18 March 1959

Dear Dick:

When I saw you in Washington last week, you asked for a proposed second phase engineering program and cost analysis on the airplane we currently designate as the A-11. I am attaching two copies of a complete report on the A-11, which pretty well summarizes all the basic features of the aircraft. A separate report will be furnished shortly on the radar aspects of the type.

We currently have authorization to do a certain amount of engineering on the A series airplanes, through 31 March 1959. For the period 1 April 1959 to 1 October 1959, I would propose the following work be done:

1. Design engineering. This includes the basic engineering required to carry on wind tunnel testing, major component layouts, and provide basic information for structural testing.

2. Structural tests. We are to the point where it is necessary to do a substantial amount of testing on titanium structures. We already have $10,000 worth of titanium material, some of which has been used and tested, but our investigation of this material would have to be greatly accelerated in the next few months.

3. HEF testing. We discussed the outlines of this program briefly with you and Mr. Kiefer during my last visit to Washington. It would envision setting up a basic part of the aircraft fuel system, shrouding the pertinent parts in ovens capable of simulating temperatures up through Mach #4.0, and a considerable amount of work of a chemical nature on such things as tank sealing material, seals, metals, etc. The basic problem of how to handle safely the appropriate HEF to be used would be studied.

4. Wind tunnel tests. It is vital that wind tunnel tests on both subsonic and supersonic models be run as a next step in the program. These tests would investigate lift, drag, stability, control problems, and obtain basic load data for design. A good deal of work would be done on the nacelle design but, in all likelihood, this phase could not be completed within the six months period referred to. The low speed tests would be run in the Lockheed wind tunnel, while the supersonic tests are proposed for the Ames Laboratory of the NASA.

5. Wing temperature tests. A scale model of as large a section of the wing as practical would be constructed and method provided for applying at least 1 g flying loads. It is proposed to put this model into an NASA tunnel at Langley Field or one of the Tullahoma tunnels of the Air Force, to get some indication of wing deflection and smoothness under flight load conditions at Mach # 3.2.

6. Cockpit and equipment bay temperature model. The largest feasible model of this part of the airplane would be constructed and instrumented completely, to determine heat transfer data to the critical areas of the model. Data would be obtained for windshield design, cockpit insulation and equipment bay environment.

7. Mockup. A full scale mockup of the nose section of the fuselage, including the equipment bay and the nose landing gear, would be constructed. A separate mockup would also be made for the power plant and main landing gear section. This work would not be completed within six months, but should be about 80 to 85 percent complete in this period.

8. Flutter analysis. The basic aircraft flutter modes would be investigated theoretically and computed data obtained to indicate the structural safety for the design flight conditions.

9. Antenna model and tests. Due to the expense of flying the A-11 type aircraft, and the importance of good communications and navigation, a model would be constructed so that the antennas could be developed to give optimum performance.

10. Shop layout and tool planning. A small amount of work of a planning nature would be undertaken to determine the optimum way to tool the airplane, determine the heat treat facilities required for the titanium, and investigate the availability of critical items from a time standpoint.

The over-all price for the above six months study is $1,722,000. As you can see, there is no proposal to construct any part of the airplane except models and test structures. Whereas I think we could schedule a period of 18 months time to an initial flight test with a full go-ahead, if we apply the following phase approach a somewhat longer period is necessary -- probably about 20 to 22 months. Of course, all the items proposed would necessarily have to be done under any program to construct the A-11.

You will note that there is no proposal to build a radar model as such. We are, of course, willing to make such a model, should you find it desirable, but my own current thinking is that our scale model approach, considering all the factors involved, would give us sufficient information to cover the desirable aspects of the problem.

The HEF test part of the program above is based on a price of $226,500, which does not include the cost of the material to be tested, This number is in fairly good agreement with the rough estimate I mentioned to you last, of between $200,000 and $300,000. I would be hopeful that the producers of the HEF would furnish the material to be used (600 to 1,000 gallons) in return for the results of the testing on the airplane system. It would be contemplated to build the HEF test rig in such a form that it could be transferred bodily to our engine friends' test location, as we did in the previous program, so that the total compatibility of the system, except for altitude effects, could be studied with an actual running engine.

I do not have information at this time regarding the over-all cost of 12 aircraft. We are having difficulty in evaluating the exact effect of use of the titanium in terms of our shop hours. My best horseback guess on the cost of a 12 airplane program would be between $78,000,000 and $85,000,000, in addition to the above engineering study. Of this cost, it appears that $9,000,000 to $10,000,000 is the cost of the raw titanium itself.

These costs likewise do not include an astro-inertial guidance system, which may be desirable for the type. If these units get into production, they would have costs varying between $165,000 and $300,000. If they were hand built, they approach $1,000,000 apiece.

I will provide you with the best estimate we can make on an over-all program cost prior to 26 March 1959.

STAT

Sincerely,

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Lockheed Aircraft Corporation

CALIFORNIA DIVISION

REPORT NO.	SP 114
DATE	Mar. 18, 1959
COPY NO.	Copy #1

MODEL

TITLE PROPOSAL - A-11

PREPARED B[redacted] STAT

[redacted]

[redacted]

APPROVED [redacted] STAT

[redacted]

Clarence L. Johnson
Vice President
Advanced Development Projects

REVISIONS

DATE	PAGES AFFECTED

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SUMMARY

The airplane herein proposed, is designed around two (2) Pratt and Whitney J-58 afterburning engines using HEF type fuel in the afterburners and JP-150 in the engines. The fuel load is approximately 65% HEF and 35% JP-150. Below 10,000 feet no HEF fuel is burned in order to avoid undesirable smoke and contamination.

The airplane has a 2,000 n. mi. mission radius at Mach 3.2 and crosses the target at 94,300 feet as shown in Figure 1 in the "Performance" section of this report.

Provisions are made for a crew of one and a nominal design payload of 500 lbs. The design strength is consistent with transport criteria. Modern titanium alloys are used extensively in the interest of simplicity and weight saving. The strength-temperature characteristics of these titanium alloys provide for a stretch in airplane speed to Mach 3.5. This is compatible with the J-58 engine stretch potential.

The configuration is as shown in Figure 1 in the "General Description" section of this report. It consists basically of a low aspect ratio triangular planform wing carrying a long slender fuselage and the two (2) engine nacelles underneath the wings. This arrangement

SUMMARY (Cont.)

is consistent with the maximum in structural simplicity and aerodynamic performance. In this manner the size and weight of the airplane is held to the minimum consistent with mission requirement.

In the section entitled "Alternate Fuel" it is shown that the same airplane can use JP-150 entirely and accomplish the same 2,000 n. mi. mission radius at approximately 1,500 feet less altitude.

GENERAL DESCRIPTION

This airplane is an extremely high altitude Mach 3.2 reconnaisance vehicle designed to carry a crew of one and a nominal 500 pound payload.

The configuration is characterized by a long thin fuselage, a very thin triangular wing, and under-wing mounted engines. Cross-sectional area of the fuselage is determined by the reconnaisance equipment, its length by fuel volume and balance considerations. The wing, while only 2½% thick, has such a large root chord that its physical thickness results in large internal fuel volume and relatively low loads. Engine positioning under the wing results in excellent accessibility and serviceability, and also contributes to the lightness of the airplane by relieving wing bending loads and eliminating long intake ducts and tailpipes. The conventional tricycle landing gear is forward-retracting and is designed to free-fall.

The military equipment bay, immediately aft of the pilot's compartment, is 72 inches long and is accessible through two removable doors, one top and one bottom, each approximately 58 inches long. Equipment is installed and removed through the bottom door, approximately 4 feet wide. Thus the provisions for payload are equal to or better than those presently existing in the U-2 airplanes.

Airplane structure is almost entirely of B-120VCA, a new titanium alloy having very high strength-to-weight ratio at elevated temperatures, and good formability. Extensive use will be made of spotwelding in sub-assemblies. The fuselage is of skin, ring, and longeron design.

GENERAL DESCRIPTION (CONT.)

The wing and vertical tail consist of multiple beams, widely spaced ribs, and a covering of skin stiffened by corrugated inner skins.

Airplane systems, such as controls, hydraulic, cooling and pressurization, are discussed in succeeding sections of this report.

Following is a brief weight summary:

Weight Empty	35,815
Oxygen, Oil, Unusable Fuel	200
Pilot	285
Payload	500
Zero Fuel Weight	36,800 lbs.
Fuselage Fuel	30,925
Wing Fuel	17,100
Takeoff Weight	84,825 lbs.

WING-

AREA	1605.56 SQ.FT.
ASPECT RATIO	2.00
TAPER RATIO	0.00
ROOT CHORD	56.67 FT. (680.00 IN)
TIP CHORD	0.00
SPAN	56.67 FT. (680.00 IN)
DIHEDRAL	0°
INCIDENCE	0°
AIRFOIL - MOD. BI-CONVEX	2.5%
SWEEPBACK AT .25C	47.73°
M.A.C. AT WING STA. 113.33	453.33 IN.

ELEVON-

AREA (TOTAL)	138.75 SQ.FT.
TRAVEL	UP 23° DOWN 18°

VERTICAL TAIL-

AREA	279.07 SQ.FT.
ASPECT RATIO	.302
TAPER RATIO	.298
ROOT CHORD	36.33 FT. (436.00 IN)
TIP CHORD	10.83 FT. (130.00 IN)
SPAN	11.83 FT. (142.00 IN)
AIRFOIL - MOD. BI-CONVEX	ROOT 3% TIP 2%
SWEEPBACK AT .25C	62.13°
M.A.C. AT FIN STA. 55.21	310.57 IN.

RUDDER-

AREA	30.00 SQ.FT.
TRAVEL	27.50°

FUSELAGE-

MAXIMUM DEPTH	5.29 FT. (63.50 IN)
MAXIMUM WIDTH	5.29 FT. (63.50 IN)
OVERALL LENGTH	103.00 FT. (1236.00 IN)
FRONTAL AREA	21.99 SQ.FT.

