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TURBOJET COMBUSTOR PERFORMANCE WITH NATURAL GAS FUEL

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16. Abstract

Natural gas fuel for advanced turbojet combustor applications was investigated with two swirl-can modular-combustors. In tests at a combustor inlet pressure of 45 psia (31 N/cm^2), inlet temperatures of 540^o and 1140^o F (556 and 889 K) and reference velocities up to 190 ft/sec (57.9 m/sec). The efficiency of both combustors was near 100 percent for inlet air temperatures of 1140^o F (889 K). However, at inlet air temperatures of 540^o F (556 K), the efficiency of one design decreased rapidly with decreasing fuel-air ratio. Acoustical instabilities, encountered with both combustors as initially designed, disappeared when the blockage at the module exit plane was sufficiently increased. Altitude blowout and ignition performance was poorer than that of a JP-fueled combustor of similar design. Vitiated preheat inlet air had an important adverse effect on combustion efficiency.

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SUMMARY

Natural gas as a fuel for advanced turbojet combustor applications was investigated in tests with two swirl-can modular-combustor designs. The combustors from the diffuser inlet to the exhaust nozzle were 34 inches (86 cm) long. The diffuser was 15 inches (38 cm) long with a 33° included angle. Burning length in the combustors was approximately 20 inches (51 cm). Tests were conducted over a range of fuel-air ratios at a pressure of 45 psia (31 N/cm^2), combustor inlet temperatures of 540° and 1140° F (556 and 889 K) and combustor reference velocities up to 190 feet per second (57.9 m/sec).

With an inlet air temperature of 1140° F (880 K) the efficiency of both combustors was near 100 percent over a range of fuel-air ratios. However, with an inlet air temperature of 540° F (556 K), the efficiency of one design decreased rapidly with decreasing fuel-air ratio. Altitude blowout and ignition performance of both combustors was poorer than that of a JP-fueled combustor of similar design. Acoustical instabilities were encountered in both gas-fueled combustor designs. The use of absorbing liners only partially dissipated the instabilities. The instabilities did disappear when the blockage at the module exit plane was significantly increased. Vitiated preheat inlet air was found to have an important adverse effect on combustion efficiency.

INTRODUCTION

Recent studies have shown that liquefied natural gas fuel offers significant advantages over JP-type fuels in some turbojet engine applications (e.g., refs. 1 to 3). For example, a 31-percent payload improvement was calculated for a Mach 3 supersonic transport application (ref. 1) assuming the increased heat sink capacity of the liquefied natural gas was used to allow higher turbine inlet temperatures. Alternatively, the extra heat sink capacity could be used to maintain lower turbine metal temperatures and thereby improve turbine life and reliability. Other advantages of liquefied natural gas fuel include a higher heating value than JP-fuels, less tendency to smoke, lower flame radiation and greatly reduced tendency to fuel decomposition (ref. 3).

The use of liquefied natural gas fuel introduces some special considerations. Although stored as a liquid on board the aircraft, it will enter the combustor as a gas. Experience with gaseous fuels in turbojet combustors is limited. A study of the results of early investigations (refs. 4 to 8) has shown that, while higher combustion efficiencies could usually be obtained with gaseous fuels, the way in which the fuel was injected had a major influence on the efficiency and on the lean and rich stability limits. This influence was most pronounced at lower pressures. Liquefied natural gas (approx. 90 percent methane), having narrower flammability limits than propane or other commonly used gaseous fuels (ref. 9) may be more critical in this regard. The tests described in reference 8, comparing natural gas to propane in an annular turbojet combustor, do show a narrowing of the limits of stable operation when natural gas was used. This appeared to be due partly to the poorer burning characteristics of natural gas and partly to the change in fuel distribution caused by the difference in density between natural gas and propane. When the fuel nozzle orifice size was enlarged for the natural gas, to match the momentum of the propane at the same fuel flow, performance was substantially improved.

It can be expected from the limited information available in the literature that some problems might be experienced in getting adequate efficiency and stability range in a natural-gas-fueled turbojet combustor. Therefore, a project was undertaken to test such a combustor over a range of operating conditions. An advanced turbojet combustor was designed for the use of natural gas. A modular combustor design was used which had been developed in the past for gaseous fuels (refs. 10 to 12). A similar modular approach was also used recently with liquid JP-fuel in a high temperature advanced combustor design (ref. 13). The primary purpose of the present study was to determine the performance problems of an advanced design turbojet combustor operating on natural gas fuel.

TEST INSTALLATION

A 12- by 30-inch (30.5- by 76.2-cm) test section, housing the combustor array, was installed in a closed-duct test facility (fig. 1) connected to the laboratory air supply and exhaust systems. Combustion air at pressures up to 150 psia (103.5 N/cm^2) was passed through a nonvitiating preheater which was capable of heating the air to 600° F (589 K). For those conditions requiring a combustor-inlet temperature of 1140° F (889 K), the air was preheated further by a vitiating preheater consisting of 10 J71 single combustor cans. A set of baffles was installed downstreamof the J71 cans to ensure a uniform temperature profile at the combustor inlet. Airflow rates and combustor pressures were regulated by remotely controlled valves upstream and downstream of the test section.

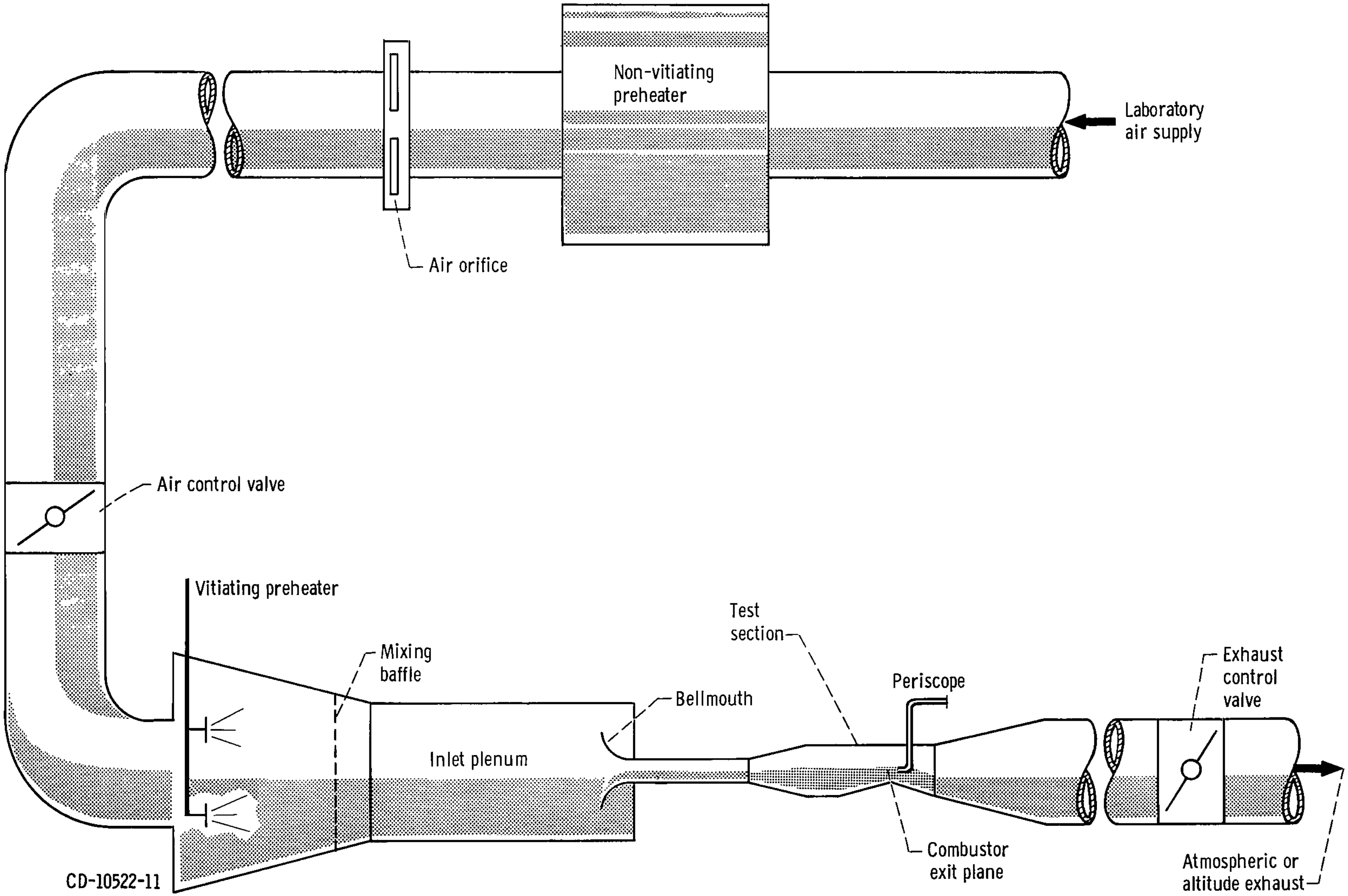

Figure 1. - Test facility and auxiliary equipment.

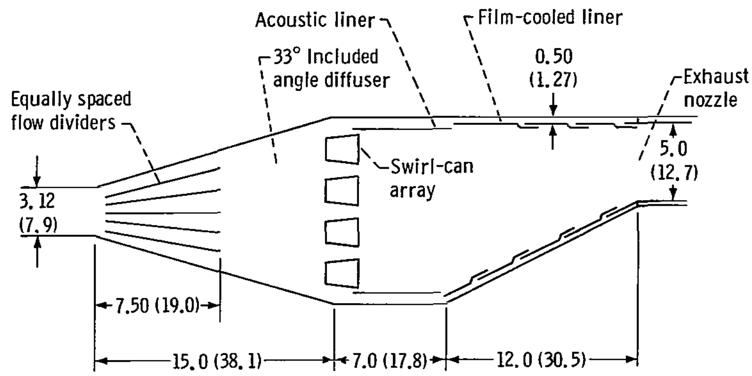

Figure 2. - Combustor installation in test section. (Dimensions are in inches (cm).)

The test sections (fig. 2) were scaled to simulate a 90° sector of a full annulus of a turbojet engine combustor with a 57-inch (145-cm) diameter outer casing, a length of 34 inches (86 cm), and a duct height of 12 inches (30 cm). For ease of fabrication, the test sections were made rectangular in cross section. The diffuser had a 33° included angle and was 15 inches (38 cm) long. Because of the steep diffuser angle, equally spaced flow dividers were installed to provide a uniform velocity profile and to prevent flow separation at the walls. The inlet sections of the combustor modules were located at the end of the diffuser. A film-cooled liner extending from the downstream end of the combustor modules to the exhaust nozzle was used to protect the outer housing. A description of the test instrumentation is presented in the appendix.

TEST COMBUSTORS

The combustors designed for the natural gas tests were based on the modular approach introduced in reference 12. In this approach, the combustor consists of an array of small swirl-can combustor modules. Each module acts as an individual flameholder. Two module designs were used in the present tests: model I, a 2-inch (5.08-cm) diameter module (fig. 3); and model II, a 3.5-inch (8.9-cm) nominal diameter scalloped module (fig. 4).

The original version of the model I combustor array is shown in figure 5. The two strips between two modules in adjacent rows were to assist in cross-firing. The four half-modules shown in figure 5 were nonburning and were installed to provide uniform blockage. The original version of the model II combustor is shown in figure 6.

In both models, the swirl-cans were supported by the fuel supply manifold. Fuel injection holes were designed so that the fuel would be injected tangentially at sonic velocity at most operating conditions, assuring even fuel distribution through each row. The

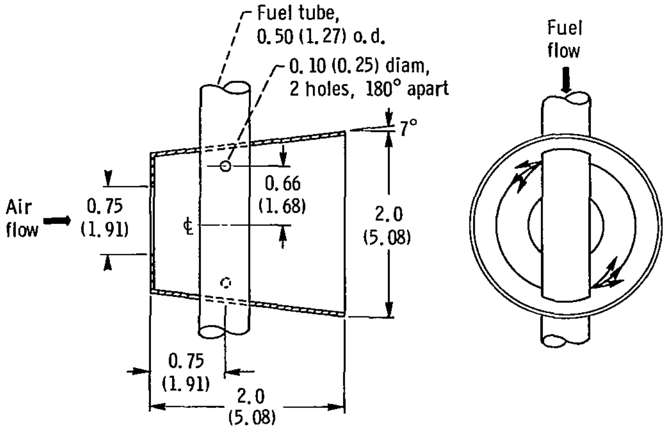

Figure 3. - Model I swirl-can combustor module. (Dimensions are in inches (cm).)

