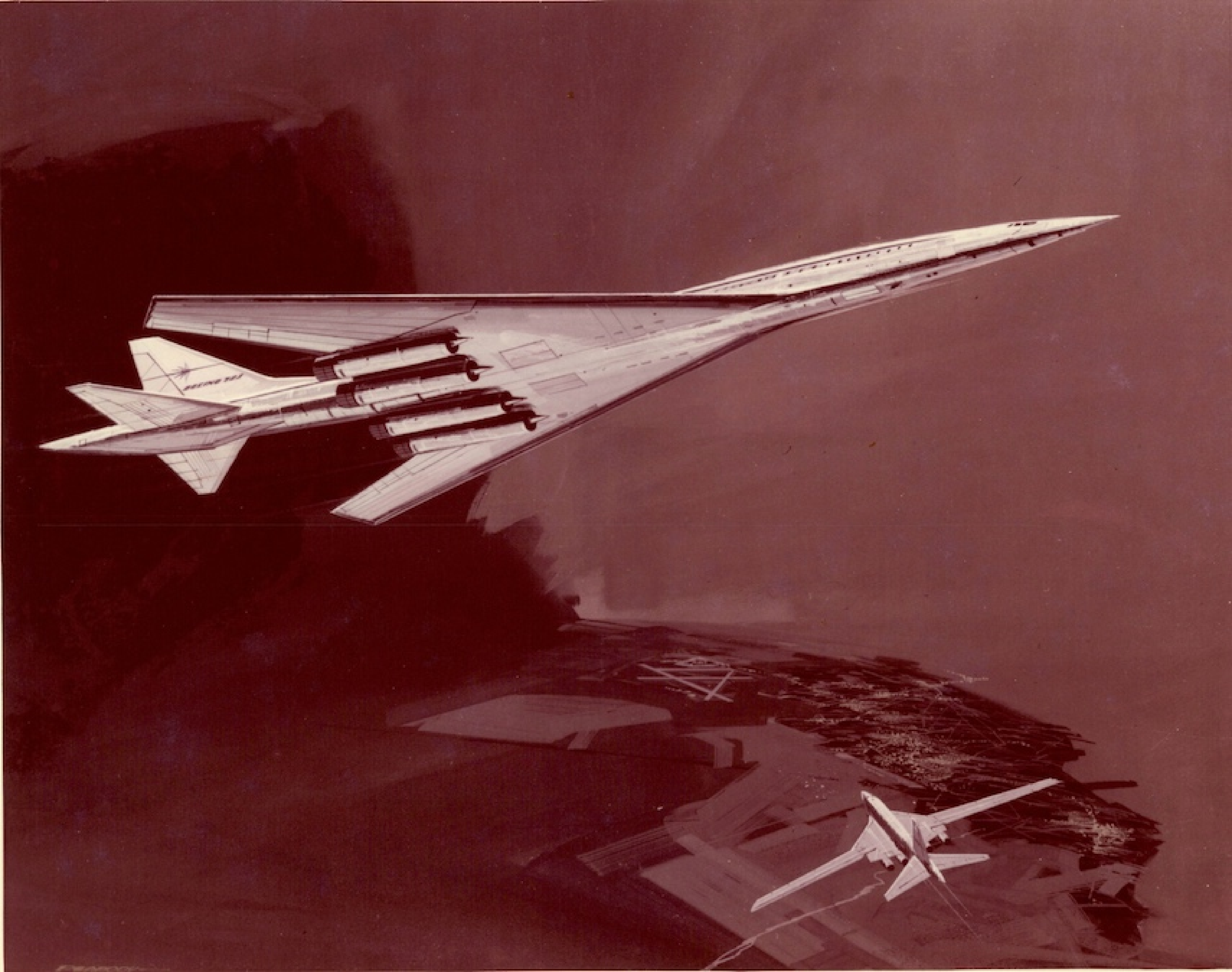
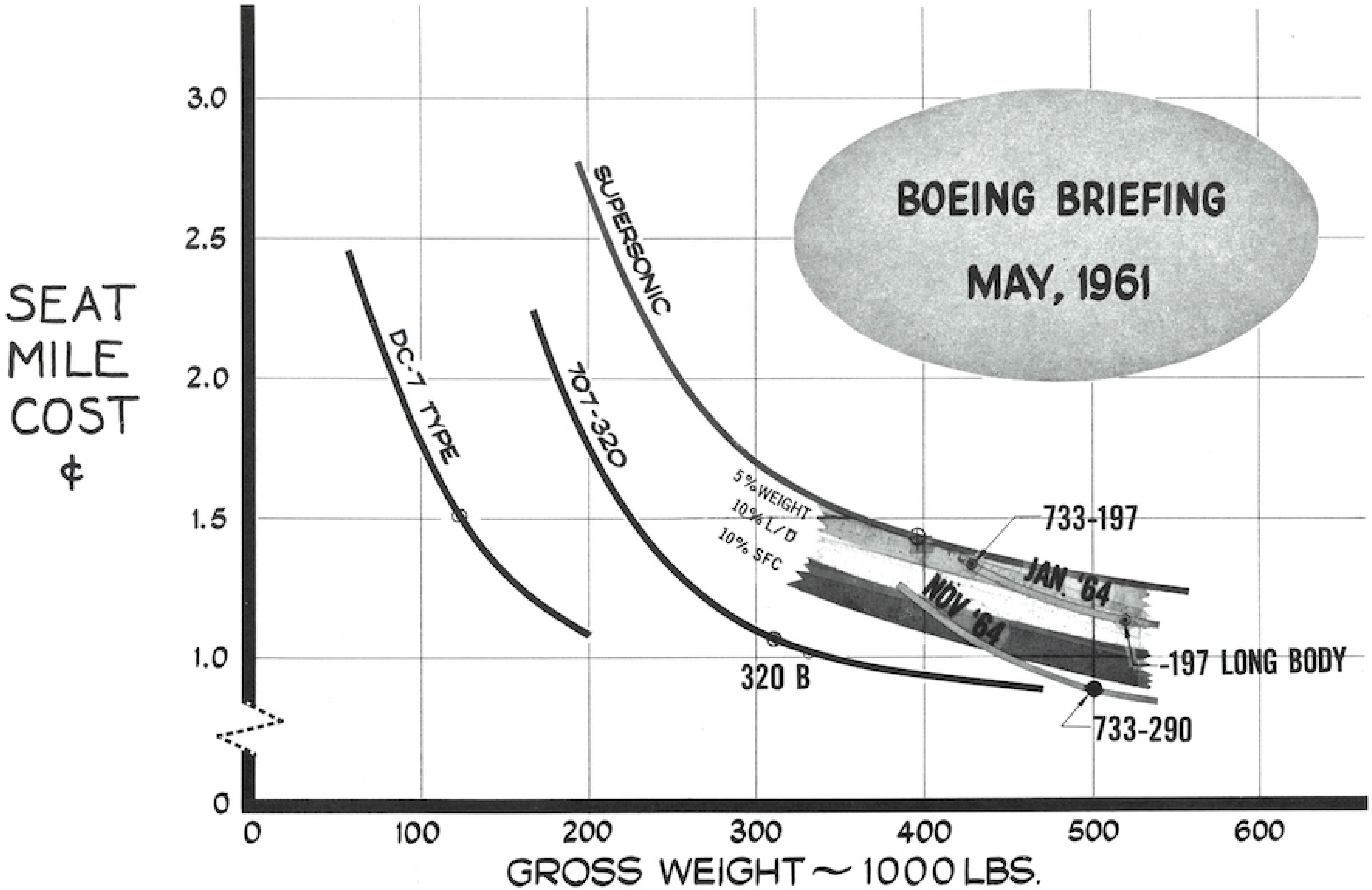
BOEING SST

MODEL 733

CIVIL AERONAUTICS BOARD – MARCH 9, 1965



OPERATING COST COMPARISON

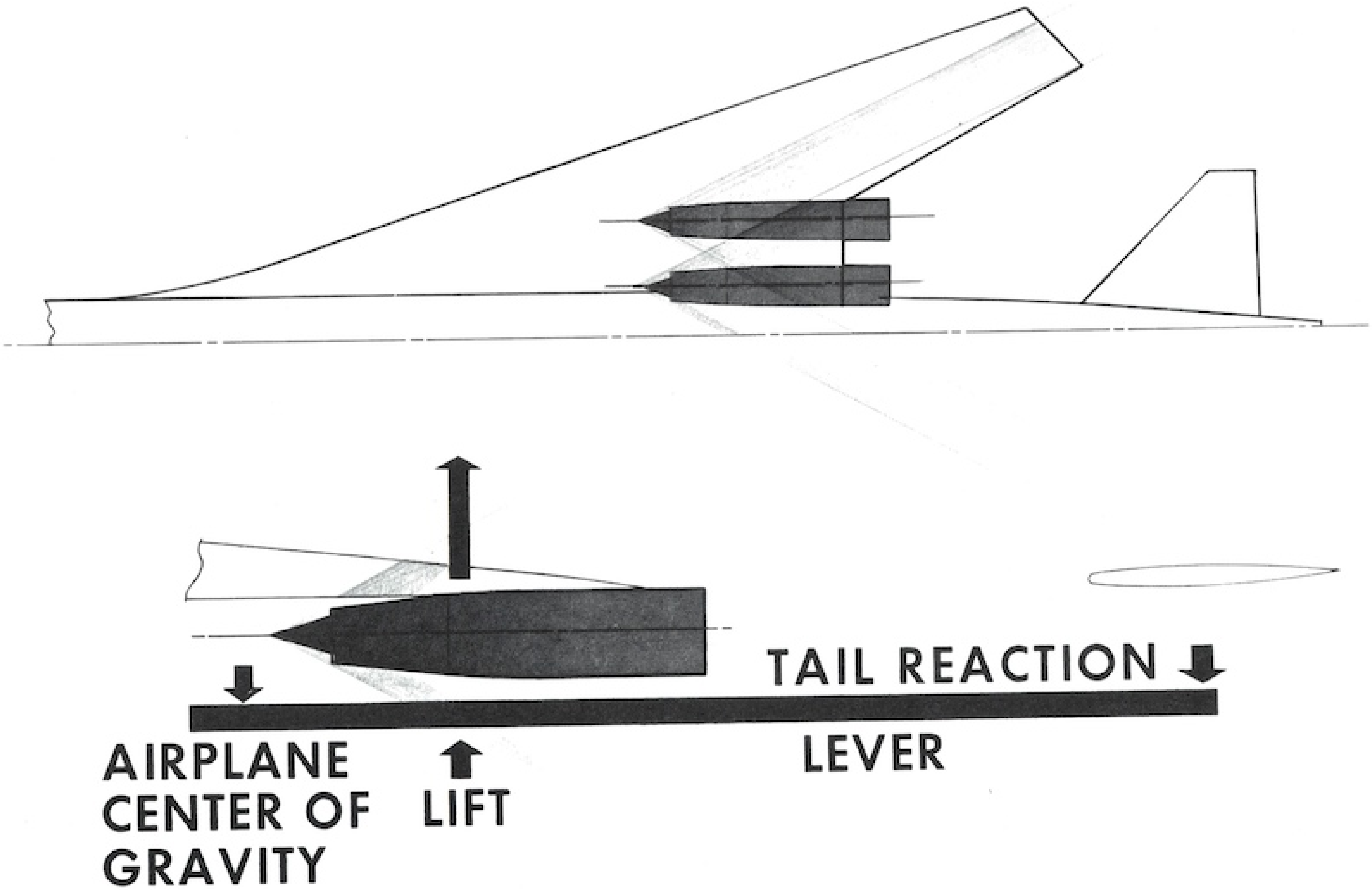


CE 6595 5-65

Extensive wind tunnel testing has shown that nacelles placed at the rear of the twisted and cambered arrow wing produces a favorable lift interference on the wing. If this wing is properly shaped, this lift interference will also produce an overall reduction in the wave drag of the wing nacelle combination. Boeing has developed extensive analytical and experimental methods to optimize location of the engine nacelles on the wing-body combination.

CE 6575

NACELLE INTEGRATION

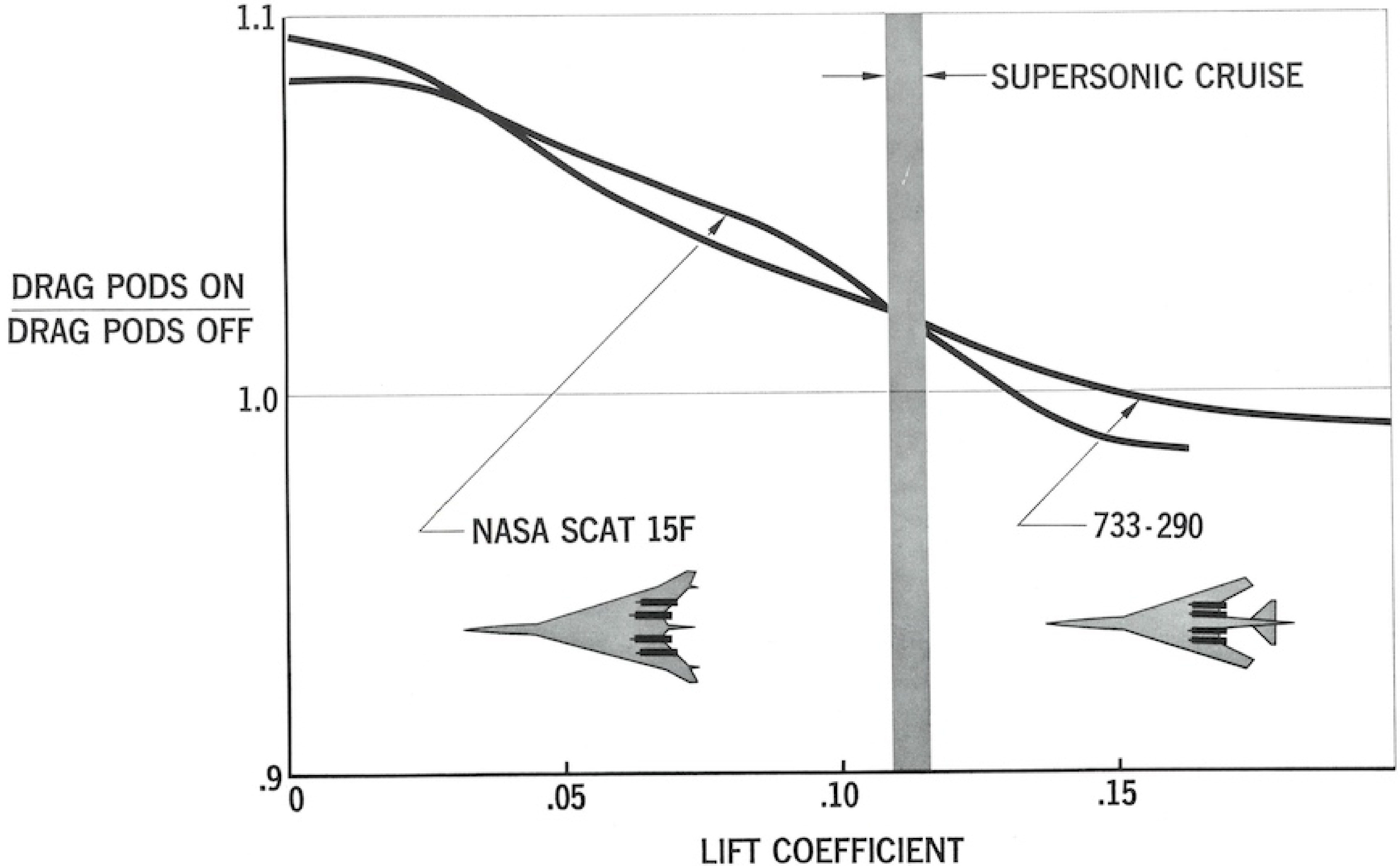


CE6575 11-64

This chart shows a comparison of the drag of the total wing body combination with pods on and pods off. A comparison between the Boeing supersonic transport model and the NASA SCAT 15 aerodynamic configuration shows that the four engine pods have been added to both configurations for practically no drag.

CE 6506

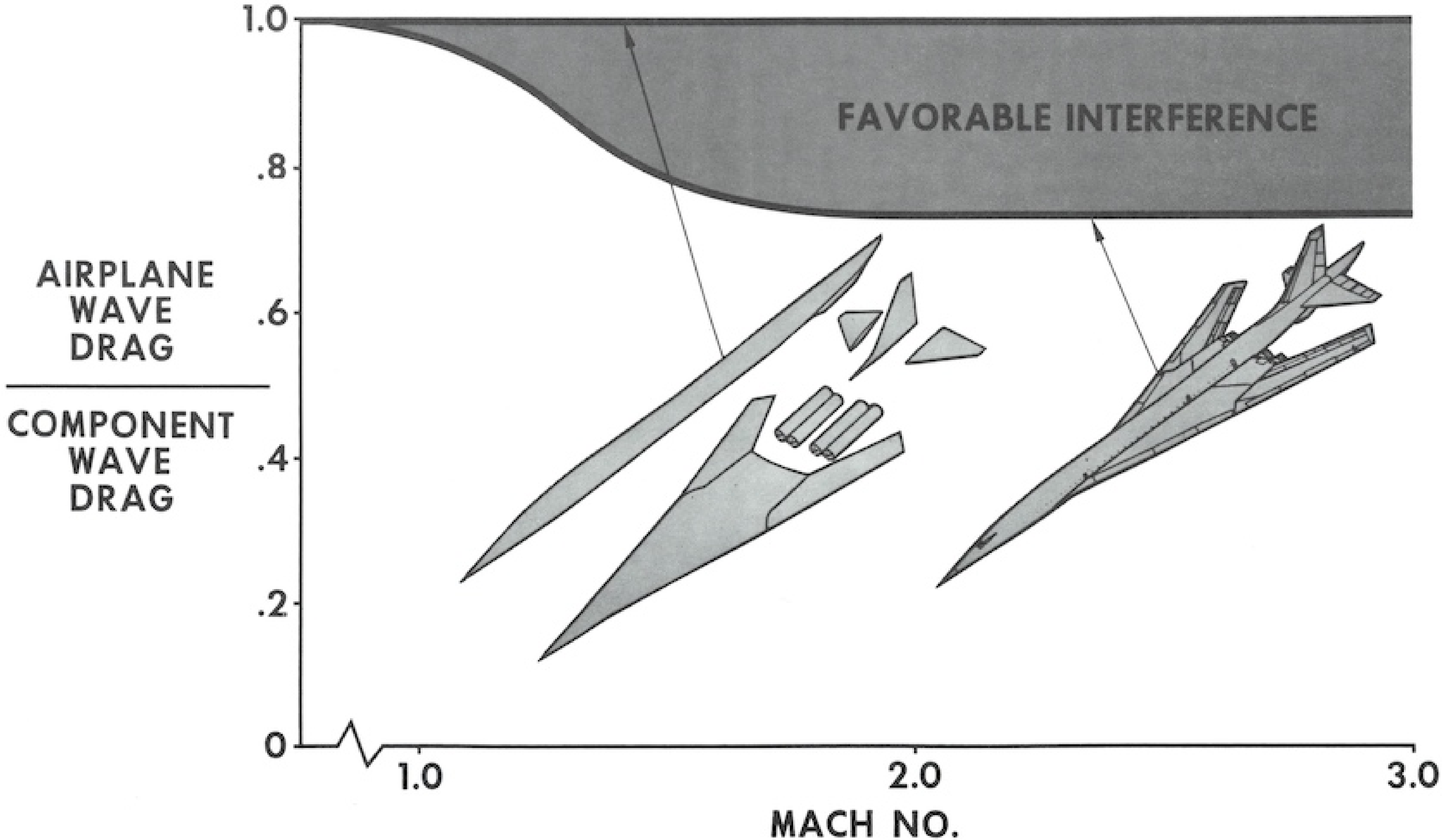
NACELLE FAVORABLE INTERFERENCE



This chart shows the total airplane wave drag divided by the wave drag of the individual components as a function of Mach number. The data indicate that the Boeing configuration has been integrated to such a degree that the airplane wave drag is only 75% of the wave drag of the individual pieces. This represents a significant area of favorable interference for the arrow wing configuration.

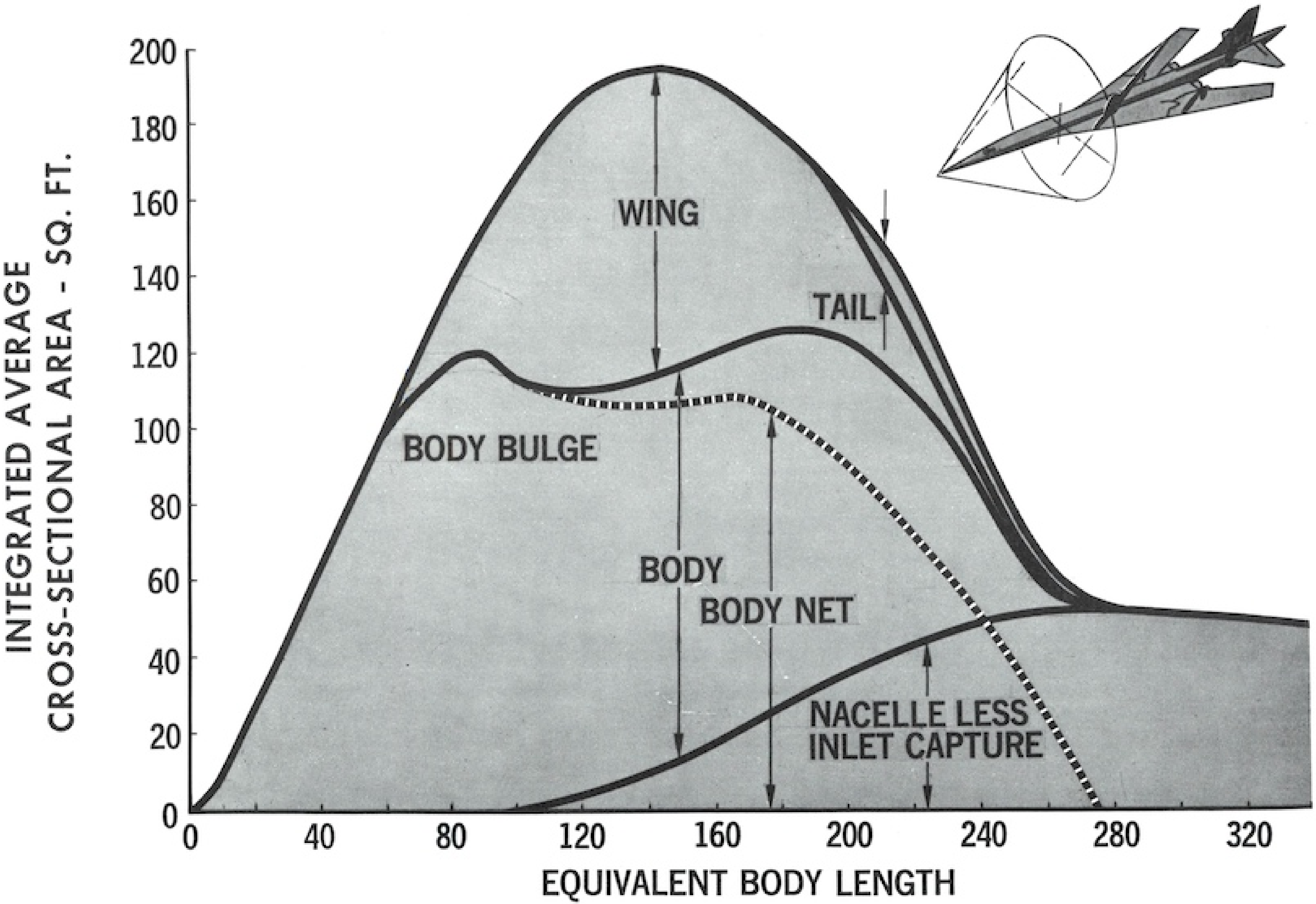
CE 6515

COMPONENT INTEGRATION



Total cross sectional area of the configuration is shown as a function of body length. The important point of this plot is that the wing and nacelles have been added to the body in such a manner that the distribution of area with length is very smooth. The slope of the forebody and the aft body are low and the total cross sectional area is low. Wave drag is a function of the slopes times the maximum cross sectional area and is quite low for the configuration. This chart also explains why the Boeing supersonic transport configuration is bulged ahead of the wing. The bulge smoothes the area plot and it has a beneficial effect on the sonic boom signature of the configuration.

CROSS-SECTION DISTRIBUTION AT MACH 2.7

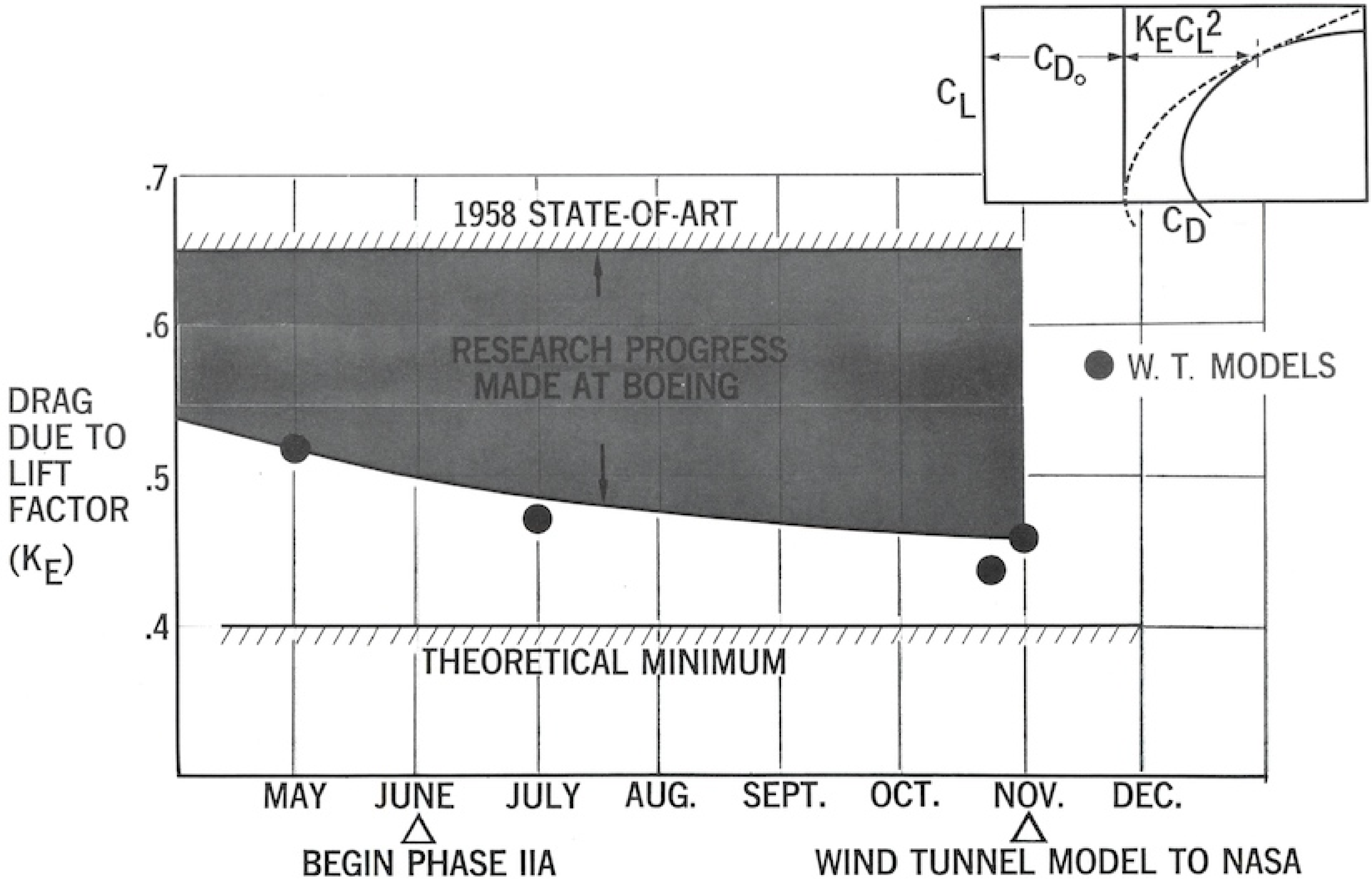


CE6551 11-64

Boeing recognized in 1958 that the twisted and cambered arrow wing had a great deal of potential for an efficient supersonic transport. The drag due to lift factor K_e is shown here as a function of time. Drag due to lift is shown on the plot of drag coefficient vs. lift coefficient in the upper right hand corner and is merely the drag that is associated with the airplane flying at an angle of attack. The theory for the arrow wing indicated a very low value of K_e was possible.

Early test data as represented by the 1958 state-of-the-art line were only slightly better than a flat wing. Through analytical techniques and the wind tunnel testing of sixty individual wings and many modifications on each of these, Boeing has reduced the drag due to lift for the twisted and cambered arrow wing nearly to the theoretical minimum available.