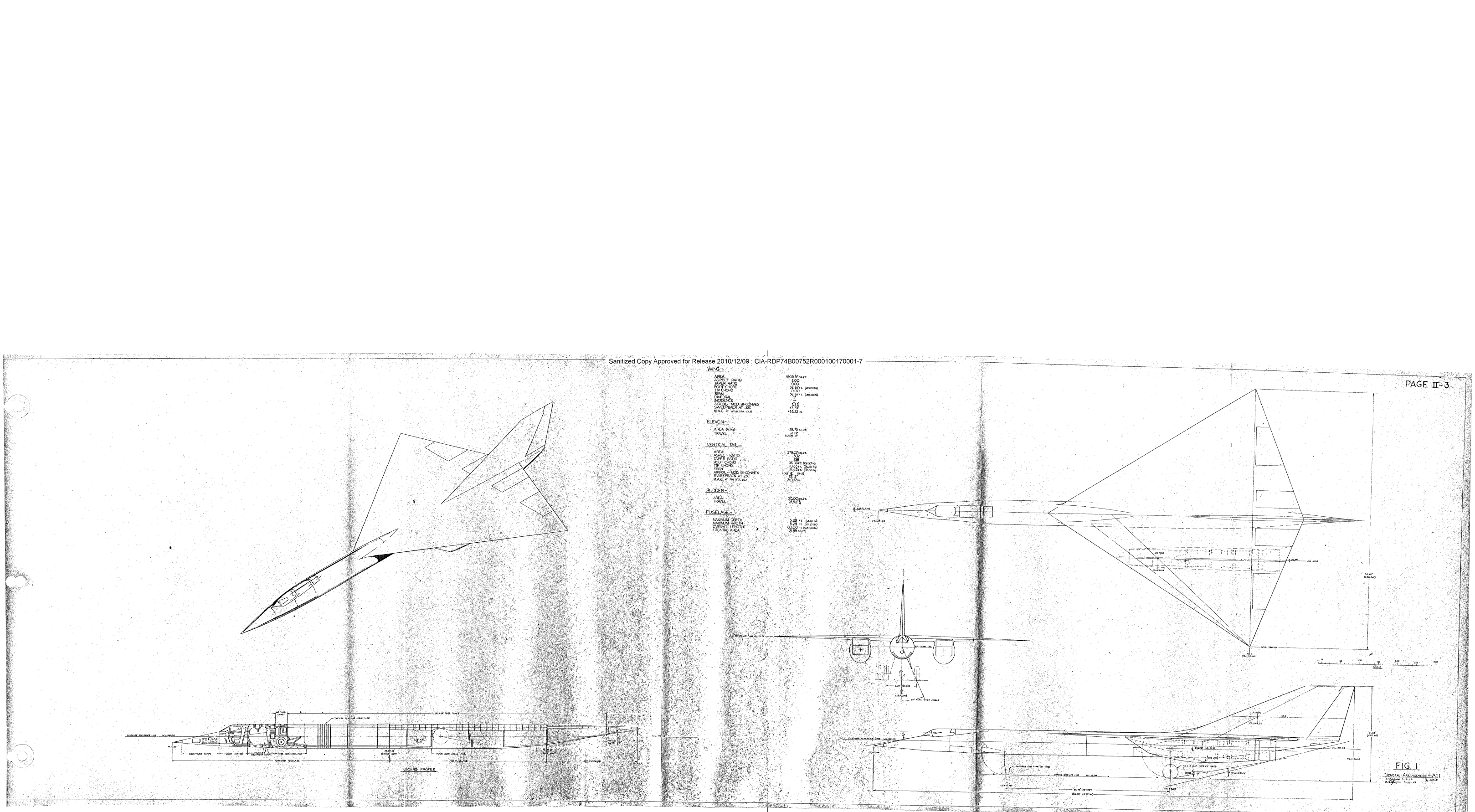


FIG. 1

GENERAL ARRANGEMENT - A11

1/32 SCALE

3-12-59

3-16-59

PERFORMANCE

The A-11 configuration is capable of 2,000 n. mi. radius mission cruising at Mach 3.2 at altitudes from 88,700 feet to 100,000 feet. The mission is summarized on Figure 1 and a distance-weight profile is shown on Figure 2. Airplane performance is summarized on Figure 3.

The mission comprises a full power take-off, climb and cruise. Fuel allowance for take-off and acceleration to climb speed is one minute at full power.

The climb performance is shown on Figure 4. The sea level rate of climb is 23,900 feet per minute and decreases with altitude to about 4,000 feet per minute at 74,000 feet. This part of the climb is made at a constant EAS of 400 Knots and an increasing true speed. Consequently a large part of the excess thrust is required for acceleration. Above 74,000 feet the climb is made at a constant Mach 3.2 and all of the excess thrust is available for climb. At 74,000 feet the rate of climb exceeds 30,000 feet per minute and thereafter decreases rapidly to zero at 88,700 feet, the start of cruise. The climb uses 9,000 pounds of fuel, covers 220 n. mi., and requires 10.67 minutes.

The climbing cruise is made at maximum power at Mach 3.2. The cruise time is 2.1 hours including a 180 degree turn at the target point 2,000 n. mi. from take-off at an altitude of 94,300 feet. The end of cruise is at 100,000 feet over the base at Mach 3.2. An actual mission would include an idle

PERFORMANCE (CONT.)

power descent starting 150 to 200 n. mi. from the base and would use less fuel than continuing the cruise to the base at altitude. Idle power operation of the engines at altitude is not yet established making the descent characteristics difficult to define. A reserve allowance is included for a single engine 30-minute loiter at subsonic speeds at 35,000 feet altitude.

The take-off and the landing ground roll are 2,400 and 2,700 feet respectively. Speeds required for take-off and landing are based on an angle of attack of 11 degrees, which is the clearance angle with the main gear struts compressed. This provides an adequate ground clearance margin over the 15 degrees provided with the gear struts extended. Single engine safety during take-off is excellent since the total airplane drag is less than 20,000 pounds including dead engine and trim drag and the operating engine provides about 30,000 pounds of thrust. Single engine performance during landing is, of course, better due to the reduced weight.

In the event of an engine failure at some point during a mission, two courses of action are open to the pilot. He can descend to about 50,000 feet and subsonic speed and return to base from any point during the mission. Or, he can maintain his speed at Mach 3.2 and descend to 72,000 feet. At this flight condition, he can return to base if the engine failure occurs not over 1,800 n. mi. from base on the outbound leg or not over 1,400 n. mi. on the return leg of the mission. Between these points the airplane cannot return to base. If the engine failure occurs at the target, the airplane

PERFORMANCE (CONT.)

will run out of fuel 350 n. mi. short of the base. The single engine supersonic return capability is shown on Figure 5. If penetration is assumed to occur at the end of climb, 220 n. mi. from base, then the airplane can make a supersonic single engine return to the penetration point from all points during the mission except within a distance of 70 n. mi. before reaching the target and 220 n. mi. after passing the target. Thus a high degree of multi-engine reliability is assured.

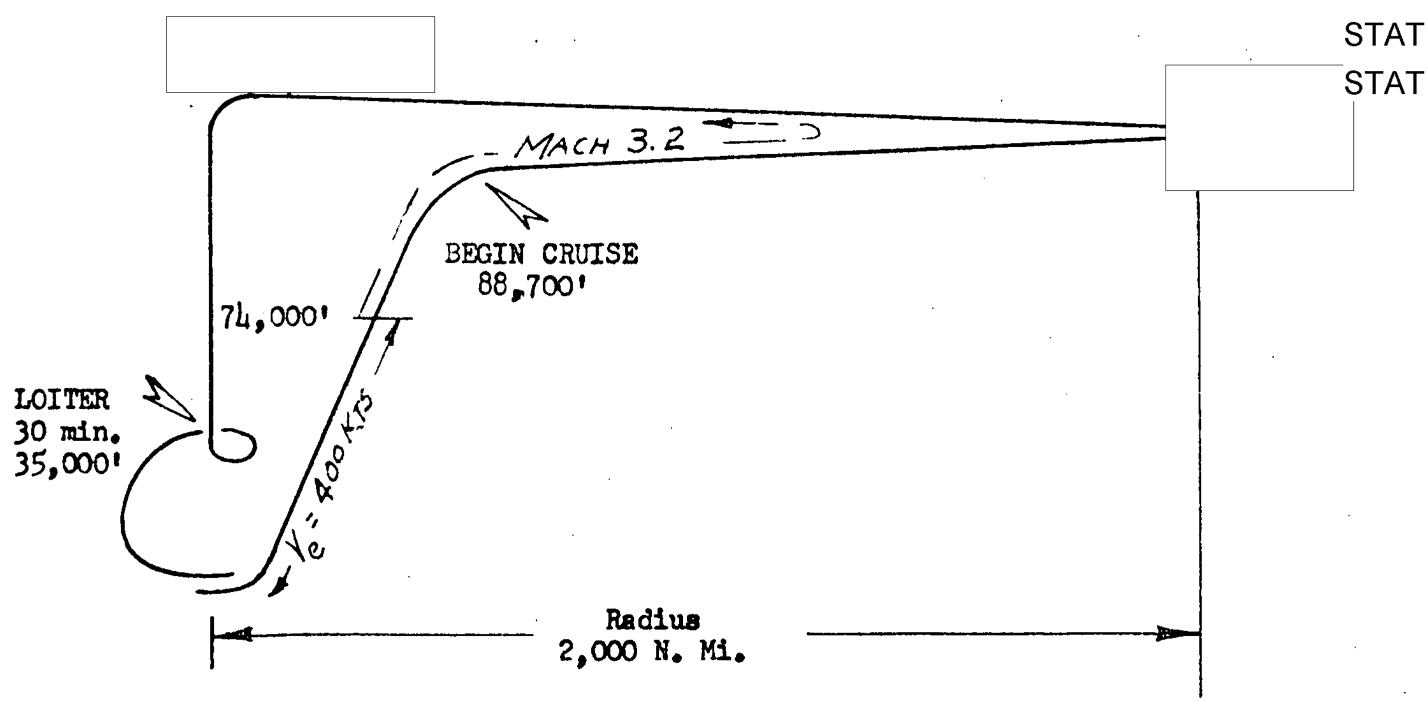
FIGURE 1

A-11 MISSION SUMMARY

	Weight Lbs.	Fuel Lbs.	Dist. N.Miles	Alt. Ft.
T. O.	84,800	1,700	0	S.L.
Climb	83,100	9,000	220	S.L.
Cruise Out	74,100	19,600	1,780	88,700
Target	54,500	-	-	STAT
Cruise Back	54,500	15,900	2,000	
Reserve (30 min.)	38,600	1,800	-	35,000
ZFW	36,800	-	-	-

Radius 2,000 N. Mi. (180° turn at target)

Fuel 48,000 lbs. Total
(31,000 lbs. HEF used in afterburner,
17,000 lbs. JP150 used in primary)