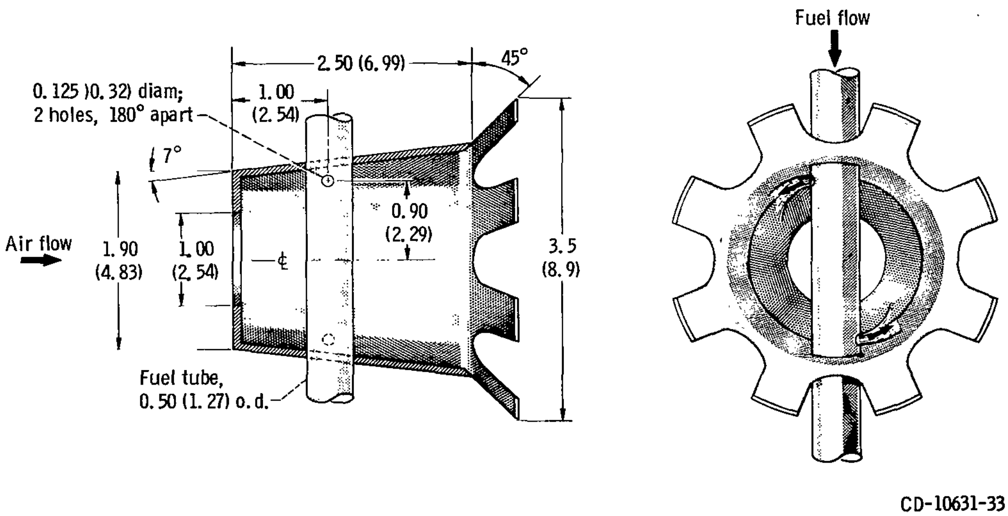

Figure 4. - Model II swirl-can combustor module. (Dimensions are in inches (cm).)

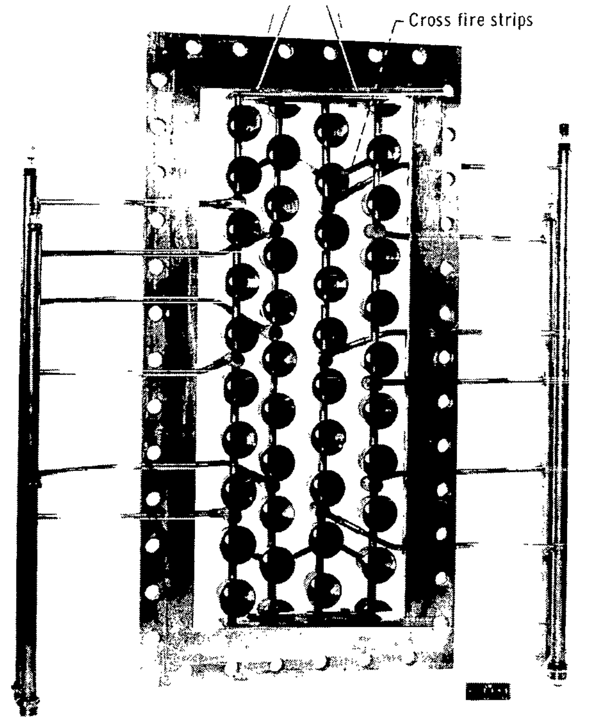

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Figure 5. - Original array of Model I swirl-can combustor modules (looking upstream).

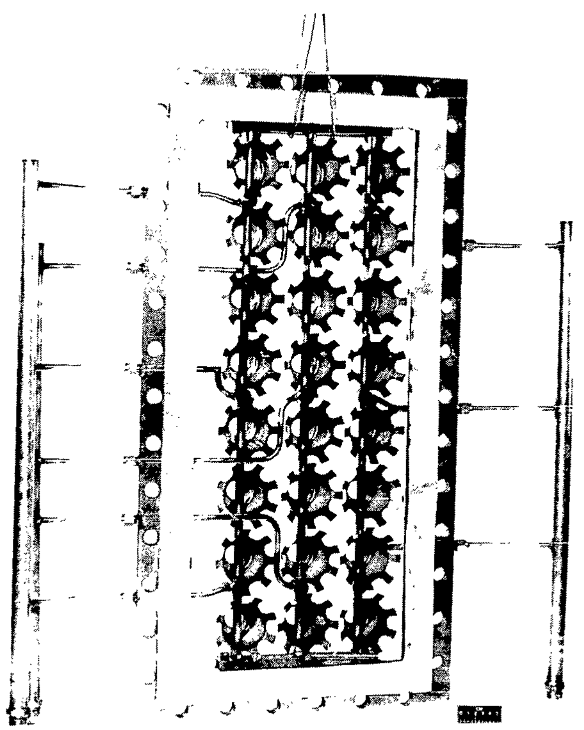

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Figure 6. - Original array of Model II swirl-can combustor modules (looking upstream).

individual manifold fuel flows were varied in some cases to improve the outlet temperature profile.

RESULTS AND DISCUSSION

The two combustors initially exhibited very low pressure loss, acoustic instability, poor temperature profiles, and poor ignition characteristics. The arrays were modified by the addition of tabs between cans which improved temperature distribution and ignition. Also blockage was added to the periphery to improve temperature profiles. The resulting combustor configurations of the model I and model II combustors are shown in figures 7 and 8, respectively. The hardware additions resulted in an increase in pressure loss and elimination of the instability.

Tests of the two combustors were conducted at the conditions shown in table I. Data are presented in tables II to VI.

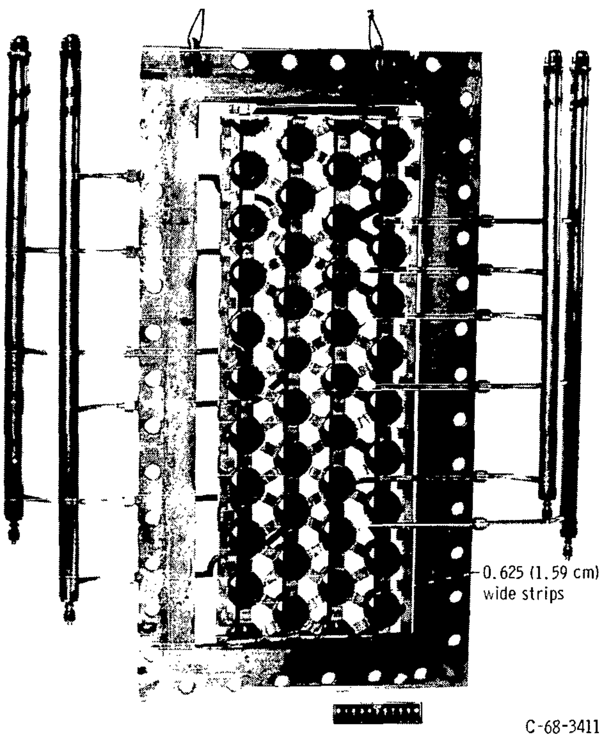

Figure 7. - Final array of model I swirl-can combustor modules (looking upstream).

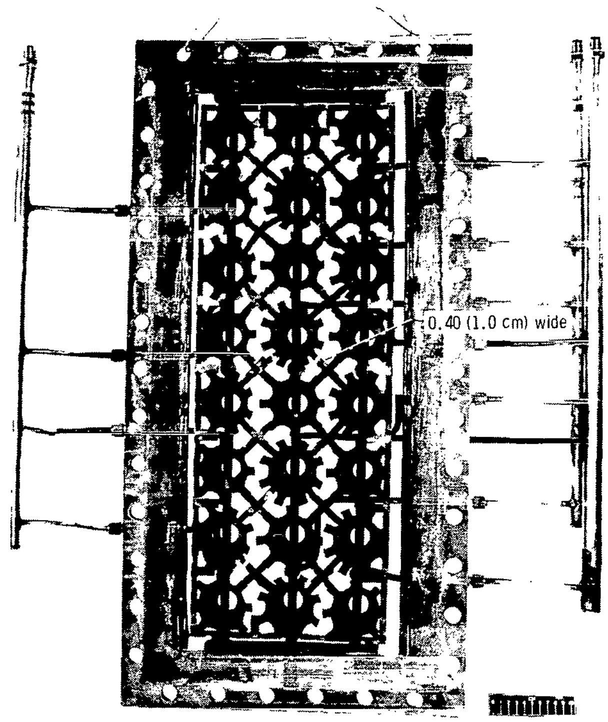

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Figure 8. - Final array of Model II swirl-can combustor modules (looking upstream).

TABLE I. - COMBUSTOR TEST CONDITIONS

[Desired average combustor exit temperature, 2200° F (1480 K); combustor inlet total pressure, 45 psia (31 N/cm^2).]

Test condition	Combustor inlet temperature		Combustor reference velocity[a]		Combustor inlet Mach number
	°F	K	ft/sec	m/sec	
1	540	556	95	29.0	0.243
2	540	556	120	36.6	.313
3	540	556	150	45.7	.405
4	540	556	190	57.9	.526
5	1140	889	90	27.4	.182
6	1140	889	120	36.6	.246
7	1140	889	150	45.7	.313
8	1140	889	190	57.9	.412

[a]Based on maximum cross-sectional area of combustor housing and total pressure and temperature at diffuser inlet.

TABLE II. - FINAL PERFORMANCE DATA FOR MODEL I COMBUSTOR

Run	Nominal reference velocity		Diffuser inlet Mach number	Air flow rate		Fuel-air ratio	Combustor inlet total pressure		Combustor inlet total temperature		Combustor exit total temperature		Combustor efficiency, percent	Total pressure loss, percent
	ft/sec	m/sec		lb/sec	kg/sec		psia	N/cm^2	^oF	K	^oF	K		
1	95	28.9	0.239	28.4	12.9	0.0168	45.2	31.2	539	555	1804	1258	105.5	5.36
2			.238	28.5	12.9	.0188	45.2	31.2	541	557	1946	1337	106.3	5.41
3			.241	28.3	12.8	.0214	44.8	30.9	542	557	2098	1421	105.1	5.61
4			.239	28.3	12.8	.0141	45.0	31.0	539	555	1619	1155	105.5	5.37
5			.240	28.9	13.1	.0107	45.0	31.0	540	556	1388	1027	106.7	5.28
6			.239	28.7	13.0	.0062	44.0	30.3	540	556	941	778	104.8	5.19
7	120	36.6	0.315	36.3	16.5	------	45.1	31.1	531	551	----	----	-----	8.19
8			.312	36.2	16.4	.0149	45.2	31.2	534	552	1602	1155	100.9	8.97
9			.310	36.5	16.6	.0169	45.0	31.0	535	553	1761	1234	102.0	9.17
10			.319	36.3	16.5	.0187	45.0	31.0	538	554	1870	1294	101.2	9.24
11			.319	36.5	16.6	.0198	44.7	30.8	537	554	1954	1341	102.7	9.45
12			.315	36.6	16.6	.0204	45.0	31.0	538	554	1988	1365	103.0	9.43
13			.320	36.4	16.5	.0217	45.3	31.3	539	555	2089	1416	103.7	9.30
14			.309	36.2	16.4	.0138	45.1	31.1	538	554	1543	1113	100.3	9.06
15			.313	36.4	16.5	.0123	45.1	31.1	539	555	1453	1064	101.7	8.95
16			.316	36.2	16.4	.0107	45.0	31.0	540	556	1341	1000	100.7	8.97
17			.314	36.4	16.5	.0088	45.0	31.0	541	556	1208	927	100.9	8.94
18			.313	36.5	16.6	.0068	45.0	31.0	541	556	1054	841	98.7	8.84
19	150	45.7	.414	45.4	20.6	------	45.1	31.1	538	554	----	----	-----	13.0
20			.413	45.8	20.8	.0146	45.2	31.2	535	553	1540	1111	95.1	14.1
21			.407	45.5	20.6	.0166	45.2	31.2	537	554	1669	1183	95.5	14.1
22			.414	45.6	20.7	.0182	44.9	31.0	537	554	1774	1241	96.4	14.5
23			.411	46.0	20.9	.0193	45.0	31.0	536	553	1843	1279	96.8	14.4
24			.409	45.7	20.7	.0202	45.3	31.3	538	554	1914	1318	97.8	13.9
25			.406	46.0	20.9	.0122	45.3	31.3	539	555	1406	1036	96.6	14.0
26			.411	45.5	20.6	.0104	44.9	31.0	537	554	1290	972	96.6	13.9
27			.412	45.5	20.6	.0085	45.0	31.0	537	554	1139	888	93.6	13.9
28			.412	45.7	20.7	.0069	45.0	31.0	539	555	998	832	87.5	13.7
29			.409	45.4	20.6	.0208	45.0	31.0	535	553	1953	1340	98.0	16.0
30	177	54.0	.520	55.3	25.1	.0170	46.2	31.9	531	551	1493	1085	79.4	25.8
31	95	28.9	.196	18.2	8.3	.0171	44.9	31.0	1116	875	2297	1532	107.4	3.6
32	120	36.6	.238	22.5	10.2	.0167	45.4	31.3	1109	871	2233	1496	104.5	5.36
33			.248	22.4	10.2	.0104	44.7	30.8	1122	879	1885	1302	106.5	5.54
34			.245	22.7	10.3	.0065	44.9	31.0	1099	880	1631	1162	102.7	5.34
35	150	45.7	.323	28.5	12.9	.0167	45.0	31.0	1118	877	2242	1501	104.5	9.58
36			.321	28.5	12.9	.0107	45.1	31.1	1123	880	1886	1303	106.5	9.35
37			.316	28.8	13.0	.0072	45.2	31.2	1099	866	1605	1147	102.7	8.84
38	190	57.9	.413	36.7	16.6	.0163	44.6	30.8	1091	862	2272	1518	107.3	16.0