DRAG DUE TO LIFT PROGRESS

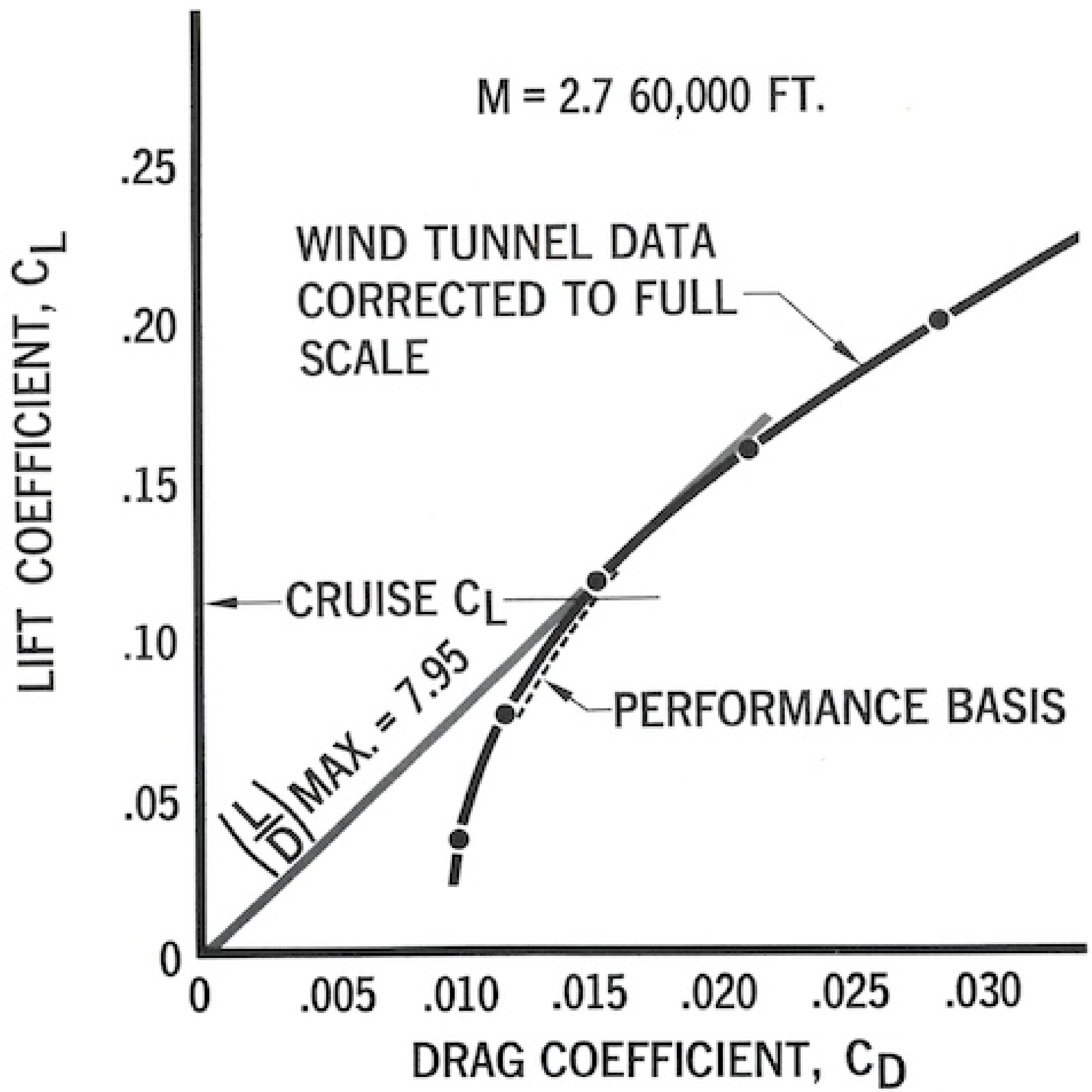


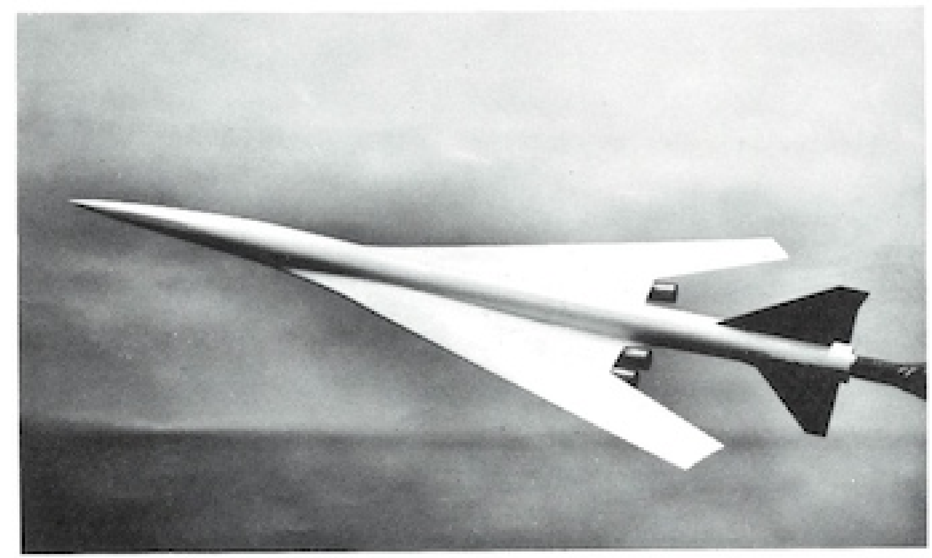
CE6522 11-64

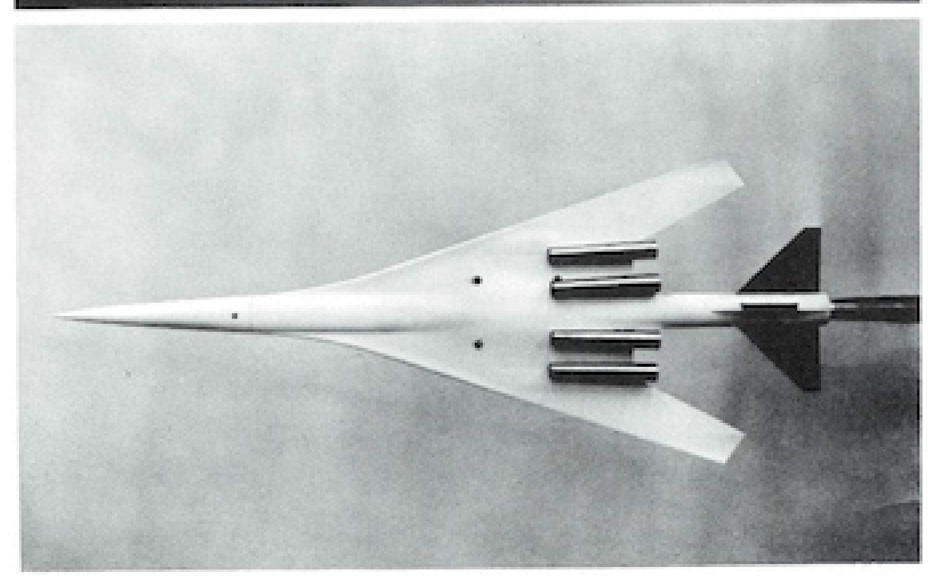
Lift coefficient is plotted against drag coefficient for Boeing's supersonic transport wind tunnel model. The data indicate that we have achieved a maximum lift to drag ratio of 7.95. It is important to note that the performance which has been quoted today is based on the dashed line or a lift to drag ratio of 7.82. The NASA wind tunnel tests in November, 1964 of this configuration validated a maximum L/D of 8.2.

In summary we have shown that the wave drag of a cambered and twisted arrow wing can be very low, that the drag due to lift is nearly the minimum available, and since the friction drag of an arrow wing transport and any delta wing transport would be nearly the same, the arrow wing approach will yield the highest supersonic efficiency. The addition of variable sweep to the arrow wing allows the supersonic transport to achieve this high level of supersonic efficiency and yet take off and land with less noise, lower speeds, and shorter field lengths than the present subsonic jets.

L/D SUBSTANTIATION





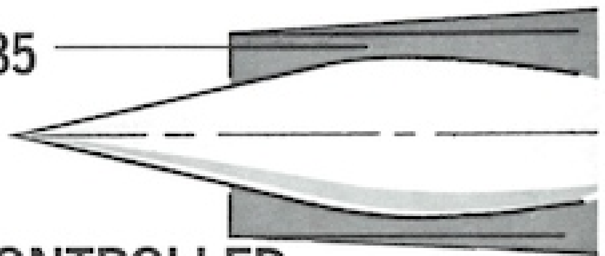


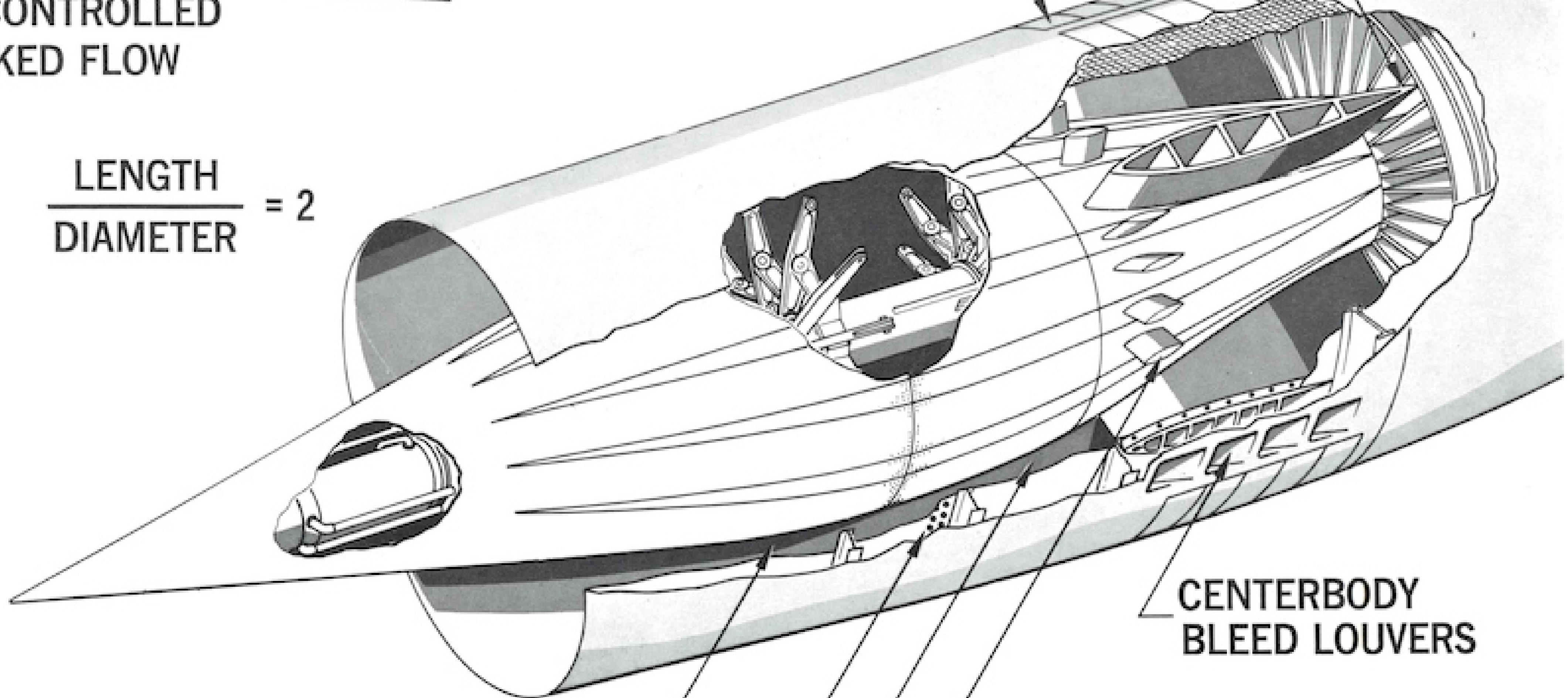
The supersonic inlet does most of the compression work at Mach 2.7. To make it efficient at lower supersonic speeds, the throat area must vary.

We vary the throat by changing the diameter of the centerbody. At subsonic speeds during approach to the airport, the diameter can be expanded and allow the inlet air to be accelerated to nearly the speed of sound at the throat section and then slowed down to enter the compressor at the proper speed. Compressor noise tries to get out the inlet but since air is coming in at the speed of sound, the noise is trapped in the inlet.

CE-6555

BOEING SST INLET DESIGN



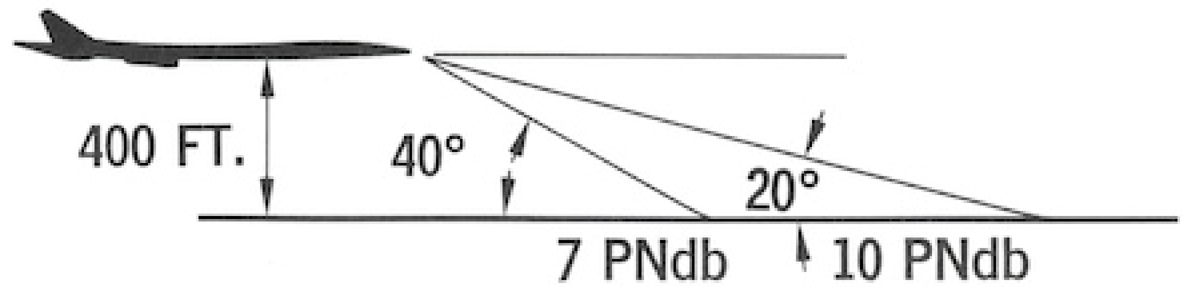


The test you will witness is of a J-75 jet engine with a simulated inlet duplicating the approach condition of the SST. The bellmouth allows simulation of approach speed flow conditions at the lip on the static test rig. No bellmouth will be needed on the actual airplane.

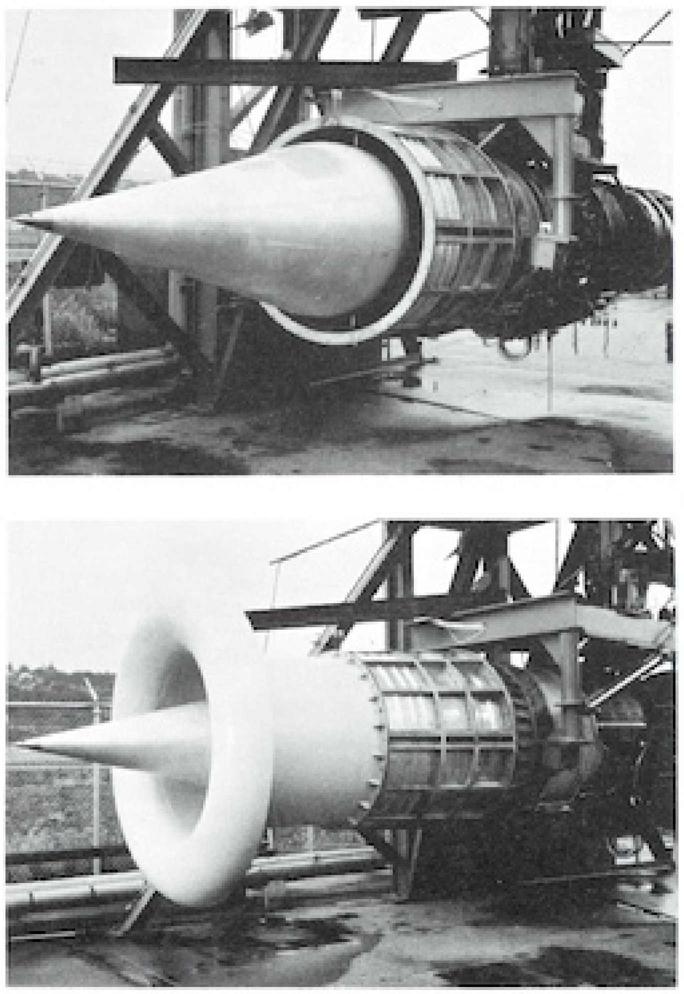
The engine is started and warmed up at idle condition, then the power raised to approximately 25 percent at which time the velocity at the inlet throat is approxi-mately sonic. This is about the power setting required for approach condition on the SST.

Microphones placed ahead of the inlet have measured sound pressure levels against frequency in cycles per second. At lower than 25 percent power, the air flow is unchoked and you hear the compressor noise quite plainly. As the power is raised to the choked condition, compressor noise will drop as shown on the chart. The effective noise suppression from choking the inlet is illustrated by the airplane approach at 400 feet altitude. When the airplane is far away at an angle of about 20 degrees above the ground observer, the suppression would be about 10 PNdb. After the airplane approaches closer, the suppression would be about 7 PNdb.

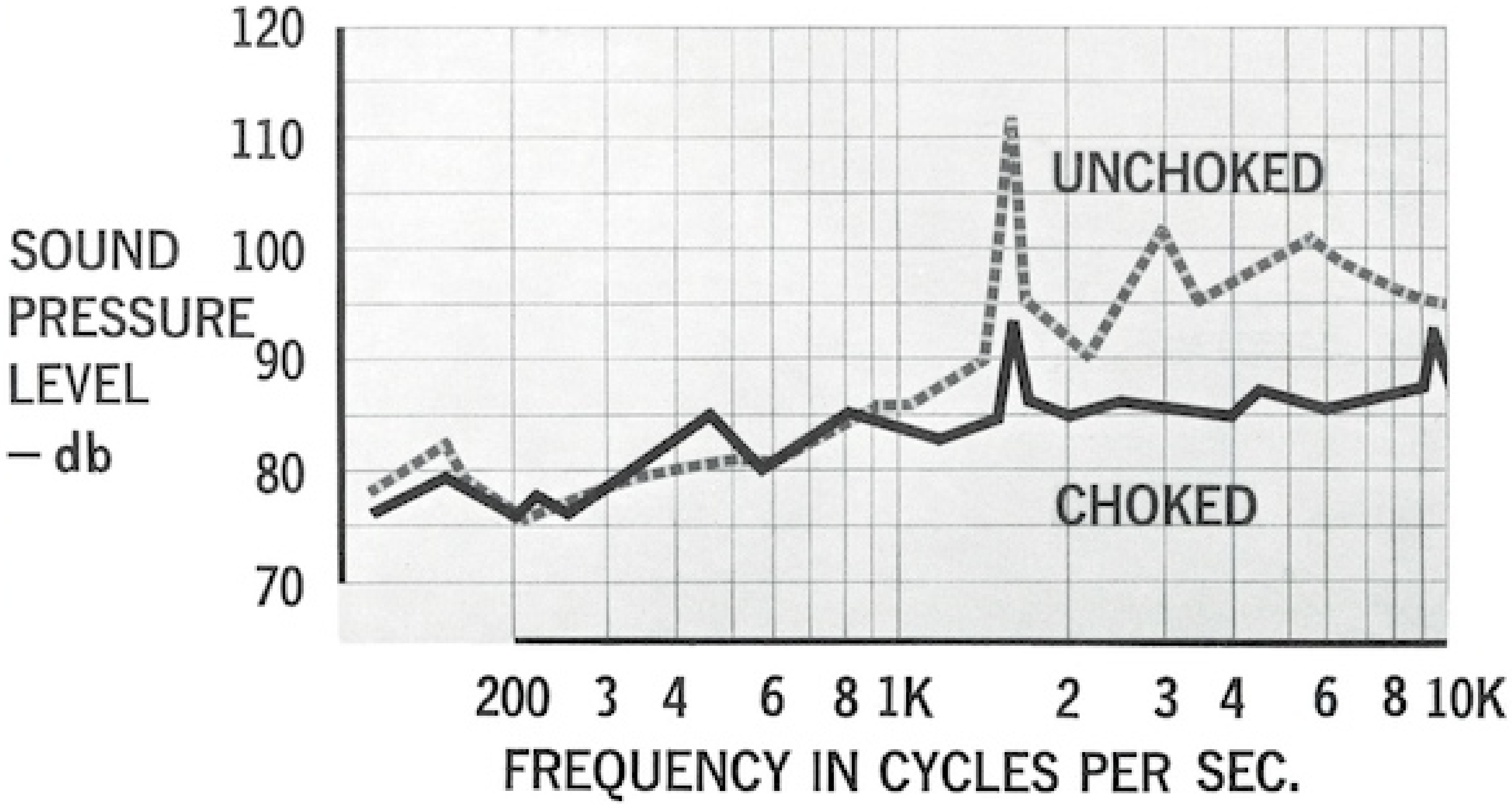
EFFECT OF INLET CHOKING ON APPROACH NOISE



NOISE SUPPRESSION FROM CHOKED INLET

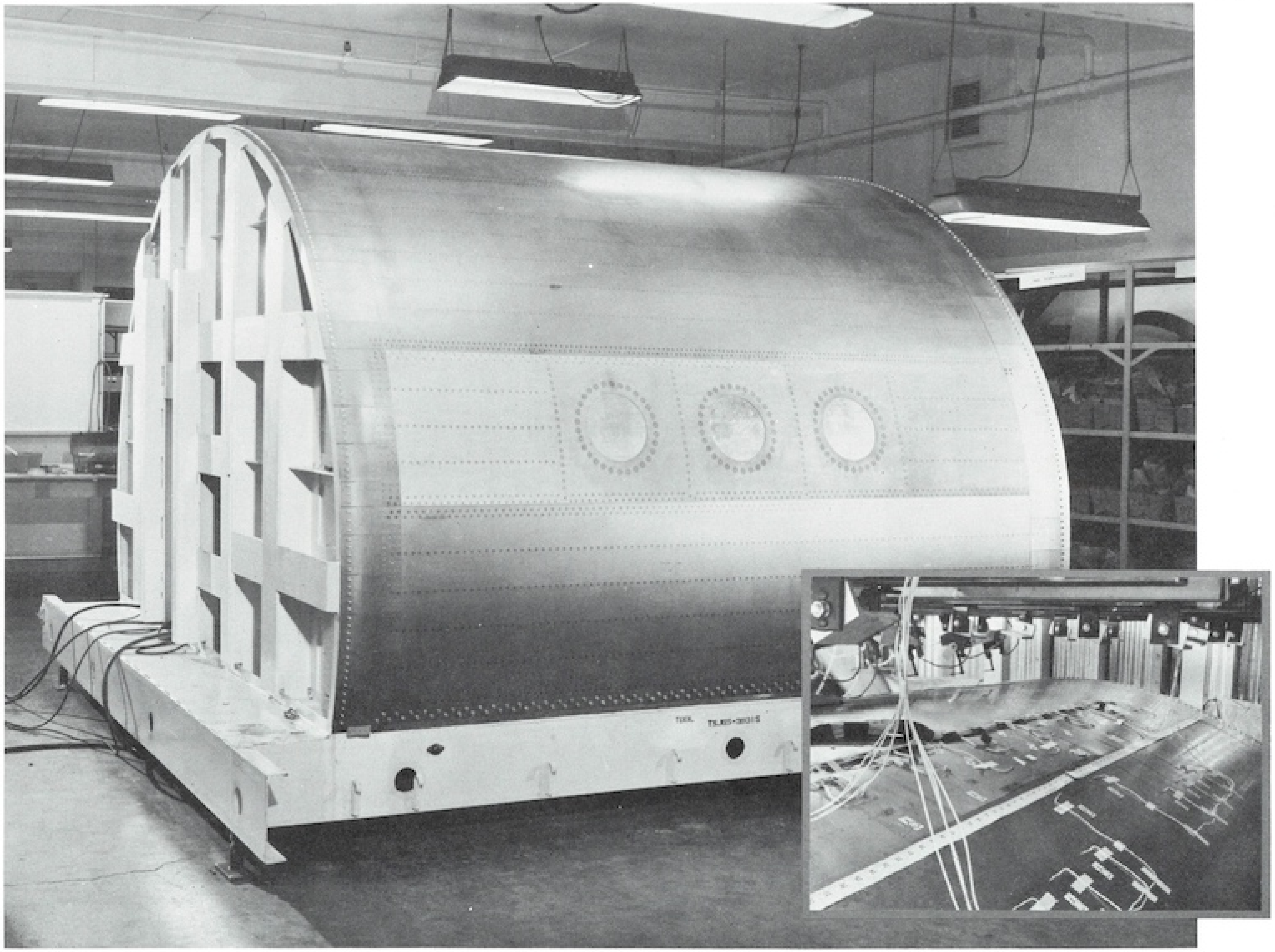


J-75 TEST RIG



CE6556 11-64

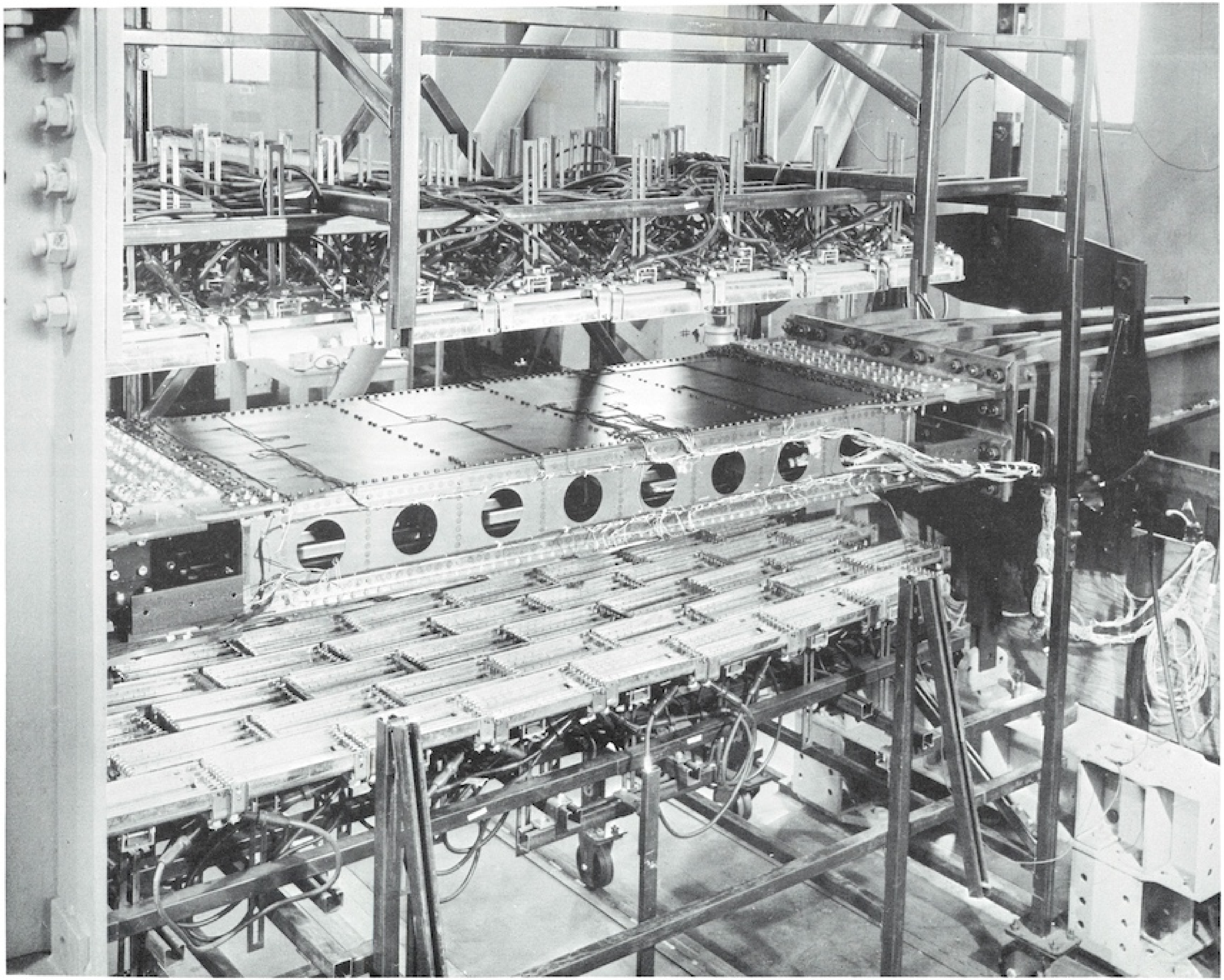
MODEL 733 TITANIUM BODY TEST SECTION



CE6437 12-64

Thermal Box Test

The structure is typical construction for the empennage area or the outer part of the wing and is subjected to an elevated temperature of 500°F while receiving a bending load. This test will provide information on the distribution of stresses under high temperature and load conditions, also panel deflection under these conditions will be determined.

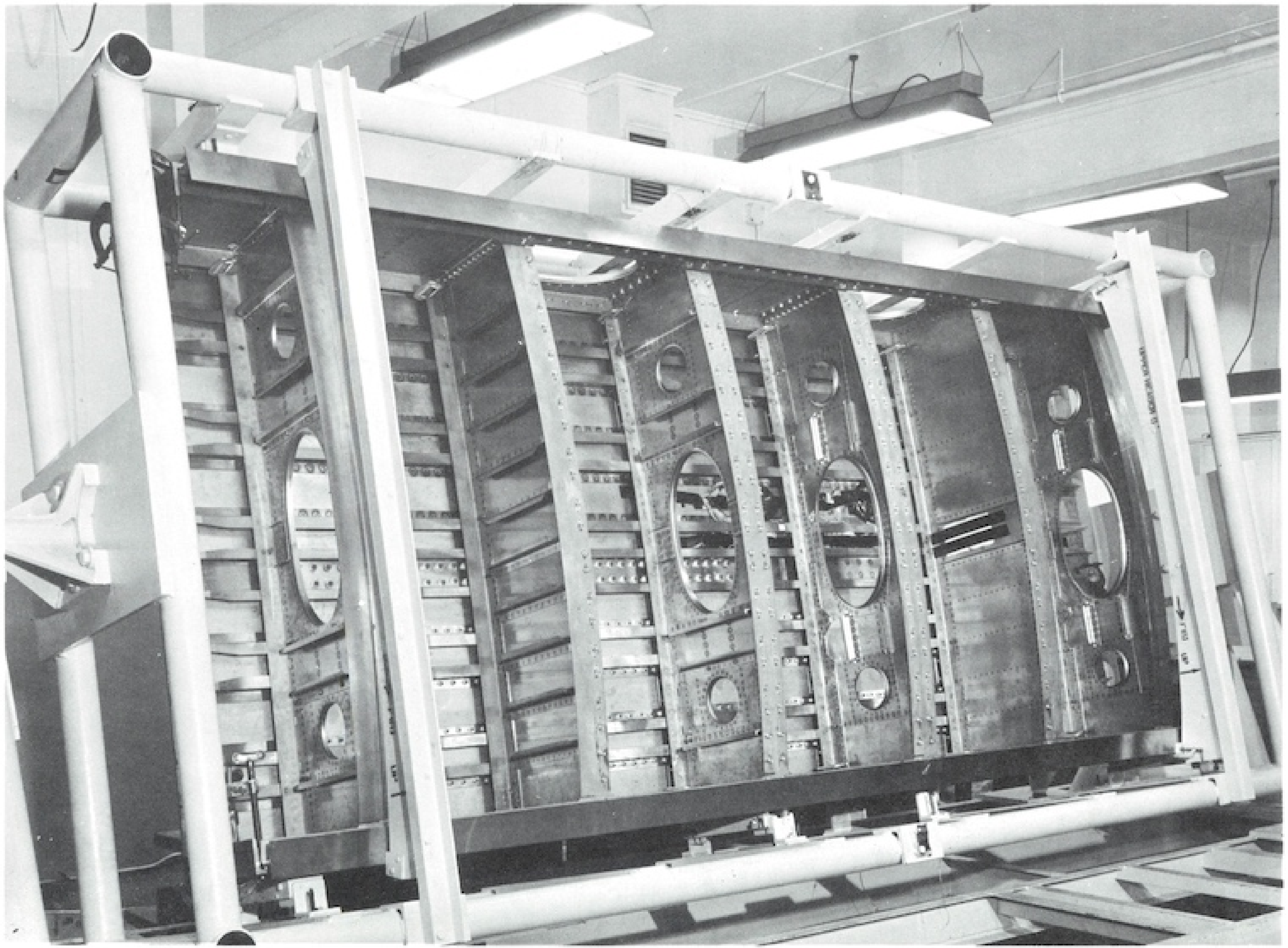


Wing Box Test Structure

An interior view is shown of our wing box structure. This duplicates the type of structure that would be used in the main wing box and will be tested at room and elevated temperatures duplicating typical wing bending loads.

CE-6447

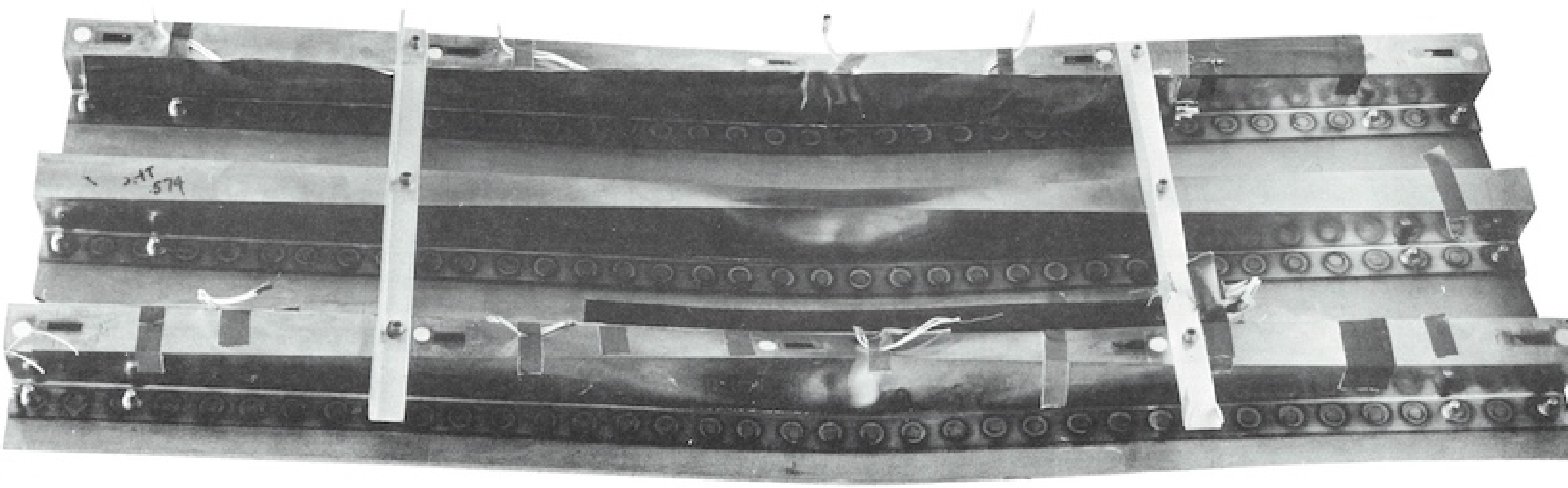
12 FT. WING BOX SECTION



Compression Panel

A spotwelded titanium alloy compression panel was loaded to ultimate and took a compressive stress of 110,000 psi. The test shows excellent performance of the spotwelds as there was no skin to stiffener separation. An aluminum alloy structure of the same type of design would take approximately 50,000 psi.