A-11

FIGURE 2

WEIGHT-DISTANCE PROFILE

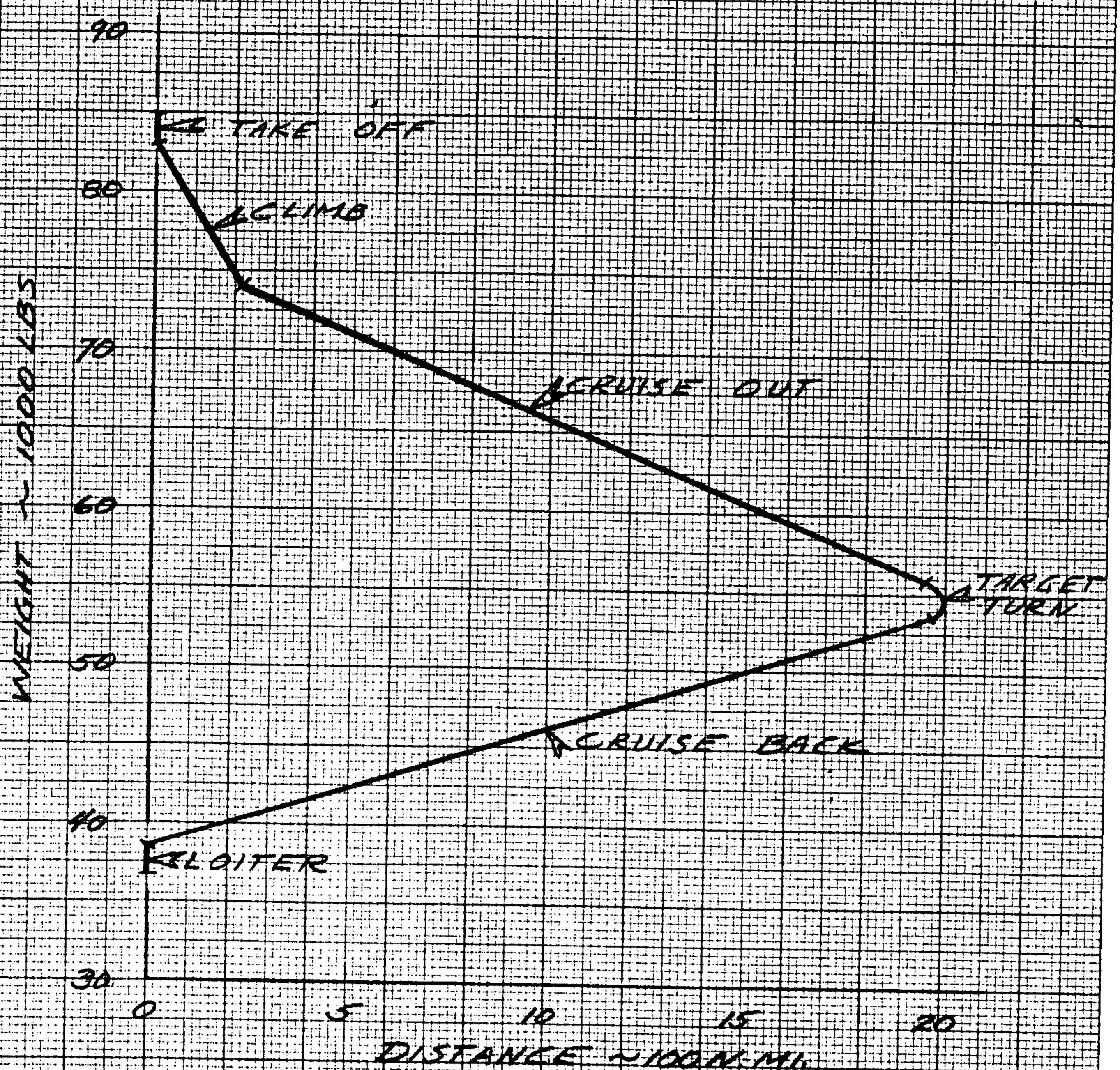


FIGURE 3

Lockheed AIRCRAFT CORPORATION
CALIFORNIA DIVISION

A-11 PERFORMANCE SUMMARY

Radius	2,000 n. mi.	
Take-off		
Weight (lbs.)	84,800	
Speed (Kts)	185	
Take-off Ground Roll (Feet)	2,400	
Rate of Climb at S.L. at 400 Kts. (Ft./Min.)	23,900	
Cruise		
Mach No.	3.2	
Speed (Kts)	1,890	
Altitude (Feet)		STAT
Target		
Altitude (Feet)	94,300	
Weight (lbs.)	54,500	
Landing		
Weight (lbs.)	38,600	
Speed (Kts)	125	
Distance (Feet)	2,700	

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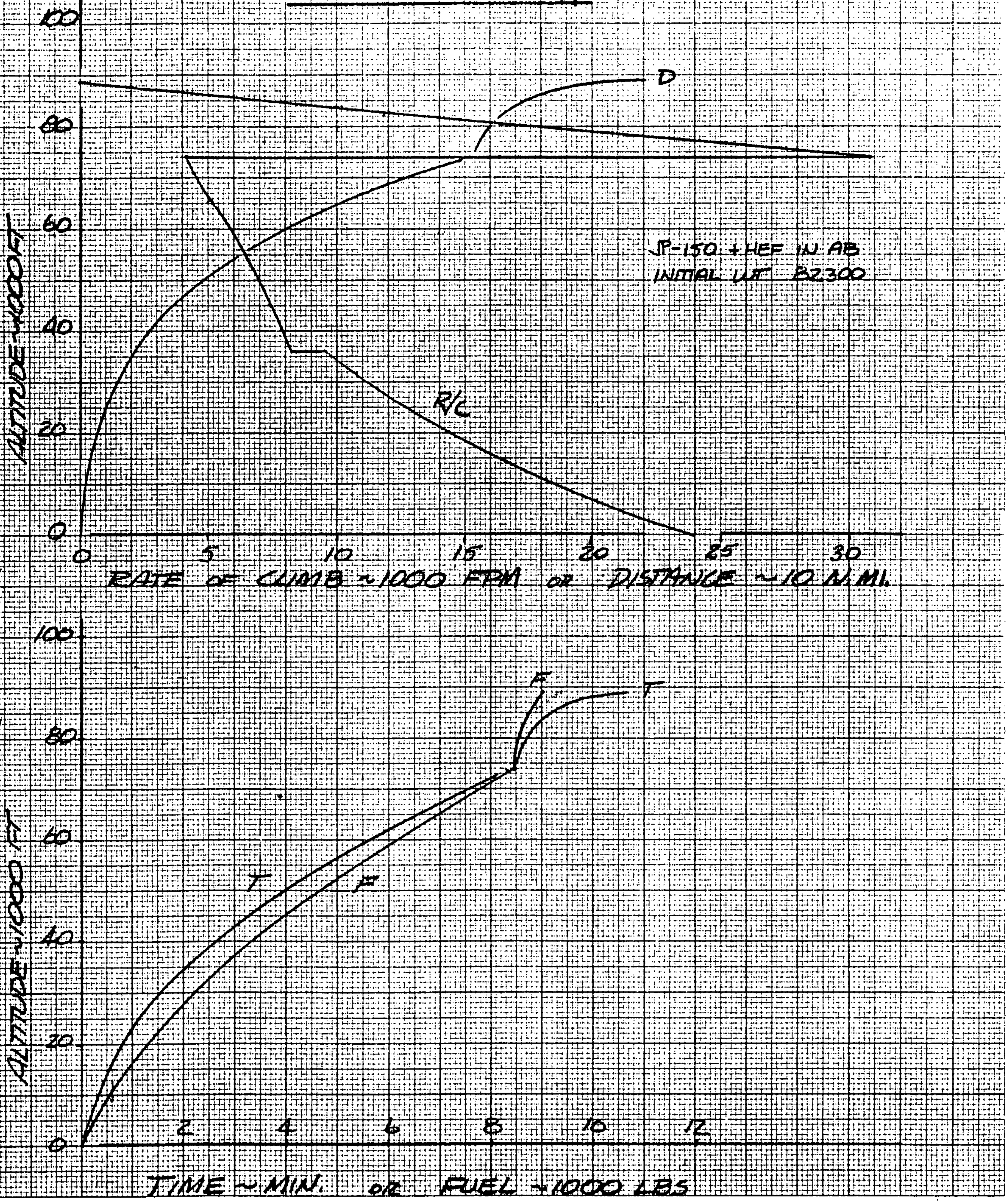
A-11

FIGURE 4

CLIMB SUMMARY



FORM 5278A

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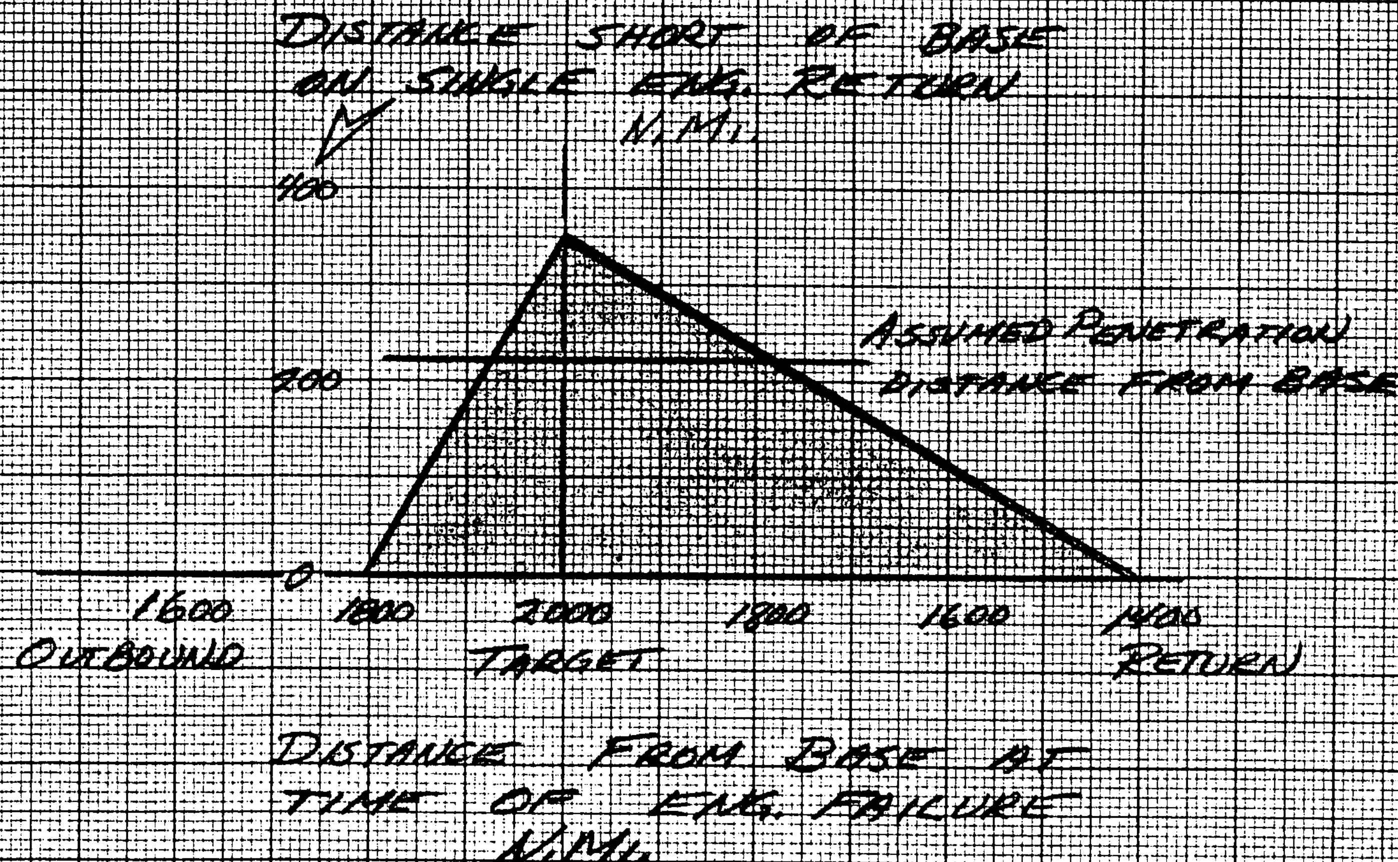
FIGURE 5

SINGLE ENGINE RETURN CAPABILITY
MACH 3.2
72000 FT.