TABLE III. - FINAL PERFORMANCE DATA FOR MODEL II COMBUSTOR

Run	Nominal reference velocity		Diffuser inlet Mach number	Air flow rate		Fuel-air ratio	Combustor inlet total pressure		Combustor inlet total temperature		Combustor exit total temperature		Combustor efficiency, percent	Total pressure loss, percent
	ft/sec	m/sec		lb/sec	kg/sec		psia	N/cm^2	^oF	K	^oF	K		
1	92	28.0	0.229	27.7	12.6	0.0155	44.8	30.9	542	557	1700	1200	103.8	4.57
2			.234	27.3	12.4	.0175	44.6	30.8	543	557	1814	1263	102.7	4.70
3			.231	27.5	12.5	.0130	45.2	31.2	541	556	1547	1115	99.4	4.49
4			.232	28.2	12.8	.0116	45.0	31.0	542	557	1397	1032	99.7	4.48
5			.228	27.7	12.6	.0099	45.3	31.3	543	557	1249	949	95.7	4.42
6			.233	28.1	12.7	.0079	45.0	31.0	542	557	1101	867	93.0	4.43
7			.229	27.7	12.6	.0051	45.2	31.2	540	556	869	738	83.4	4.35
8	120	36.6	.321	37.0	16.8	.0151	45.2	31.2	541	556	1585	1136	96.2	8.28
9			.319	36.4	16.5	.0171	44.8	30.9	543	557	1725	1214	97.4	8.55
10			.315	36.8	16.7	.0172	45.1	31.1	540	556	1757	1232	99.8	8.41
11			.316	36.9	16.7	.0144	44.9	31.0	541	556	1524	1102	94.8	8.34
12			.314	36.6	16.6	.0132	45.3	31.3	543	557	1425	1047	91.6	8.13
13			.314	36.7	16.6	.0113	45.1	31.1	541	556	1271	962	87.2	8.10
14			.32-	36.5	16.6	.0094	45.2	31.2	542	557	1125	880	82.9	8.05
15			.313	36.8	16.7	.0077	45.4	31.3	542	557	1012	818	79.9	7.92
16			.325	36.8	16.7	.0052	45.0	31.0	543	557	854	730	77.7	7.93
17	150	45.7	.412	46.5	21.1	.0138	44.8	30.9	534	552	1324	991	78.6	14.6
18			.417	46.3	21.0	.0144	45.0	31.0	539	555	1394	1030	81.9	14.5
19			.411	46.3	21.0	.0158	45.2	31.2	537	554	1536	1109	88.0	14.4
20			.429	45.8	20.8	.0106	44.5	30.7	538	554	1099	866	70.9	14.4
21			.421	45.7	20.7	.0071	44.9	31.0	539	555	865	736	59.8	14.0
22	90	27.4	.180	16.4	7.4	.0132	44.7	30.8	1153	896	2024	1380	100.8	2.44
23			.178	16.5	7.5	.0154	44.6	30.8	1153	896	2148	1449	100.1	2.44
24			.183	16.8	7.6	.0185	44.3	30.5	1149	894	2374	1574	104.2	2.57
25			.178	16.5	7.5	.0114	44.8	30.9	1152	895	1908	1315	100.4	2.52
26			.184	16.8	7.6	.0088	44.6	30.8	1146	892	1744	1224	101.1	2.41
27			.179	16.9	7.7	.0072	44.7	30.8	1151	895	1640	1167	99.9	2.39
28	120	36.6	.242	22.6	10.3	.0123	44.9	31.0	1144	891	1993	1363	104.6	4.33
29			.242	22.0	10.0	.0148	44.7	30.8	1149	894	2124	1435	101.7	4.44
30			.244	22.1	10.0	.0167	44.4	30.6	1155	897	2229	1494	100.5	4.49
31			.243	22.8	10.3	.0104	44.8	30.9	1151	895	1876	1298	104.1	4.32
32			.244	22.2	10.1	.0089	44.6	30.8	1150	894	1759	1233	101.5	4.36
33			.244	22.7	10.3	.0066	44.6	30.8	1146	892	1598	1143	100.4	4.30
34	150	45.7	.316	28.1	12.7	.0127	44.8	30.9	1150	894	1964	1347	97.9	7.94
35			.312	27.8	12.6	.0146	44.9	31.0	1153	896	2104	1424	100.1	7.95
36			.314	27.7	12.6	.0167	44.8	30.9	1151	895	2213	1485	99.2	8.03
37			.315	29.0	13.2	.0171	44.4	30.6	1141	889	2274	1519	103.3	8.21
38			.320	28.6	13.0	.0106	44.7	30.8	1143	890	1874	1297	103.7	7.80
39			.315	28.1	12.7	.0086	44.8	30.9	1144	891	1708	1204	97.9	7.70
40			.318	28.3	12.8	.0062	44.7	30.8	1144	891	1556	1120	96.7	7.61
41	190	57.9	.419	36.5	16.6	.0142	44.7	30.8	1111	873	2025	1380	98.2	13.5
42			.420	37.2	16.9	.0166	44.4	30.6	1106	870	2227	1493	104.4	13.8
43			.416	36.8	16.7	.0175	44.7	30.8	1104	869	2259	1510	102.5	13.5
44			.417	36.5	16.6	.0079	44.7	30.8	1104	869	1618	1154	95.4	12.9

TABLE IV. - EFFECT OF VITIATION ON MODEL II COMBUSTOR EFFICIFNCY

Run	Nominal reference velocity		Diffuser inlet Mach number	Air flow rate		Fuel-air ratio	Combustor inlet total pressure		Combustor inlet total temperature				Combustor exit total temperature		Combustor efficiency, percent	Total pressure loss, percent
									Vitiated		Nonvitiated					
	ft/sec	m/sec		lb/sec	kg/sec		psia	N/cm^2	^oF	K	^oF	K	^oF	K		
1	95	28.9	0.244	28.1	12.7	0.0158	44.9	31.0	577	576	---	---	1655	1175	97.5	4.93
2			.244	28.5	12.9	.0187	45.0	31.0	564	569	---	---	1896	1309	103.3	4.96
3			.242	28.6	13.0	.0110	44.7	30.8	564	569	---	---	1196	920	77.5	4.67
4			.243	28.8	13.1	.0154	44.8	30.9	---	---	534	552	1643	1168	100.3	4.82
5			.239	28.5	12.9	.0184	45.1	31.1	---	---	535	553	1870	1294	102.9	4.93
6			.242	28.8	13.1	.0196	44.9	31.0	---	---	534	552	1983	1357	105.6	5.04
7			.240	29.2	13.2	.0101	44.8	30.9	---	---	534	552	1246	948	94.1	4.69
8	120	36.6	.340	36.0	16.3	.0150	45.0	31.0	552	562	---	---	1281	967	69.0	8.10
9			.340	35.7	16.2	.0189	44.7	30.8	552	562	---	---	1373	1018	66.4	8.15
10			.324	35.8	16.2	.0190	44.7	30.8	554	563	---	---	1482	1079	71.1	8.21
11			.326	36.9	16.7	.0104	44.8	30.9	558	566	---	---	997	809	57.6	7.87
12			.315	36.1	16.4	.0157	44.8	30.9	---	---	537	554	1620	1155	96.0	8.16
13			.315	36.0	16.3	.0189	45.0	31.0	---	---	537	554	1856	1287	98.8	8.50
14			.316	36.6	16.6	.0103	44.9	31.0	---	---	541	557	1188	916	83.9	7.86
15			.314	36.8	16.7	.0103	44.8	30.9	---	---	541	557	1181	912	83.6	7.79
16	150	45.7	.404	45.5	20.6	.0151	44.3	30.5	508	538	---	---	567	571	5.4	12.4
17			.414	45.8	20.8	.0148	44.8	30.9	---	---	532	551	1433	1052	84.2	13.6
18			.412	46.1	20.9	.0176	44.8	30.9	---	---	531	551	1693	1196	93.4	14.0
19			.413	46.1	20.9	.0101	44.6	30.8	---	---	530	550	1062	846	70.1	13.2

TABLE V. - BLOWOUT AND IGNITION LIMITS OF MODEL I COMBUSTOR

Run	Combustor reference velocity		Diffuser inlet Mach number	Air flow rate		Fuel-air ratio	Combustor inlet total pressure		Combustor inlet total temperature		Ignition	Blowout
	ft/sec	m/sec		lb/sec	kg/sec		psia	N/cm^2	^oF	K		
1	86	26.2	0.306	30.2	13.7	0.0106	27.6	19.0	75	297	Yes	Yes
2	83	25.3	.296	25.2	11.4	.0121	23.9				No	Yes
3	79	24.1	.281	19.4	8.8	.0144	19.5	13.4			No	Yes
4	79	24.1	.273	24.9	11.3	.0124	24.9	17.2			Yes	No
5	70	21.3	.240	19.7	8.9	.0179	22.4	15.4			Yes	No
6	73	22.2	.253	19.4	8.8	.0141	21.0	14.5			Yes	No
7	66	20.1	.240	10.1	4.6	.0163	12.1	8.3			No	Yes
8	56	17.1	.212	5.9	2.7	.0148	8.4	5.8			No	Yes
9	46	14.0	.166	3.3	1.5	.0162	5.8	4.0			No	Yes
10	56	17.1	.149	10.0	4.5	.0151	14.0	9.7			Yes	No
11	56	17.1	.159	10.0	4.5	.0146	14.3	9.9				
12	47	14.3	.147	5.9	2.7	.0183	10.0	6.9				
13	40	12.2	.123	3.2	1.5	.0158	6.2	4.3				
14	144	43.9	.448	16.7	7.6	.0164	14.3	9.9	350	450	No	Yes
15	71	21.6	.204	2.9	1.3	.0168	5.1	3.5			No	No[a]
16	110	33.5	.315	16.5	7.5	.0166	18.4	12.7			Yes	No
17	61	18.6	.126	2.9	1.3	.0178	6.0	4.1			Yes	No
18	170	51.8	.491	14.6	6.6	.0170	12.8	8.8	540	556	No	Yes
19	150	45.7	.450	9.6	4.4	.0166	9.6	6.6			No	Yes
20	85	25.9	.274	3.0	1.4	.0144	5.3	3.7			No	No[a]
21	150	45.7	.403	14.5	6.6	.0171	14.8	10.2			Yes	No
22	133	40.5	.303	9.7	4.4	.0149	11.0	7.6			Yes	No
23	89	27.1	.221	3.1	1.4	.0181	5.2	3.6			Yes	No

[a]Altitude limit of facility.

