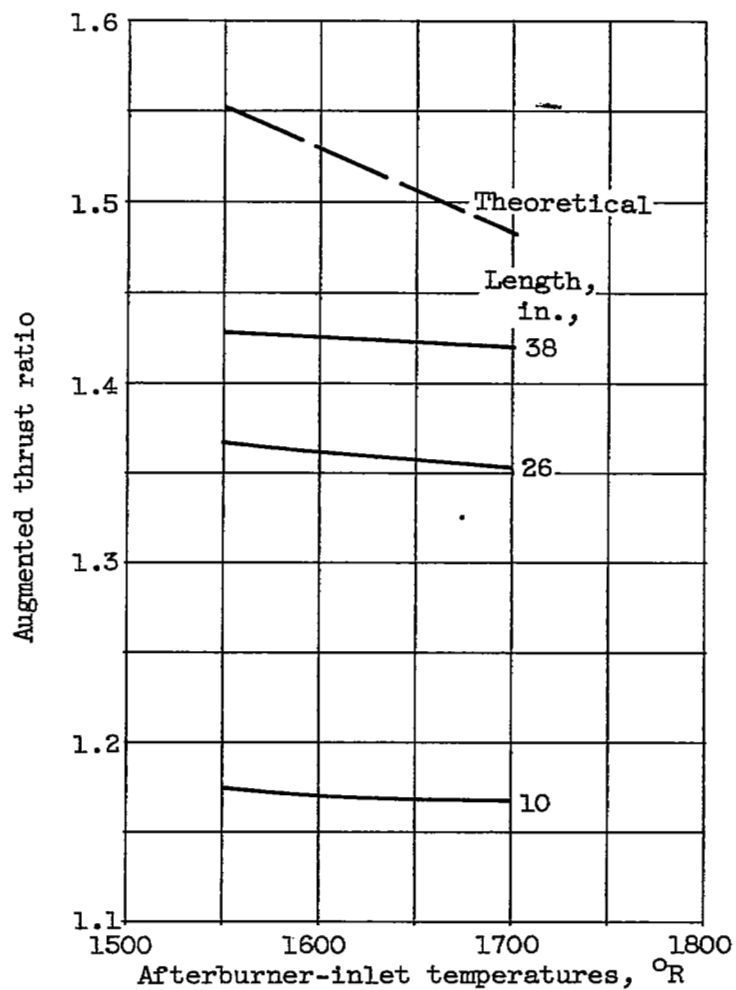


Figure 7. - Effect of afterburner-inlet temperature on augmented thrust ratio for several combustion-chamber lengths. Fuel-air ratio, 0.055.

UNCLASSIFIED

NASA Technical Library



3 1176 01435 4451

UNCLASSIFIED

A thick, black, horizontal redaction bar covering several lines of text at the bottom of the page.