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Lockheed AIRCRAFT CORPORATION
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SECTION IV - STRUCTURAL DESCRIPTION

FORM 8767A-1

WEIGHT AND BALANCE

This section contains a brief discussion of the weight estimate and the airplane balance. The configuration achieves by structural simplicity the lightest airplane to perform the mission. The weight estimate is based on the use of present day production techniques and good weight control activity in design. Sufficient analyses have been made of the structure and major aircraft systems to determine the validity of the component weights; these analyses are the basis for the weight estimate.

The airplane balance is shown on Figure 1. The center of gravity envelope is tailored to give minimum trim penalty during the supersonic position of the mission, while retaining reasonable c.g.'s for take-off and landing. The most forward c.g. is at take-off, as fuel is used the c.g. moves aft to give the most aft c.g. at the mid-point of the mission and then forward for landing.

Page 3 contains the weight summary followed by a brief discussion of the component weights on pages IV-5 to IV-8.

Lockheed AIRCRAFT CORPORATION
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WEIGHT SUMMARY

Wing	9,430
Fin	1,450
Fuselage	4,550
Landing Gear	1,900
Surface Controls	1,120
Nacelles	1,900
Propulsion Group	13,110
Instruments	110
Hydraulics	550
Electrics	300
Electronics	425
Furnishings	150
Air Conditioning	750
Tail Parachute	70
Weight Empty	35,815
Oxygen	40
Oil	60
Unusable Fuel	100
Pilot	285
Payload	500
Zero Fuel Weight	36,800
Fuselage Fuel	30,925
Wing Fuel	17,100
Take-off Weight	84,825

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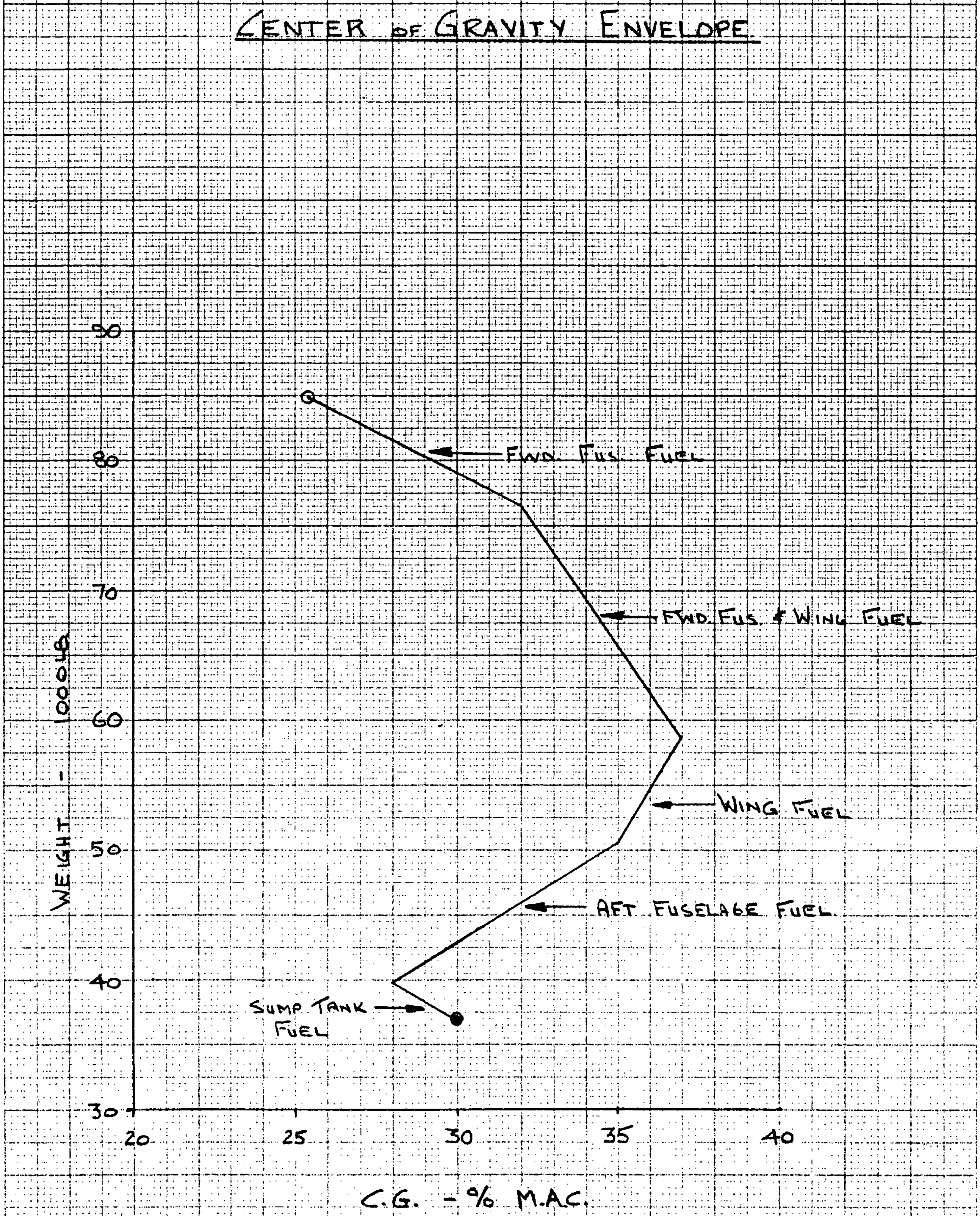

FIGURE 1

FORM 5278

Lockheed AIRCRAFT CORPORATION

CALIFORNIA DIVISION

WEIGHT AND BALANCE

Component Weight

The wing and fuselage weights are derived from the structural analyses briefly presented in this section of the report. The fin structure will be the same type as the wing, reduced in weight due to the lower load intensities.

Wing

Box Beam

Skin Panels	3,000
Beam Caps	1,390
Beam Webs	780
Ribs	1,150
Joints etc.	380
	6,700
Leading Edge	1,020
Trailing Edge	1,480
Fillets-Wing to Fus.	230
Total	9,430

Fin

Fin	1,450

Fuselage

Skin	1,225
Longerons	670
Frames	705
Wing & Fin attachments	350
Landing gear support structure	250
Bulkheads	190
Joints etc. in Shell	340
Windshield & Canopy	250
Doors - Equip. Bay, Gear, etc.	470
	4,550

FORM 5767A 1

WEIGHT AND BALANCE

Component Weight (Cont.)

Landing Gear

Main

Wheels and Tires	380
Brakes	320
Struts, Retraction, etc.	850
	1,550

Nose

Wheel and Tire	110
Strut	180
Steering and Retraction	60
	350

Surface Controls

The surface control weight is based on full powered irreversible systems. An allowance of 50 lbs. is included in the autopilot weight to provide any stability augmentation that may be required.

Cockpit Controls	45
Autopilot	150
Elevon System	675
Rudder System	250
	1,120

Nacelles

The total weight of this group is 1,900 lb. and includes the air intake system and engine cowl. The engine cowl, that is the portion aft of the front face of the engine, is estimated to weigh 900 lb. The air intake

FORM 8787A 1

WEIGHT AND BALANCE

Component Weight (Cont.)

Nacelles (Cont.)

system as drawn is somewhat tentative since the inlet configuration will probably require some development, however, the weight of 1,000 lb. allowed seems adequate for anything that can be envisaged at this time.

Propulsion Group

The J-58 engine weight of 5,950 lb. each includes starting provisions and self contained oil system. The fuel is contained in integral wing and fuselage tanks, the simultaneous use of JP-150 and HEF will require some ingenuity in the design of the fuel system plumbing to minimize the weight penalty for this feature. The additional weight of 200 lb. carried for the HEF system is based on some duplication of pumps, distribution and transfer systems.

Engines		11,900
Engine Controls		50
Fuel System		1,160
Tank Sealing	350	
Basic System	610	
HEF Increment	200	
		13,110

Instruments

Flight Instruments	25
Engine Instruments	40
Misc. & Installation	45
	110

WEIGHT AND BALANCE

Component Weight (Cont.)

Hydraulics	550
Electrics	300

Electronics

This group includes the navigational and communication equipment described in Miscellaneous Systems section together with the wiring and supports required to install these systems in the airplane.

ARC 62 Command Set	75
ARN 44 Radio Compass	85
Inertial Navigation System	200
Driftsight	35
MA1 Compass	30
	425

Furnishings

Ejection Seat	100
Oxygen System (fixed items)	15
Misc. Consoles & Trim	35
	150

Air Conditioning

The air conditioning problem is discussed in Cockpit Environment section. The weight allowance of 750 lb. for this system is a reasonable estimate at this stage.

FORM 5787A-1

DESIGN LOADS

Loads used for the structural design of this airplane are based on the requirements of Military Specification MIL-S-5700 with modified gust criteria. The gust criteria modification refers to the variation of gust velocities with altitude as shown by Figure 4.

Figure 3 shows the variation of design speeds with altitude. Above 72,000 feet, maximum speed is limited to M = 3.2. From 72,000 feet to sea-level the maximum design speed is 425 knots, EAS. The design level flight speed of 370 knots, EAS shown on this chart was selected for combination with a ± 50 fps. gust. Calculated aileron reversal speeds are also shown on Figure 3. Adequate wing stiffness within the design speed range is indicated by these reversal speeds.

V-n diagrams for gust and maneuver are shown by Figure 2. For the maneuver envelope maximum accelerations of +2.5 g and -1.0 g are used. The gust envelope shown is conservatively based on zero-fuel weight of 36,800 lbs. and therefore, results in the maximum design gust load factors.

Ultimate design loads for the various airplane components are included in the pertinent sections of this report. Except for the forward part of the fuselage, a 2.5 g sub-sonic maneuver @ T.O. weight of 85,000 lbs. produces critical loads on both the wing and fuselage. The +50 fps. gust condition @ 36,800 lbs. produces slightly higher loads on the forward

DESIGN LOADS (Continued)

part of the fuselage. A 2.5 g maneuver @ M = 3.2 is not critical because fuel used to climb reduces the gross weight to 75,000 lbs. Wing loads for this condition are approximately 86% of the "cold" condition loads. Fuselage loads for this condition are not critical because the fuel used is removed from the forward fuselage tanks.