TABLE VI. - BLOWOUT AND IGNITION LIMITS OF MODEL II COMBUSTOR

Run	Combustor reference velocity		Diffuser inlet Mach number	Air flow rate		Fuel-air ratio	Combustor inlet total pressure		Combustor inlet total temperature		Ignition	Blowout
	ft/sec	m/sec		lb/sec	kg/sec		psia	N/cm^2	^oF	K		
1	50	15.2	0.164	8.8	3.99	0.0115	14.7	10.1	50	283	Yes	No
2	45	13.7	.165	4.0	1.81	.0136	6.0	4.1	80	300	No	Yes
3	75	22.9	.234	13.7	6.21	.0120	15.9	11.0				
4	80	24.4	.289	16.1	7.30	.0100	15.9	11.0				
5	84	25.6	.310	17.8	8.07	.0120	16.9	11.7				
6	101	30.8	.368	25.0	11.3	.0110	25.0	17.2				
7	44	13.4	.165	4.0	1.81	.0136	6.4	4.4			Yes	No
8	44	13.4	.162	4.6	2.09	.0140	8.0	5.5				
9	60	18.3	.210	11.8	5.35	.0137	15.4	10.6				
10	72	21.9	.267	20.0	9.07	.0116	22.1	15.2				
11	70	21.3	.254	20.0	9.07	.0117	23.3	16.0				
12	75	22.9	.245	5.3	2.40	.0096	6.9	4.8	210	372	No	Yes
13	103	31.4	.336	17.4	7.89	.0121	17.0	11.7			No	Yes
14	51	15.5	.186	3.0	1.36	.0150	5.4	3.7			Yes	No
15	72	21.9	.229	8.7	3.95	.0117	11.8	8.1			Yes	No
16	91	27.7	.293	17.3	7.84	.0121	19.0	13.1			Yes	No
17	138	42.1	.425	19.0	8.62	.0105	16.7	11.5	330	439	No	Yes
18	120	36.6	.337	5.4	2.45	.0098	5.8	4.0	400	477	No	Yes
19	86	26.2	.277	6.0	2.72	.0133	8.0	5.5	400	477	Yes	No
20	112	34.1	.357	15.5	7.03	.0128	16.0	11.1	400	477	Yes	No
21	152	46.9	.408	6.0	2.72	.0126	6.0	4.1	600	589	No	Yes
22	181	55.2	.522	9.6	4.35	.0150	8.1	5.6			No	Yes
23	194	59.1	.571	13.0	5.90	.0111	10.3	7.1			No	Yes
24	116	35.4	.295	6.0	2.72	.0126	8.0	5.5			Yes	No
25	142	43.3	.384	7.6	3.45	.0121	10.2	7.7				
26	140	42.7	.377	9.6	4.35	.0106	10.4	7.9				
27	137	41.8	.365	10.5	4.76	.0116	11.7	8.1				

Acoustic Instability

As initially designed, both combustors exhibited an audible combustion instability, confirmed by high frequency pressure transducers in the combustion chamber. The mode of oscillation was transverse at approximately 520 hertz. At a reference velocity of 120 feet per second (36.6 m/sec) this instability was present in the model I combustor at an average temperature rise of 900° F (500 K) and above. At a reference velocity of 190 feet per second (57.9 m/sec), the instability was present at an average temperature rise as low as 600° F (333 K). In the model II combustor, the instability was present only at average temperature rises over 1200° F (666 K).

Because the instability was at a level that damaged the test hardware and prevented testing to the desired average exit temperature of 2200° F (1480 K), measures were taken to eliminate it or lessen its intensity. Screens were installed in the diffuser, perpendicular to the direction of flow, but they did not affect the screech frequency or intensity. Perforated metal liners were installed in place of the solid liners for 6 inches (15 cm) downstream of the swirl cans. The perforations helped lessen the intensity of the screech. These liners were replaced by corrugated and perforated liners for 8 inches (20 cm) downstream of the swirl cans, but these produced no further effect and were removed for the balance of testing. The final acoustic liner configuration is shown in figure 9.

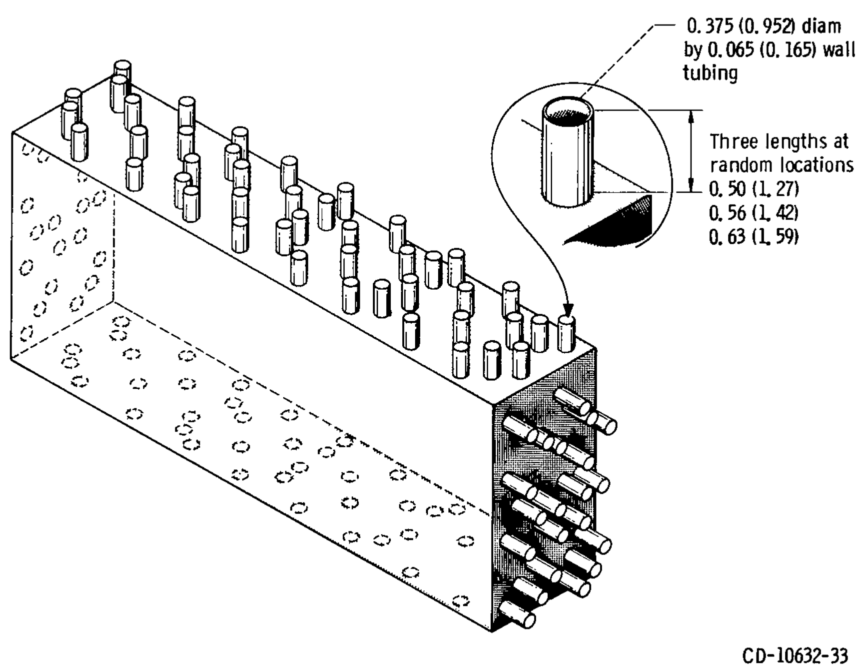


Figure 9. - Acoustic liner. (Dimensions are in inches (cm).).

While decreasing the intensity of the instability to a sufficient degree to permit continued testing, the perforated liners also allowed the liner cooling air to come out into the combustion zone along all four walls. The resulting thick layer of cold air along the walls did not mix sufficiently with the hot gases. Therefore, the exit temperature was very low at the walls. This condition worsened when additional blockage was later added to the modules to improve ignition characteristics. Increasing the pressure loss across the combustor forced even more air through the liners. Although the purpose of the extra blockage was to improve ignition, it also caused the instability to stop entirely. The effect of combustor blockage and temperature rise on acoustic instability has been noticed before (ref. 14). The blockage in the model II combustor rose from 46 percent in its initial design to 66 percent in its final design where the instability disappeared.

Combustion Efficiency

Combustion efficiency was defined as the ratio of actual temperature rise to theoretical temperature rise. The oxygen depletion resulting from vitiation of the combustion air was considered in the combustion efficiency calculations.

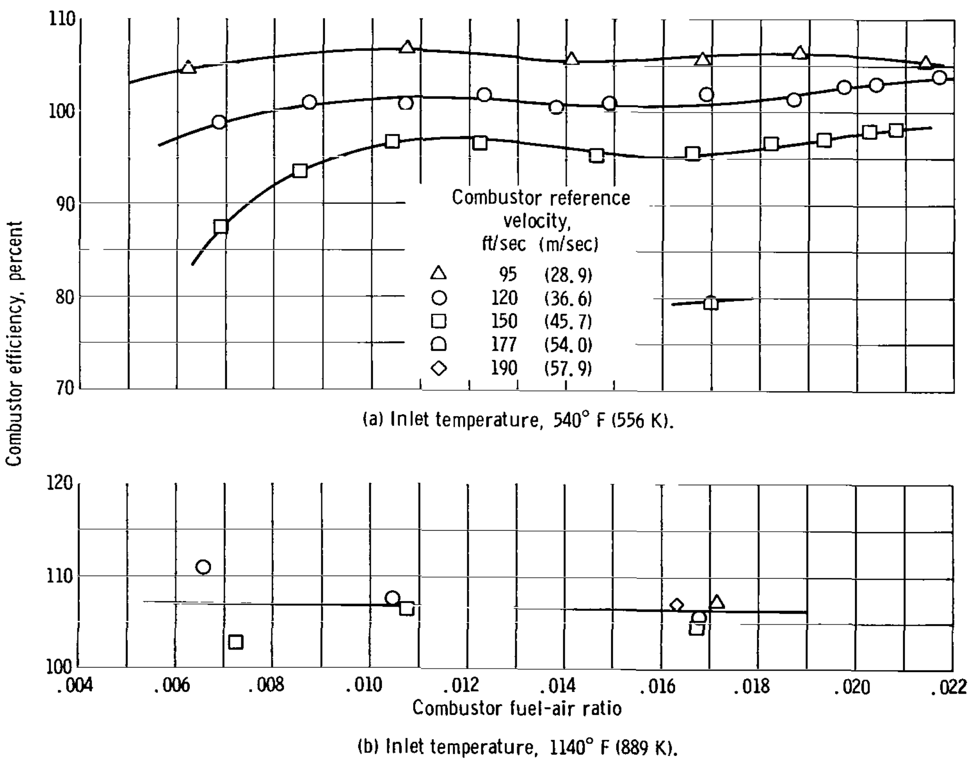
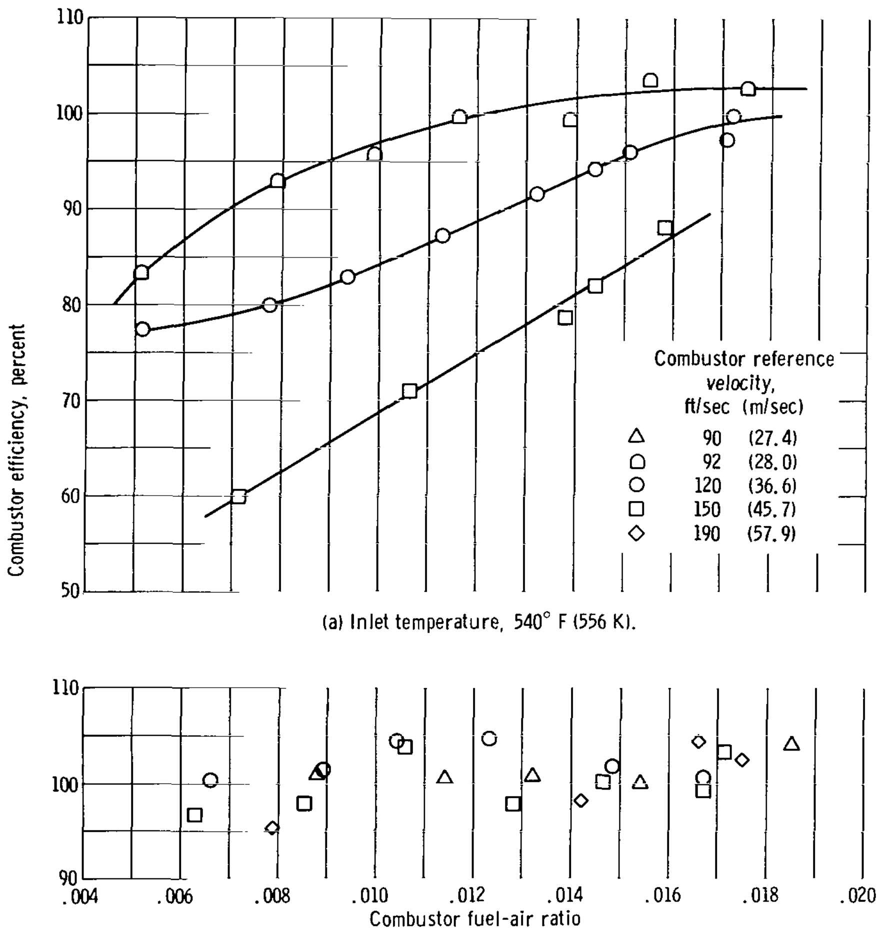


Figure 10. - Effect of combustor inlet conditions on model I combustor efficiency. Inlet pressure, 45 psia (31 N/cm^2).