6A65410-1

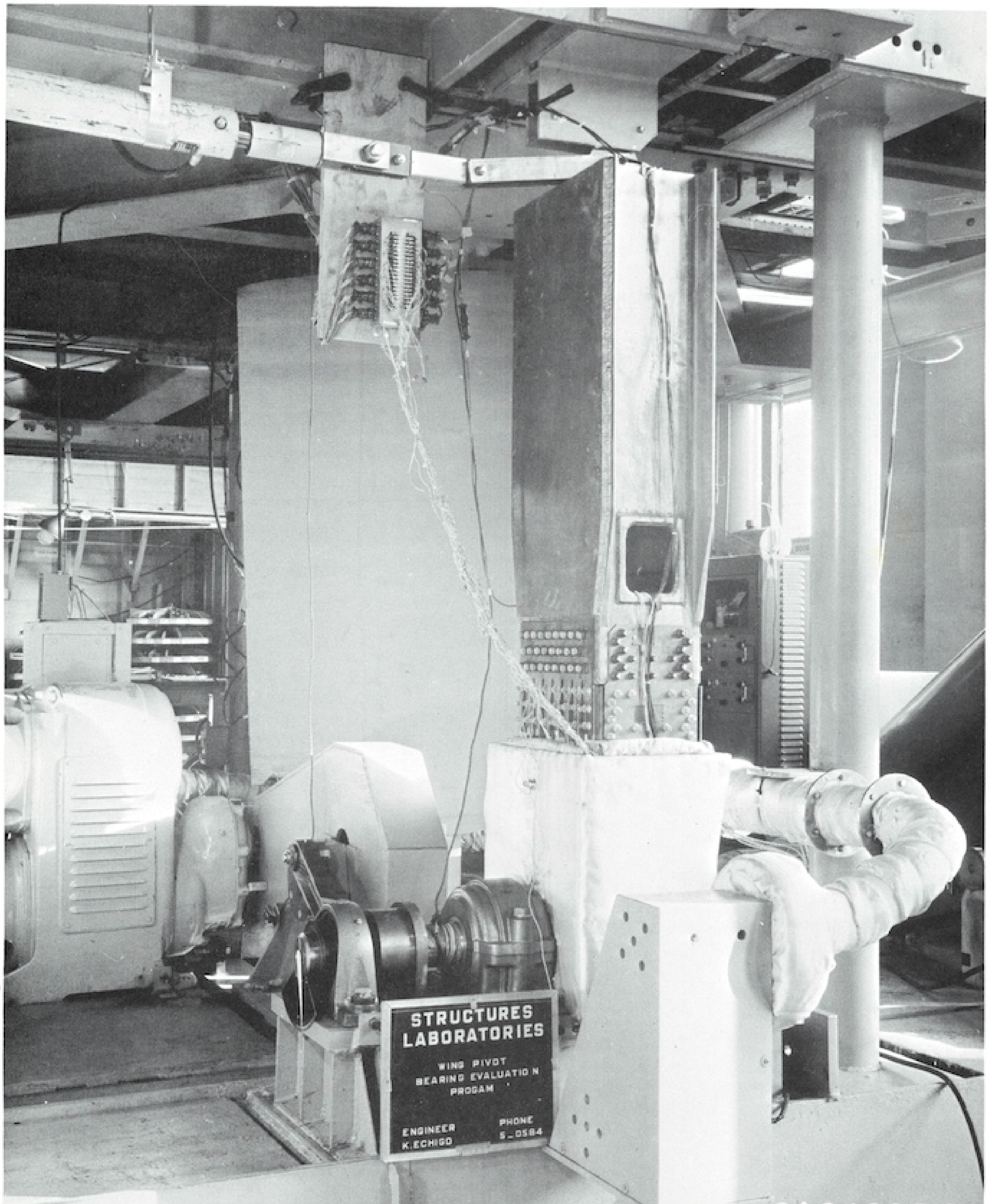


6A65410-1

Quarter Size Bearing Test

In this test we duplicate the typical loading that will occur on the main pivot bearings. The teflon bearing was subjected to 215,000 cycles at 300°F before failure which is more than the life of the airplane.

6A64695-1


STRUCTURES
LABORATORIES
WING PIVOT
BEARING EVALUATIO N
PROGAM
ENGINEER
K. ECHIGO
PHONE
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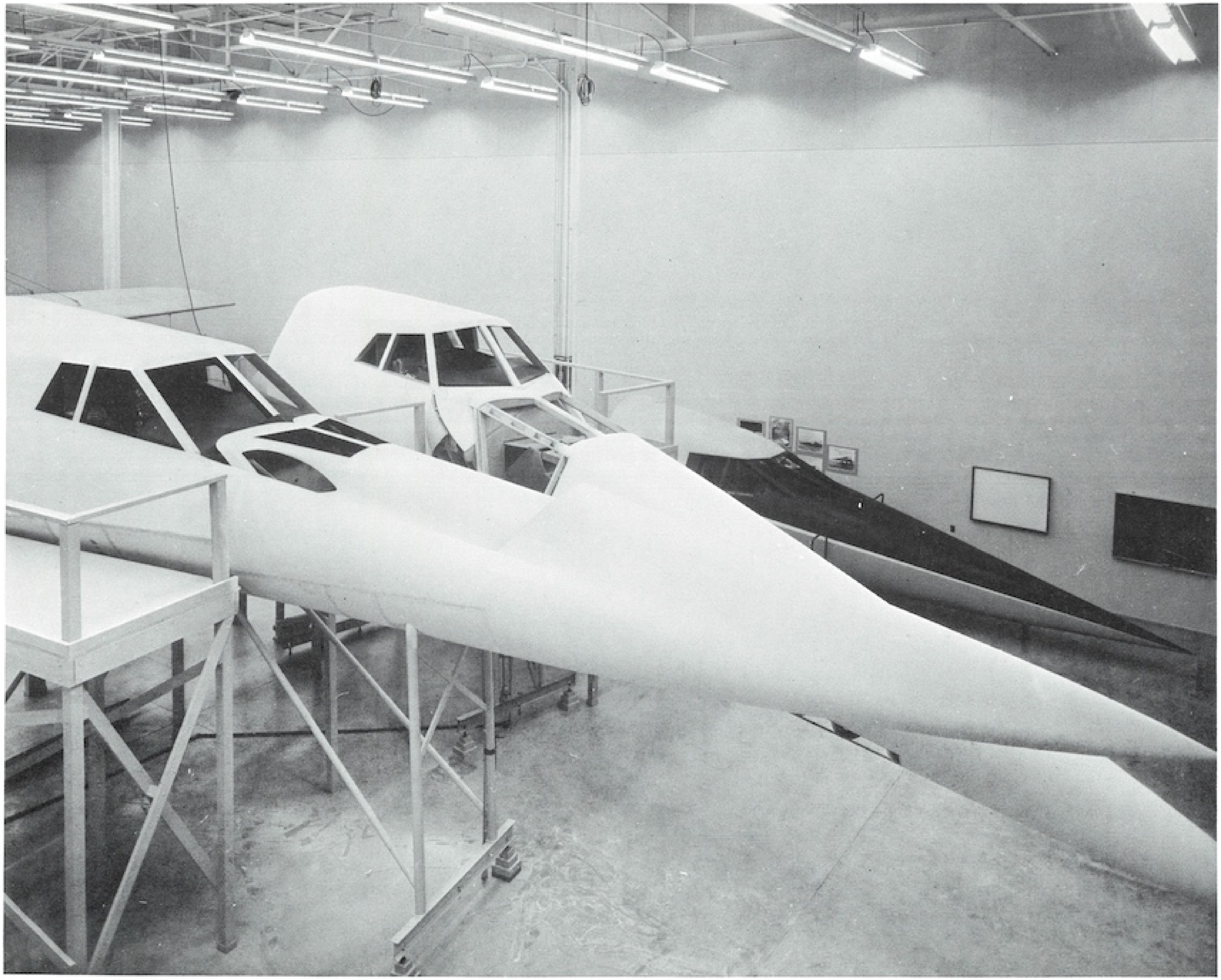
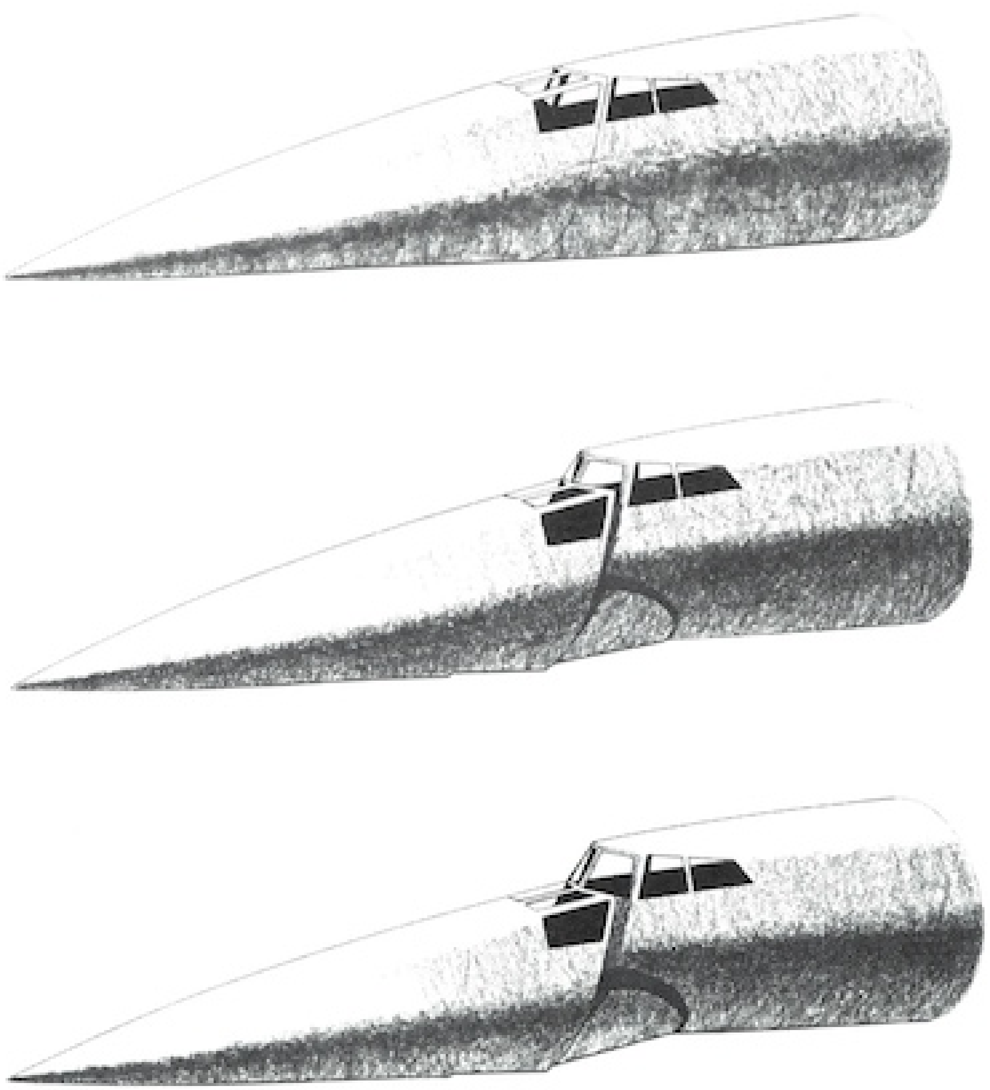
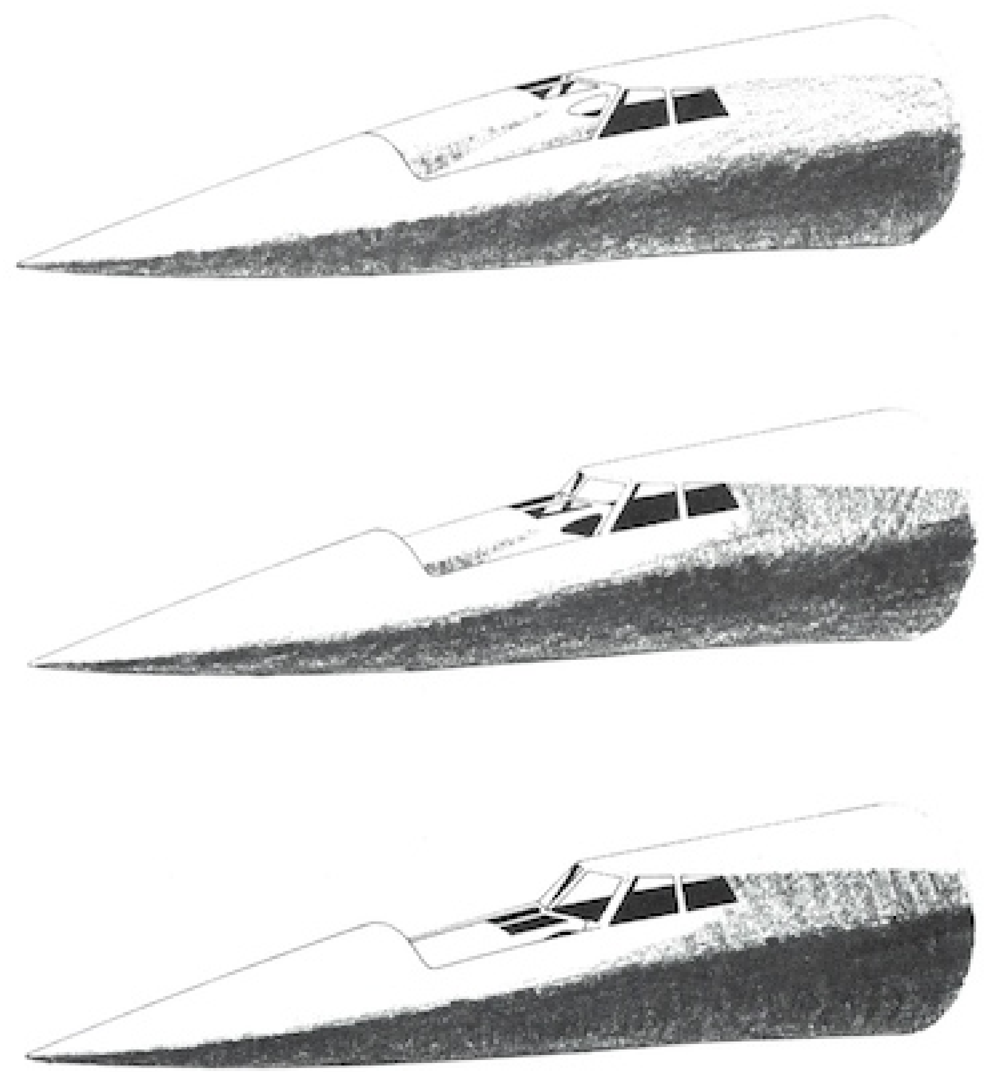


NOSE CONFIGURATIONS

TRANSLATING (6 TO 1 OGIVE)



WINDSHIELD VISOR (MODIFIED OGIVE)



Full Size Wing Pivot Test

The full size wing pivot bearing is being tested at temperatures of -40°F, at room temperature, and 200°F. During actuation, the bearing receives a load that varies from 900,000 pounds to 1,800,000 pounds. Every 10,000 cycles the bearing receives a static load of 3,600,000 pounds. This test should demonstrate that the bearing will be satisfactory for the life of the airplane.

CE-6440

FULL-SIZE WING PIVOT TEST

Differences between the Phase I (733-197 basic and long body versions) and the 733-290 are highlighted in this figure. The most obvious changes are quickly apparent. The body of the -290 is considerably longer and the cabin is wider. The increased length and width of the body enlarge the payload capability of the -290 airplane.

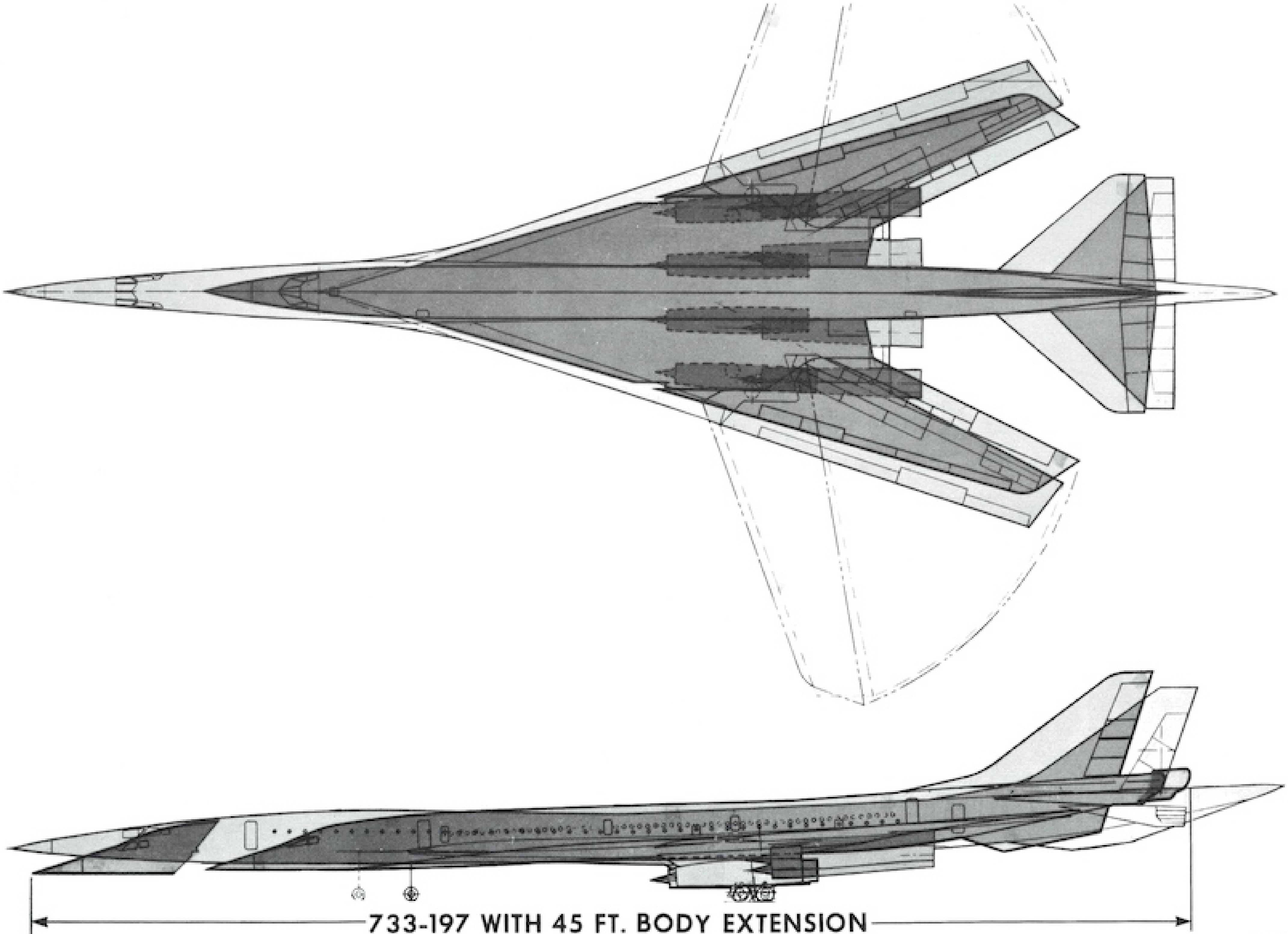
Aerodynamic improvements have changed the airplane at a number of important locations. The wing of the -290 has slightly less sweep when in the full-back position. Less sweep increases the supersonic span slightly and improves the lift/drag ratio. The engine nacelles are located farther aft and their shape has changed from cylindrical to slightly conical to accommodate a more fully expanded nozzle. This relocation and conical shape increases the favorable pressure reaction on the wing which improves the lift/drag ratio at supersonic speeds.

The portion of the body near the wing leading edge intersection has been bulged to create a favorable pressure reaction with the wing which further improves the supersonic lift/drag ratio. The nose shape of the airplane has been slimmed and lengthened for optimum supersonic cruise drag. The tail of the body is also more pointed to improve the over-all lift/drag ratio.

Wing span with the movable sections forward is practically identical with the earlier -197. This span remains constant because the pivot has moved slightly outboard and back for improved weight, balance, and other characteristics.

CE-6577

CONFIGURATION COMPARISONS



733-197 WITH 45 FT. BODY EXTENSION

CE6577 11-64

A further comparison between the 733-197, long body 733-197 and 733-290 Intercontinental airplane and the long body 733-197 and 733-291 Transcontinental airplane is shown on the table on the opposite page.

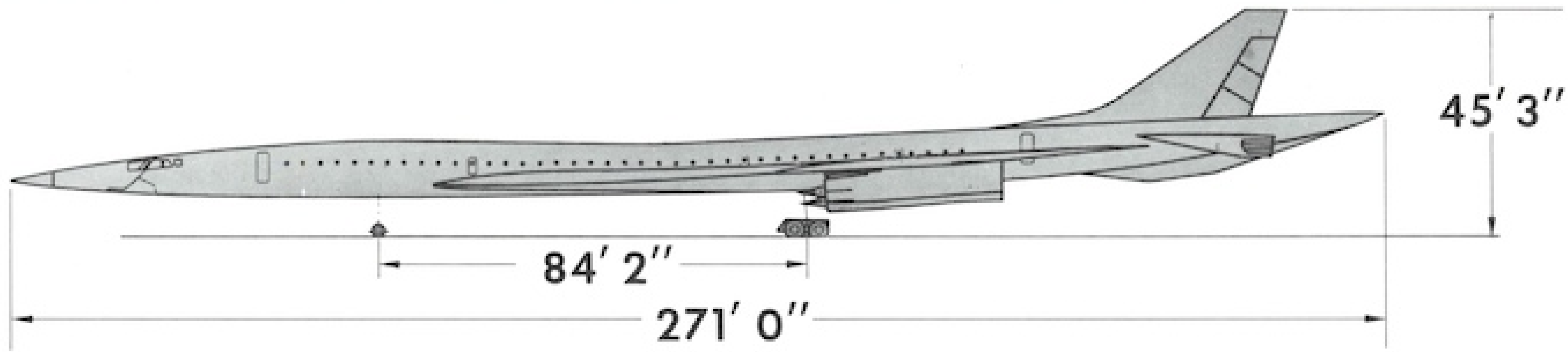
MODEL COMPARISON

	JANUARY FAA 733-197	JANUARY AIRLINES 733-197 LONG BODY	NOVEMBER 733-290	JANUARY DOMESTIC AIRLINES 733-197 LONG BODY	NOVEMBER 733-291
WING AREA-FT^2	4680	4680	5020	4680	5020
GROSS WEIGHT-1000 LB	430	520	500	408	425
NO. PASSENGERS (MIXED CLASS)	145	203	230	194	209
ENGINE G. E. 4- (SIZE-LB/SEC)	J4C 415		J5G 475	J4C 415	J5G 475
MAIN GEAR- TIRES	8-49″		16-40″	8-46″	16-38″

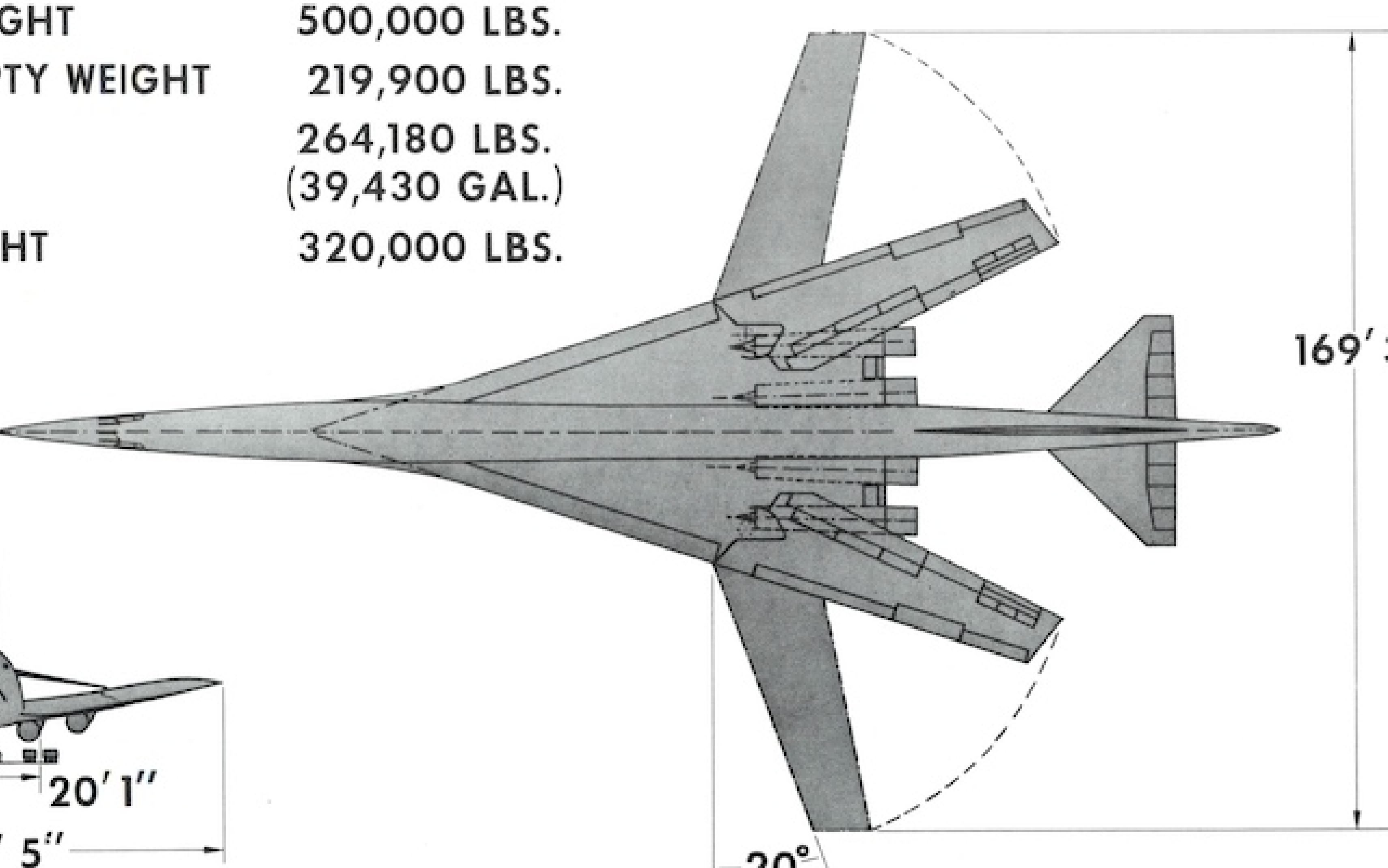
Two versions of the basic airplane are proposed, the 733-290 Intercontinental shown on the opposite page and the 733-291 Transcontinental shown overleaf. Body length and diameter, engines and wing planform for the two versions are identical.

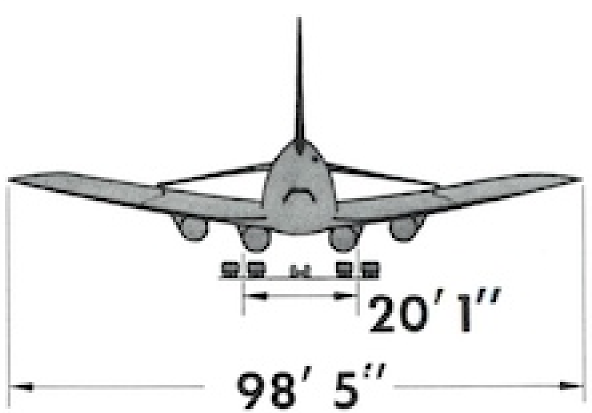
While the configuration has been extensively revised from the Phase I proposal (733-197) it is still relatively close to the ground with the forward entry door only ten feet six inches above the ground line in the parked position. This door sill height is comparable to present subsonic jets.

733-290 GENERAL ARRANGEMENT



MAX. RAMP WEIGHT	500,000 LBS.
OPERATING EMPTY WEIGHT	219,900 LBS.
FUEL CAPACITY	264,180 LBS. (39,430 GAL.)
MAX. LDG. WEIGHT	320,000 LBS.



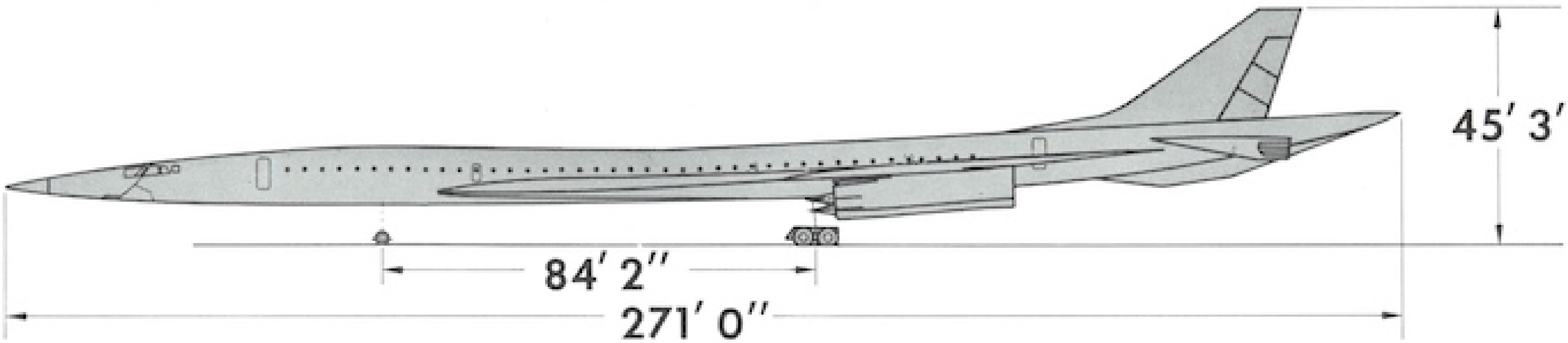


CE6531 11-64

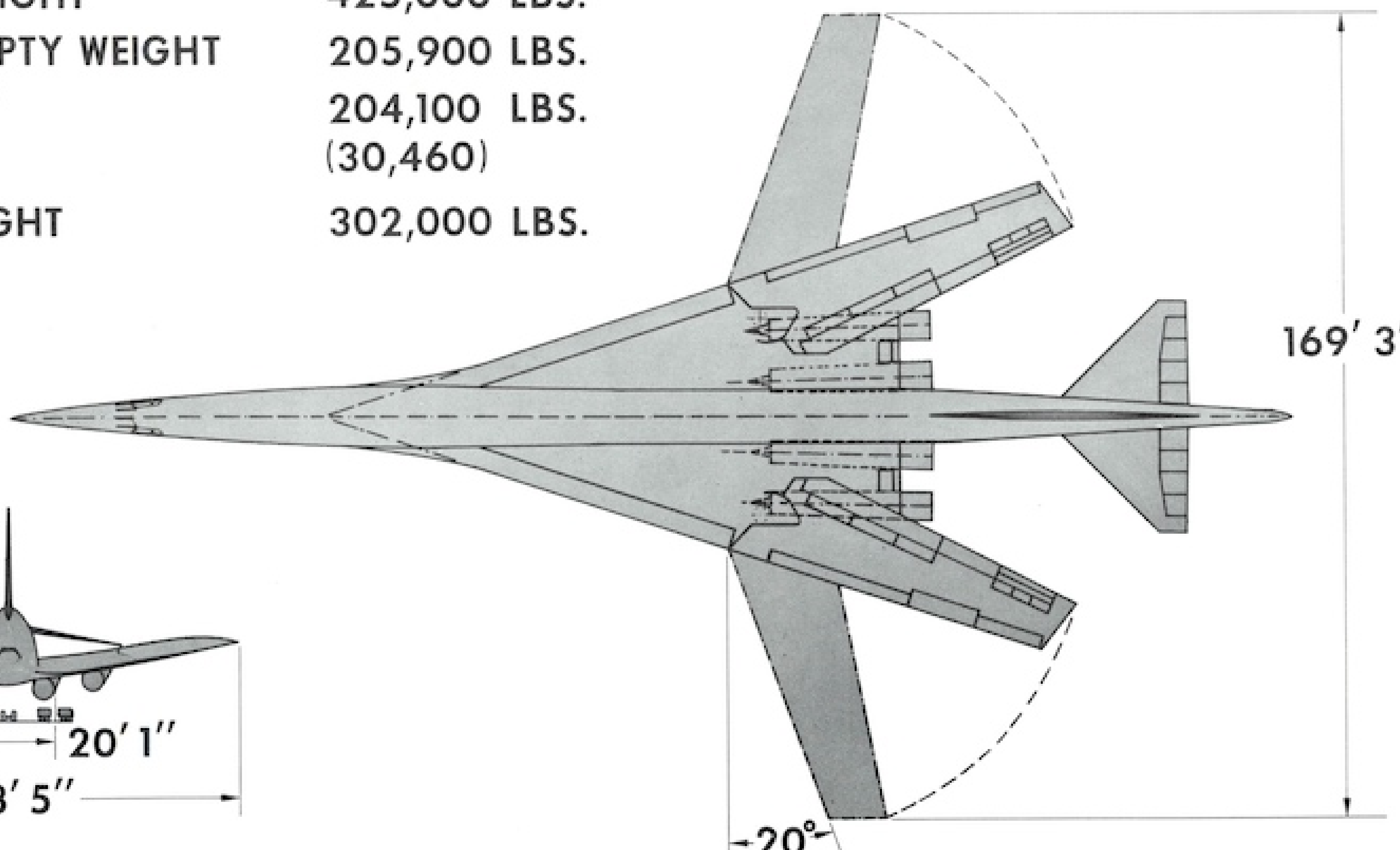
The 733-291 Transcontinental airplane differs mainly from the 733-290 Intercontinental in the amount of fuel carried. Because of this and the resultant reduction in gross weight, we plan to lighten the structure in much the same manner as was done for the Model 720 when it was derived from the original basic 707 design.