V-n DIAGRAMS

GUST ENVELOPE FOR 36800 LBS @ 25000 FT

MANEUVER ENVELOPE FOR 36800 LBS

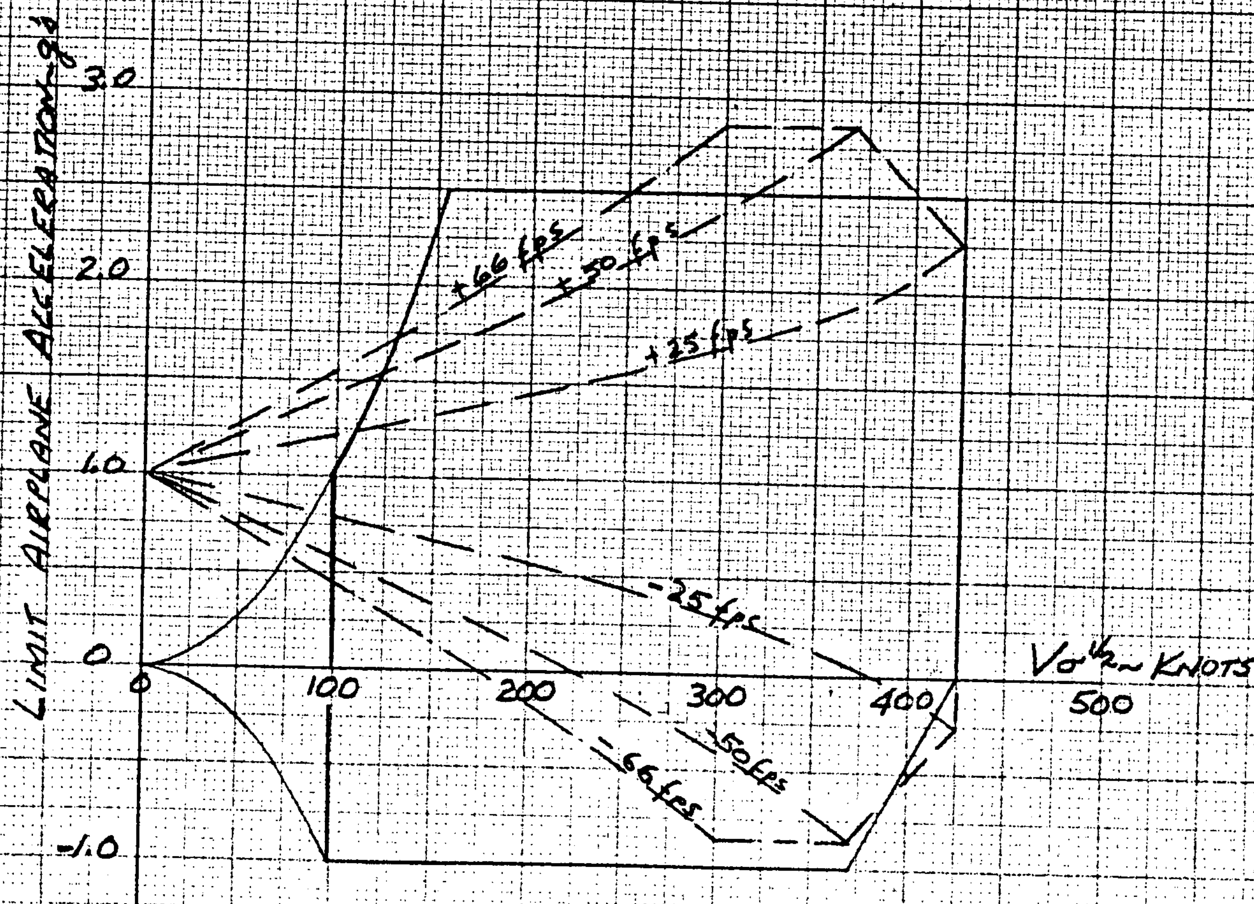

FIGURE 2

SPEED - ALTITUDE CHART

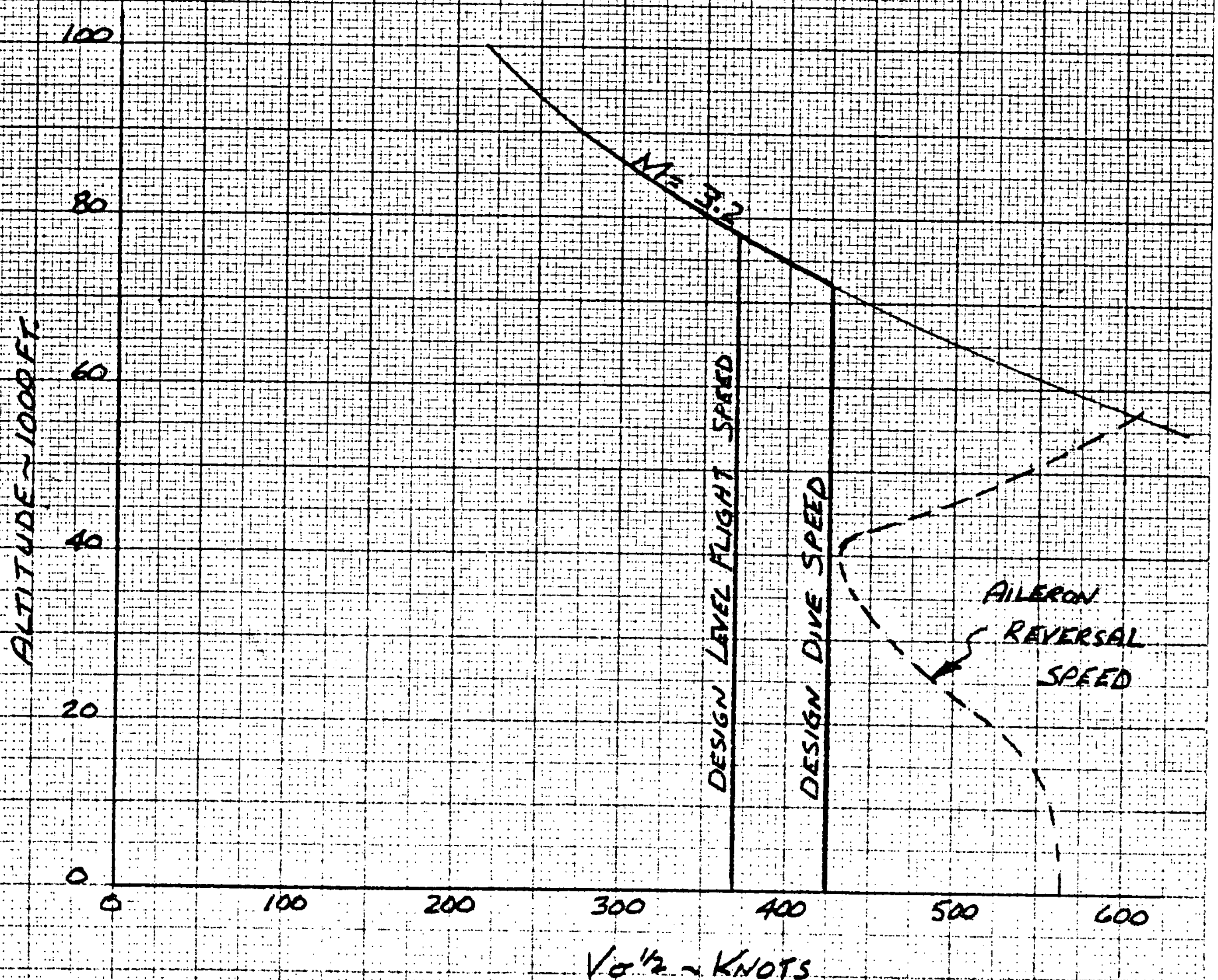

FIGURE 3

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DESIGN GUST VELOCITIES

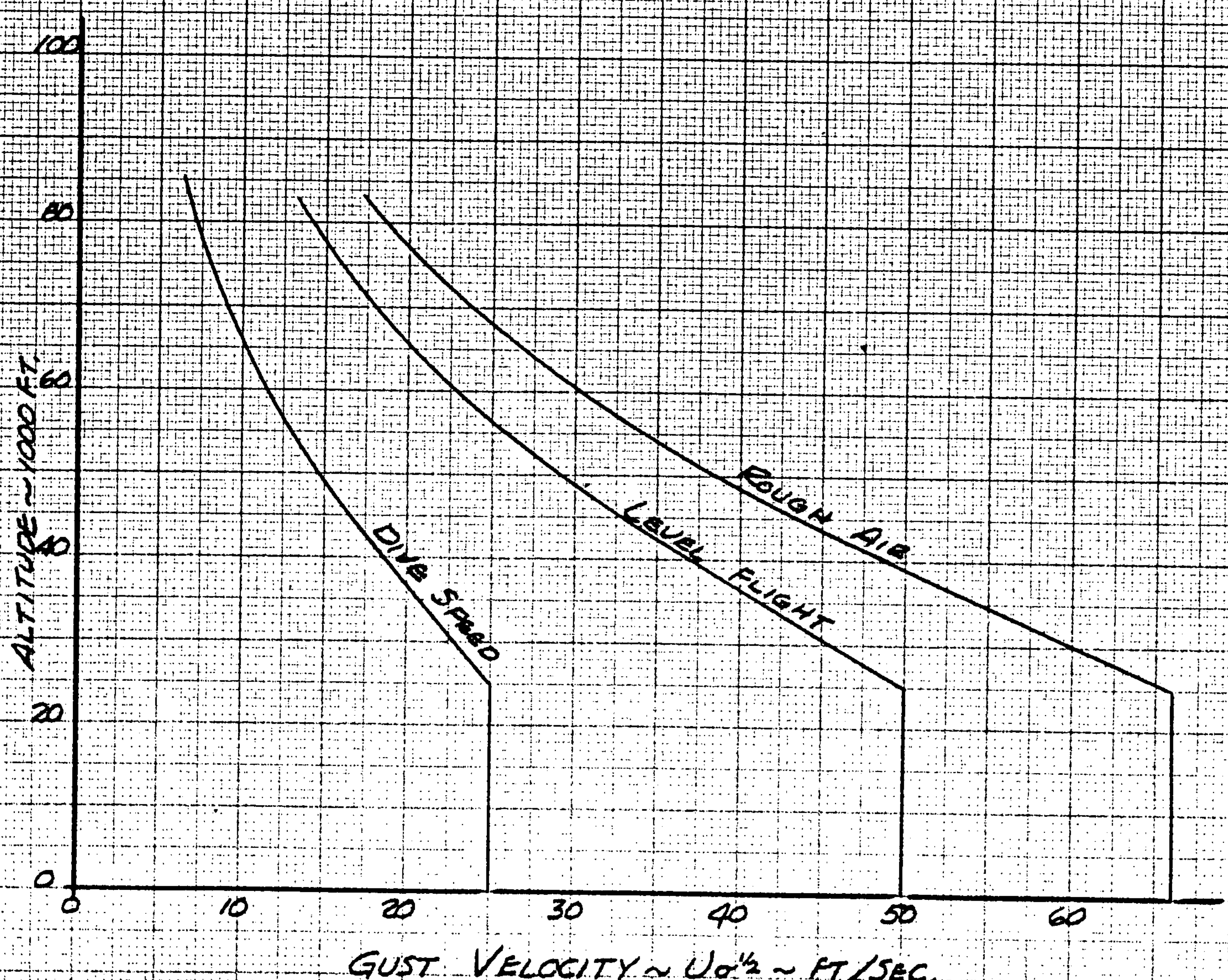


FIGURE 4

FORM 5278

MATERIAL SELECTION

Investigation was made into new experimental materials available and those still being developed in the laboratory. All of the common and exotic metals and modifications thereof were considered. These were compared to each other on strength/density basis, for ultimate, yield and modulus of elasticity, for all temperatures up to 1200°F. For temperatures up to 800°F titanium alloys indicated as good as or better strength/density capabilities. Of the titanium alloys considered MST 185 and B-120VCA were shown to be most promising.