Low combustion efficiency was obtained with the original model I combustor at an inlet-air temperature of 540° F (556 K). By adding cross strips to the modules the efficiency at the 540° F (556 K) inlet-air temperature was substantially improved so that the efficiency of the final configuration was over 95 percent at reference velocities of 120 feet per second (36.6 m/sec) or less (fig. 10(a)). A dropoff in efficiency with increasing reference velocity is noticeable. At an inlet-air temperature of 1140° F (889 K) the combustion efficiency was 100 percent at all reference velocities (fig. 10(b)). Values of efficiency over 100 percent are believed to be due to air leakage at the flanges and to bent thermocouple probes in the region of large temperature gradients near the walls.

Combustor efficiency data obtained with the original model II combustor at an inlet-air temperature of 540° F (556 K) were consistently low at low fuel-air ratios and had an almost linear increase in efficiency with increasing fuel-air ratio. Attempts to increase the efficiency at the low fuel-air ratios included several changes of the swirl-can inlet-



(a) Inlet temperature, 540° F (556 K).

(b) Inlet temperature, 1140° F (889 K).

Figure 11. - Effect of combustor inlet conditions on model II combustor efficiency. Inlet pressure, 45 psia (31 N/cm²).

orifice diameter and the addition of air swirlers to the inlet orifice. Increasing the inlet-orifice size decreased the efficiency further; however, decreasing the size beyond the initial design size did not produce any notable increase in efficiency. The addition of air swirlers did not significantly affect efficiency. Neither did the addition of cross tabs (initially added to aid altitude relight) significantly improve the efficiency. Efficiency curves for the final model II configuration were very similar to curves for the initial configuration and are shown in figure 11. At the 1140° F (889 K) inlet air temperature, the efficiency was close to 100 percent at all fuel-air ratios and reference velocities.

At an inlet air temperature of 540° F (556 K) the rapid dropoff in efficiency with decreasing fuel-air ratio with the model II combustor did not occur with model I. This result is attributed to geometric differences in the swirl-can modules which affect mixing and quenching. The result cannot be simply attributed to differences in lip velocity since the lip velocity is calculated to be lower for the model II combustor than for the model I combustor for the same reference velocity. This is supported by the fact that while the projected blockages are nearly equal for the two combustors (about 55 percent), the blockage for the model I combustor is essentially all in one plane, as it is not for the model II combustor. In addition, the pressure loss for the model I combustor was higher than that for the model II combustor. The effect of swirl-can module geometry on efficiency performance appears to warrant further investigation.

Effect of Inlet Air Vitiation

The efficiency data presented above at the 540° F (556 K) inlet air temperature conditions were taken with the inlet air heated by the nonvitiating preheater. Because of the long warmup time required for the nonvitiating preheater, it was advantageous on certain occasions to use the vitiating preheater to obtain a few data points to check out the test installation. In these instances the combustion efficiency of the test combustor was lower than expected. Therefore, a test was conducted to determine the effect of inlet air vitiation on combustion efficiency at the 540° F (556 K) inlet air temperature condition. The test was conducted with the model II combustor using first the vitiating preheater and then the nonvitiating preheater. The results of these tests are shown in figure 12. There is a remarkable decrease in combustion efficiency when vitiated preheat air is used. The difference is greatest at high reference velocities and low fuel-air ratios. The low efficiency at the high reference velocities was undoubtedly due to some modules not staying lit at these conditions; although all modules were lit before each condition was set. All modules remained lit when the nonvitiating preheater was used. The effect of vitiated preheat air on combustion efficiency is probably due to the depletion of oxygen (depleted approx 9 percent) in the airflow through the combustor. The effect of oxygen concentration on combustion efficiency has been noted before using

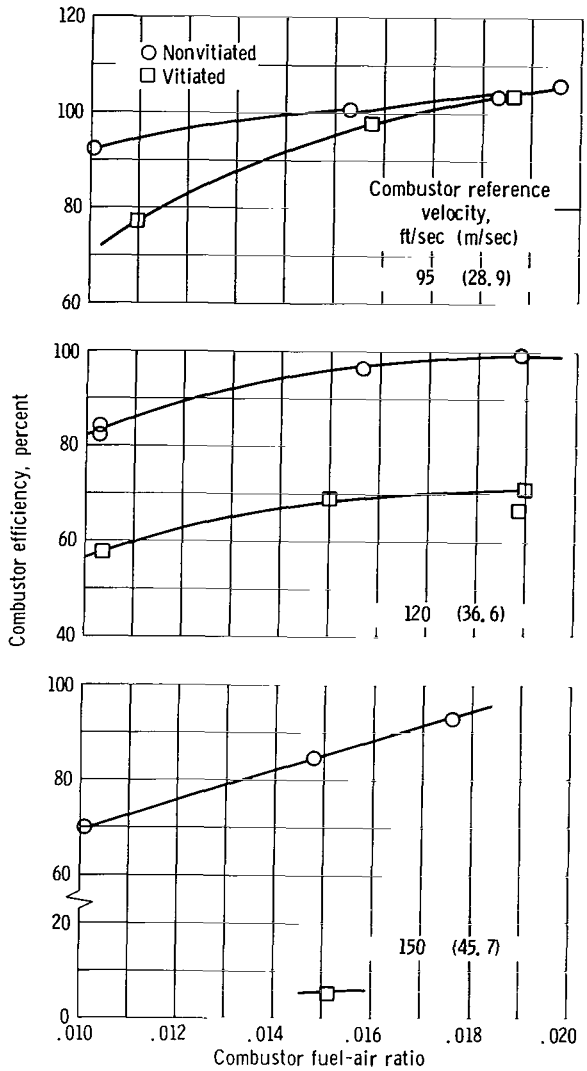
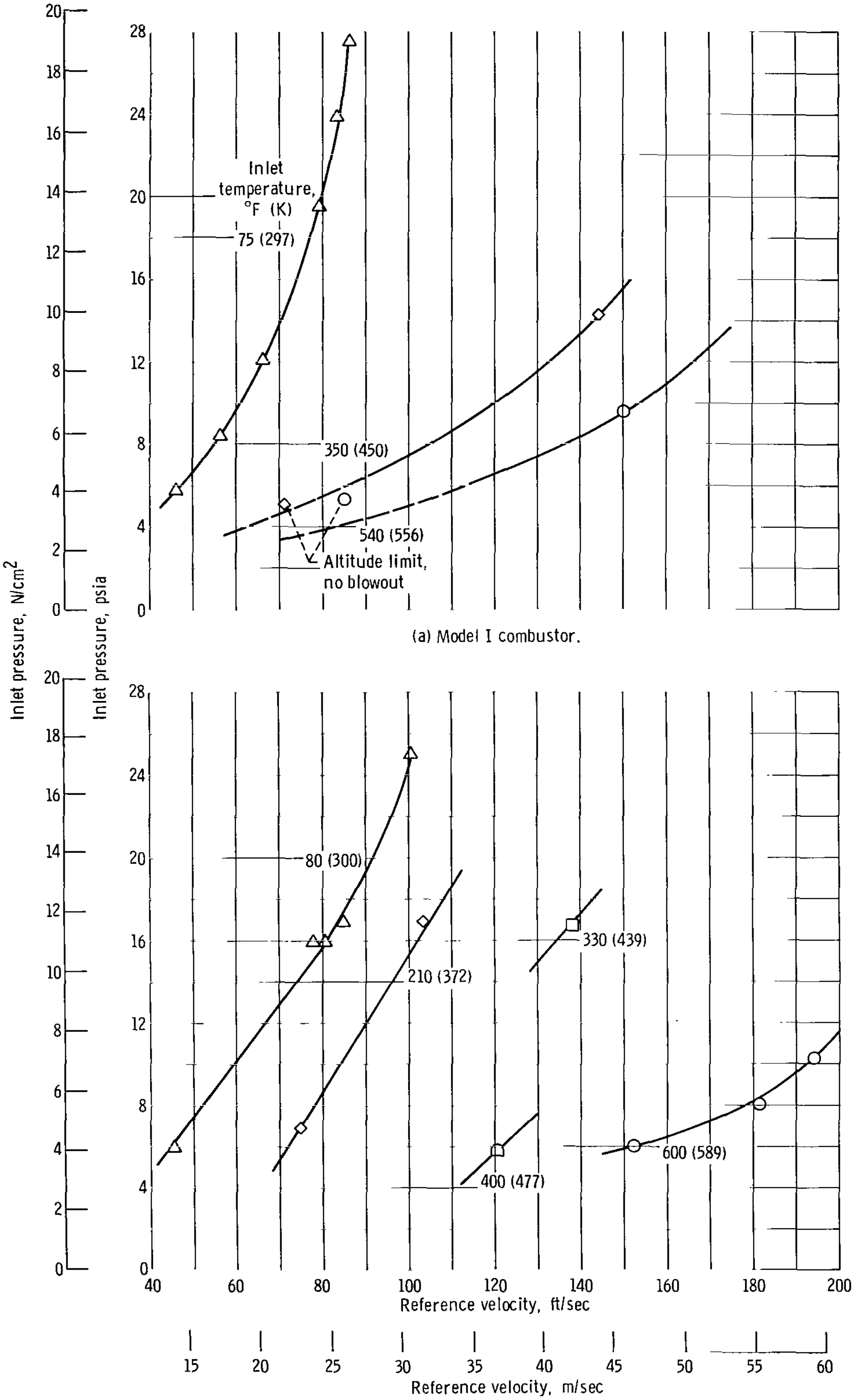


Figure 12. - Effect of vitiation on model II combustor efficiency. Inlet pressure, 45 psia (31 N/cm^2); inlet temperature, 540° F (556 K).

other hydrocarbon fuels (e.g., ref. 15). However, the effect observed with other fuels was much smaller than the effect observed herein with natural gas fuel.

Blowout and Ignition

Blowout data were obtained at a fuel-air ratio of approximately 0.015 at various inlet air temperatures for both combustors and are shown in figure 13. The nonvitiating preheater was used to set the inlet air temperature conditions. In general, the blowout per-



(a) Model I combustor.

(b) Model II combustor.

Figure 13. - Blowout limits.