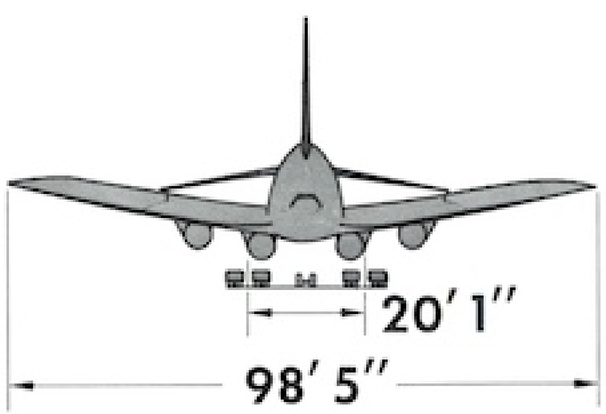
CE-6597

733-291 GENERAL ARRANGEMENT



MAX. RAMP WEIGHT	425,000 LBS.
OPERATING EMPTY WEIGHT	205,900 LBS.
FUEL CAPACITY	204,100 LBS. (30,460)
MAX. LDG. WEIGHT	302,000 LBS.



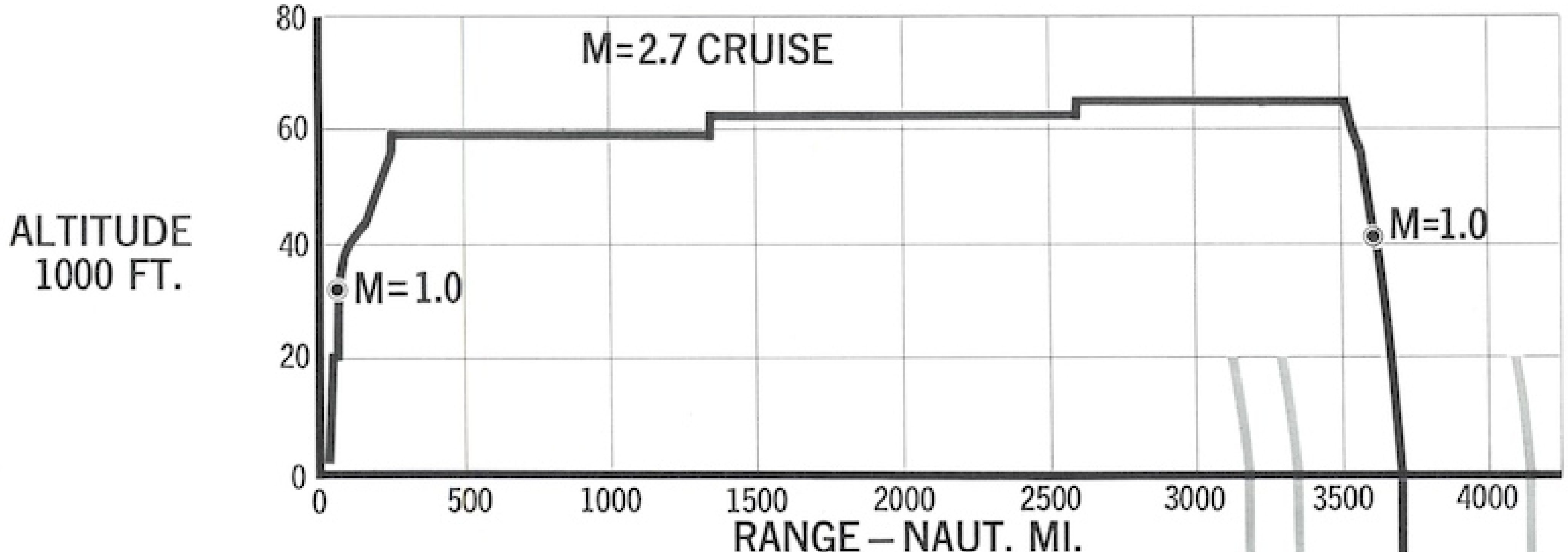


CE6597 11-64

The range-payload capability of the Boeing 733-290 is greatly increased over that of the 733-197. An intensive program of aerodynamic development has produced a remarkable improvement in performance and economy of operation. The 733-290 and -291 can carry up to 65 percent greater payloads than could the -197 over representative intercontinental and transcontinental distances.

CE-6552

MISSION PERFORMANCE
733-290



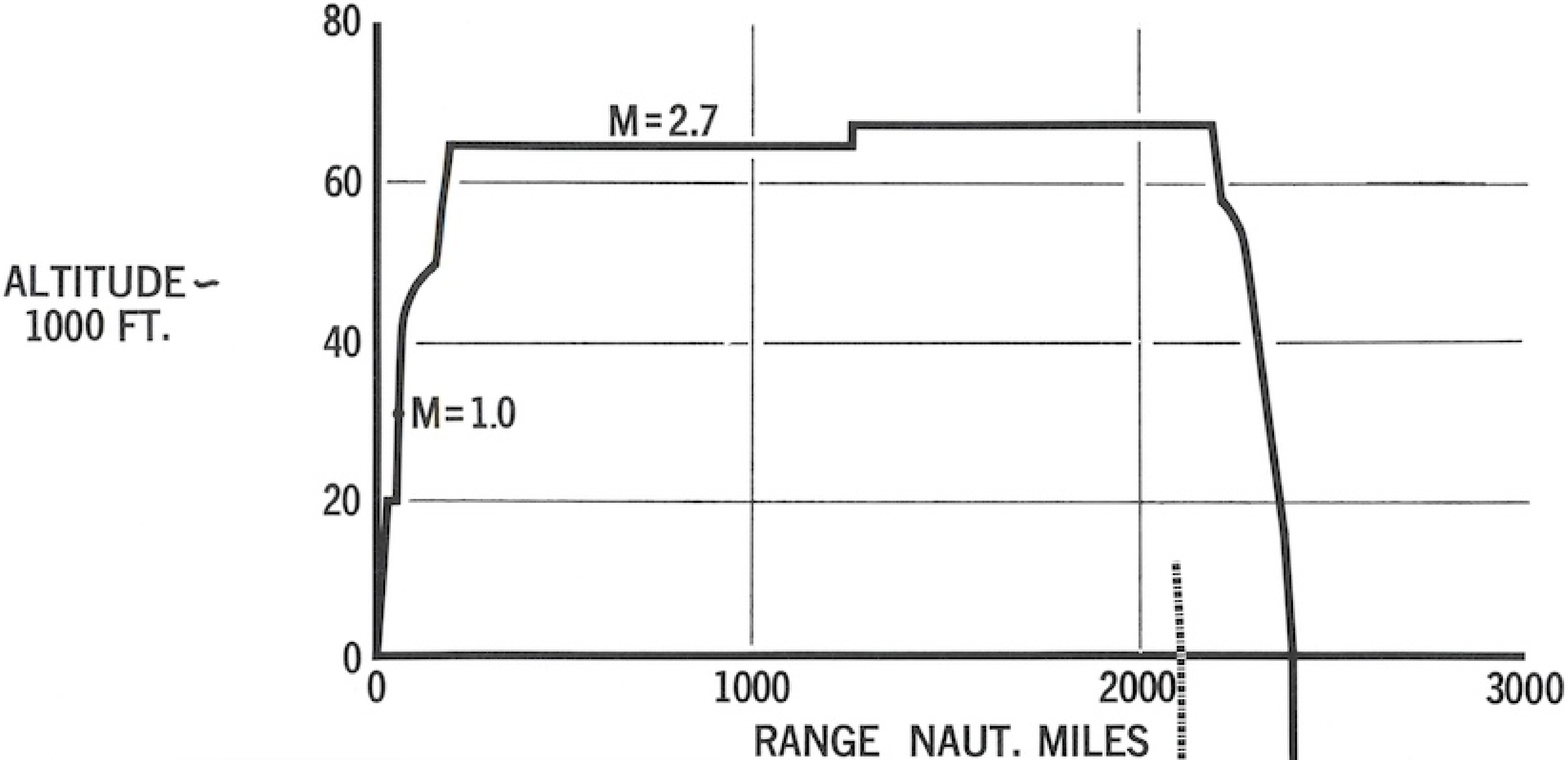
ROUTE	RANGE* NAUT.MI.	RAMP G.W., LB.	PAYLOAD LB.	BLOCK TIME, HR.
N.Y. TO PARIS	3160	494 000	52 865	2.74
N.Y. TO FRANKFURT	3340	500 000	50 500	2.89
N.Y. TO ROME	3680	500 000	37 000	3.07
TOKYO TO SEATTLE	4150	500 000	21 000	3.40

*AIRLINE DISTANCES

Typical mission capabilities of the 733-291 Transcontinental are shown here.

CE-6630

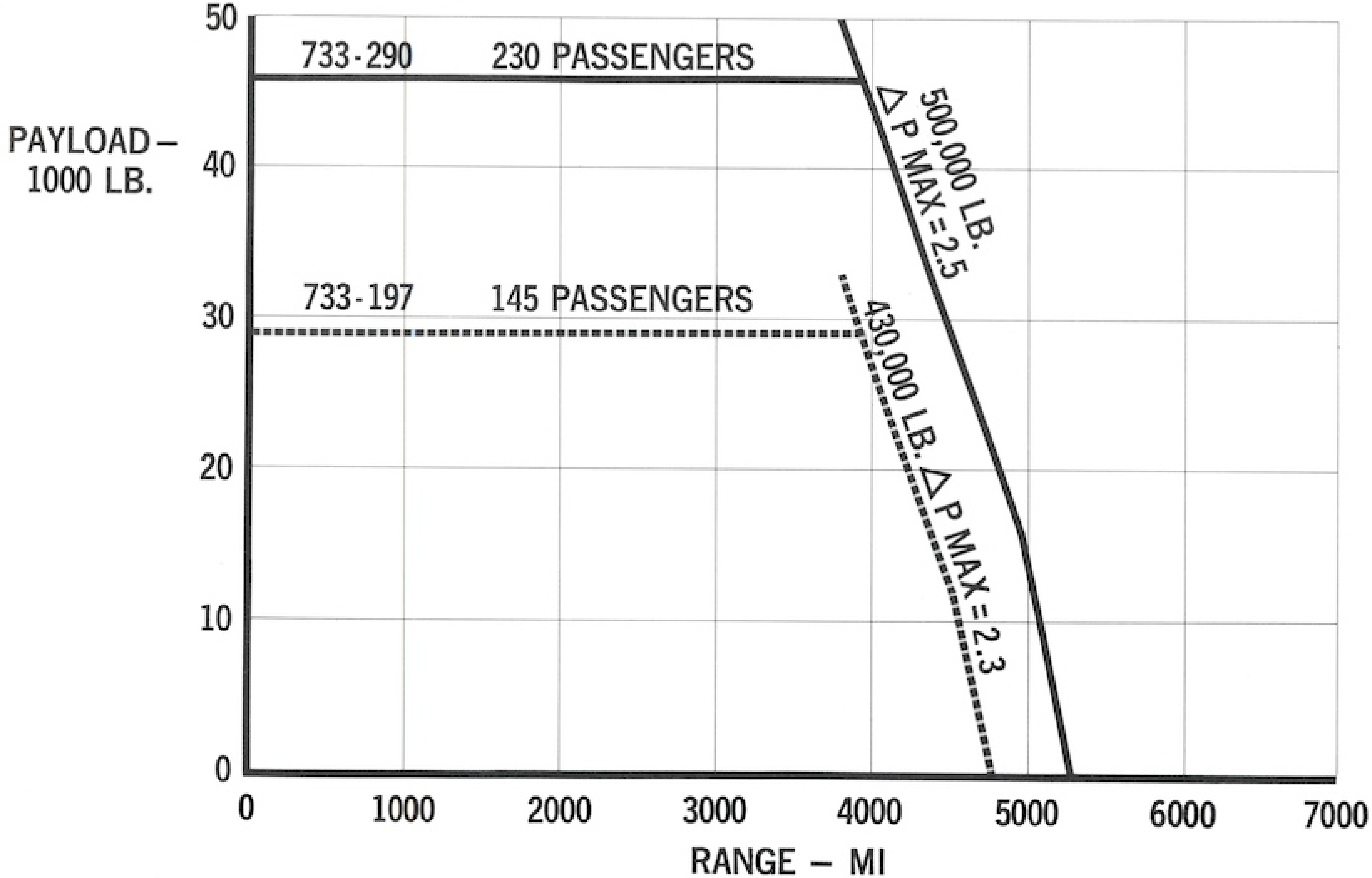
MISSION PERFORMANCE
733–291



ROUTE	RANGE * NAUT. MI.	RAMP G.W. LB.	PAYLOAD LB.	BLOCK TIME HR.
N.Y. TO L.A.	2095	413,000	52,865	2.04
SEATTLE TO HONL.	2387	425,000	50,200	2.24

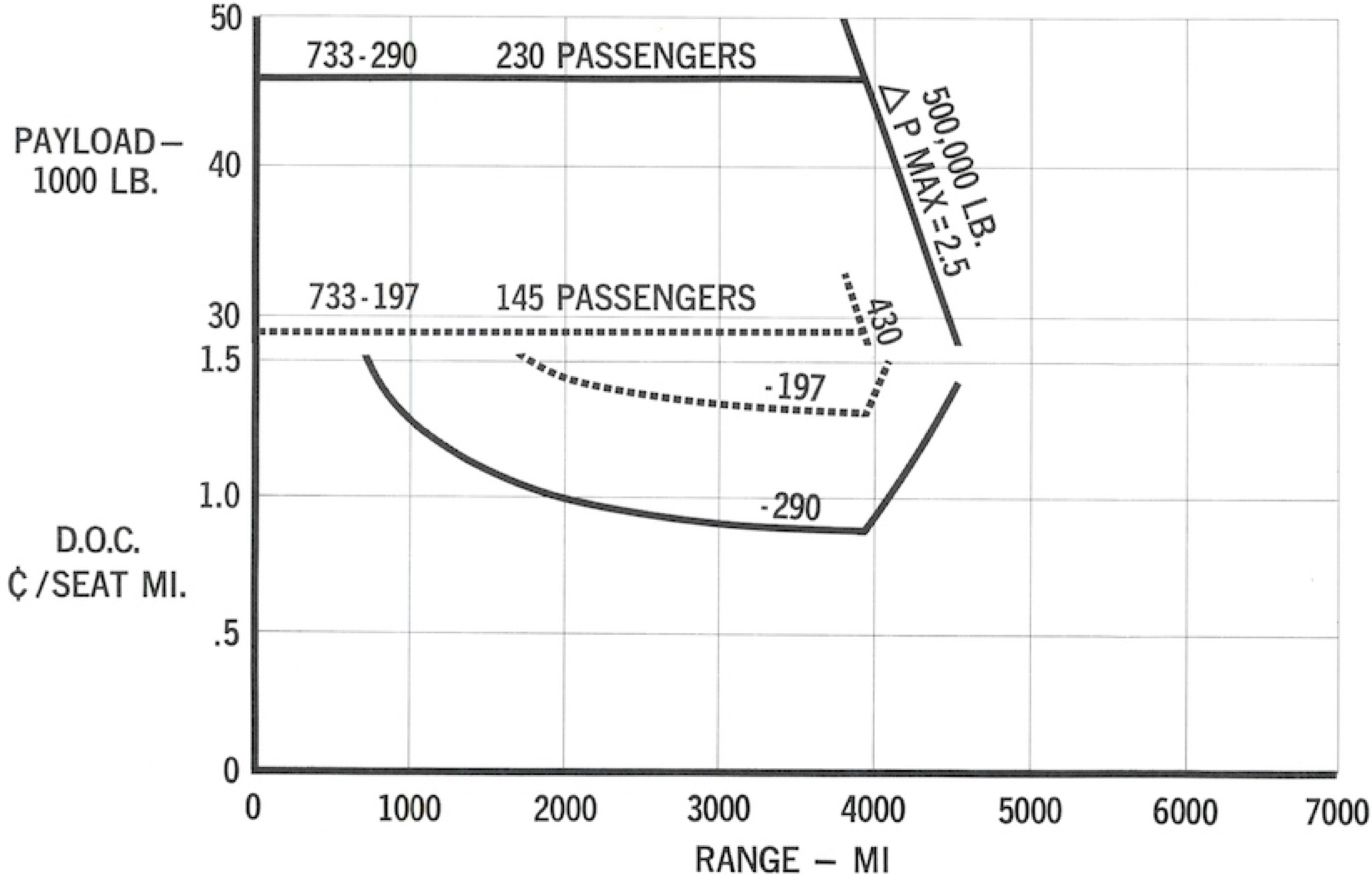
* AIRLINE DISTANCE

COMPARISON OF IMPROVED AIRPLANE 733-290 WITH JANUARY AIRPLANE 733-197



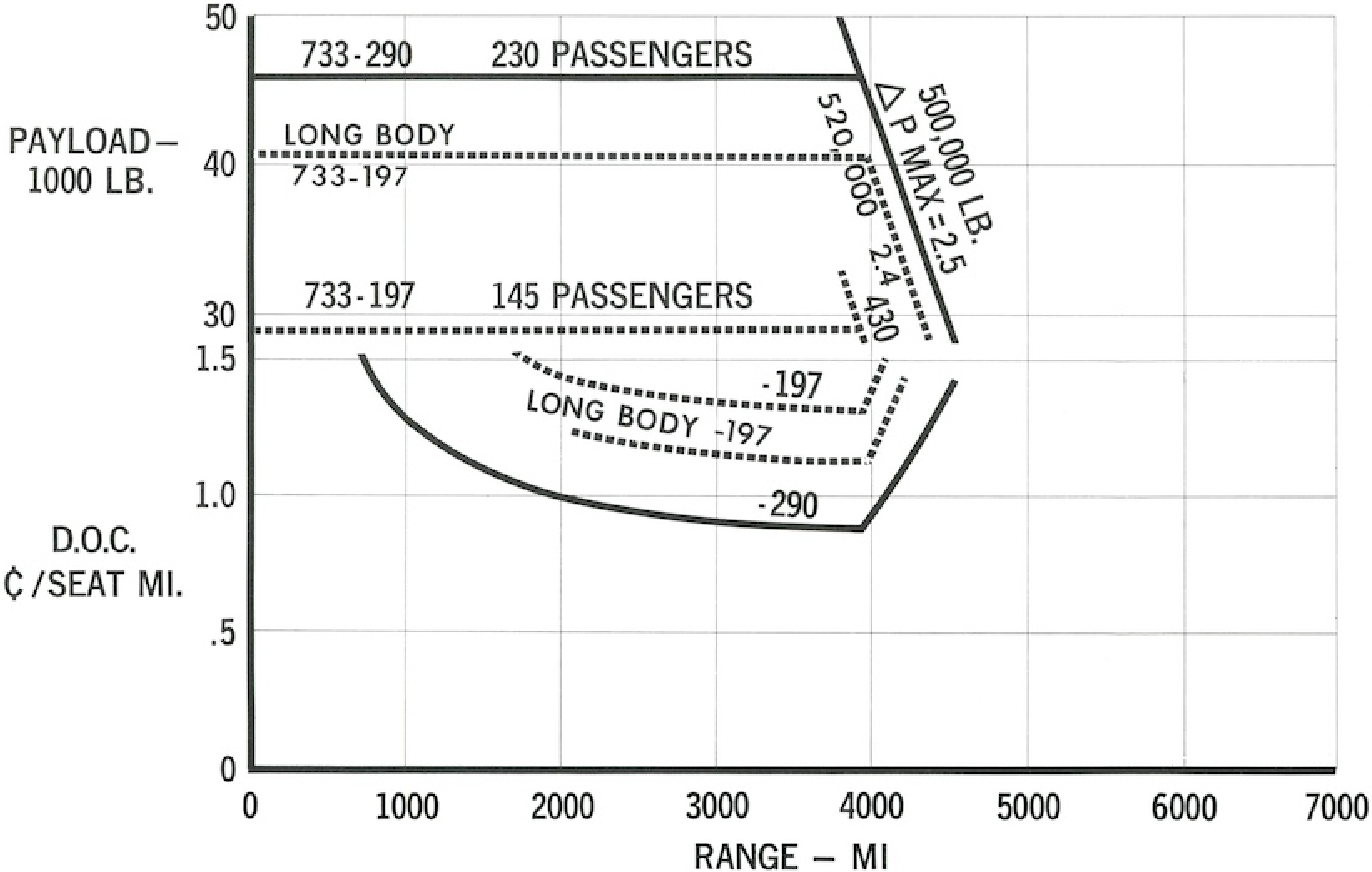
CE6536 11-64

COMPARISON OF IMPROVED AIRPLANE 733-290 WITH JANUARY AIRPLANE 733-197



CE6537 11-64

COMPARISON OF IMPROVED AIRPLANE 733-290 WITH JANUARY AIRPLANE 733-197

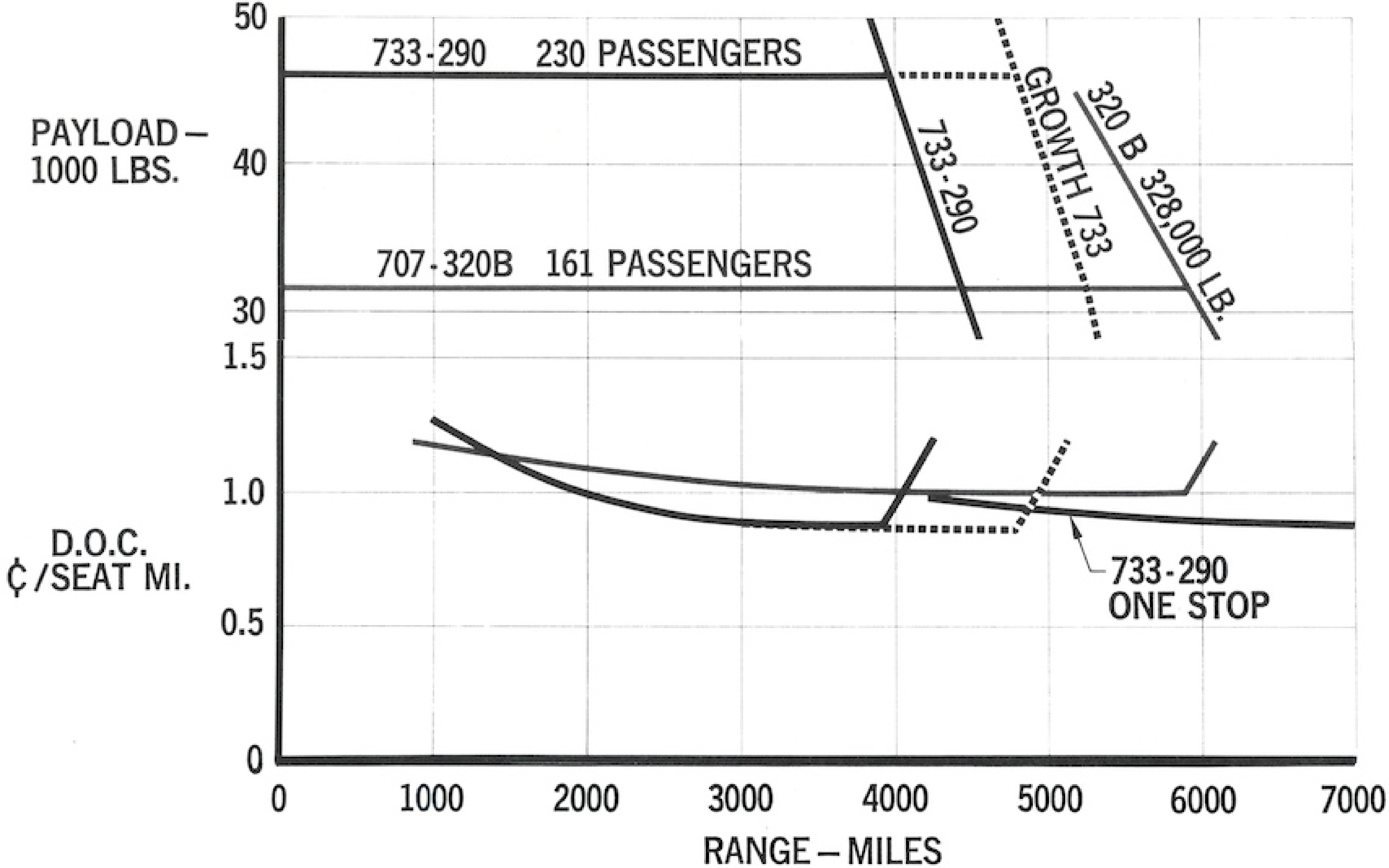


CE6580 11-64

Payload-range performance of the 733-290 and 707-320B are compared here. The loading mix and seat spacing used on both aircraft are the same: 10 percent of the seats are in a four-abreast first class section at 40 inch pitch, and the remaining 90 percent are in the tourist cabin at 34 inch pitch.

The "Growth 733" curve represents a reasonable estimate of the extra range which is potentially available due to anticipated improvements in aerodynamic, propulsion system and structures efficiencies.

COMPARISON WITH 707-320B

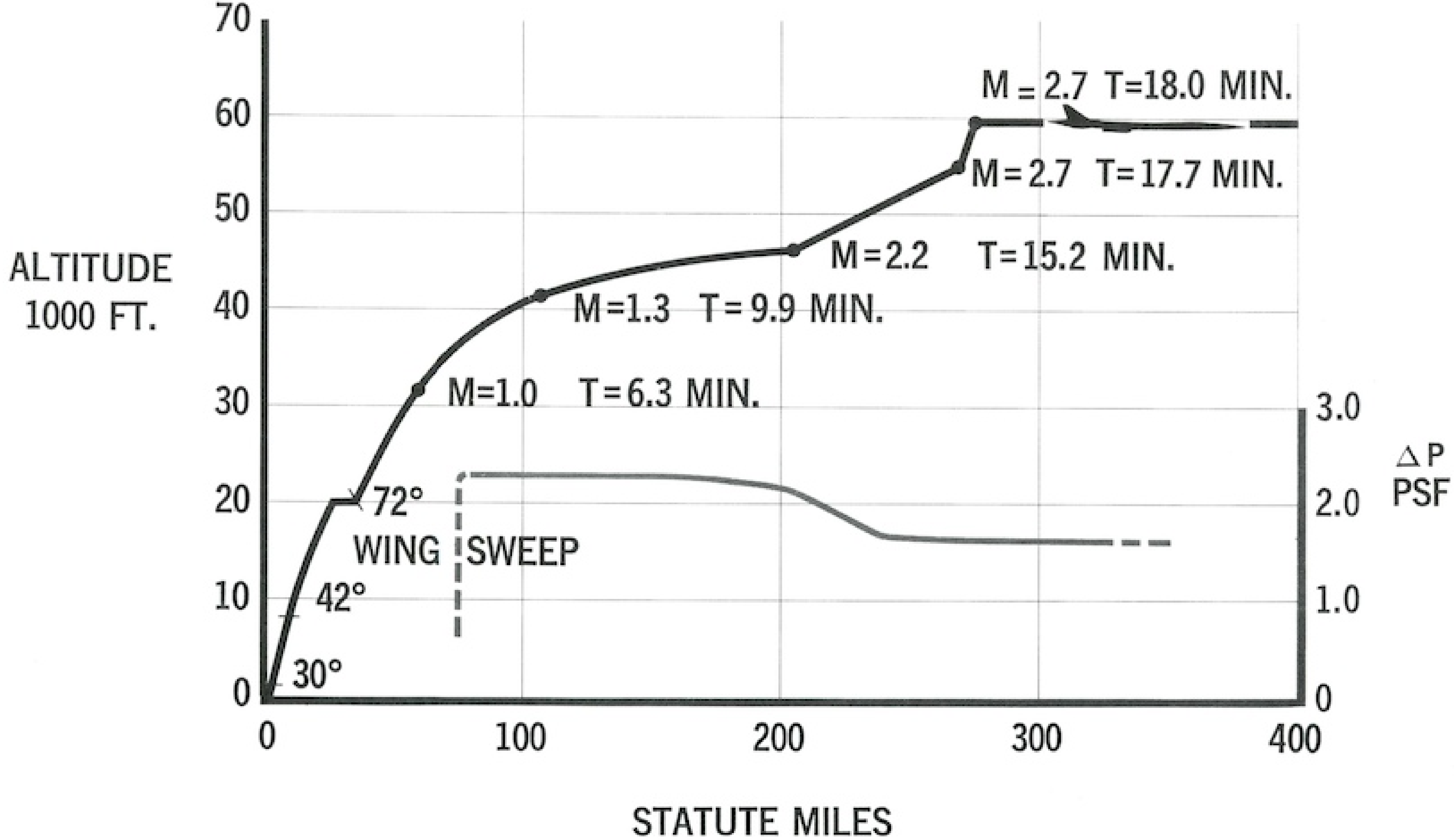


CF6535 11-62

A typical 733-290 climb and acceleration schedule will permit attaining cruise altitude and speed in approximately 18 minutes from brake release at maximum gross weight. This schedule is based on maintaining a maximum sonic boom overpressure level of 2.3 psf.