From feasibility and producibility aspects B-120VCA is the most practical and the most efficient in strength at all temperatures up to 800°F. The material selected is manufactured by Crucible Steel Corporation, Pittsburgh, Pennsylvania, and is basically an all Beta titanium alloy. Its elements are 13% vanadium, 11% chromium and 4% aluminum. It can be purchased in the annealed, aged, or cold worked and aged conditions. Aging is a simple heating procedure (800°F - 1000°F) for extended periods of time ranging from 8 to 100 hours, followed by air cooling.

This material indicates the following characteristics:

1. Good bendability and formability.
2. Good weldability.
3. Non-directional characteristics.
4. Ability to be brazed.

MATERIAL SELECTION (Continued)

5. Cold headability.
6. Readily machined.
7. Exceptionally low creep rates at elevated temperatures.

The physical properties of solution treated or annealed material are as follows:

1. Density: 4.82 GMS./c.c. (0.175 lbs./cu.in.).
2. Specific Heat: .131 BTU/lb./°F.
3. Thermal Expansion: 5.2×10^{-6} in./in./°F (68 - 200°F)
4. Thermal Conductivity: 3.90 BTU/hr./Ft.^{2}/°F/Ft.

The mechanical properties furnished by material vendor are as follows:

Annealed	Room Temp.	600°F
F_{t_u} - psi	152,000	109,000
F_{t_y} - psi	151,000	103,000
% Elong.	12	21
Elastic Modulus - psi	14.3×10^6	13.2×10^6

Aged	Room Temp.	600°F
F_{t_u} - psi	200,000	175,000
F_{t_y} - psi	190,000	145,000
% Elong.	5	9
Elastic Modulus - psi	15.3×10^6	13.8×10^6

FORM 3767A-1

MATERIAL SELECTION (Continued)

The above values have been verified by a number of coupon tests in the Lockheed Research Laboratory.

General temperatures expected throughout the airplane structure are expected to be 500°F with peak temperatures on leading edges equal to 780°F. The above allowables indicate this material has good mechanical properties in this range.

WING

Description

The construction of the wing is as shown in Figure 5. The structural box extends from 15 percent to 80 percent of the wing chord. Forward of 15 percent, the leading edge consists of a solid leading edge arrowhead and skins supported by multiple ribs and stiffeners perpendicular to the swept leading edge. The structural box itself consists of multiple beams sapced at 16 inches along the chord. Beams are built up of beam caps, webs and stiffeners. Caps are located under contour in order to allow for the passage of surface corrugations in a chordwise direction. Shear attachment of beams to outside skin is accomplished by tabs between corrugations. The beams are designed to carry the wing beam bending load and vertical shear.

The surfaces of the box consist of an outer skin and an inner corrugated skin with corrugations running in a chordwise direction. This surface structure is designed to carry normal pressures to the beams and to resist wing torsional moment. This type of surface design, acting together with intercostal ribs spaced approximately 40 inches along the span, provides good chordwise form stiffness.

Aerodynamic heating of the structure results in a temperature gradient from outside skin to inside structure. This gradient can be accommodated by this type of structure easily since expansion of the outside surface results only in buckling or waving between corrugations in the streamwise direction.

WING (Continued)

Description (Continued)

Hence, the stresses due to temperature gradient are held to a minimum and aerodynamic smoothness is maintained.

For producibility and transportability, a joint in the wing is provided just outboard of the engine nacelle as shown in Figure 5. The trailing edge structure from 80% to 100% of chord consists mainly of control surfaces.

Material throughout the wing is B-120VCA titanium in various forms.

Design Loads

Ultimate wing shear, bending moment and torsion is shown in Figure 6 for critical 2.5 g heavy weight condition. This is a room temperature condition at M = 0.8. Supersonic "hot" conditions are 14% less and are not critical on the box structure since the material reduction factor at 500°F is only 10%.

Section Properties

The airfoil section is presented graphically in Figure 7. Using this section and the wing basic dimensions, the structural section properties are calculated and presented graphically in Figure 8.

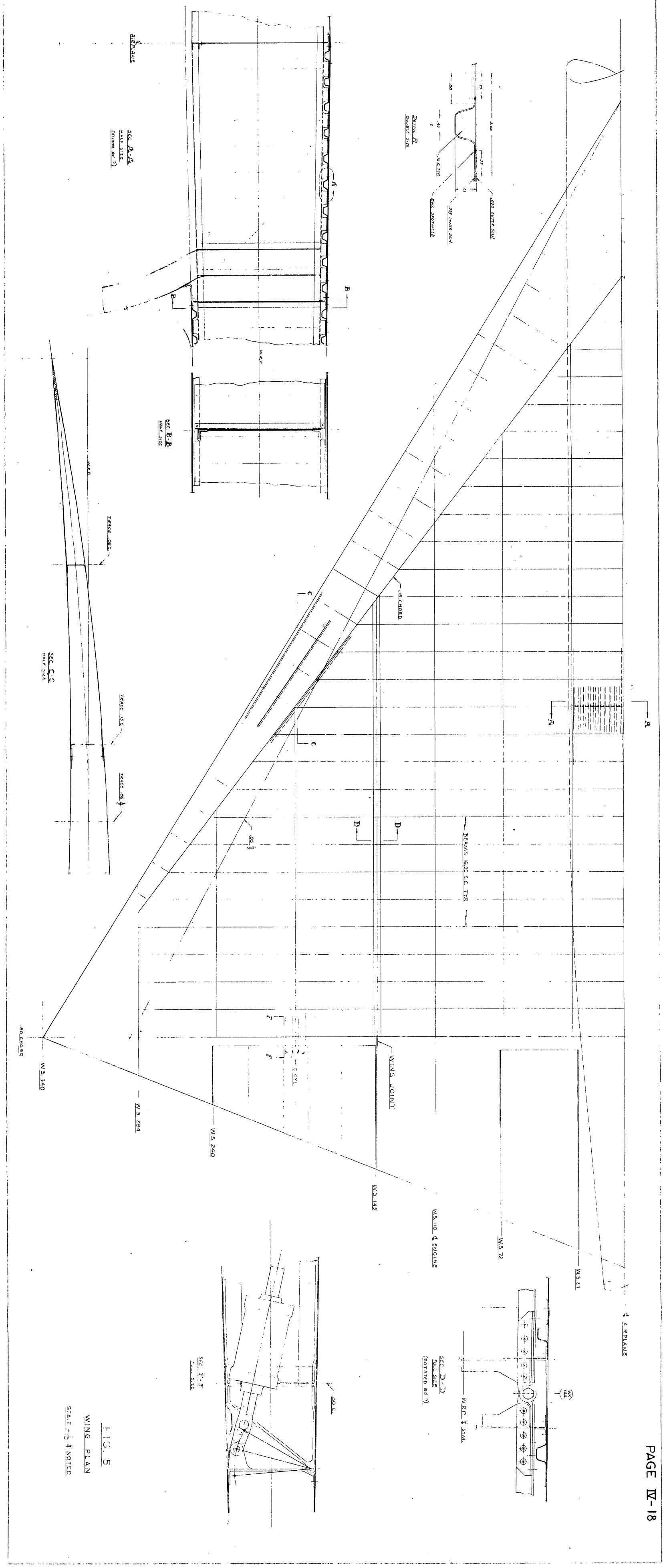
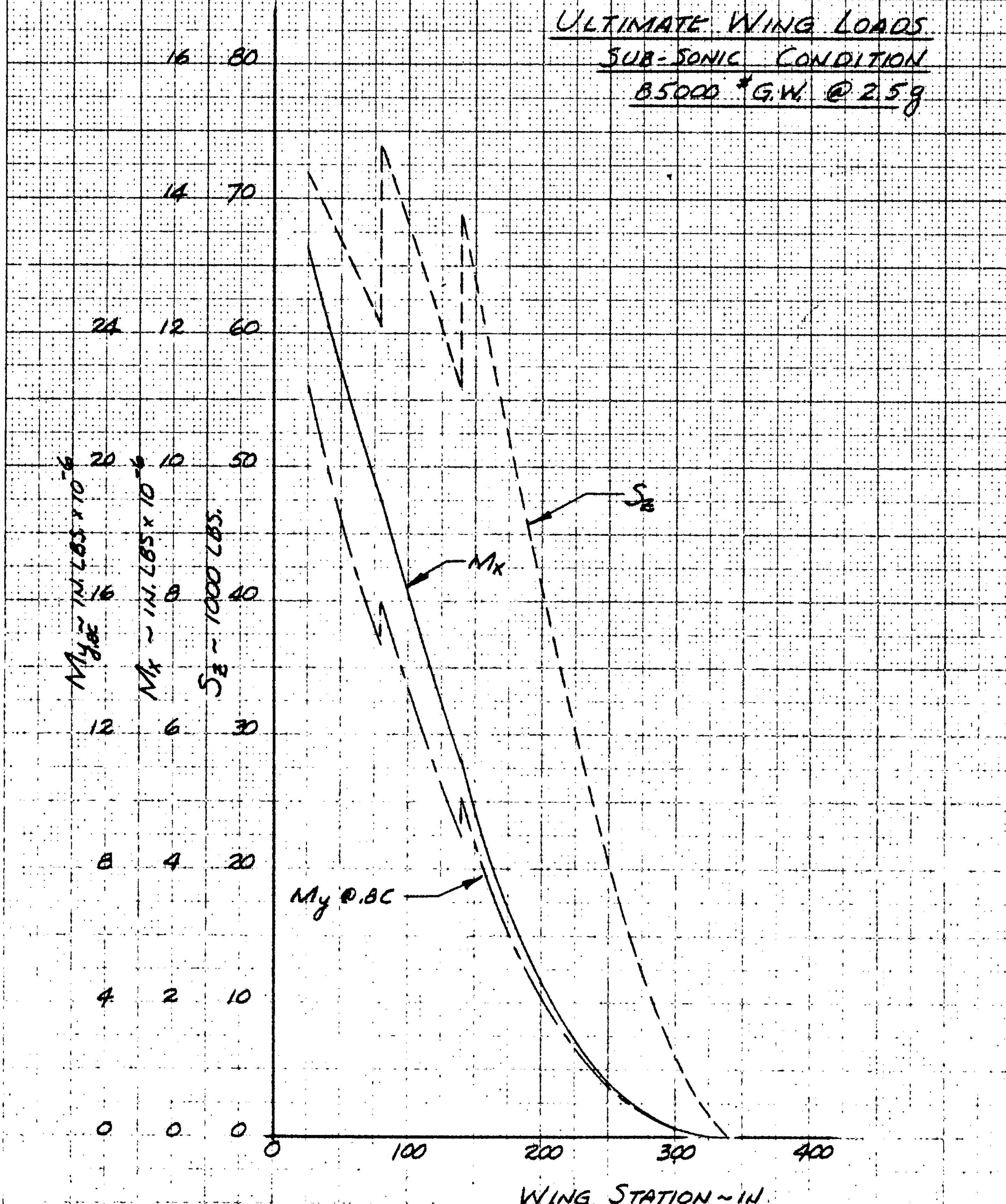