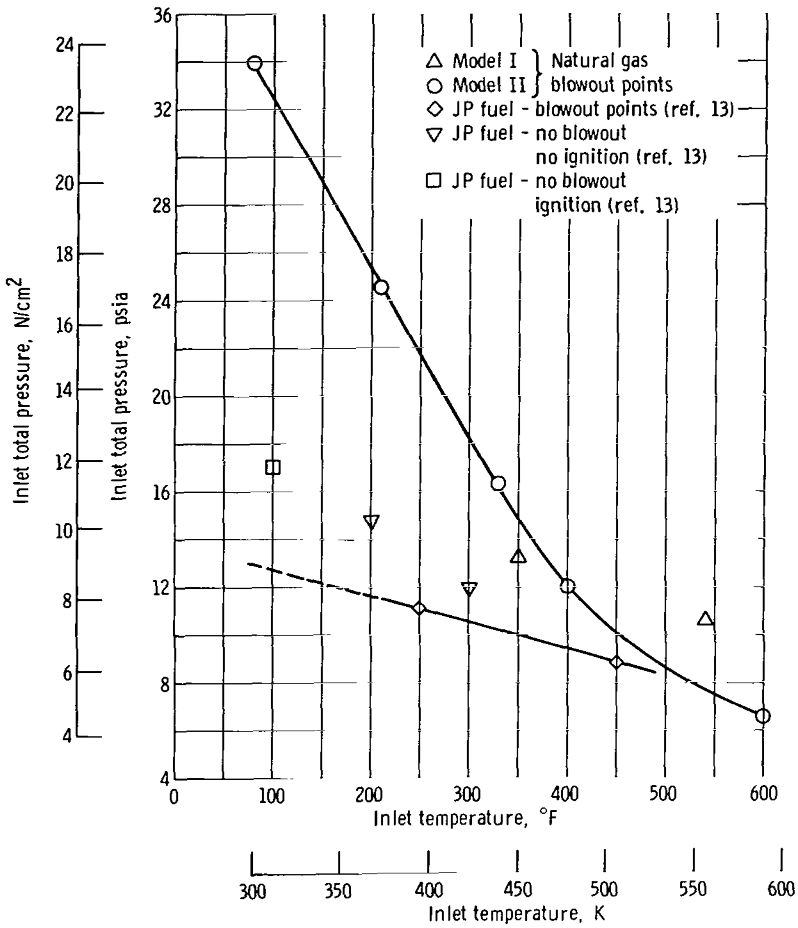
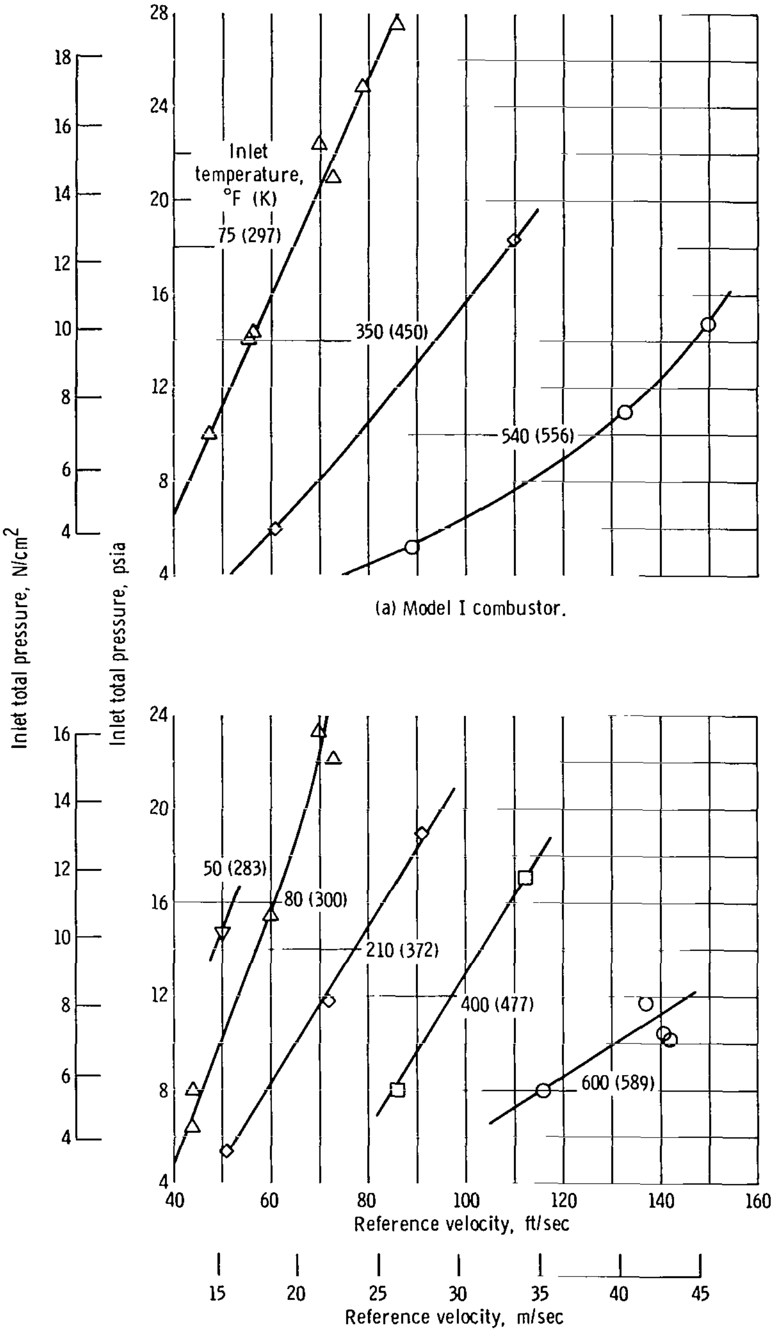


Figure 14. - Comparison of blowout characteristics for methane and JP fueled modular combustors. Reference Mach number, 0.1.

formance of both combustors was poor. In figure 14, the data from figure 13 were extrapolated to a reference Mach number of 0.1 and compared with blowout data taken with a modular combustor burning JP-fuel (ref. 13) at the same reference Mach number. At an inlet air temperature near 500° F (533 K) both fuels behave similarly. But as the inlet air temperature decreases, the natural gas combustors require a greater increase in pressure than the JP-combustor to remain lit.

Figure 14 shows a large difference in the blowout characteristics of the natural gas and JP-combustors at low inlet air temperatures. This difference might be due to several reasons: the geometric design differences of the modules; the fundamental burning differences of the two fuels; or a difference in the local fuel-air ratio patterns in the primary zone as a result of differences in fuel injector geometry, fuel state, and momentum.

Ignition data are presented for both combustors in figure 15. Again, the desired inlet temperature was obtained by use of the nonvitiating preheater. In all cases, ignition



(a) Model I combustor.

(b) Model II combustor.

Figure 15. - Ignition limits.

occurs at somewhat higher values of pressure than those at which the combustor blew out. No effort was made to improve ignition capability by relocating the spark plug or by increasing spark energy. Cross-fire tabs were added between modules early in the program because the flame would not propagate without some bluff body protection.

Generally speaking, the blowout and ignition characteristics of the two natural gas-fueled combustors were poor. Even with the addition of strips between the modules, failure to cross fire was a problem at many operating points. Some change in the method of fuel introduction might have improved the situation but was not undertaken for this study. Special attention to ignition problems appears to be indicated for natural gas-fueled combustors.

Pressure Loss

Combustor pressure loss was defined by the following expression:

$$\frac{\Delta P}{P} = \frac{(\text{Average diffuser inlet total pressure}) - (\text{Average combustor exhaust total pressure})}{\text{Average diffuser inlet total pressure}}$$

Thus, the pressure loss includes the diffuser pressure loss.

Values of pressure loss $\Delta P/P$ for the initial combustor configurations were approximately 4 percent at a diffuser inlet Mach number of 0.313. Due to the instability and poor ignition characteristics, cross-firing tabs and additional flameholding area were added to the arrays. These appendages increased the blocked area to such an extent that further increases in blockage produced large increases in pressure loss. Values of the pressure loss are plotted against diffuser inlet Mach number in figure 16 for the final configurations of the model I and model II combustors.

Temperature Distribution

To describe the quality of the combustor-outlet temperature profile, the following temperature distribution parameters were established:

$$\delta_{stator} = \frac{\left(T_{R,local} - T_{R,design}\right)_{max}}{T_{av}}$$

where $\left(T_{R,local} - T_{R,design}\right)_{max}$ is the largest temperature difference between the

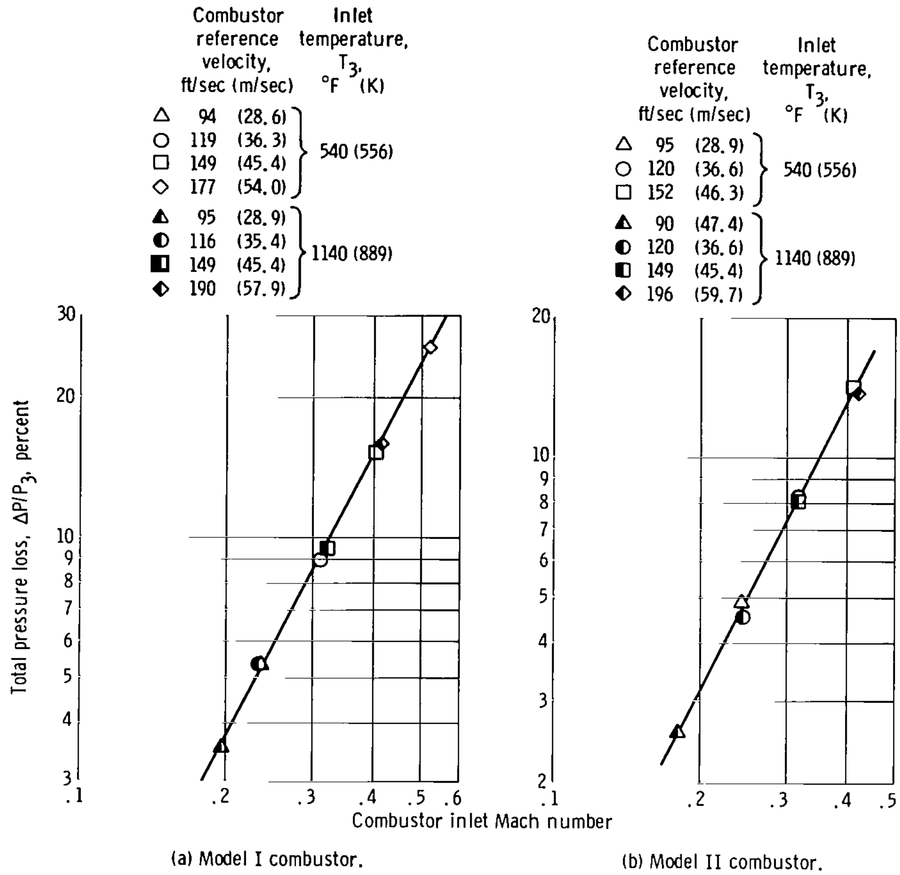


Figure 16. - Total pressure loss of final combustor configurations. Temperature rise in combustors, 1100 F° (610 K).

highest local temperature on any radius and the design temperature for that same radius, and T_{av} is the average temperature rise across the combustor.

$$\delta_{rotor} = \frac{\left(T_{R,av} - T_{R,design}\right)_{max}}{T_{av}}$$

where $\left(T_{R,av} - T_{R,design}\right)_{max}$ is the largest temperature difference between the average circumferential temperature at any radius and the design temperature for that same radius. The terms radial and circumferential are used as though the test section were a sector of an annulus. The design radial temperature profile for simulated sea-level takeoff, as well as that for cruise condition, is typical of those encountered in advanced supersonic engines (ref. 16). The shape of the radial profile is generally dictated by the requirements of the turbine stator and rotor. In addition to the factors

δ_{stator} and δ_{rotor}, another parameter, used in the aircraft industry and based only on maximum and average temperature rise, was employed. This parameter, the pattern factor is defined as follows:

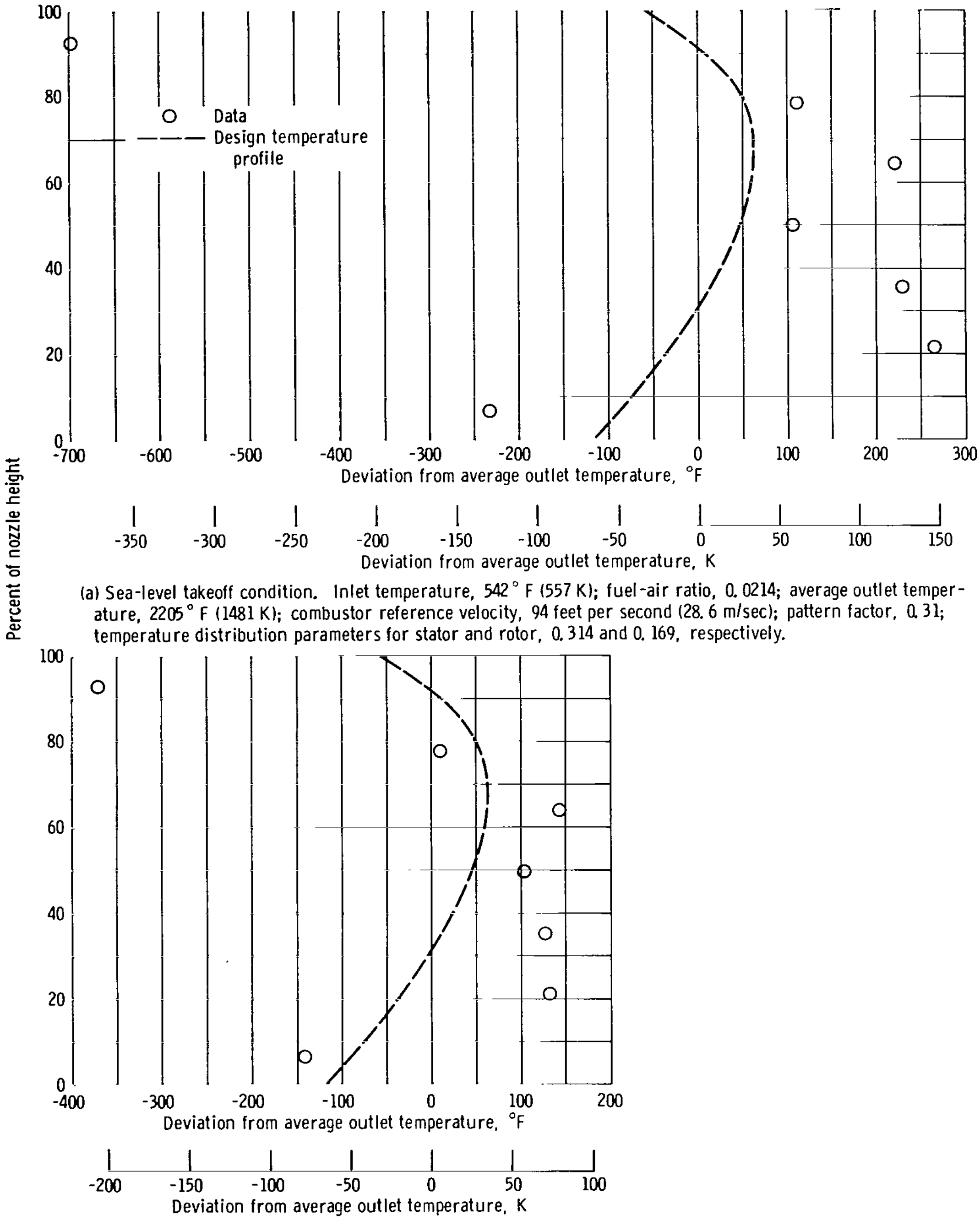
$$\bar{\delta} = \frac{\text{(Maximum local combustor outlet temperature)} - \text{(Average combustor outlet temperature)}}{\text{(Average combustor outlet temperature)} - \text{(Average combustor inlet temperature)}}$$

For the combustion efficiency calculations the combustor outlet temperatures were mass weighted and the average was based on the total number of readings taken in the survey. For the temperature profile calculations, the actual nonweighted temperatures were used; approximately 10 percent of the readings at each side were disregarded to eliminate sidewall effects which would not be present in a complete annular combustor.

Average exhaust temperature profiles in the radial and circumferential directions were determined at various fuel-air ratios and inlet temperature conditions.

The initial model I combustor configuration exhibited a center-peaked radial profile that was unacceptable. This condition was further aggravated by the use of perforated liners to reduce the acoustic instability. Moving the rows of swirl-cans farther apart made ignition difficult and did not help the instability. It was found that increased blockage in the heat release zone increased the efficiency of this combustor, smoothed out the peaks in temperature due to the individual cans, and reduced or eliminated the instability. However, the air flowing adjacent to and through the liners increased as combustor blockage increased and this caused high temperature gradients at the periphery of the combustor. Also, gaps in the cooling liner, caused by thermal distortion, increased the open area along the periphery of the liner and allowed more air to flow along the liners, accentuating the thermal gradients at the periphery. The fuel-flow to the two middle rows of cans was reduced in an attempt to raise the temperature on the top and bottom. The radial profile for the final model I combustor is shown in figure 17 for the sea-level takeoff and Mach 3 cruise conditions. The radially-averaged circumferential profile for the final model I combustor is shown in figure 18.

Subsequent to obtaining the final data for the model I combustor, an attempt was made to decrease the excess air along the top and bottom of the combustor by adding blockage and by welding shut some of the film cooling slots. The results (fig. 19) show that sufficient blockage was provided on the top of the combustor to give a good radial profile in that region and that only slightly more blockage was needed on the bottom to make the profile close to the design temperature profile. However, these small changes increased the combustor pressure loss from 9.0 to 9.8 percent at an inlet Mach number of 0.311. Since additional blockage would further raise the pressure loss, the program was not pursued. It is felt that the combustor could be redesigned to give a good temperature profile with an acceptable pressure loss.



(a) Sea-level takeoff condition. Inlet temperature, 542° F (557 K); fuel-air ratio, 0.0214; average outlet temperature, 2205° F (1481 K); combustor reference velocity, 94 feet per second (28.6 m/sec); pattern factor, 0.31; temperature distribution parameters for stator and rotor, 0.314 and 0.169, respectively.

(b) Mach 3 cruise condition. Inlet temperature, 1116° F (875 K); fuel-air ratio, 0.0171; average outlet temperature, 2372° F (1573 K); combustor reference velocity, 95 feet per second (28.9 m/sec); pattern factor, 0.32; temperature distribution parameters for stator and rotor, 0.335 and 0.118, respectively.

Figure 17. - Radial profile for final model I combustor. Inlet pressure, 45 psia (31 N/cm^2).

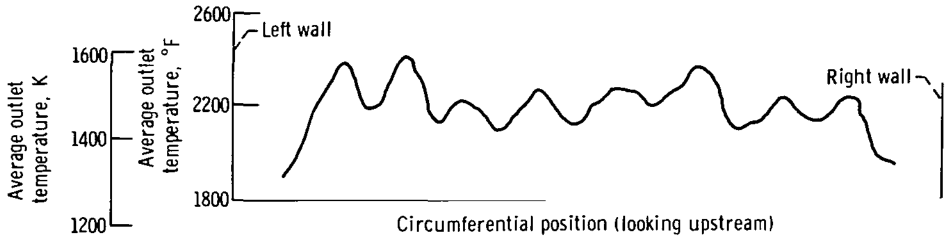


Figure 18. - Circumferential profile for final model I combustor. Inlet temperature, 542° F (557 K); inlet pressure, 45 psia (31 N/cm^2); fuel-air ratio, 0.0214; average outlet temperature, 2205° F (1481 K); combustor reference velocity, 94 feet per second (28.6 m/sec); pattern factor, 0.31; temperature distribution parameters for stator and rotor, 0.314 and 0.169, respectively.

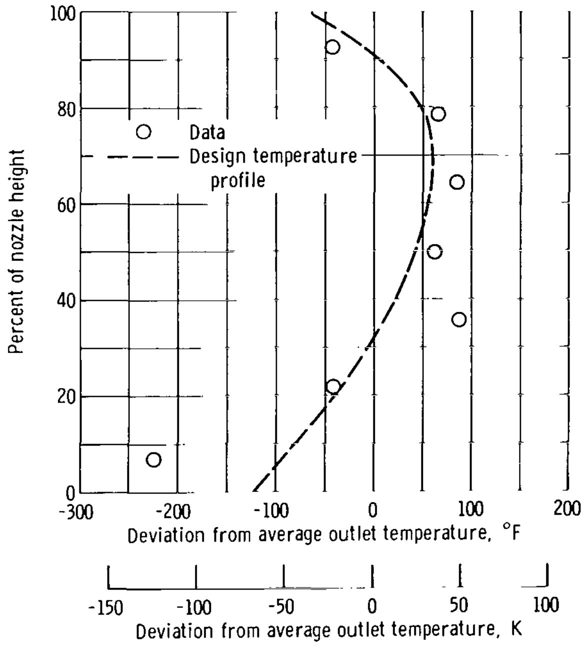


Figure 19. - Radial profile of model I combustor with peripheral blockage. Inlet temperature, 552° F (562 K); inlet temperature, 45 psia (31 N/cm^2); fuel-air ratio, 0.0182; average outlet temperature, 1882° F (1301 K); combustor reference velocity, 95 feet per second (28.9 m/sec); pattern factor, 0.40.

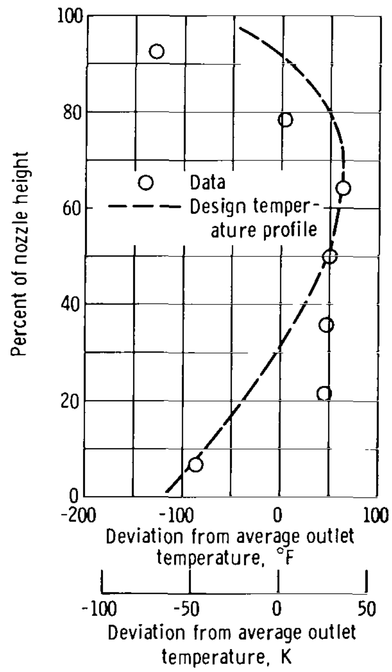


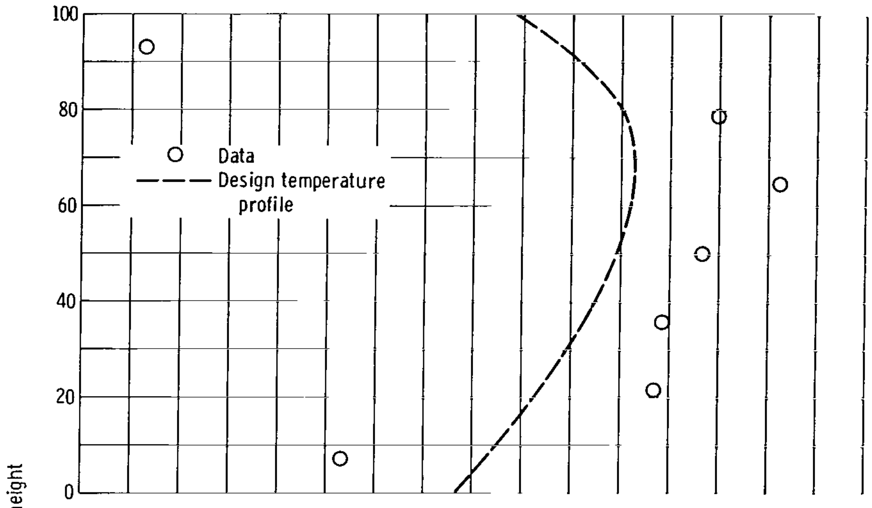
Figure 20. - Radial profile for model II combustor with controlled liner air. Inlet temperature, 542° F (557 K); inlet pressure, 45 psia (31 N/cm^2); fuel-air ratio, 0.0226; average outlet temperature, 2262° F (1512 K); combustor reference velocity, 120 feet per second (36.6 m/sec); pattern factor, 0.19.

The initial model II combustor configuration exhibited good profile characteristics in the radial and circumferential directions. Because of the acoustic instability and poor ignition characteristics of this combustor, crosses were added between modules. The resulting radial profile of the final configuration was very close to the design at the sea-level takeoff condition (fig. 20). Subsequent tests have not been able to reproduce these results, probably because the combustor liners have deteriorated to the extent that they allow excessive amounts of liner cooling air into the combustor along the periphery. As explained previously, excess cool air along the periphery causes steep temperature gradients in this region. In recent tests model II combustor exhibited steep temperature gradients on the top and bottom of the radial profile as seen in figure 21.