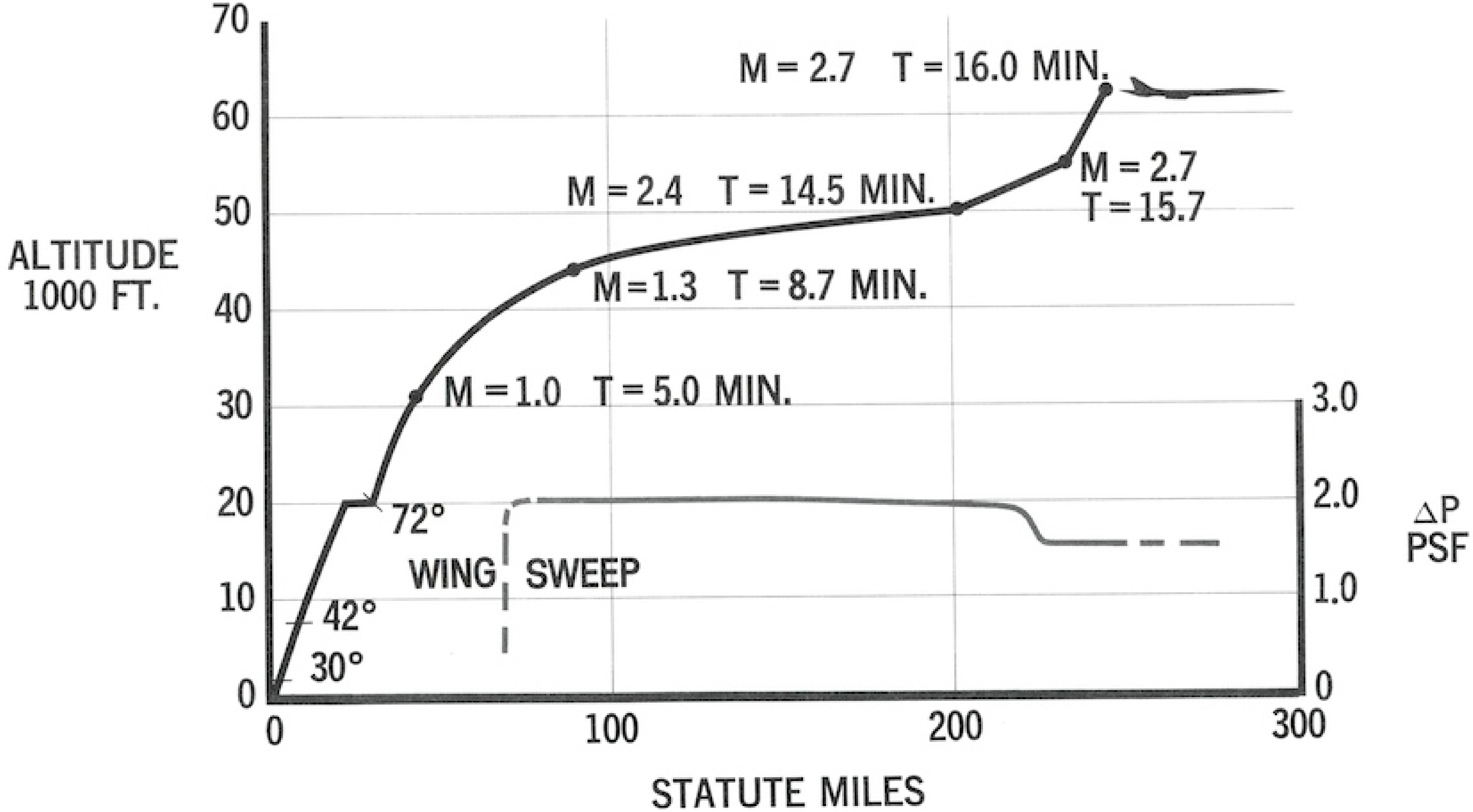
Wing sweep rate has been increased to allow a more expeditious management of the wing sweep and more flexibility in climb speed selection. At approximately 240 knots IAS or as soon as practical after flaps have been retracted, the thrust is set at maximum dry power and 42° of wing sweep selected. Acceleration to 350 knots IAS and climb to 20,000 feet at this IAS is made, followed by wing sweep to 72° and acceleration to 375 IAS. As the airplane approaches Mach 0.9 (20,000 to 25,000 feet), maximum augmented power is applied. The airplane climb angle is reduced slightly at 30,000 feet to accelerate along the selected sonic boom overpressure path. This path is followed out to approximately 630 IAS (40,000 to 50,000 feet) which is then the climb speed until intersecting a final segment at Mach 2.7 (55,000 feet). The Mach 2.7 climb continues until level off at cruise altitude (60,000 feet).

CLIMB & ACCELERATION
733-290



The climb, acceleration and sonic boom schedule for the 733-291 is shown on the opposite page. This profile is based on a maximum gross weight takeoff and maintaining a 2.0 psf sonic boom overpressure.

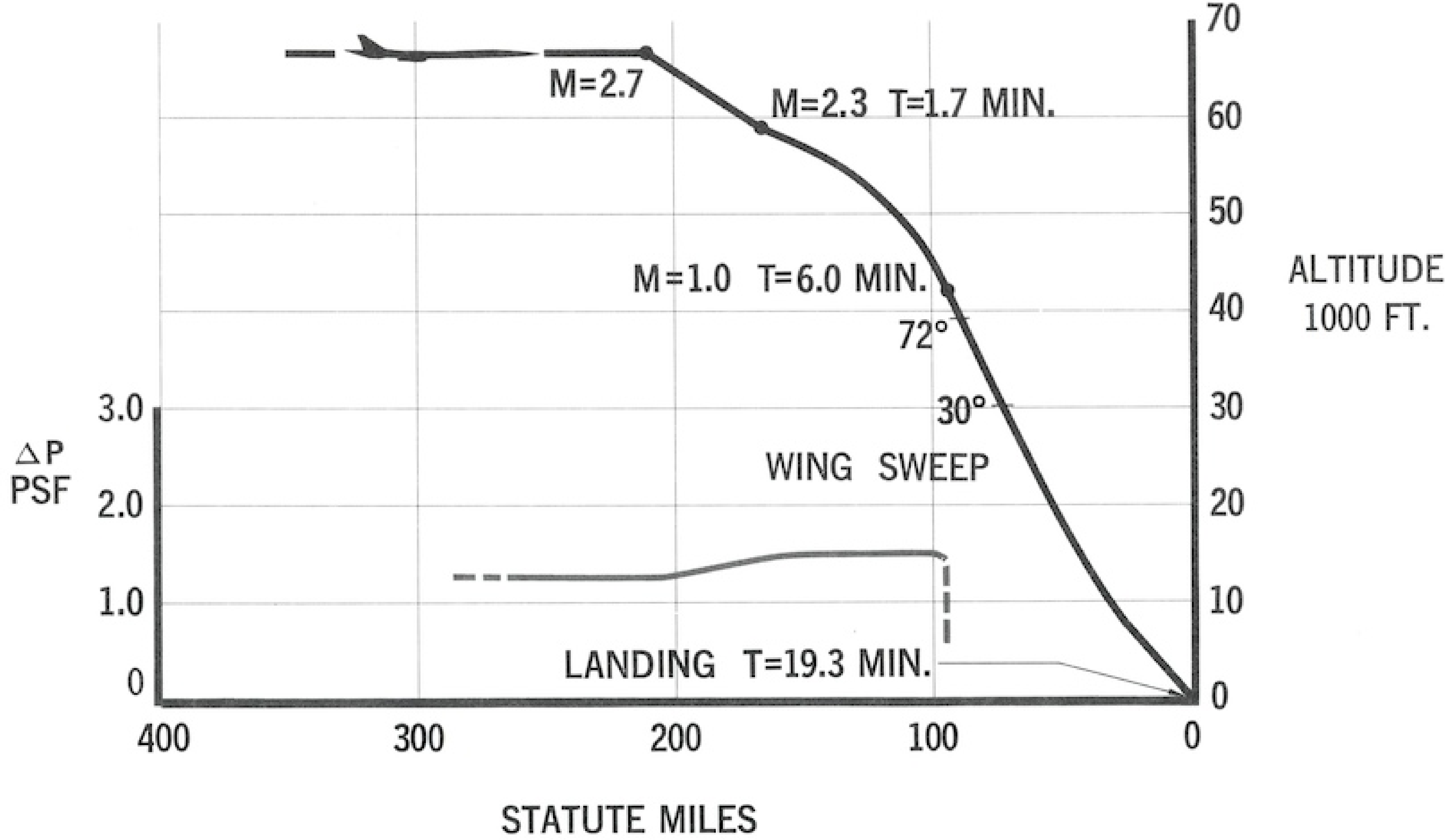
CLIMB AND ACCELERATION
733-291



CE6618 11-64

The Model 733 can slow down any time without preparation or c.g. manipulation. "Low idle" is selected and the airplane started down at constant indicated airspeed (essentially the same airspeed held at the end of cruise). This is 500 knots for the normal mission. The 72° wing sweep is held until the airplane decelerates to subsonic speeds. At approximately 58,000 feet the descent path intersects the 1.5 psf sonic boom overpressure limit segment which is then followed. The airplane becomes subsonic at 42,000 feet and descends from this point in a manner similar to today's jet transports. The wing sweep is started forward so as to end up at 30° at about Mach 0.8. The normal descent times, sonic boom overpressures and distances are shown on the opposite page.

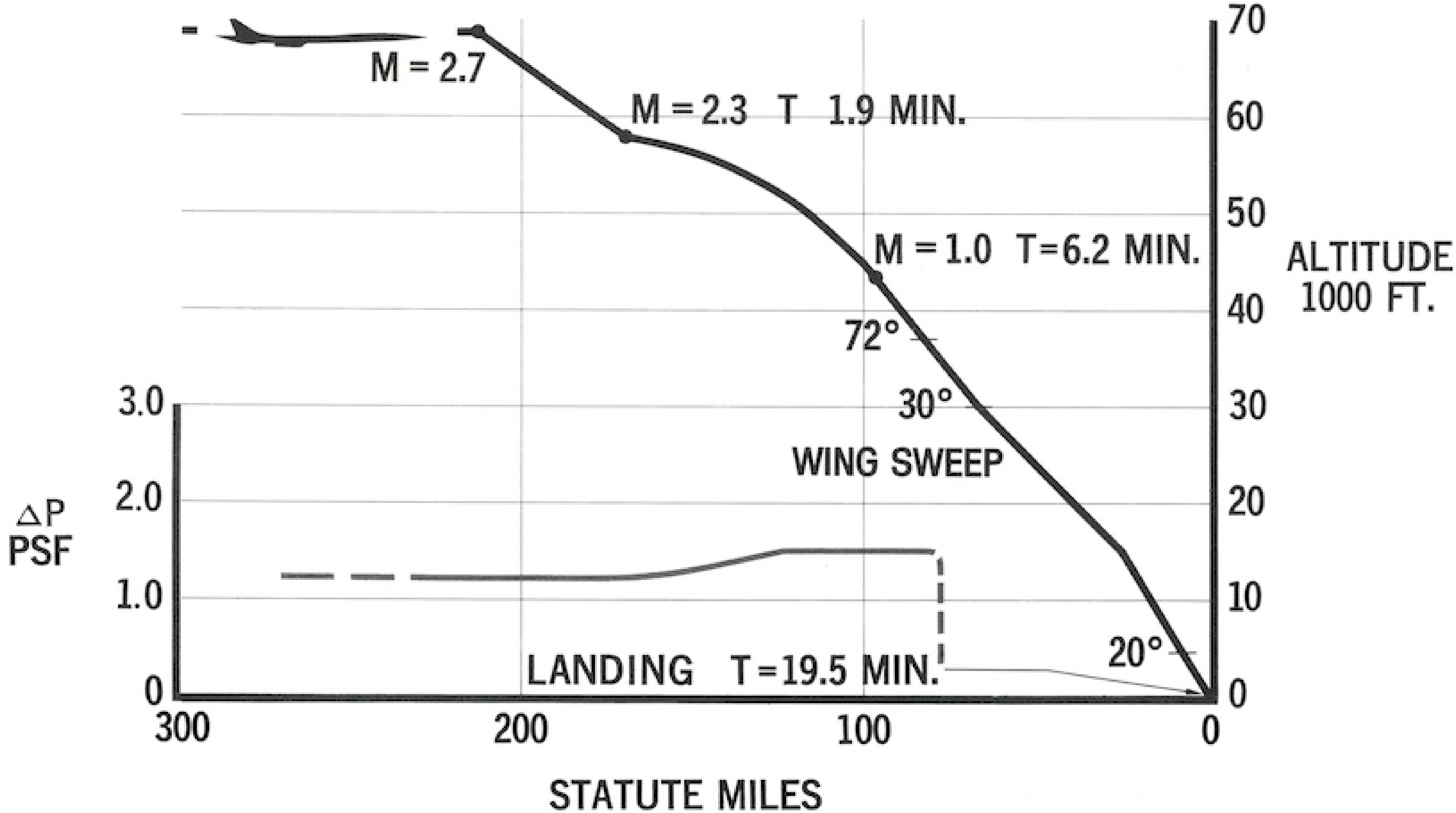
DECELERATION & DESCENT
733-290



CE6512 11-64

A typical deceleration and descent profile for the Model 733-291 is shown on the facing page.

DECELERATION AND DESCENT
733-291

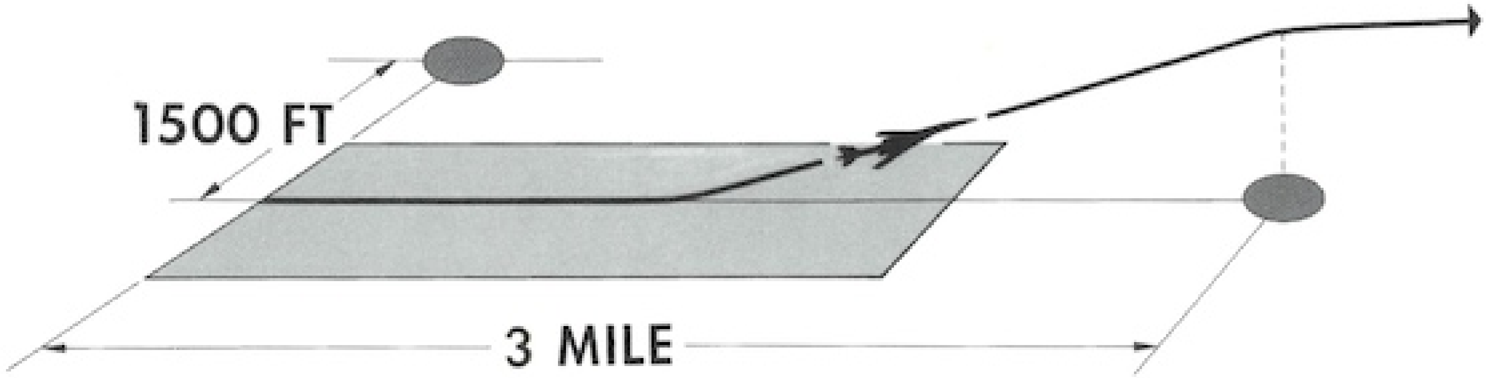


CE6616 11-64

The excellent low speed performance of the Models 733-290 and -291 helps to maintain low community and airport noise levels. At takeoff, high maximum lift coefficient and high lift/drag ratio permit takeoff in a short distance with moderate airport noise levels, and rapid climbout results in low community noise levels.

The chart opposite compares takeoff performance and noise levels for the 733, 720B, 707-320B, 727 and Concorde under standard day, sea level conditions. Takeoff speeds for the -290 and -291 for these conditions are 159 knots and 146 knots, respectively.

TAKEOFF PERFORMANCE

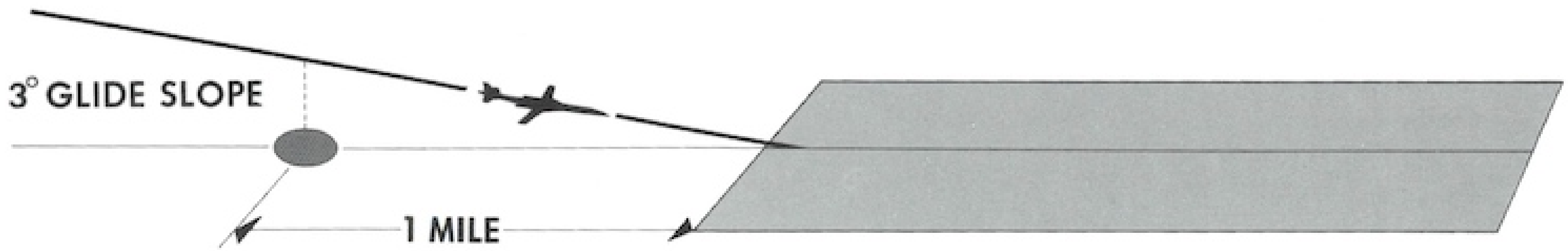


MODEL	GW (1000 LB.)	TAKE-OFF SPEED (KTS)	TAKE-OFF DISTANCE (FEET)	AIRPORT NOISE (PN db)	COMMUNITY NOISE (PNdb)	ALTITUDE (FEET)
733-290	500	159	7060	116	105.5	1570
707-320 B	327	166	10,850	108	120.5	600
733-291	425	146	5115	116	97	2340
720B	234	156	6200	107	110.5	1150
727	152	133	5850	108	100.5	1250
CONCORDE	326	205	9750 (MAX AUG)	118	106(112)	800

Excellent approach and landing performance at low noise levels is another attractive feature of the 733. The airplane's large span with wings swept forward plus an efficient high lift system provide low approach speeds at low attitudes and minimum approach noise because of the low power required. Engine compressor whine is all but eliminated by operating the engine inlet in the choked position.

Approach noise one mile from the end of the runway under the airplane flying an ILS is shown for the maximum landing weights of the -290 (320,000 pounds) and the -291 (302,000 pounds). CAR landing field lengths are also listed. Landing distances are comparable to the 707. Approach noise is substantially less.

LANDING PERFORMANCE



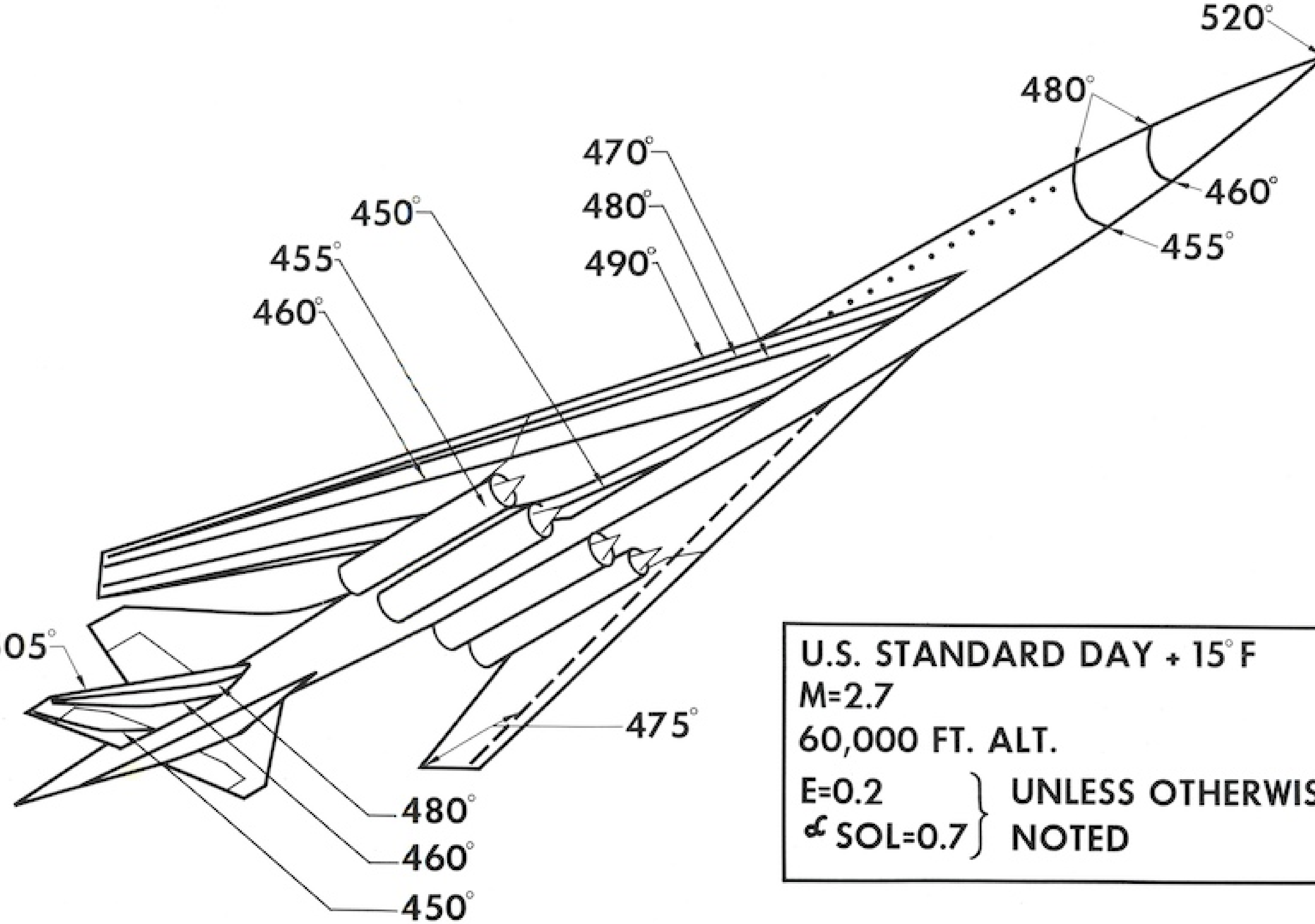
MODEL	LANDING WEIGHT (1000 LB)	APPROACH SPEED (KTS)	BODY APPROACH ATTITUDE (DEGREES)	CAR LDG DISTANCE (FT)	COMMUNITY NOISE (PNdb)
733-290	**320**	**137**	**2.5**	**6440**	**112**
707-320B	207	135	0.9	6420	122.5
733-291	**302**	**133**	**2.5**	**6150**	**111**
720B	175	134	1.6	6400	120
727	135	121	1.4	4600	117.5
CONCORDE	200	165	12	7750	110(119)

Structural Temperatures

The maximum temperatures that we expect to encounter on the exterior of the SST are shown on the chart. This is for a hot day at Mach 2.7 and at 60,000 feet altitude for an unpainted airplane. The temperatures would be reduced by approximately 50°F if we painted the airplane. The body structure has an exterior temperature of 480°F on top and 455°F on the bottom. The wing primary structure has an exterior temperature of 480°F to 475°F.

The thermal buckling test in the test area was being tested at 500°F, which is higher than the hot day temperatures and is conservative. Also, the large bearing test temperatures were based on calculations from the hot day condition.

STRUCTURAL TEMPERATURES
UNPAINTED AIRPLANE



U.S. STANDARD DAY + 15° F
M=2.7
60,000 FT. ALT.
E=0.2 } UNLESS OTHERWISE
α SOL=0.7 } NOTED

CE6674 3-65

Materials Comparison

The Materials Comparison chart shows a comparison of the structural efficiency of stainless steel and titanium 8-1-1 alloy with aluminum. Stainless steel weighs 2.9 times as much as aluminum, and titanium alloy weighs 1.6 times as much as aluminum. At room temperature, steel is 3.2 times the strength of aluminum and titanium alloy is 2.2 times the strength of aluminum. If the relative strength is divided by the relative weight, then aluminum will have a structural efficiency of 1, stainless steel will have a structural efficiency of 1.13 and titanium alloy a structural efficiency of 1.4. Thus, a structure of titanium that is the same weight as a structure of aluminum will sustain 40 percent more load than the aluminum structure. At supersonic speed aluminum cannot be used for speeds much greater than Mach 2, so if the relative strength of aluminum at Mach 2 is 1, then steel at Mach 2.7 would be 3.3 times the strength of aluminum at Mach 2, and titanium at Mach 2.7 would be 2.1 times the strength of aluminum at Mach 2. If the relative strengths are divided by the relative weight, then aluminum at Mach 2 will have a structural efficiency of 1 and stainless steel at 2.7 would have a structural efficiency of 1.18, and titanium would have a structural efficiency at Mach 2.7 of 1.33. Therefore, to obtain a structure of minimum weight we must use titanium alloy.

CE-6682

MATERIALS COMPARISON

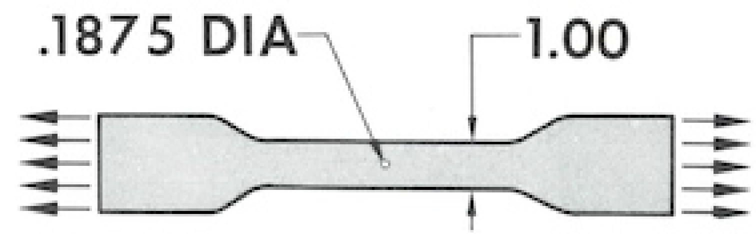
	ALUMINUM	STEEL	TITANIUM
RELATIVE WEIGHT	1	2.9	1.6
ROOM TEMPERATURE PROPERTIES			
RELATIVE STRENGTH	1	3.2	2.2
RELATIVE EFFICIENCY	1	1.13	1.4
HIGH TEMPERATURE PROPERTIES			
AT MACH NO =	2	2.7	2.7
RELATIVE STRENGTH	1	3.3	2.1
RELATIVE EFFICIENCY	1	1.18	1.33

Fatigue Life

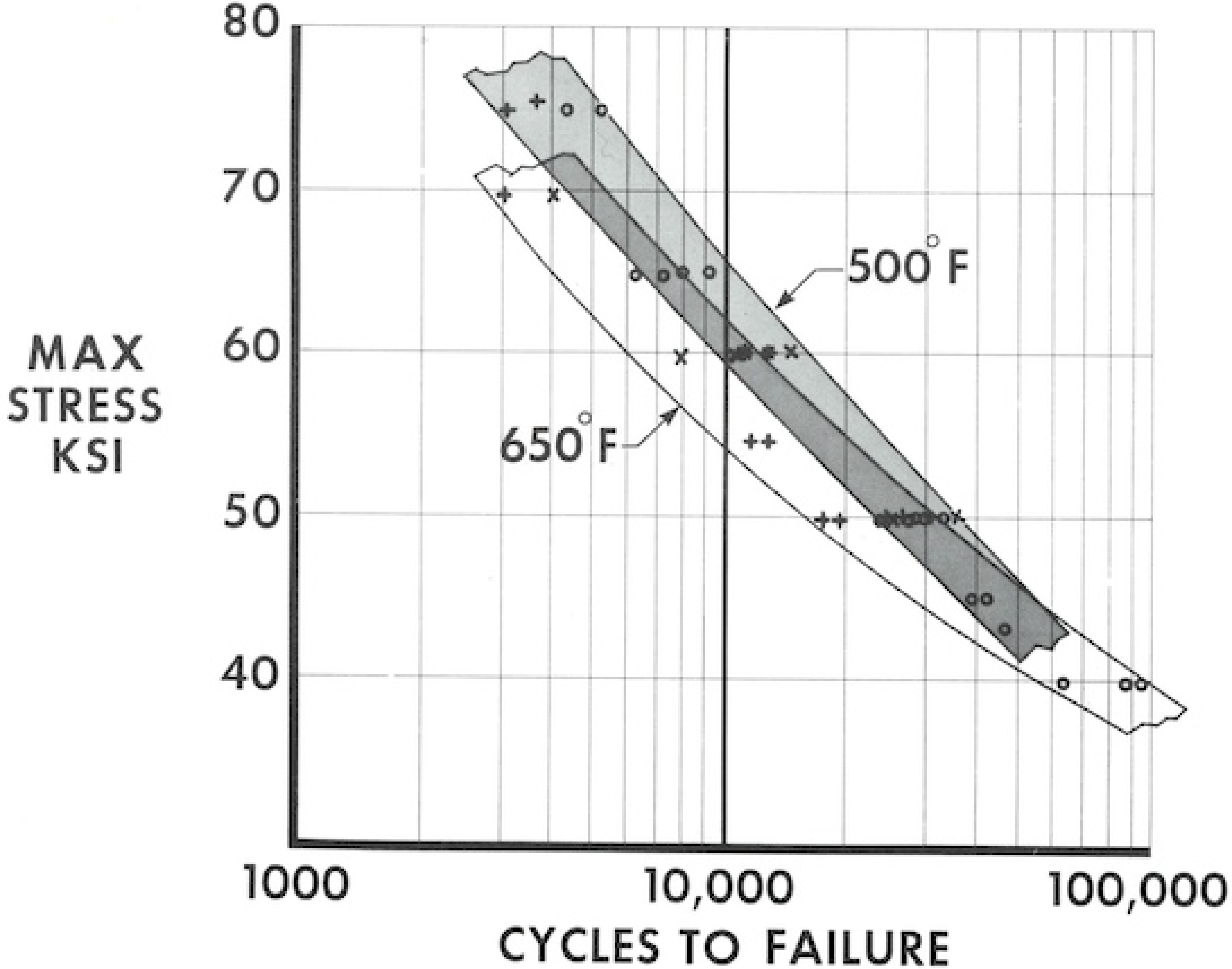
To be certain that the titanium structure will retain its structural integrity for the life of the airplane under the elevated temperature environment, we have conducted extensive fatigue tests on titanium 8-1-1 material. These tests consist of subjecting the specimen to a load of 40,000 pounds per square inch while the specimen is heated to 500°F. (In some tests a temperature of 650°F has been used.) The specimens are loaded and placed in the elevated temperature environment and exposed for an extended period of time. After 2000 hours and 5000 hours exposure test specimens were removed and fatigue tested at various stress levels.

The chart compares elevated temperature fatigue results on specimens that have not been exposed with specimens that have been exposed for 2000 hours of temperature and stress, and with specimens exposed at 5000 hours of temperature and stress. The test data shows that there was no deterioration of fatigue properties for the specimens that had been exposed for 2000 hours and 5000 hours at either the 500°F temperature or the 650°F temperature. We have just recently removed specimens which had been exposed to the elevated temperatures and stress for a period of 10,000 hours. NASA test data on materials exposed for 10,000 hours have not shown a decrease in fatigue life over unexposed specimens. We therefore conclude that we have sufficient information to design an airplane that will be satisfactory for the prototype program. As test specimens acquire approximately 9000 hours per year, in three years we will have sufficient information to evaluate this material for the life of the production airplane.

FATIGUE LIFE



TEST SPECIMEN
TITANIUM 8-1-1
DUPLEX ANNEALED
GAGE .040
R .06



EXPOSURE & TEST TEMPERATURE
500° F
650° F

SPECIMEN LEGEND
- ∘ UNEXPOSED
- + 2000 HRS
- × 5000 HRS

CE6679 3-65

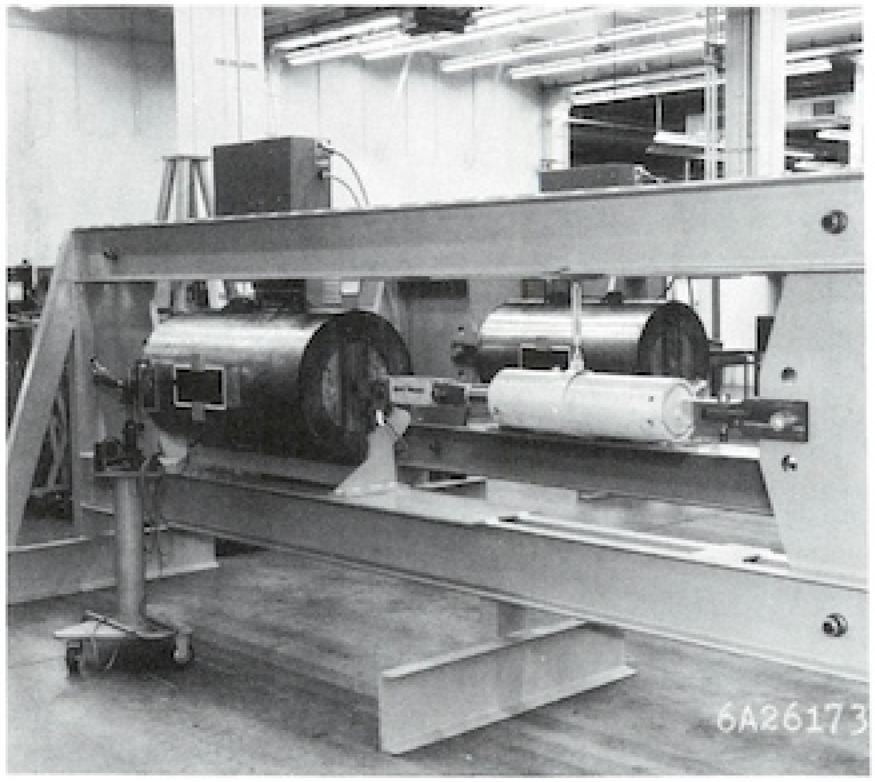
Fatigue and Fail Safe Design

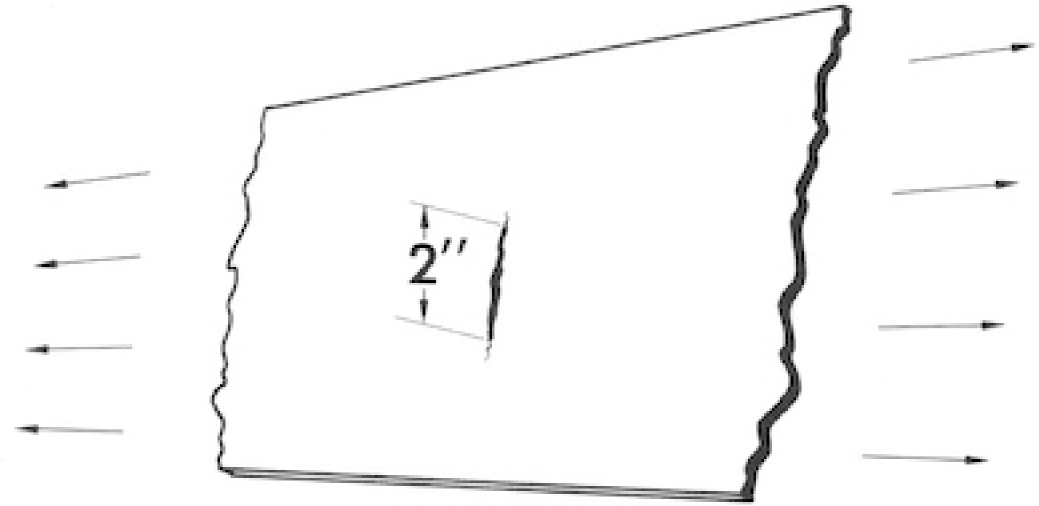
We are very much concerned about the tear resistance of titanium alloy at the SST operating temperatures. The tear resistance chart compares the tear resistance of various structural materials at sub-zero, room, and elevated temperatures. The specimen receives a 2 inch saw cut and is then fatigue tested until a crack starts from each edge of the saw cut. The specimen is then placed in the test temperature environment and subjected to a tension load. In this chart the gross stress at failure is divided by the density of the material so that the materials may be compared on a uniform weight basis. The chart shows that 24ST aluminum alloy, stainless steeel, 2618-T81 aluminum alloy (which the British are using on the Concorde) and titanium 8-1-1 have favorable test values at room temperature at Mach 2, which is roughly 250°, there is some drop in properties for 2024 aluminum alloy, but titanium 8-1-1 shows no drop in strength characteristics up to 500°F which would indicate that this is a favorable material for the supersonic airplane.