SEC A-A
SEC B-B
SEC C-C
SEC D-D
SEC F-F
DETAIL A
WING JOINT
BEAMS 16.00 C.C. TYP
W.S. 340
W.S. 284
W.S. 240
W.S. 165
W.S. 110 ℄ ENGINE
W.S. 72
W.S. 57
℄ AIRPLANE
FIG. 5
WING PLAN
PAGE IV-18

FIG. 6

ULTIMATE WING LOADS
SUB-SONIC CONDITION
85000 # G.W. @ 2.5g



WING FIG. 7

AIRFOIL

SECTION PROPERTIES

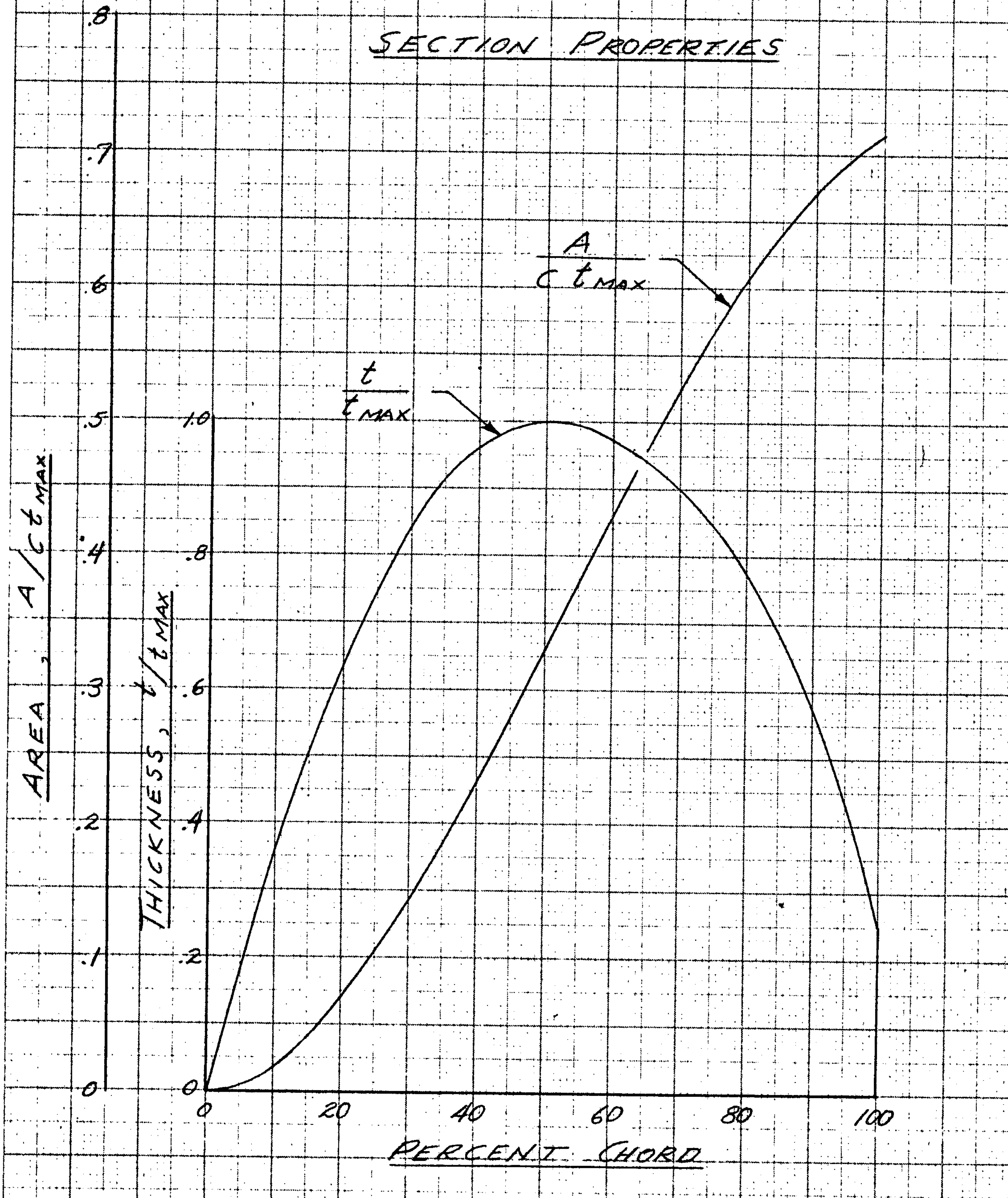
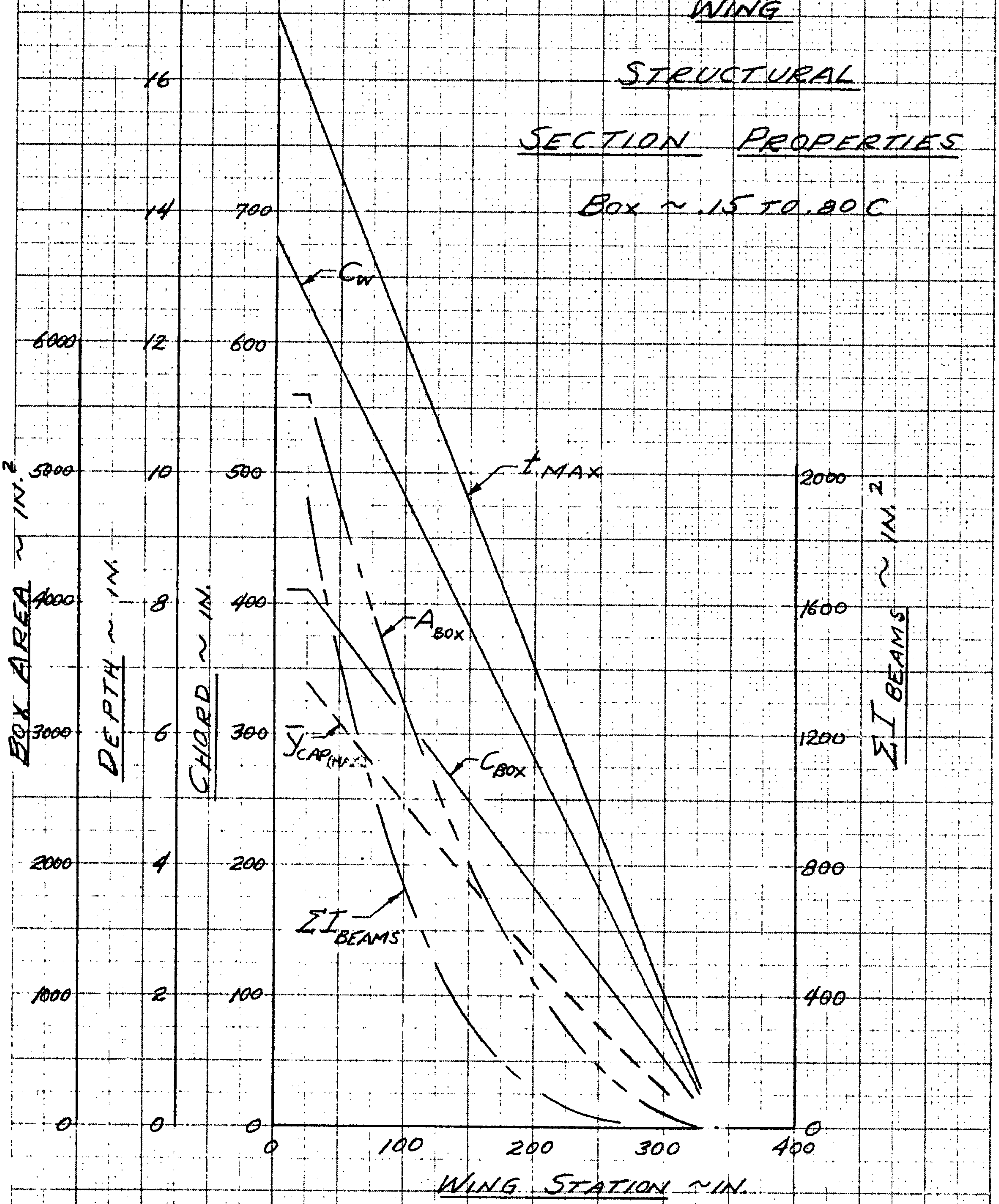


FIG 8

WING STRUCTURAL SECTION PROPERTIES

BOX ~ .15 TO .80 C



WING (Continued)

Internal Loads and Analysis

The internal loads are calculated from the critical external loads given in Figure 6 for the subsonic room temperature condition by means of the structural section properties given in Figure 8.

The beam cap design loads, stresses and cross section areas are summarized in Figure 9. The axial load shown is for the highest loaded beam. All beams are similar in cross sections and as noted in the figure have a constant area for most of their span. This makes for ease of fabrication and is efficient because tapering of material is accomplished by the number of beams decreasing with span station. Beam caps are machined from B-120VCA titanium rolled bar.

Beam web design shear flows, stresses and web gages are summarized in Figure 10. Due to the effects of beam taper, the vertical shear in the beam webs is very low and a minimum gage of .016 sheet is sufficient. Material is B-120VCA titanium cold rolled sheet. Stiffeners are sheet metal angles of the same material spaced at approximately three inches along the beams. Front and rear closing spars are of similar construction but web gage is .040 in order to maintain the torque box stiffness.

Wing upper and lower surfaces are designed by the torsion shear flows given in Figure 11 plus the effects of bending due to air loads and, in the case of the wing fuel tank region, fuel vapor pressure. The outer skin is

WING (Continued)

Internal Loads and Analysis (Continued)

.020 B-120VCA titanium cold rolled sheet and the inner skin is .025 B-120VCA titanium sheet which is formed in the annealed state and then heat treated. The depth of the corrugation varies according to the shear stability and pressure load bending requirements along the span.

The wing torsional stiffness for aileron effectiveness is presented in Figure 12.