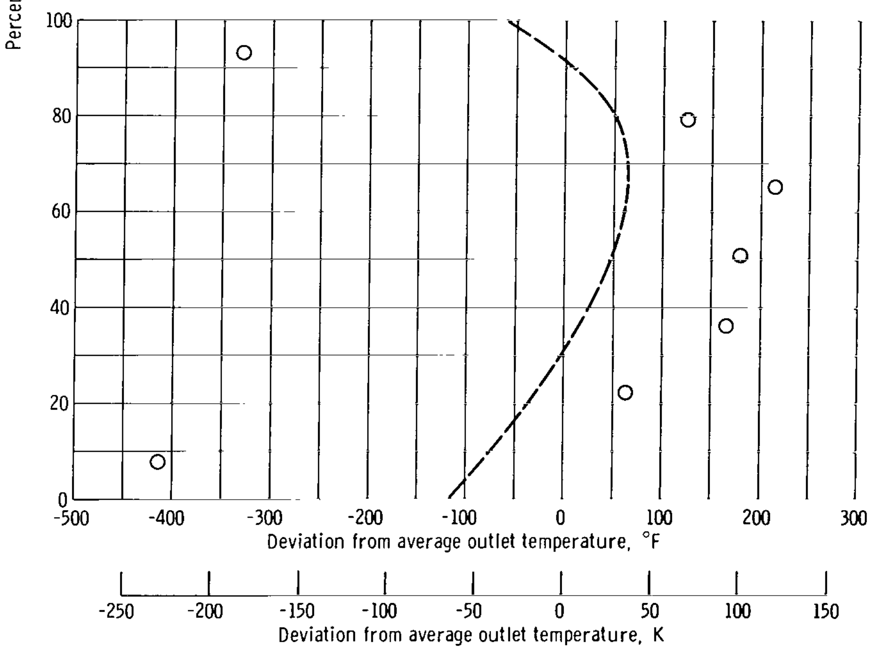
The circumferential profile of the model II combustor was fairly uniform except at the side walls. Initial test exhibited high temperatures near the sides, believed to be due to side-wall stall in the diffuser section. By blocking half the fuel flow to the end cans, the hot spots were reduced to below the average combustor exit temperature. The final radially averaged circumferential profile is shown in figure 22. We believe that the cans were too large to give good coverage of the cross-sectional area without encountering excessive end effects at the sides as well as the top and bottom.

These combustors exhibited the following values of the temperature distribution parameters at the indicated conditions for the final configurations:

Model	Nominal inlet temperature		Fuel-air ratio	Reference velocity		Pattern factor, $\bar{\delta}$	Temperature distribution parameters	
	^{o}F	K		ft/sec	m/sec		δ_{stator}	δ_{rotor}
I	540	556	0.0214	94	28.7	0.31	0.314	0.169
	1140	889	.0171	95	29.0	.32	.335	.118
II	540	556	0.0196	95	28.9	0.325	0.311	0.086
	1140	889	.0185	91	27.8	.37	.386	.119



(a) Sea-level takeoff condition. Inlet temperature, 534° F (552 K); fuel-air ratio, 0.0196; average outlet temperature, 1535° F (1108 K); combustor reference velocity, 95 feet per second (28.9 m/sec); pattern factor, 0.325; temperature distribution parameters for stator and rotor, 0.311 and 0.0860, respectively.



(b) Mach 3 cruise condition. Inlet temperature, 1149° F (784 K); fuel-air ratio, 0.0185; average outlet temperature, 2434° F (1608 K); combustor reference velocity, 91 feet per second (27.8 m/sec); pattern factor, 0.37; temperature distribution parameters for stator and rotor, 0.386 and 0.119, respectively.

Figure 21. - Radial profile of final model II combustor with excess liner air. Inlet pressure, 45 psia (31 N/cm^2).

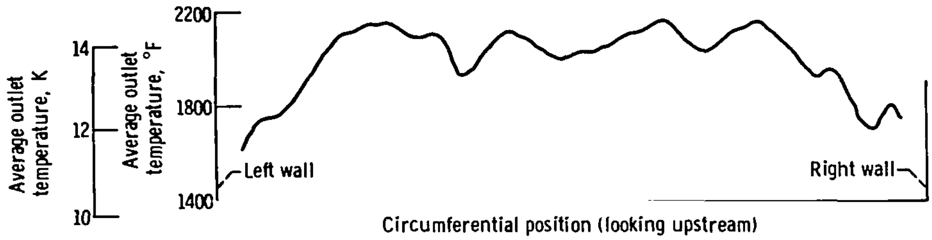


Figure 22. - Circumferential profile for final model II combustor. Inlet temperature, 534° F (552 K); inlet pressure, 45 psia (31 N/cm^2); fuel-air ratio, 0.0196; average outlet temperature, 1535° F (1108 K); combustor reference velocity, 95 feet per second (28.9 m/sec); pattern factor, 0.325; temperature distribution parameters for stator and rotor, 0.311 and 0.0860, respectively.

SUMMARY OF RESULTS

Two modular combustor designs were tested with natural gas over a range of operating conditions. The following results were obtained:

1. At an inlet air temperature of 1140° F (889 K) the combustion efficiency of both models was high over a range of fuel-air ratio. At an inlet temperature of 540° F (556 K) the efficiency of the model I combustor remained high over a range of fuel-air ratios but the efficiency of the model II combustor dropped off rapidly with decreasing fuel-air ratio.

2. Blowout and ignition characteristics of both combustor models were poor. A modular combustor, similar in design principle, but using JP-fuel, was reported in reference 13 to have much better blowout and ignition performance. It appears that the areas of ignition and low pressure and temperature performance will require special design attention for natural gas-fueled combustors.

3. Acoustical instabilities were encountered that were only partially corrected by the use of absorbing liners. The instabilities disappeared entirely when the blockage at the module exit plane was increased sufficiently.

4. The use of vitiated air was found to have a very important adverse effect on combustion efficiency at the 540° F (556 K) inlet air temperature condition.

5. The pressure loss of the model I combustor varied from 5.9 percent at an inlet Mach number of 0.25 to 8.5 percent at a Mach number of 0.30. The pressure loss of the model II combustor was 5.0 and 7.2 percent at these Mach numbers.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, November 4, 1969,
720-03.

APPENDIX - INSTRUMENTATION

Airflow and gaseous fuel flow rates were measured by square-edged orifices installed according to ASME specifications. Liquid fuel-flow was measured by a turbine-type flow-meter.

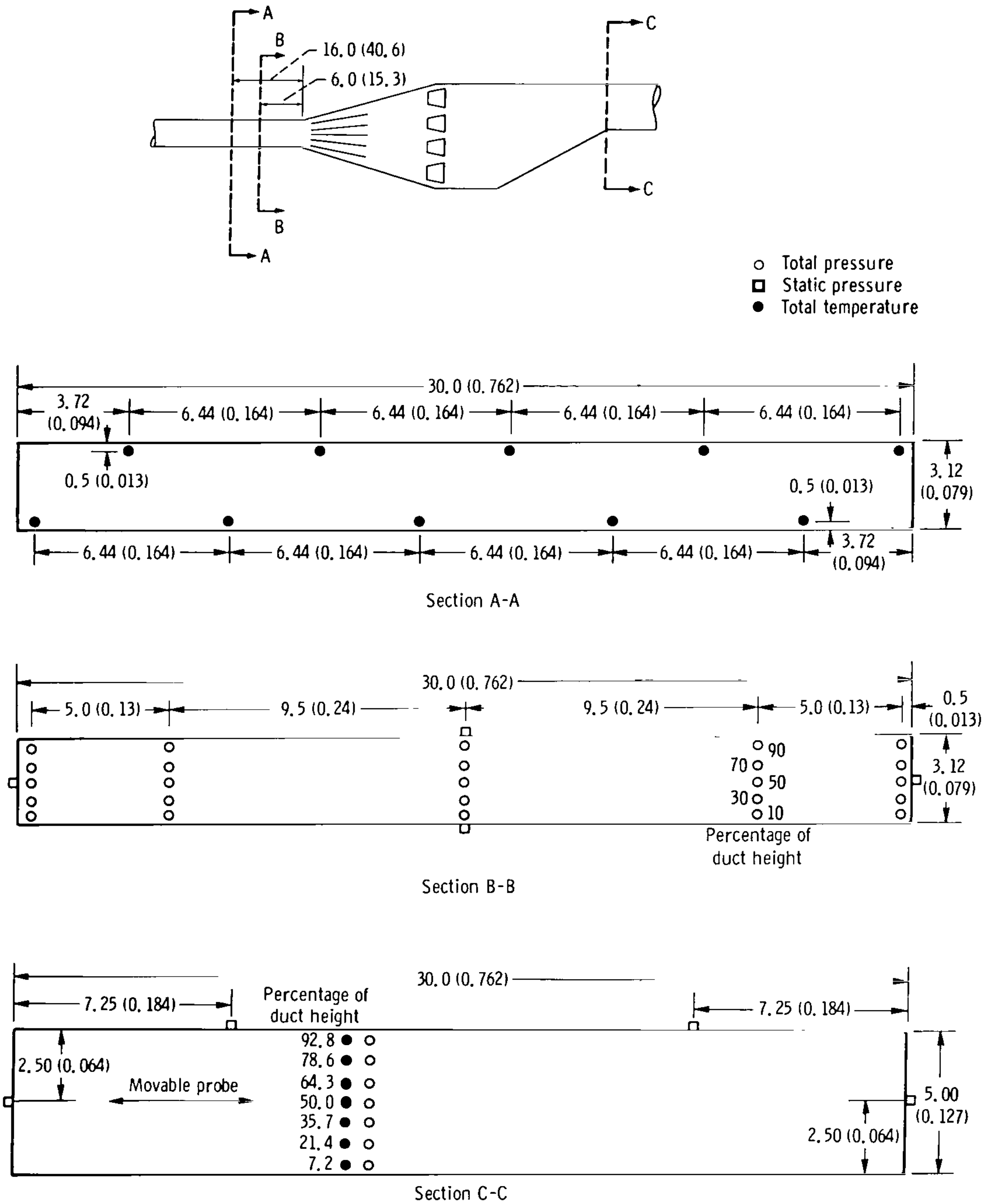


Figure 23. - Location of temperature and pressure probes in instrumentation planes. (All dimensions are in inches (m).)

The location of the pertinent instrumentation planes is shown in figure 23; the arrangement of the pressure and temperature probes is shown in figure 23. Temperatures (in the inlet section) were measured by 10 Chromel-Alumel thermocouples (section A-A, fig. 23). Pressures were measured by means of five rakes, each consisting of five-point total-pressure tubes, and by four wall static-pressure taps (section B-B, fig. 23). Combustor-outlet total pressures and temperatures were recorded by means of a movable seven-point total-pressure and seven-point total-temperature rake (section C-C, fig. 23). The temperature probes were constructed of platinum - 13-percent rhodium - platinum and were of the high-recovery aspirating type (type 6 of ref. 17). The average reading of four static-pressure taps located as shown in figure 23 was used as a measure of the static pressure at the combustor exhaust nozzle. The exhaust rake is shown in figure 24.

Temperature and pressure surveys at the combustor exit were made by moving the probe horizontally across the exhaust nozzle at a speed which produced approximately one reading every 1/2 inch (0.0127 m). A periscope mounted downstream of the exhaust nozzle provided a view of the combustor modules.

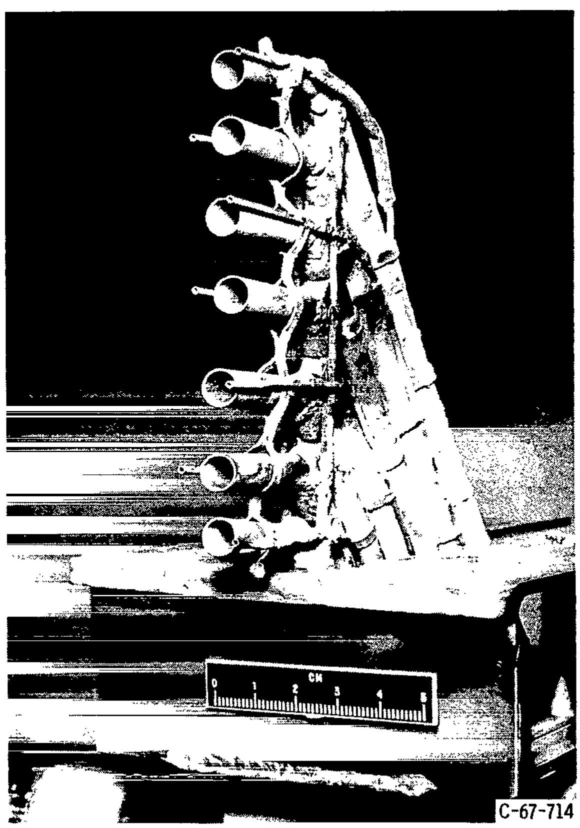


Figure 24. - Exhaust rake.

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