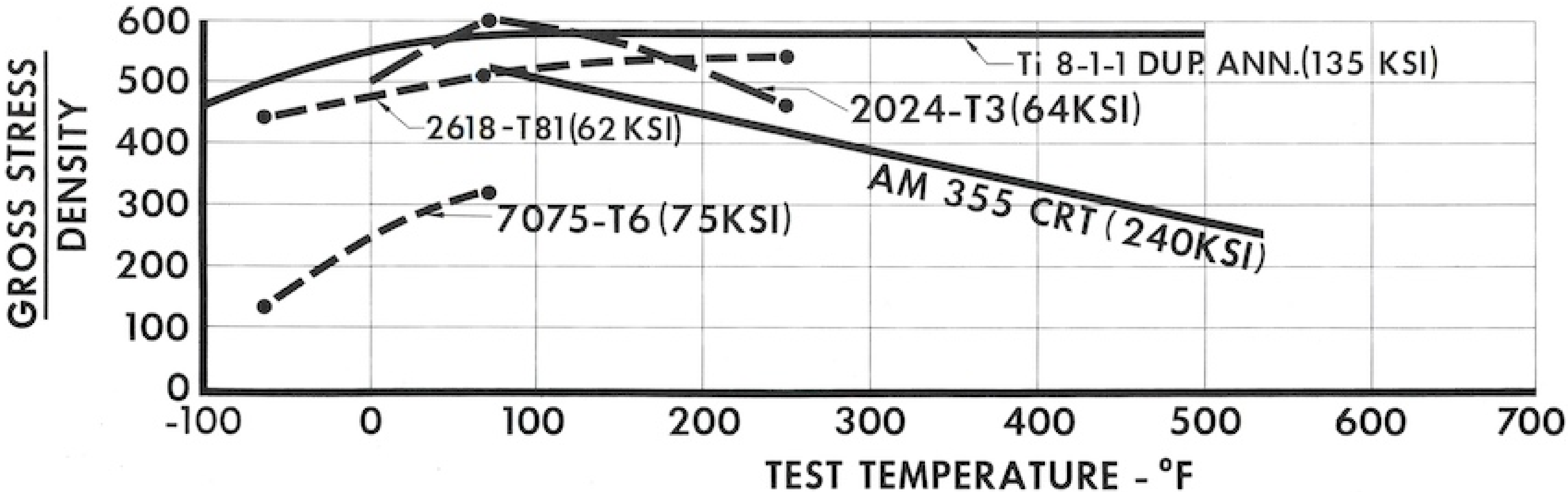
CE-6673

FATIGUE AND FAIL SAFE DESIGN

TEAR RESISTANCE TESTING







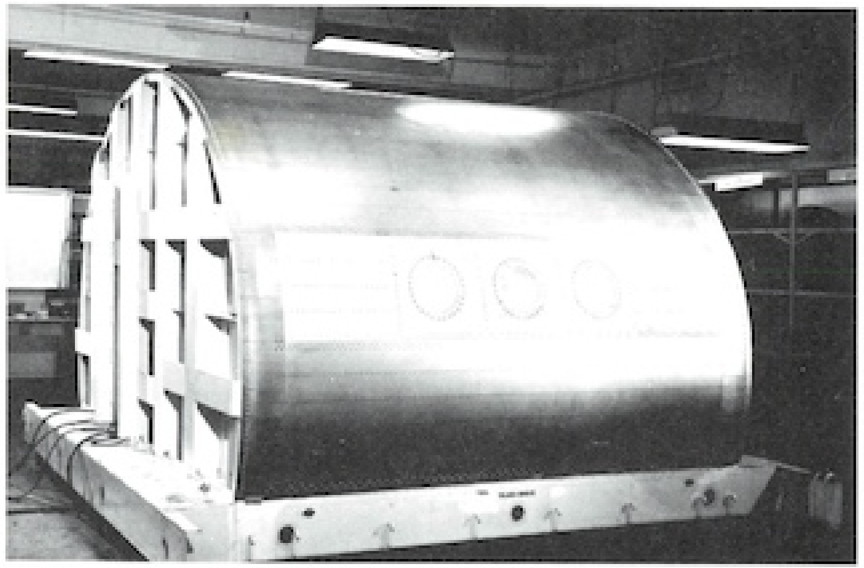
Crack Growth

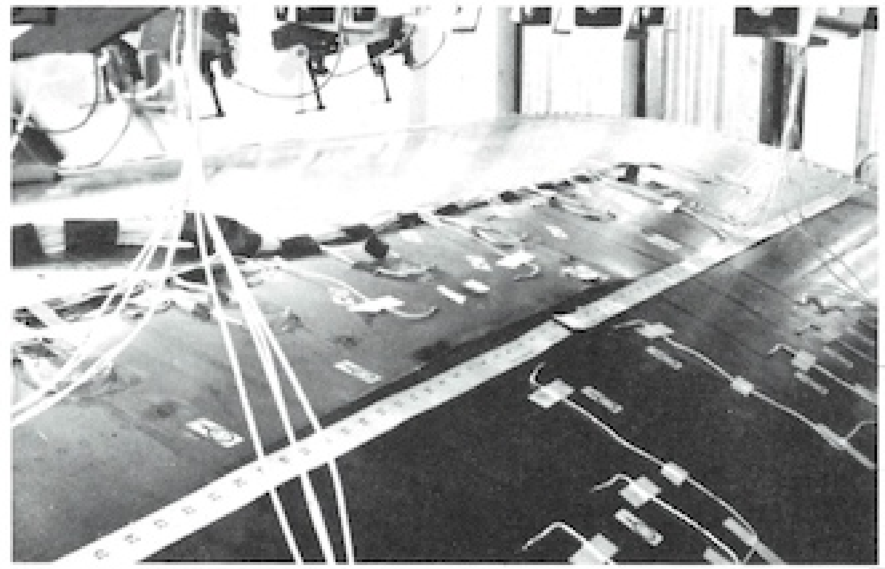
The full size body section is being tested to assure that we have fail safe body design. The chart shows a photograph of the body section before it was tested. The test consisted of cutting a 6-inch long crack in the body skin which is .045 inch thick titanium 8-1-1. We then applied 700 cycles of 12 psi pressure. As very little crack growth was obtained, the saw cut was lengthened to 8 inches and 300 additional pressure cycles were applied to the body. Again there was very little crack growth so the saw cut was lengthened to 12 inches and additional cycles were applied. The curve shows that the crack gradually increased in length to approximately 39 inches when the test was discontinued. It should be noted that even though the total crack length was in the vicinity of 30 inches, one pressured cycle increased the crack length only a very small amount. This shows that if a crack 30 inches in length existed in an airplane fuselage, the crack would not grow catastrophically but can be detected long before it extends to any serious length. Thus, we have a high degree of fail safety in our body design.

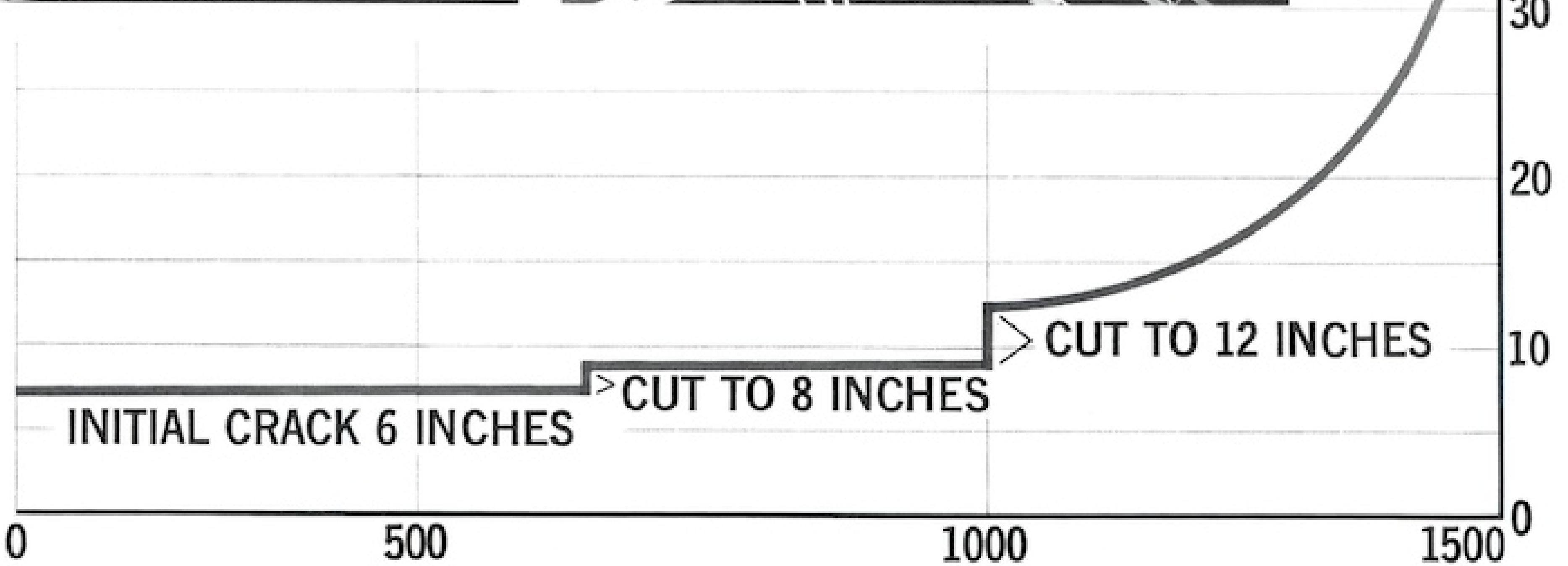
A similar test has been performed on .032 skin with tear stoppers spotwelded to the skin every 20 inches. In this case with the application of hundreds of cycles, the crack progressed slowly out to the tear stopper and did not progress beyond the tear stopper. This shows a high degree of fail safety in our minimum gage proposed body design.

CE-6540

CRACK GROWTH
TI-8-1-1 BODY SECTION





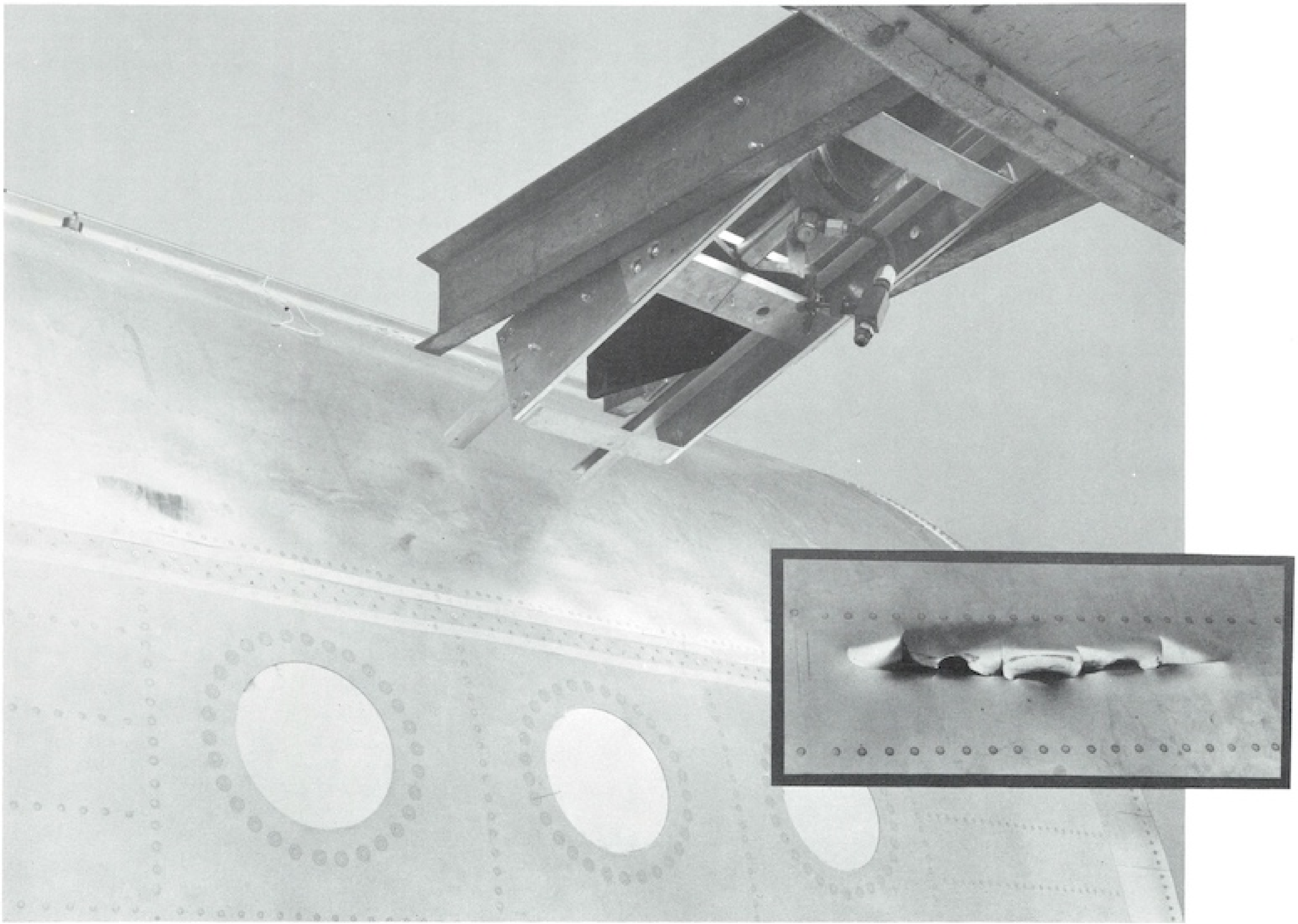


Guillotine Tests

The photograph shows the knife test in which we subjected the body to a pressure of 12 psi and then shoot a 12 inch wide knife blade into the pressure body section. Although the body skin was only .032, the cut was limited to a total length which is only slightly greater than the width of the knife blade. A second test was conducted in which the knife blade cut through a body stiffener and the same results were obtained. This shows that if the body subjected to in-flight impingement of some foreign object that catastrophic failure would not occur.

CE-6678

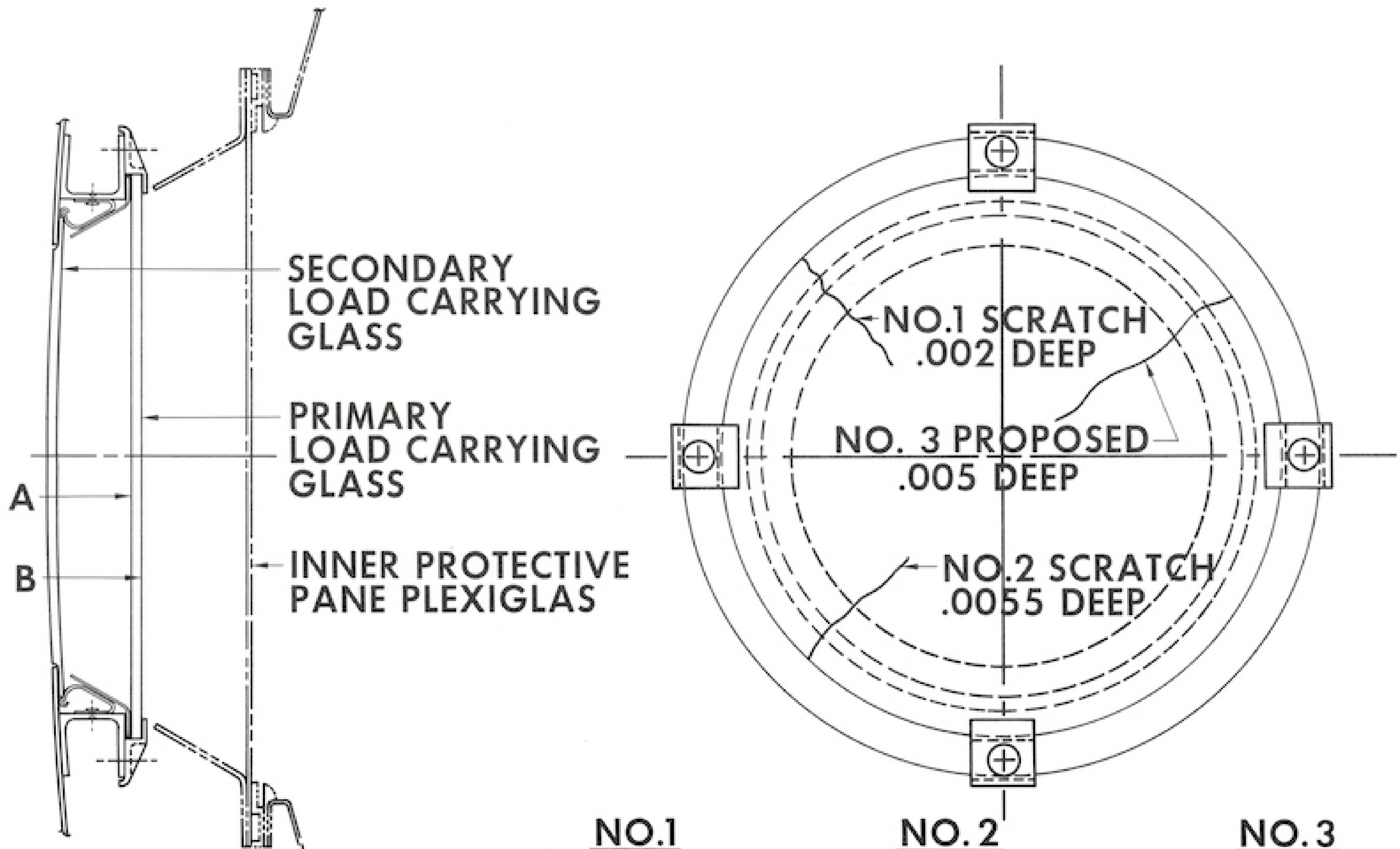
GUILLOTINE TEST



Passenger Window Scratch Test

The window design consists of an inner protective pane of plexiglass, a center pane of load carrying glass, and an outer load carrying pane of glass. A window installation consisting of the outer two load carrying panes was subjected to 36 psi which is three times the maximum pressure we anticipate with no failure. The inner primary load carrying pane was struck with a sharp point while the window installation was subjected to 12 psi and the pressure was transmitted to the outer pane and it carried the pressure. A new window was installed on the inner primary load carrying pane and was subjected to a scratch .002 deep and subjected to 1050 cycles with no indication of scratch growth. A second scratch was applied to the window that was .0055 deep and the window was subjected to 1000 cycles of 12 psi pressure with no increase in scratch length. This demonstrates that our window design has excellent damage resistance.

PASSENGER WINDOW SCRATCH TEST

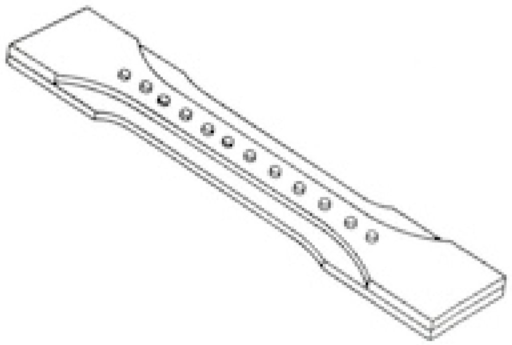


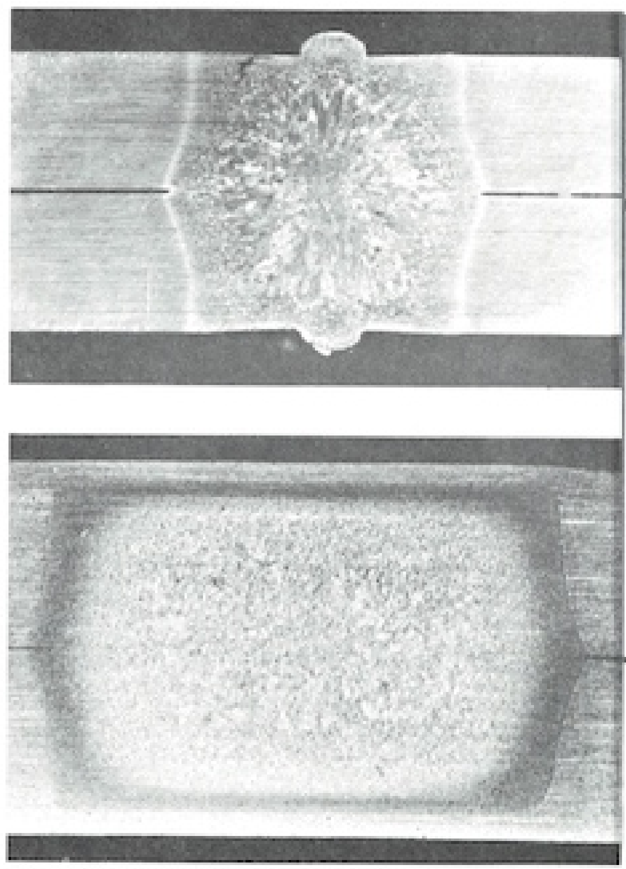
NO.1	NO.2	NO.3
1050 CYCLES +1000 CYCLES SURFACE (B)	1000 CYCLES SURFACE (B)	PROPOSED TEST SURFACE (A)

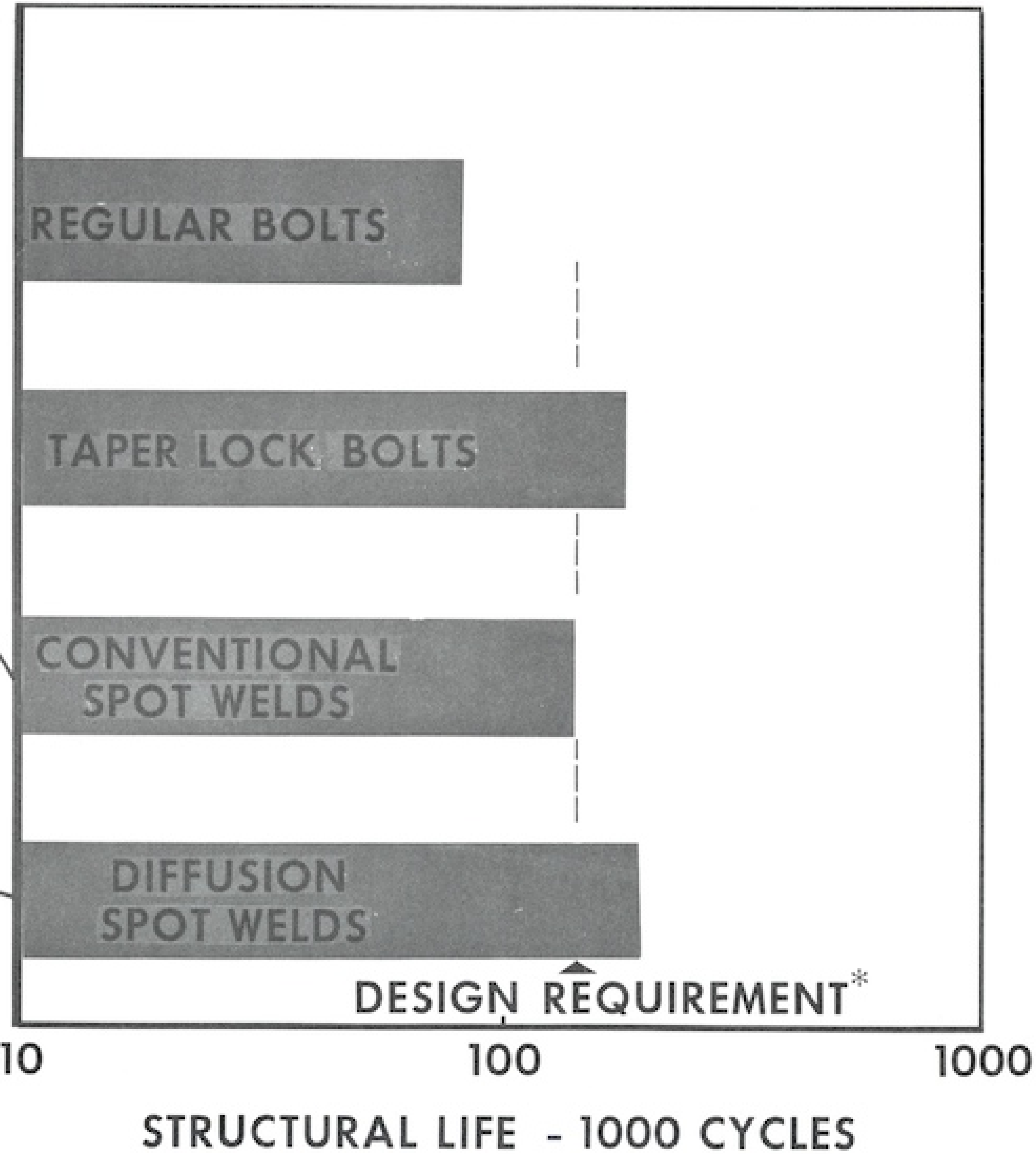
Fatigue Performance

We are performing an extensive amount of testing on various joining methods to obtain the desired fatigue performance by the most economical method. This chart shows that regular bolts do not meet our design requirements, conventional spotwelds meet our design requirements, and taper lock bolts and diffusion spotwelds exceed our requirements from a fatigue standpoint. We are also performing extensive tests on various methods of riveting. It should be noted that the design requirement is two times the airplane fatigue life. This is done to obtain high structural integrity in design.

FATIGUE PERFORMANCE
JOINING METHODS







* 2 TIMES AIRPLANE FATIGUE LIFE

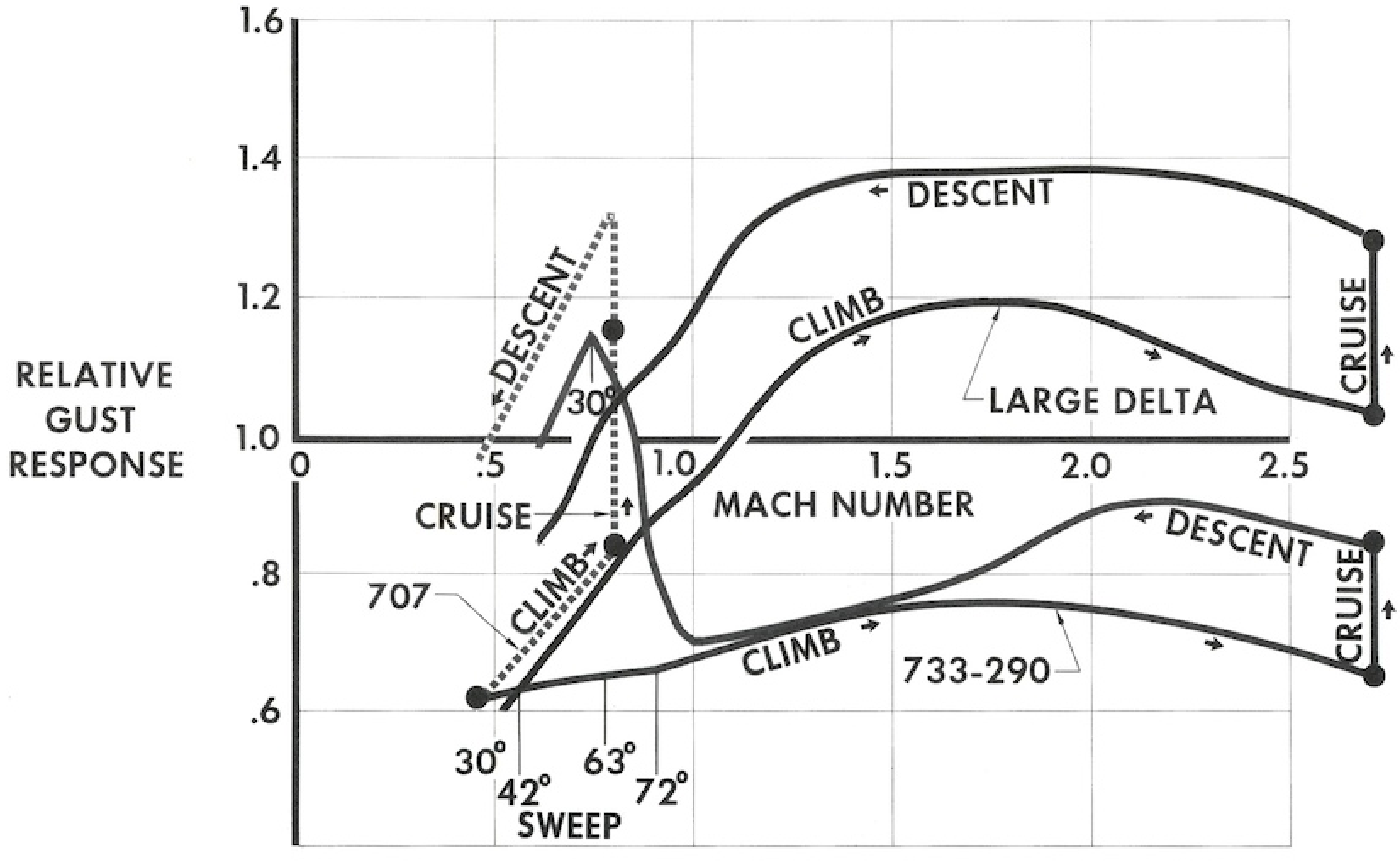
CE6547 3-65

Gust Load Factor Comparison

This chart provides a comparison of the response of three airplane configurations to a gust of the same magnitude. The response of the 707 is shown as a dotted line and is approximately .6 at takeoff, increases during climb to approximately .85, and then increases during cruise as the airplane becomes lighter. The average during the cruise condition for the 707 will be a relative gust response of 1.0. The 733-290 which is the lower line, starts out with an airplane response of approximately .6. This varies slightly during climb and then increases during cruise condition; however, the average for the cruise condition is a response of .75. In comparison with the 707, the swept wing airplane will have an effective response of 75% of the response of the 707 airplane. Thus, the swept wing airplane will provide a much smoother ride and have less cyclic fatigue damage in the cruise condition than the 707 airplane.

The upper solid line shows a large delta wing airplane response to a given gust. In this case the response is approximately .6 in the takeoff condition and increases during climb with increasing Mach number. During the cruise condition the airplane response is approximately 1.15. A large delta wing would therefore provide a ride that is 15% rougher than the 707 and approximately 50% rougher than the swept wing airplane. These values are for the same gust intensity and do not account for the reduction in equivalent gust velocity that is attained at the higher altitudes of the supersonic airplanes. This reduction in gust intensity is shown on the following chart.

GUST LOAD FACTOR COMPARISON



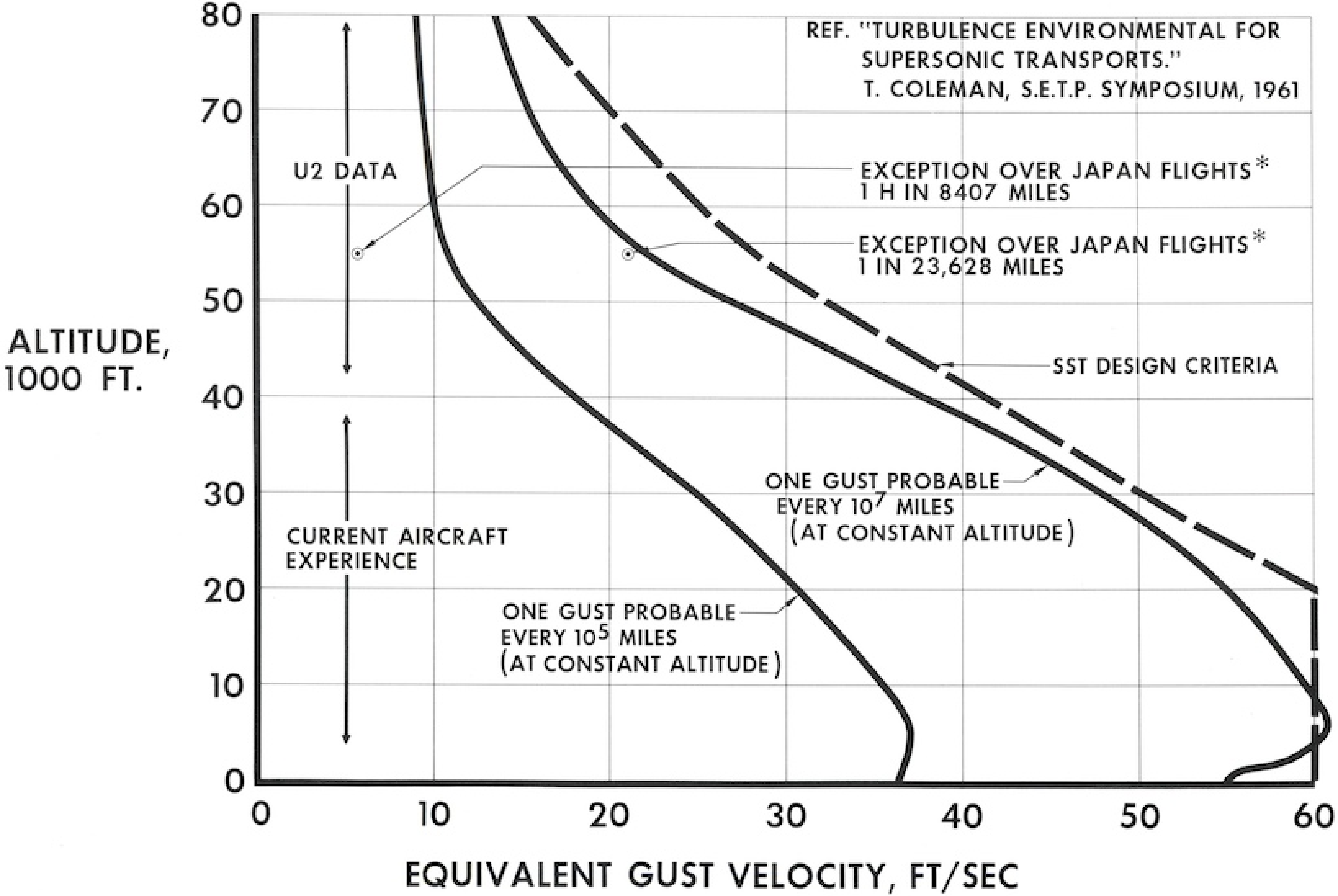
Gust Intensity

The equivalent gust velocity has been plotted for various altitudes. The high altitude information is based on U-2 flight data. This curve shows that the gust velocity is substantially reduced at the higher altitudes which represent the cruise condition for the supersonic airplanes.

Structural Testing Summary

In summary, we are performing an extensive amount of testing to assure that we will have a high degree of fail-safety and that the structure will provide satisfactory performance for the life of the airplane.

GUST INTENSITY



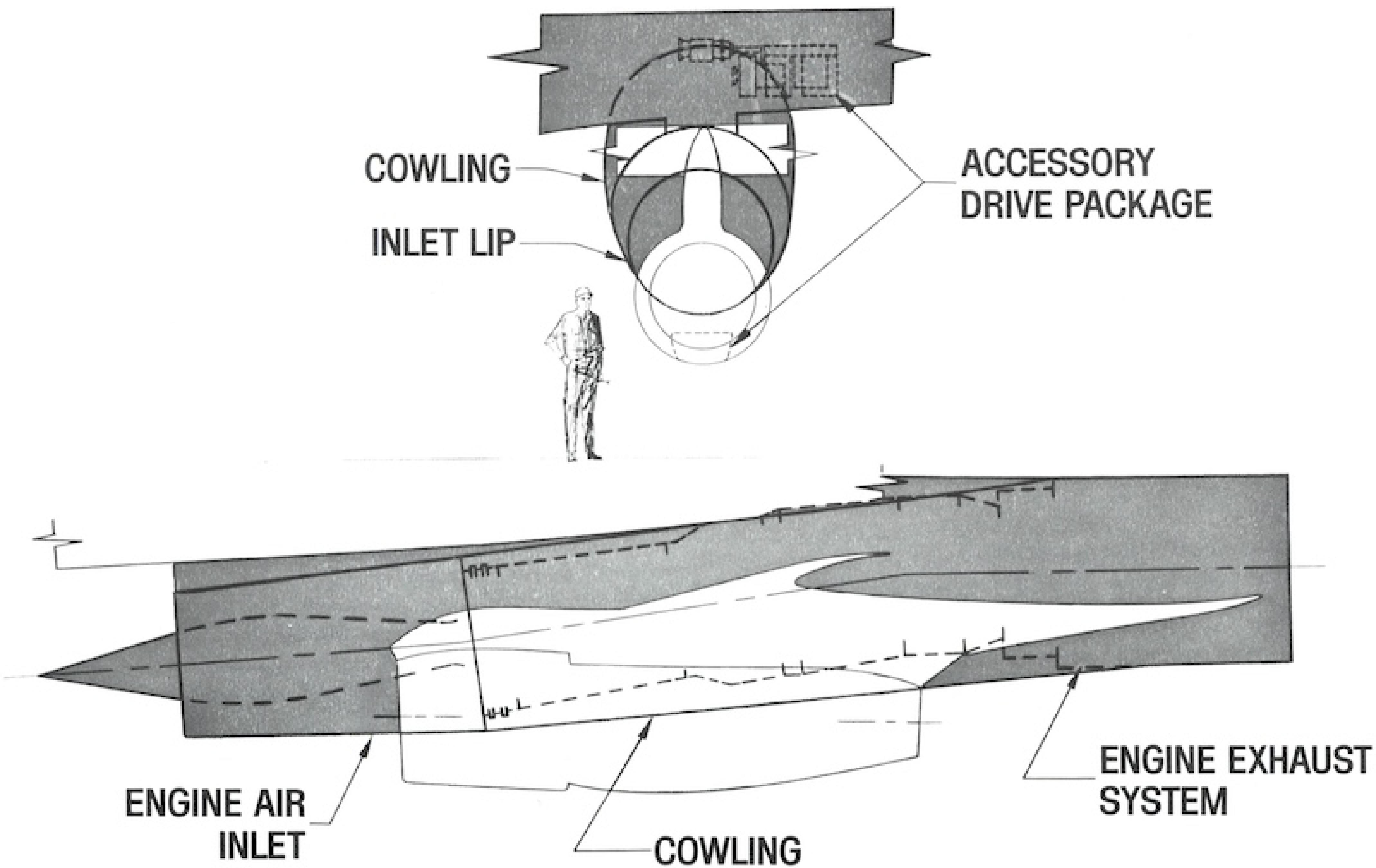
*(REF NASA TN D 548)

CE6670 3-65

The engine installation on the SST is compared on the inboard nacelle on a 707. The size differences are readily apparent. In general, the supersonic inlet is considerably longer than the subsonic inlet, and the engine exhaust nozzle system is considerably larger. The maximum static dry thrust is about twice as great. On the current subsonic aircraft all-accessory drives are on the engine and on the SST design the airplane accessory drives are in the wing. The engine accessories are placed such that no fuel accessories are on the bottom of the engine, thus, minimizing the chances of fire in case of a wheels-up landing.

CE-6634

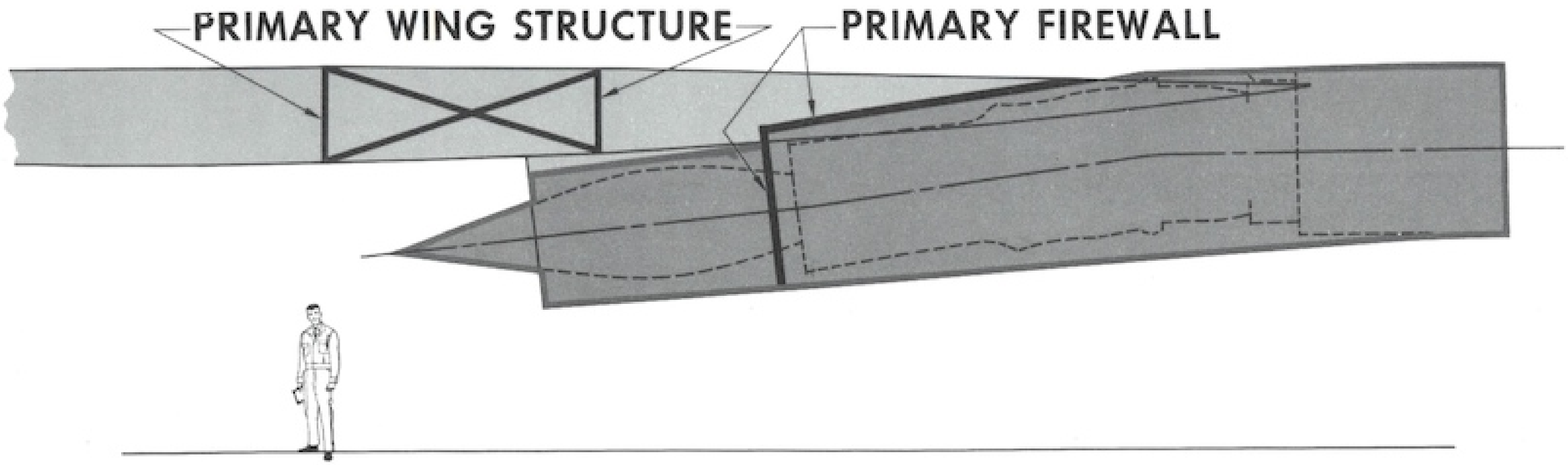
ENGINE INSTALLATION



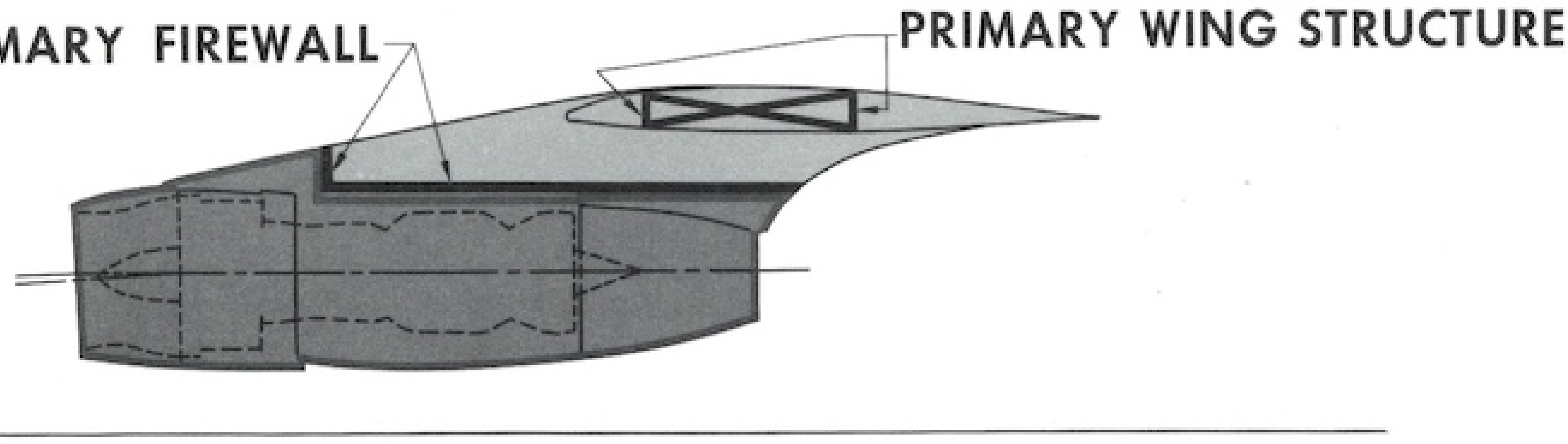
Pod safety under fire conditions appears to be similar to that found in subsonic aircraft today. As in the subsonic aircraft, the maximum wing structure is isolated from the aircraft by distance and firewall provisions.

CE-6667

POD COMPARISON-FIRE PROTECTION



733-290 POD



707-320B POD

The most dangerous hot areas on the airplane, from the viewpoint of hydraulic fluid safety, are the brakes and the engine compartment. These conditions are expected to be quite similar to conditions on the subsonic aircraft. While the vented compartments on the SST will be at about 460°, these are not considered critical in considering the available high temperature hydraulic fluids. A comparison of the characteristics of fluids suitable for subsonic aircraft and for the SST indicates very samll differences in the fire safety characteristics.

HYDRAULIC FLUID COMPATIBILITY

	SUBSONIC JETS	SST
ENVIRONMENT		
HIGH TEMP. AREAS - °F (BRAKES & ENGINES)	1200 TO 1500	1200 TO 1500
FLIGHT CONTROL AREAS - °F	-40 TO 160	-40 TO 460
FLUID PROPERTIES		
IGNITION TEMPERATURES - °F	1100 -1500	900 - 1400

The safety of the control systems for the SST is considered comparable to present subsonic aircraft. For example, the 727 has two completely independent hydraulic supply systems and actuators on the elevator control with manual reversion as a third system. Because the SST elevator hinge moments are too large for manual reversion, a third hydraulic system is incorporated.

HYDRAULIC SUPPLY

3 INDEPENDENT SYSTEMS

2 PUMPS / SYSTEM ON DIFFERENT ENGINES

SURFACES

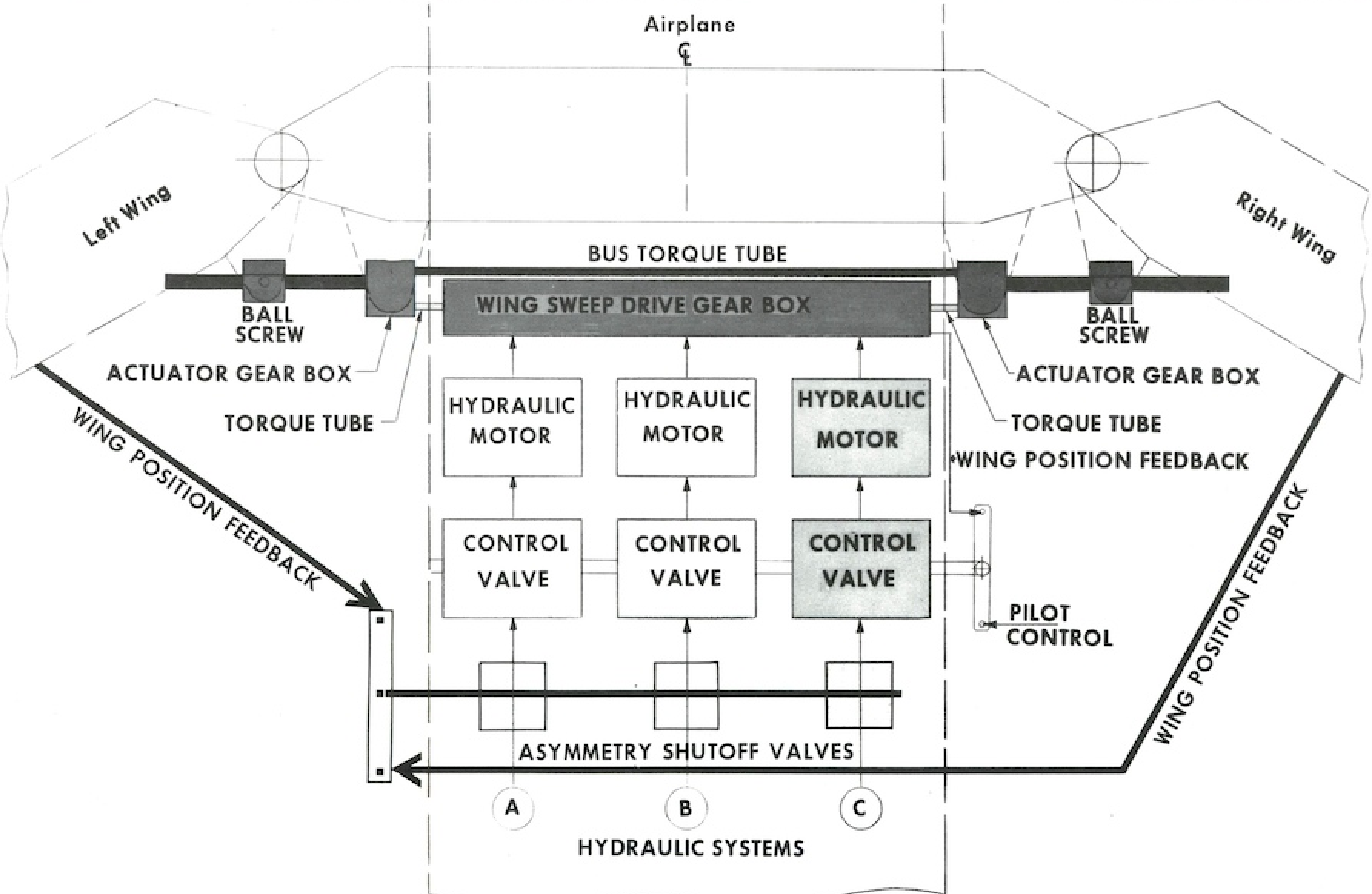
3 HYDRAULIC ACTUATORS

2 INDEPENDENT LOAD PATHS

ASYMMETRY SHUT OFF SYSTEM

The wing sweep actuation and the primary flight control systems are high redundant designs. Three hydraulic systems in the aircraft providing a triple power source drive independent hydraulic motors which drive a gearbox which in turn drives the ball screw wing sweep actuator. A buss torque tube provides for driving both actuators if one primary torque tube fails. The wing sweep drive gearbox torque tube and ball screw have dual structural load paths in each. The flight control systems to all of the primary surfaces are similar in construction, that is; triple redundancy in the hydraulic system and dual redundancy in the structural load paths.

WING SWEEP SCHEMATIC

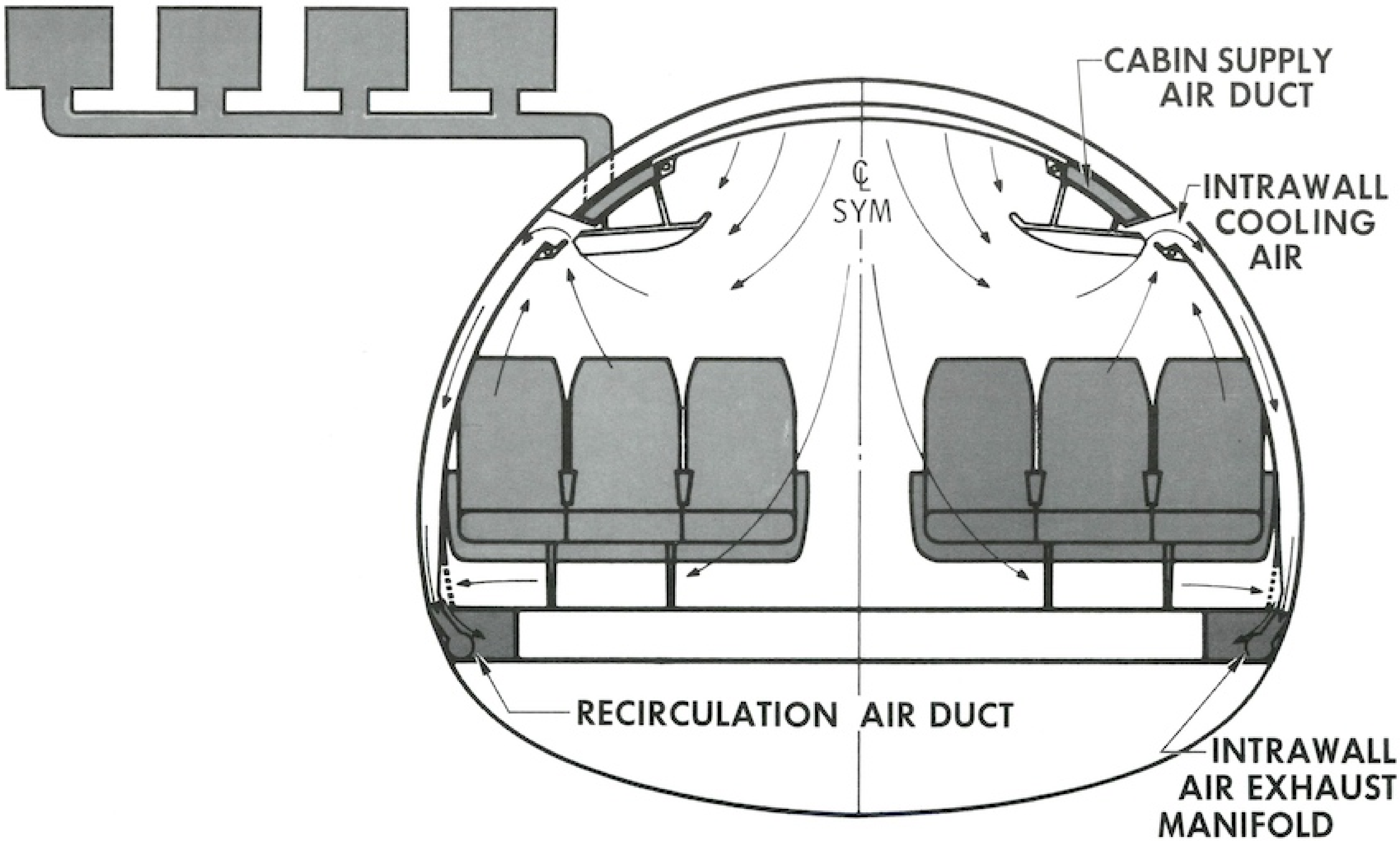


CE6028 -2 2-64

Cool air from four independent air conditioning packs is distributed through ducts and discharged into the interior through the ceiling. The air is exhausted from the cabin into a collection system under the seats and under the coat racks from whence it cools the wall and then is discharged overboard.

CE-6663

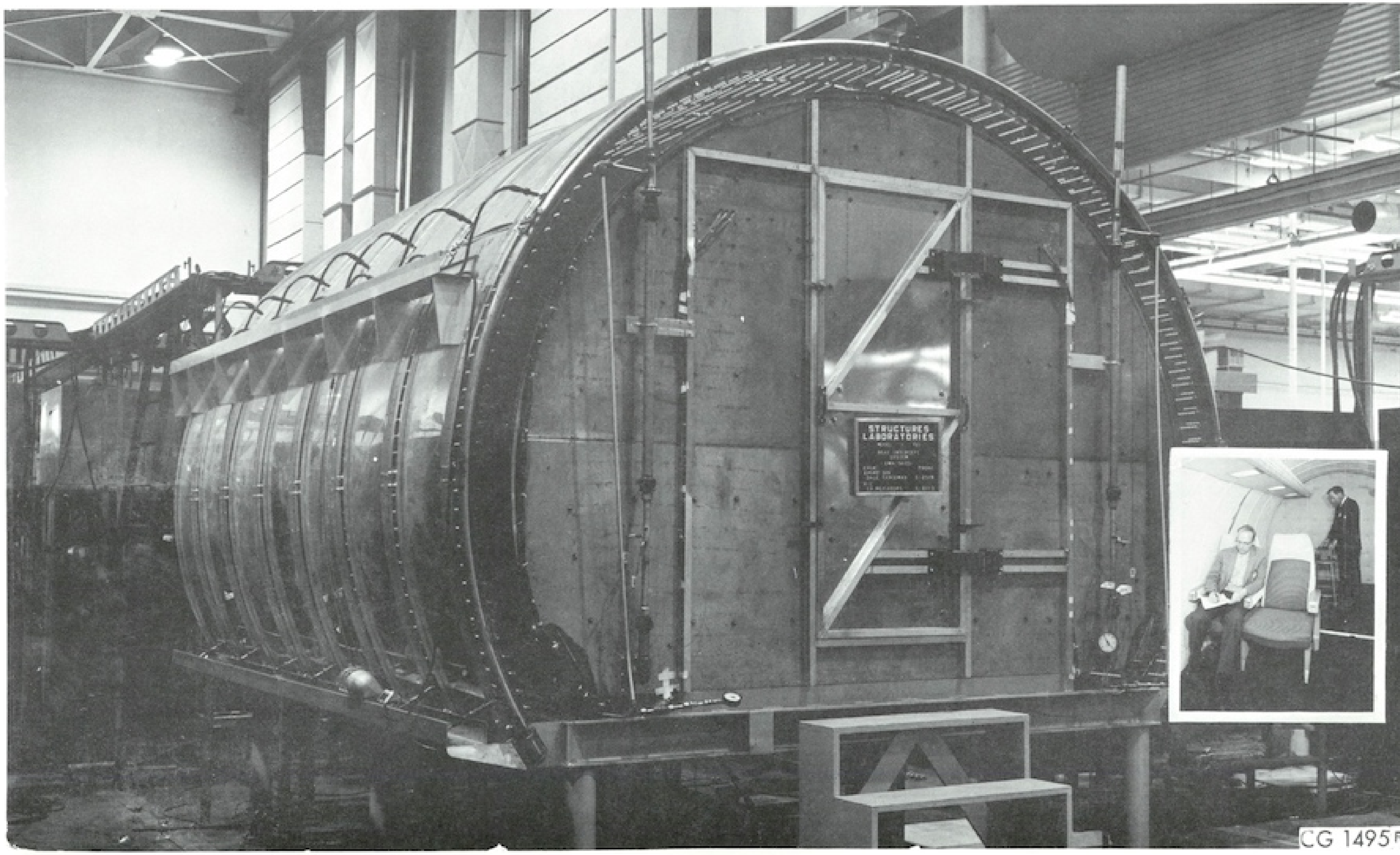
CABIN AIR DISTRIBUTION SYSTEM



This picture shows a representative section of the SST fuselage being treated by quartz lamps to an outside temperature of 500°F. The walls of the cabin are insulated and as you can see, people are inside reading data. From these tests it is expected that the thermal insulation from the outside environment will be no problem.

CG-1495R

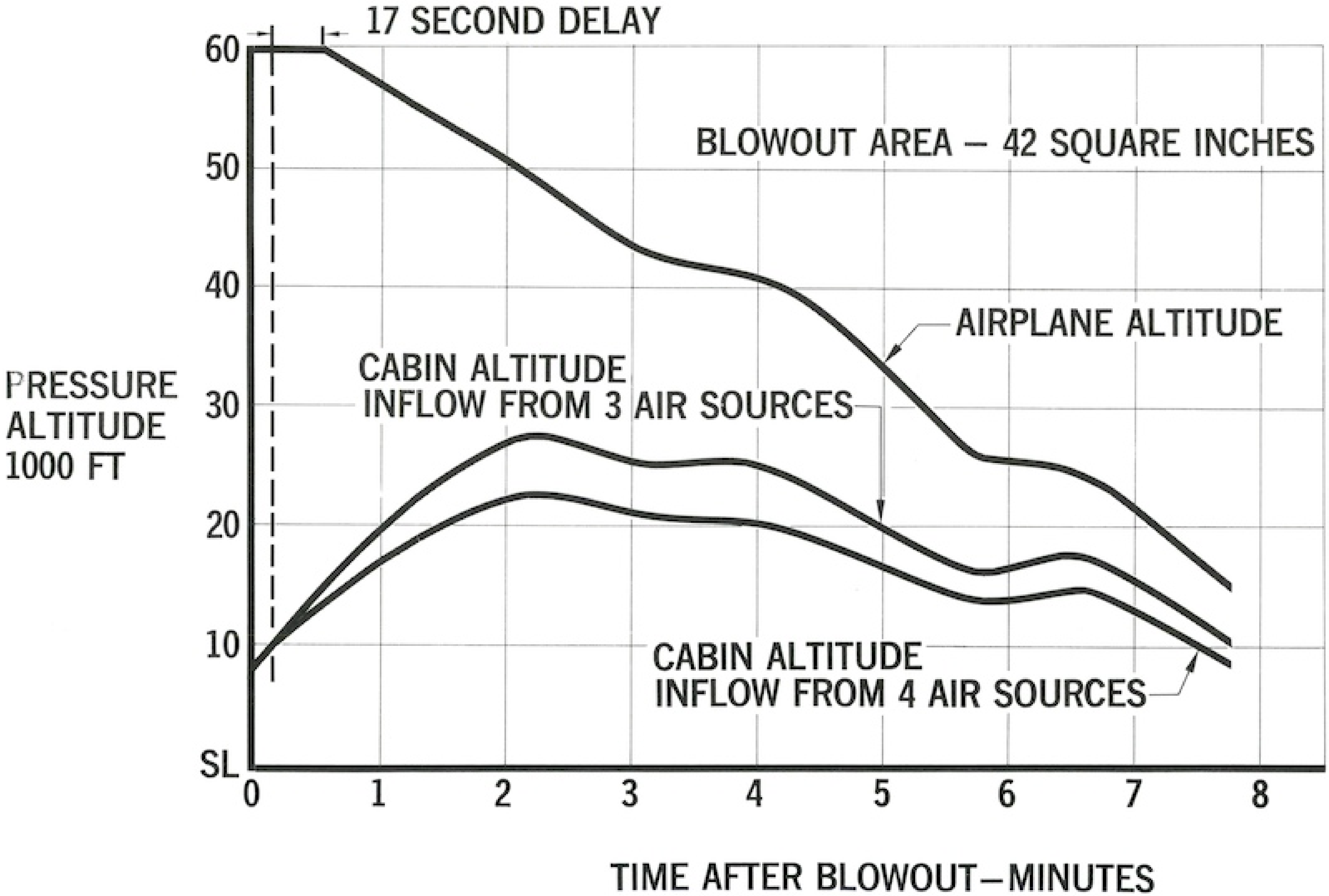
CABIN COOLING DEVELOPMENT



CG 1495R

As previously explained, titanium used in aircraft structure is a very tough material. However, should it be ruptured for any cause, stringers and frames will limit the size of the tear. The largest hole which we can foresee is approximately 42 square inches. This is larger than the window area. The windows cross-sectional area is approximately 30 square inches. With a 42 inch hole and the airplane cruising at its normal altitudes, the cabin pressure would rise in about 13 seconds to 10,000 feet at which time a warning would sound. At the end of half a minute, total assumed necessary to initiate descent, the airplane would descend along the altitude-time schedule shown. The cabin altitude would rise to approximately 23,000 feet and then descend paralleling the aircraft altitude. An oxygen system is provided for passengers and crew such that with it these cabin altitudes are considered to be quite safe even though only three air conditioning packs are functioning.

CABIN DECOMPRESSION

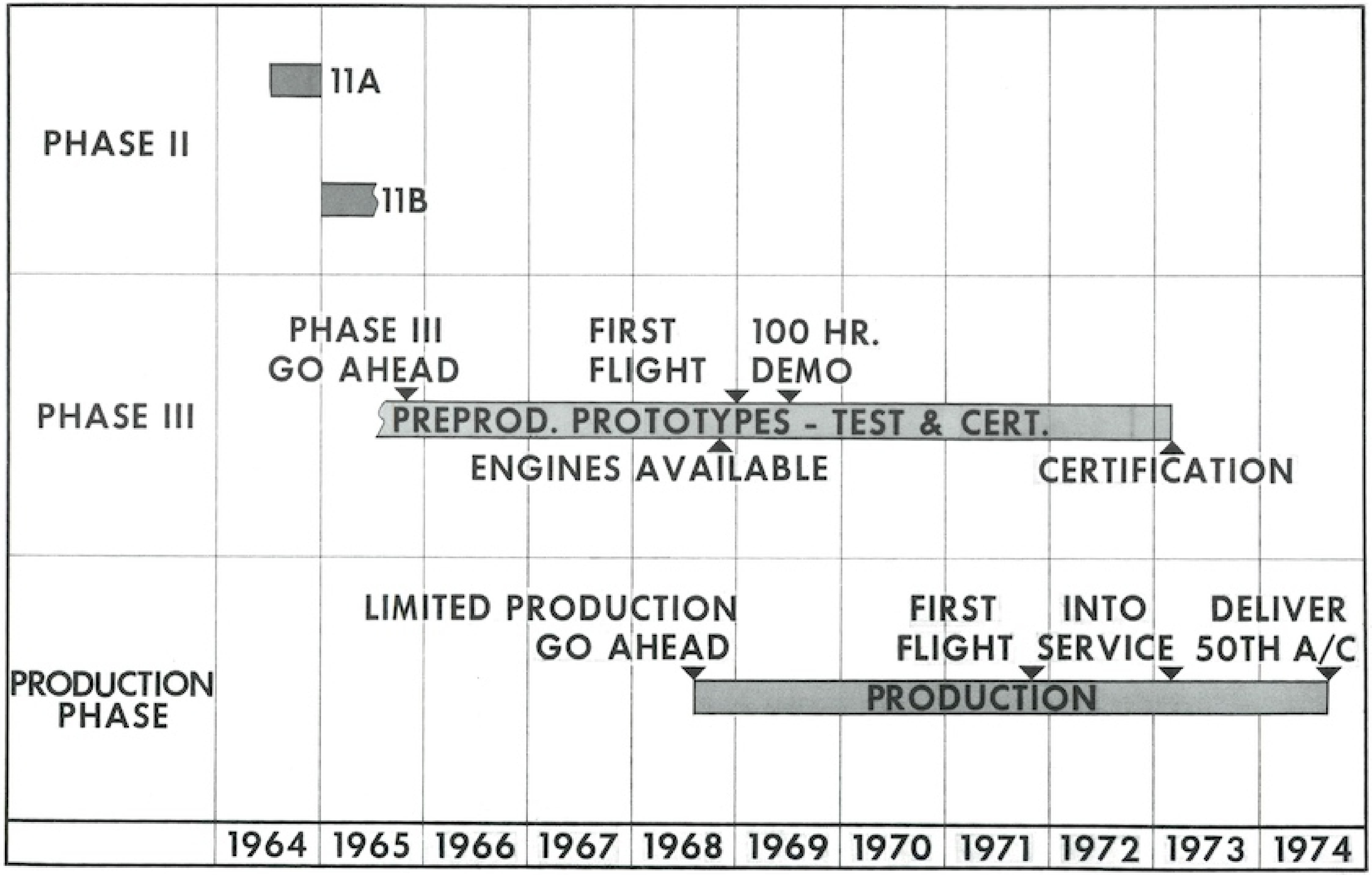


CE6609 11-64

This chart assumes go-ahead of Phase III in late 1965 with first flight of the prototype in late 1968 and first airline delivery in early 1973. It is recognized that slides may occur in the go-ahead date for Phase III but the time intervals between the milestones should remain fairly realistic. It is most important that any such slides in program timing be minimized so that introduction of the airplane into airline service will not follow the Concorde by such an extended period of time as to loose its market potential.

CE-6518R

SST PROGRAM PHASING



CE6518A 11-64

SST COST COMPARISONS

The next several pages contain unit operating cost comparisons of the airplanes shown on the facing page. To contrast the sizes the plan views are shown approximately to scale. For reference the 733-290 is 271 feet long.

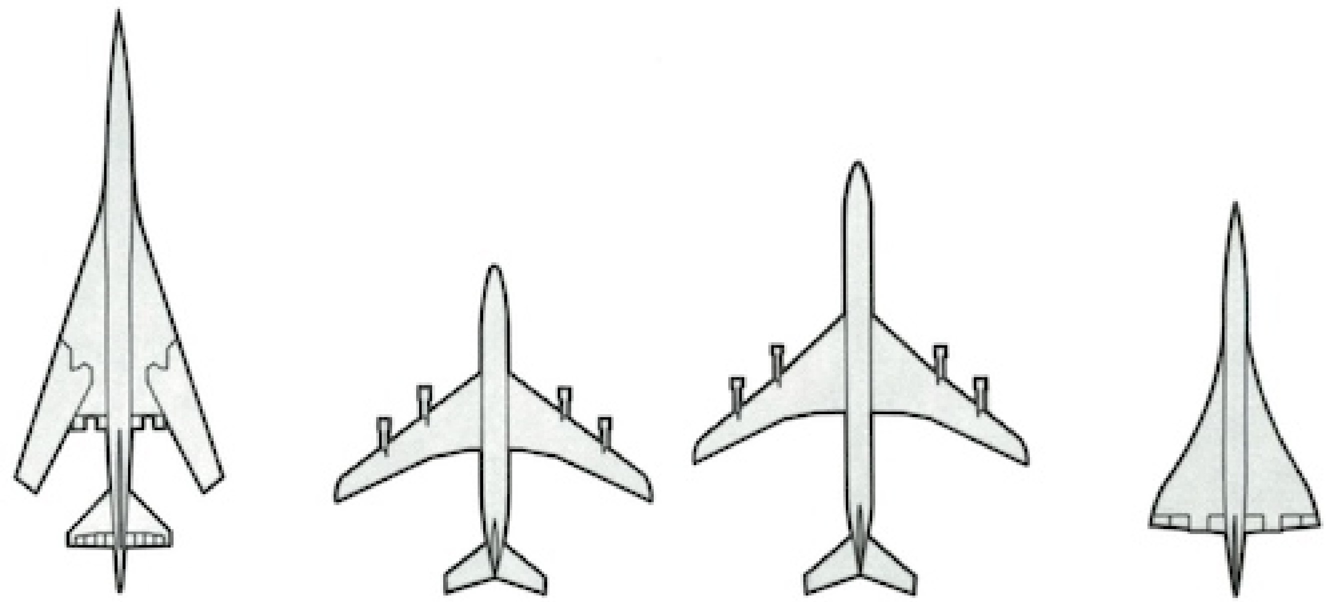
All comparisons have been made on the basis of the FAA Phase II-A international economic model. Use of the domestic model would produce similar comparisons. The FAA model called for subsonic airplane utilization of 3300 hours per year and a 12 year depreciation period. The model values for the supersonics were 3000 hours per year and 15 years.

In accord with the Phase II-A ground rules the 733-290 price is estimated on the basis of a 200 airplane production. It is given in terms of 1964 factor costs and 1964 dollars. The price does not include amortization of research and development. Amortization of research and development is included in the operating costs of the 733-290 which will follow.

The 6.96 million price of the 707-320B is based upon 1964 historical data. The price of the 707-820 is a current exercise value and the Concorde price of 14 million is our understanding of the current selling price of the airplane.

The fuel burned and block times used for the Concorde are based upon BAC/SUD data and are not the result of detailed performance analyses by Boeing.

ECONOMIC ANALYSIS ASSUMPTIONS



	SST 733-290	707-320B	707-820	CONCORDE
GROSS WEIGHT - LBS	500,000	328,000	412,000	326,000
TOTAL AIRPLANE PRICE - MILLIONS $	21.142	6.960	10.000	14.000
ENGINE PRICE - MILLIONS $/ENGINE	.980	.258	.335	.560

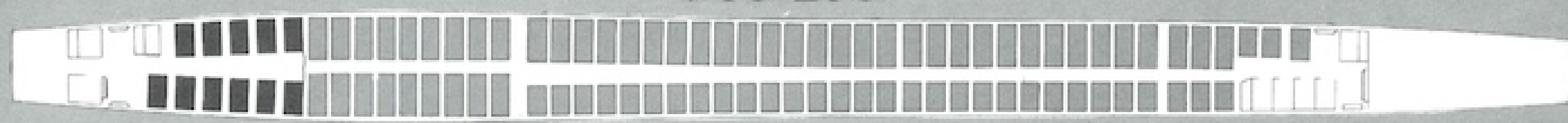
In order to compare unit economics it was necessary to put all of the interiors on a comparable basis, taking into account the differences in flight times where significant. All interiors were configured with the same seat pitches and with the same first class to tourist mix. Fewer meals were carried on the supersonics resulting in somewhat reduced galley volumes. The seat counts used are shown on the facing page.

INTERIOR ARRANGEMENT COMPARISON

INTERNATIONAL

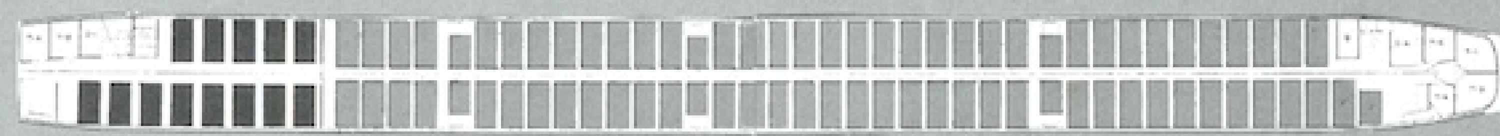
PASSENGER MIX: 10% FIRST CLASS 90% TOURIST
SEAT PITCH : 40" FIRST CLASS 34" TOURIST

733-290



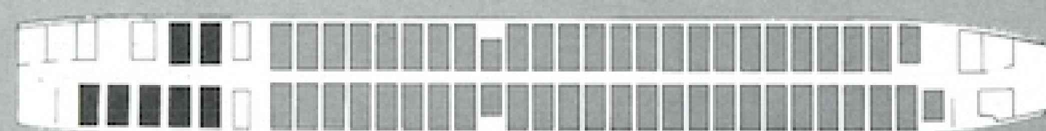
22 FIRST CLASS 208 TOURIST 230 TOTAL

707-820



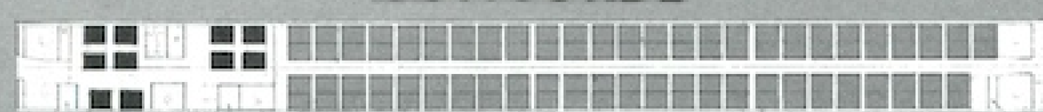
26 FIRST CLASS 221 TOURIST 247 TOTAL

707-320B



14 FIRST CLASS 141 TOURIST 161 TOTAL

CONCORDE



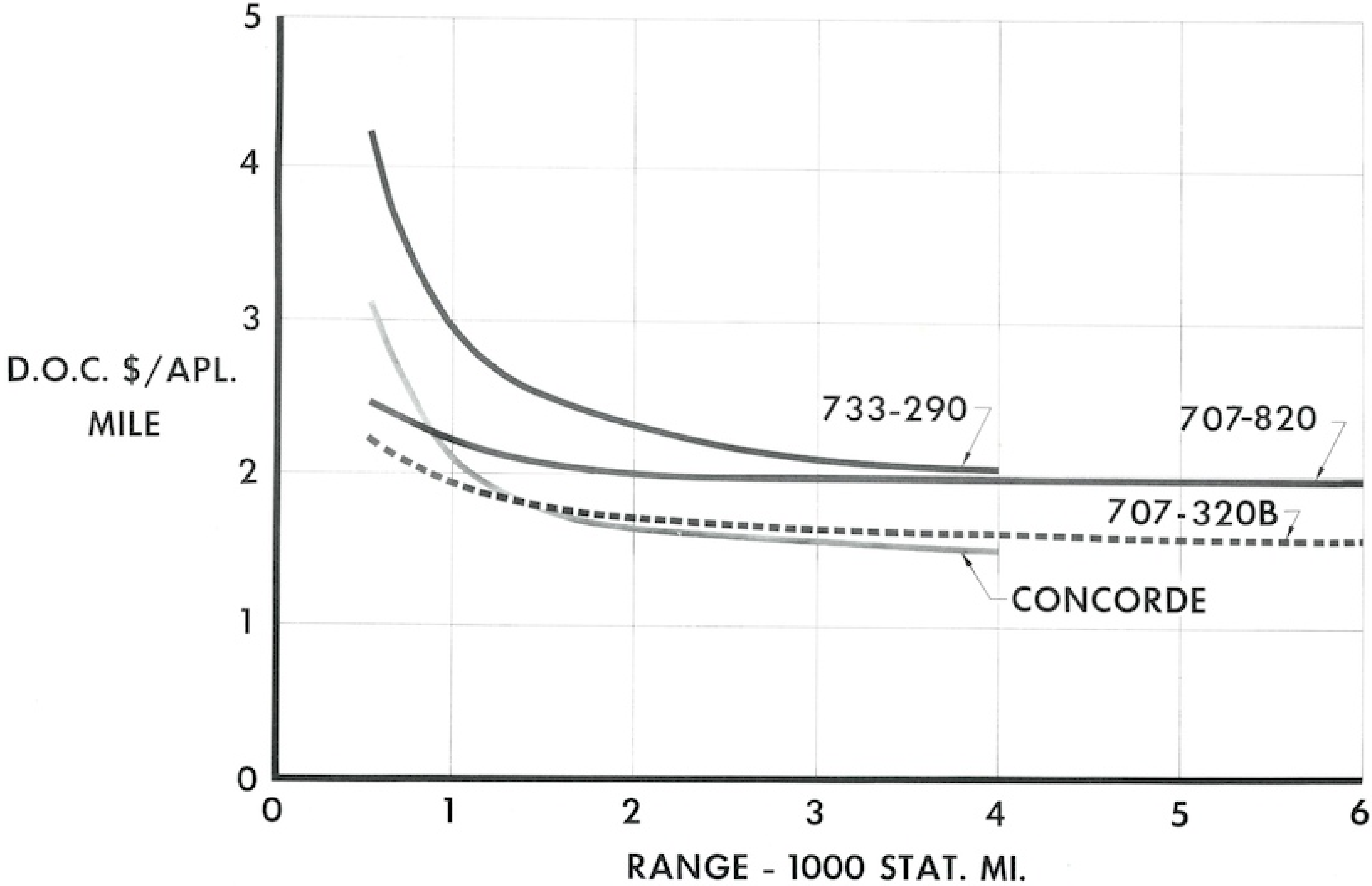
10 FIRST CLASS 102 TOURIST 112 TOTAL

The page opposite and the following two pages show direct operating cost comparisons.

The small tube Concorde has the lowest per airplane mile costs.

D.O.C. AIRPLANE MILE COMPARISON

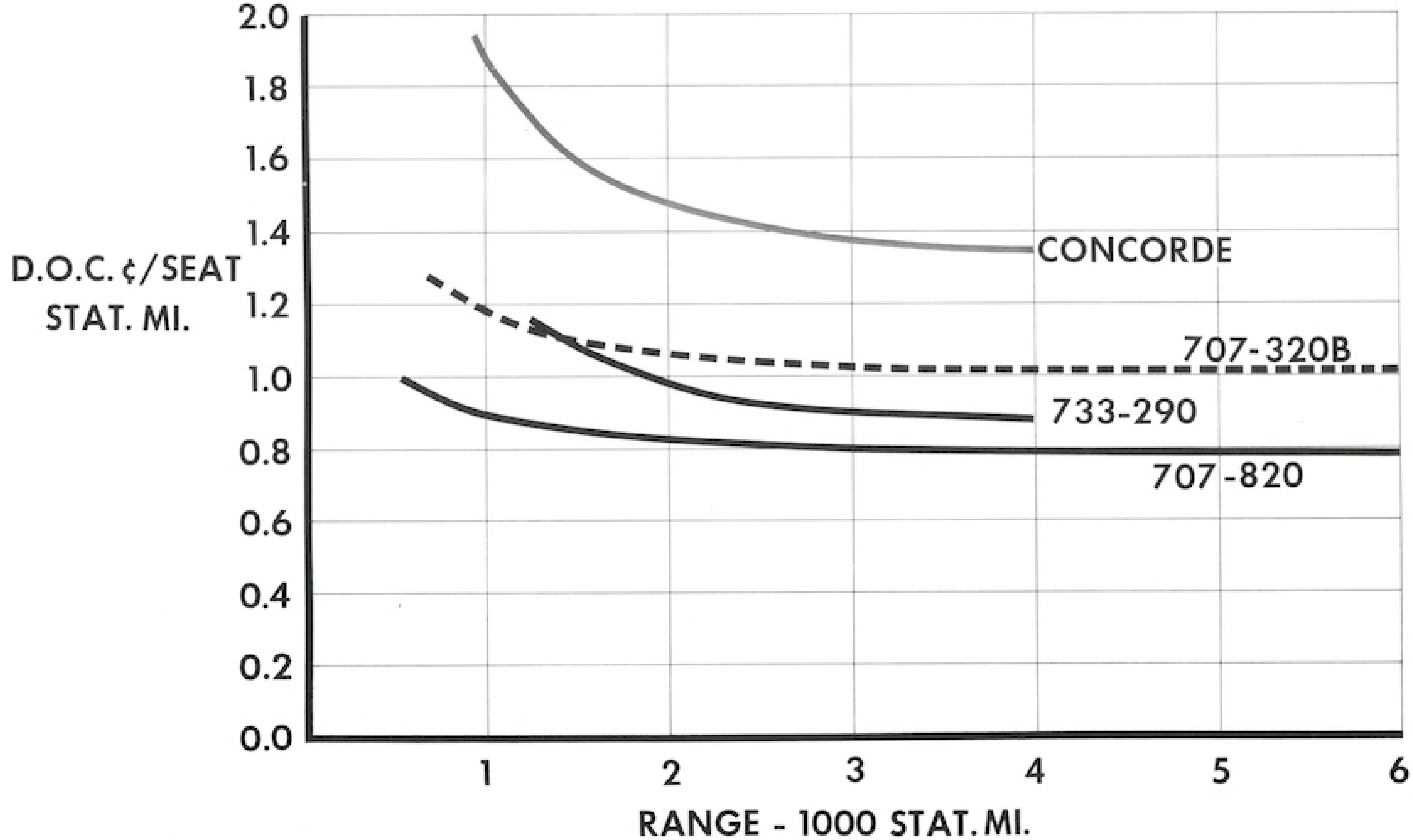
PHASE IIA INT'L. ECON. MODEL



CE6689 3-65

On a seat mile basis, however, the 707-820 is the lowest cost. It is approximately 10% lower than the 733-290.

D.O.C. - SEAT MILE COMPARISONS
PHASE IIA INT'L. ECON. MODEL

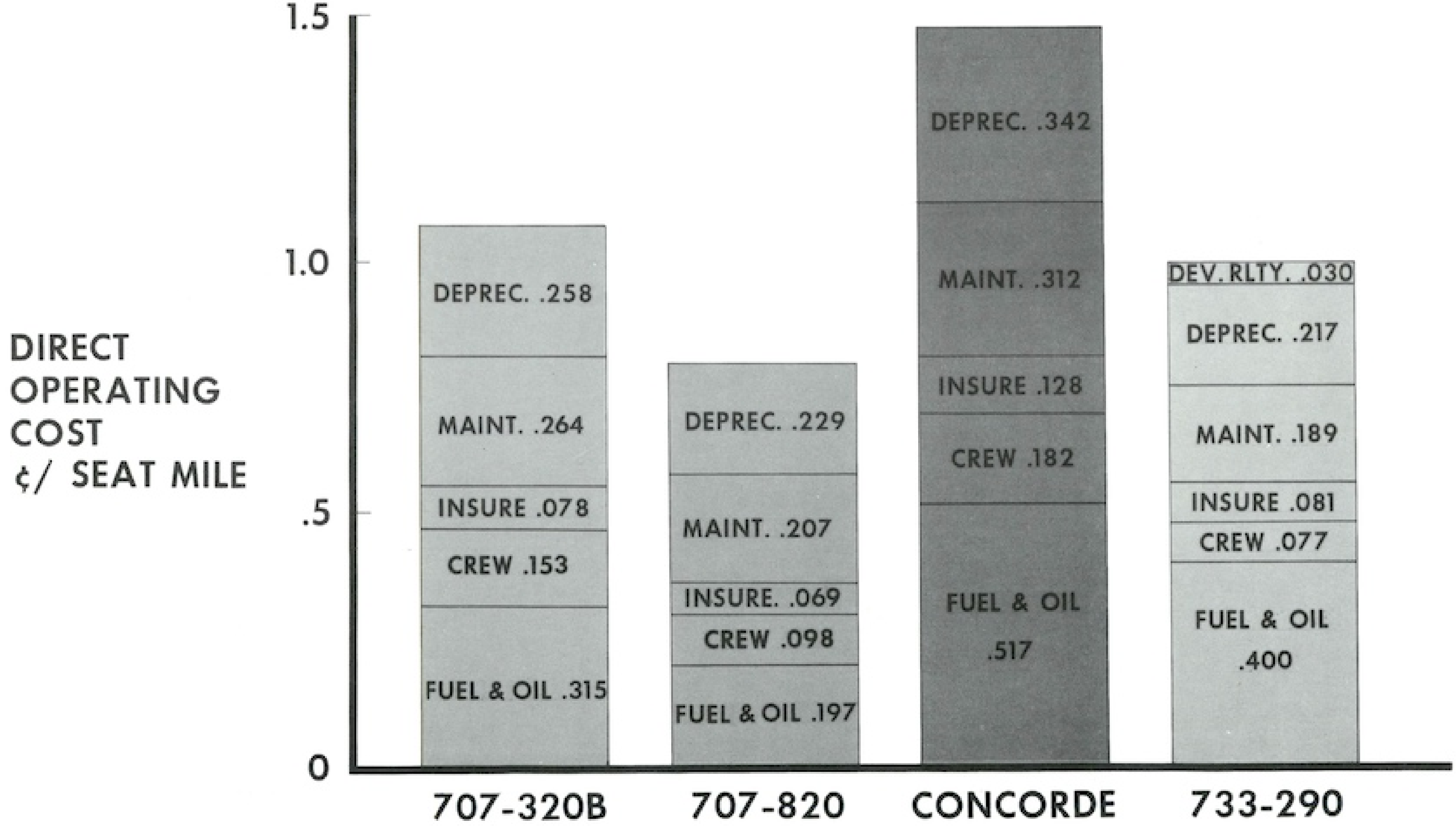


CE6690 3-65

The direct costs per seat mile are itemized on the opposite page at the economic model average range of 1980 miles. Note that the out-of-pocket costs of the Concorde are above the total costs of the 733-290 indicating the potential of the 733-290 to replace the Concorde on a cost basis alone. This assumes of course an SST program timing such that the U.S. airplane becomes available prior to the time BAC/SUD could offer an improved Concorde.

DIRECT COST PER SEAT MILE COMPARISONS

PHASE IIA INTL. ECONOMIC MODEL - RANGE 1980 ST. MILES



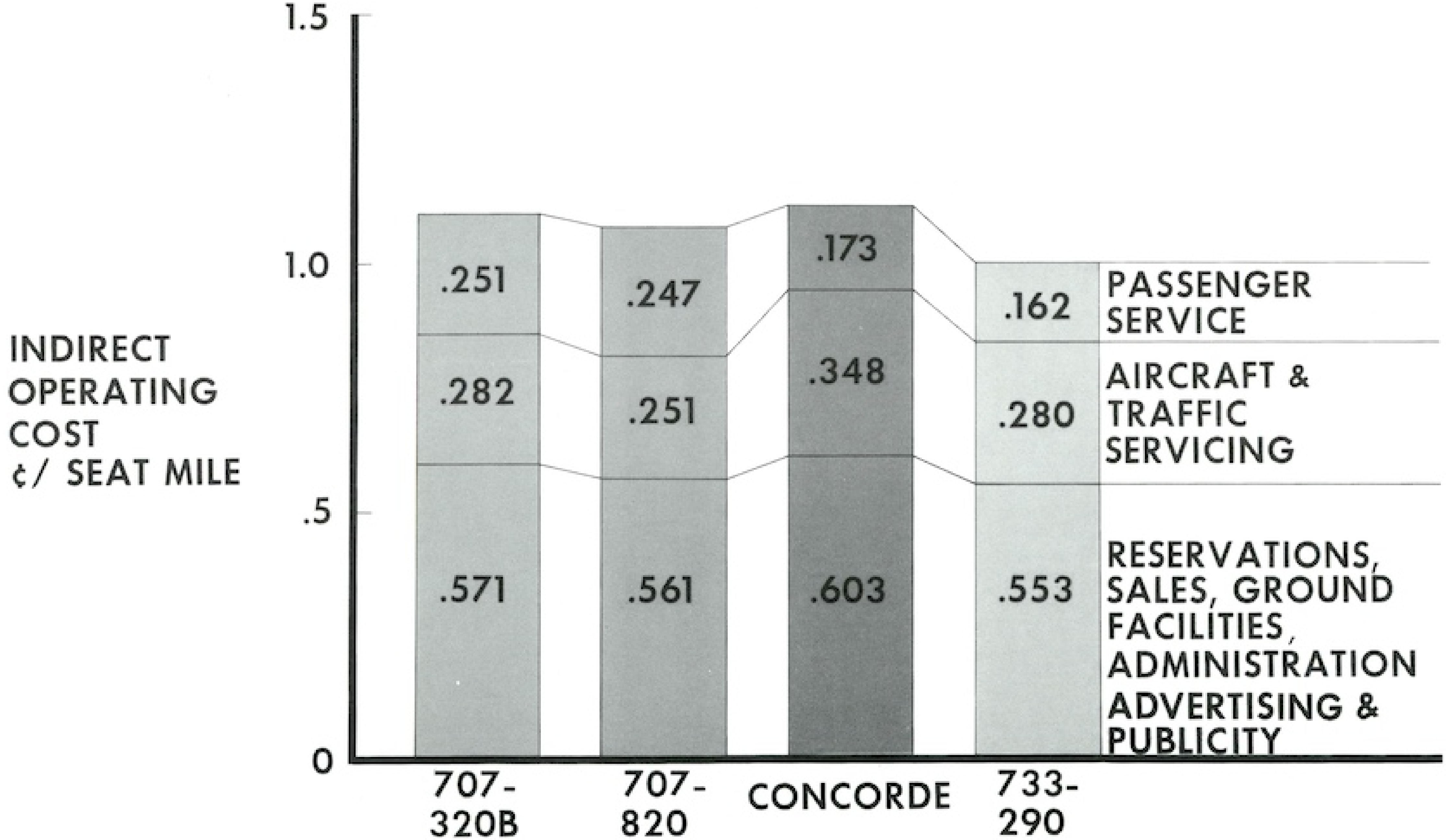
CE6684 3-65

An indirect cost allocation formula was developed by Boeing and Lockheed as a part of the work done in Phase II-A. The opposite page shows the results of applying this formula. One major factor was brought to light, namely, a saving in passenger service costs is forecast due to increased speed. This is brought about by reduced costs per trip of stewardesses (who are paid by the hour) and meals. Today there are about 2 meals per passenger carried on North Atlantic flights. With a reduction in flight time from 7 hours to 2.6 hours the second meal should be unnecessary.

Other factors of indirect costs seemed to be relatively unimportant in the comparisons except for the disadvantage of size and payload of the Concorde resulted in higher unit costs associated with landing fees and costs of communications and control.

INDIRECT COST PER SEAT MILE COMPARISONS

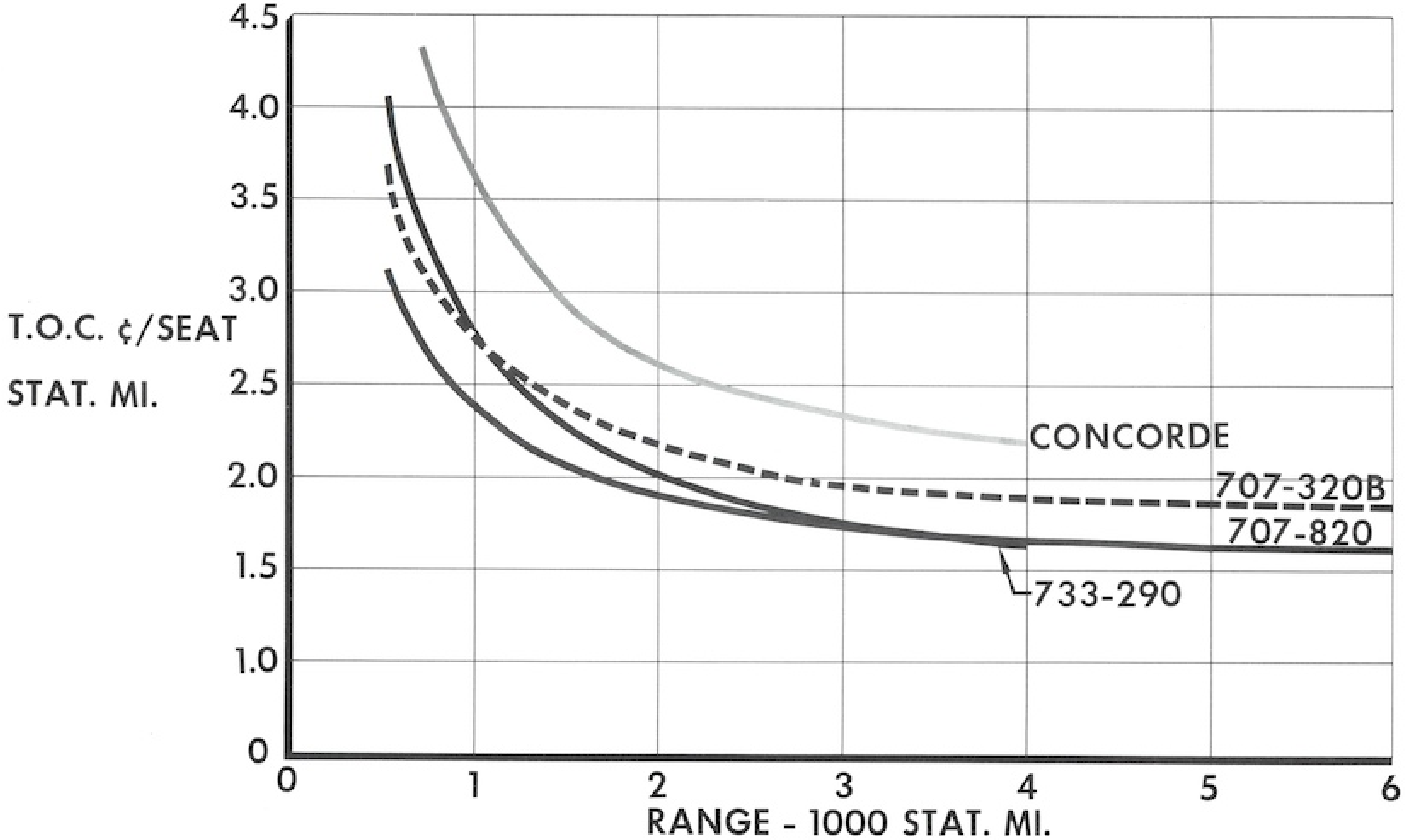
PHASE IIA INTL ECONOMIC MODEL - RANGE 1980 ST. MILES



Combining the direct and indirect costs shows a comparison which places the SST in a much more favorable light, with economics comparable to the 707-820 at ranges of 2000 to 3000 miles. Only a slight load factor advantage for the faster airplane would result in it having comparable or even superior economics at average ranges of 2000 miles.

T.O.C. - SEAT MILE COMPARISONS

PHASE IIA INT'L. ECON. MODEL



CE6685 3-65