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MODEL A-11

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FIG. 9

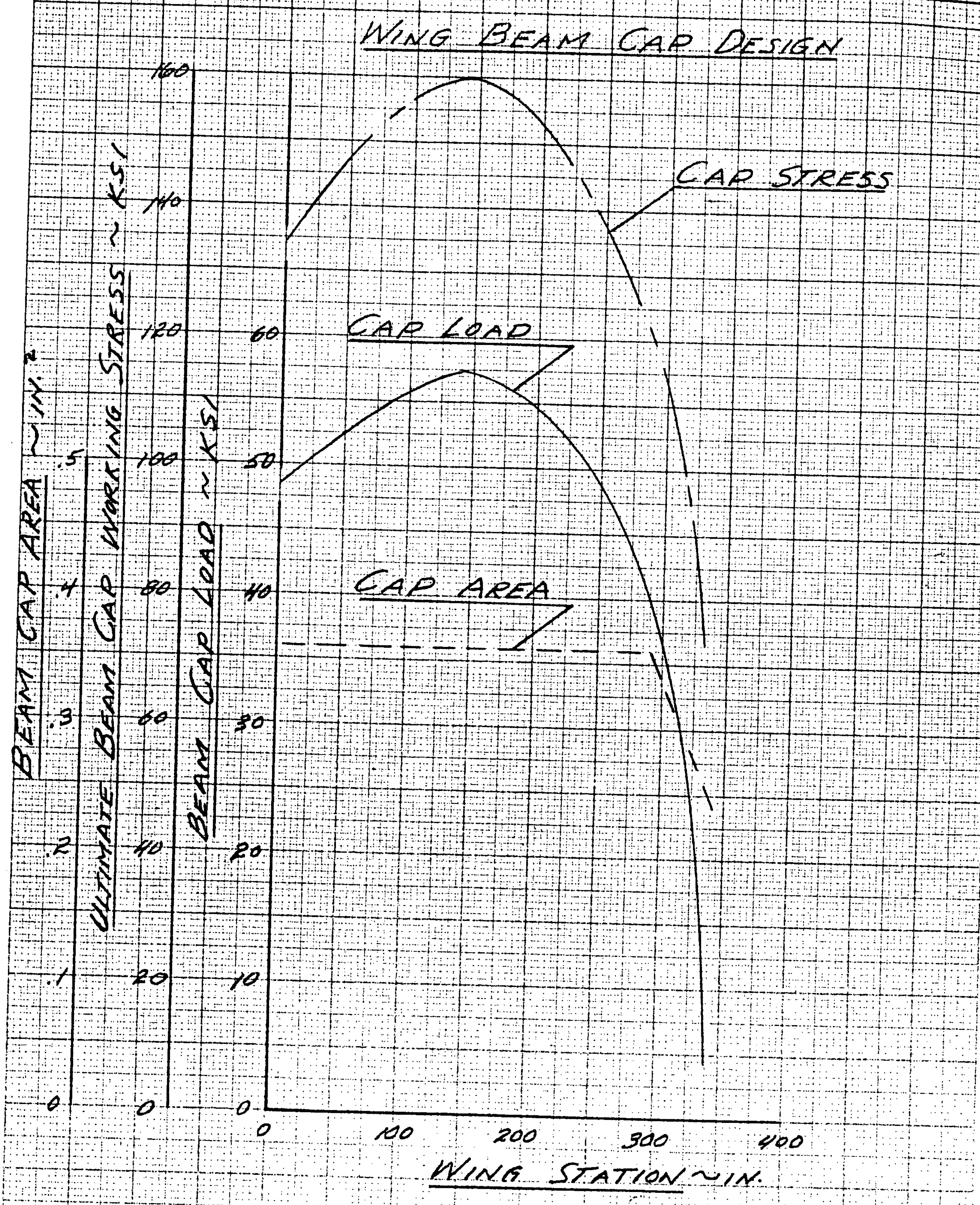

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FIG. 10

WING BEAM WEB DESIGN

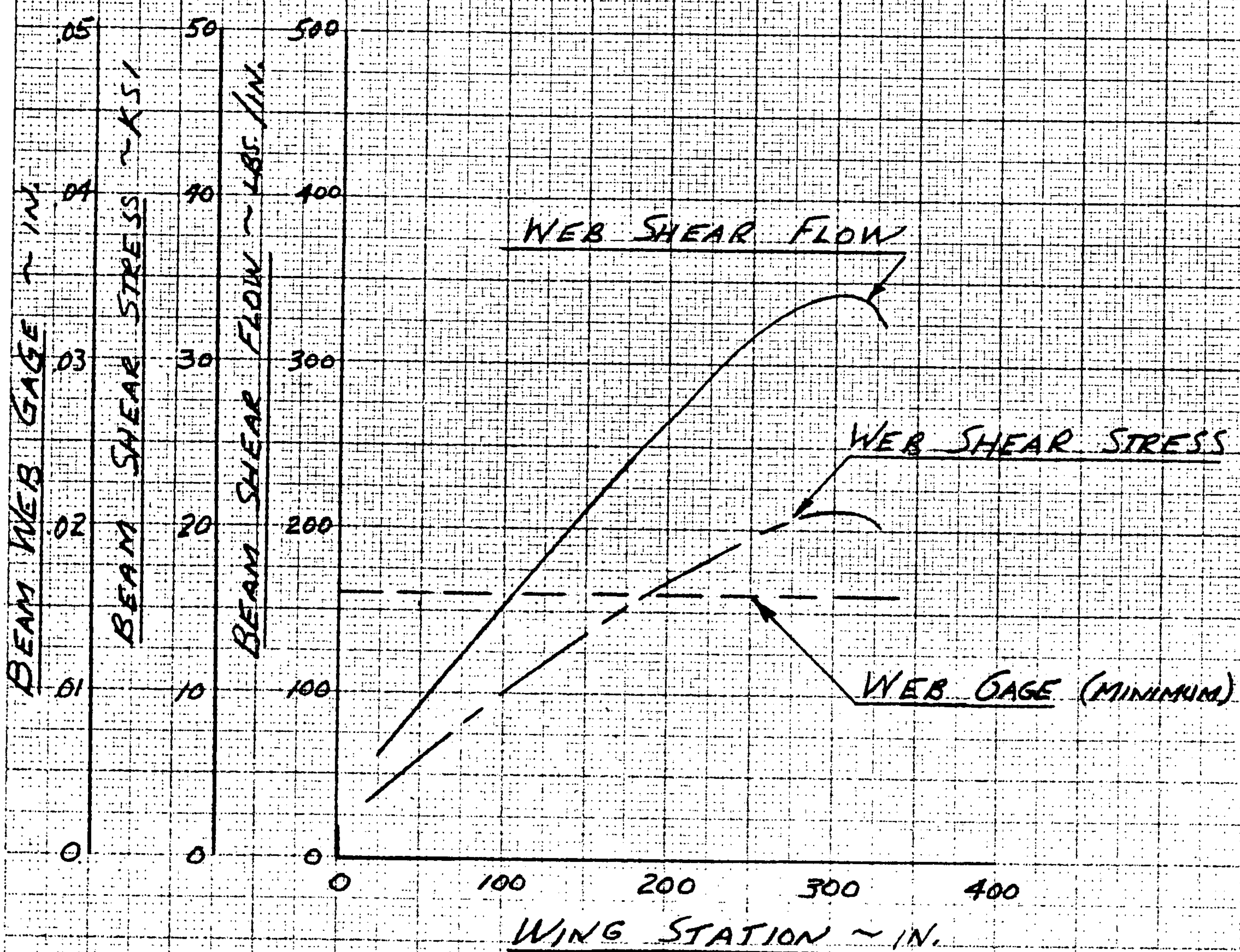
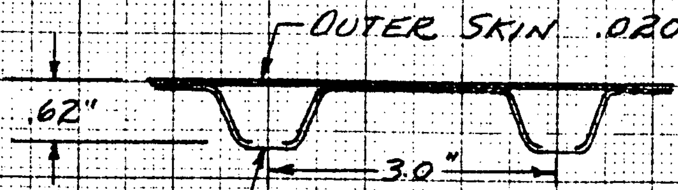


FIG. 11

WING SURFACE DESIGN



TYPICAL SECTION

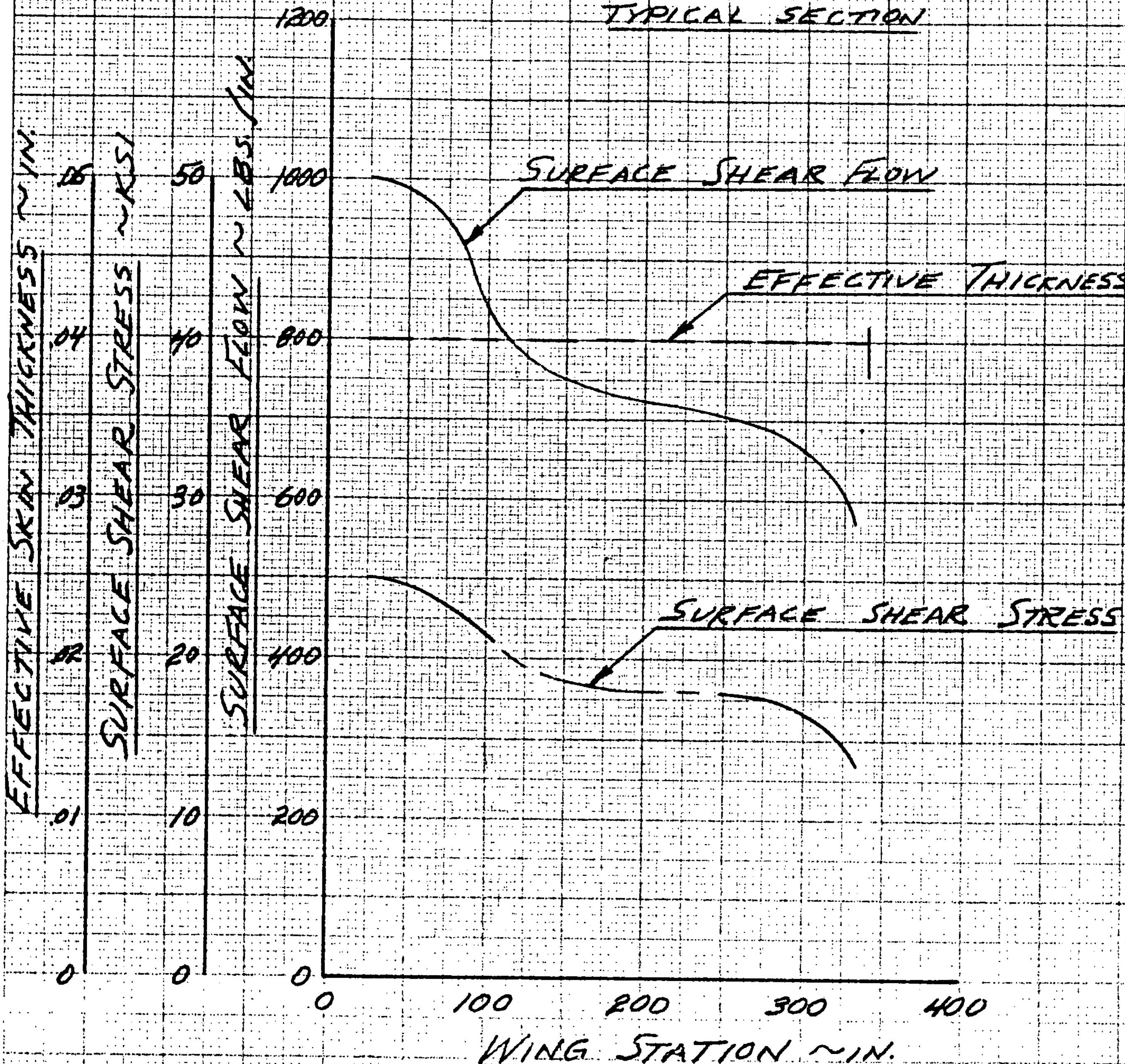


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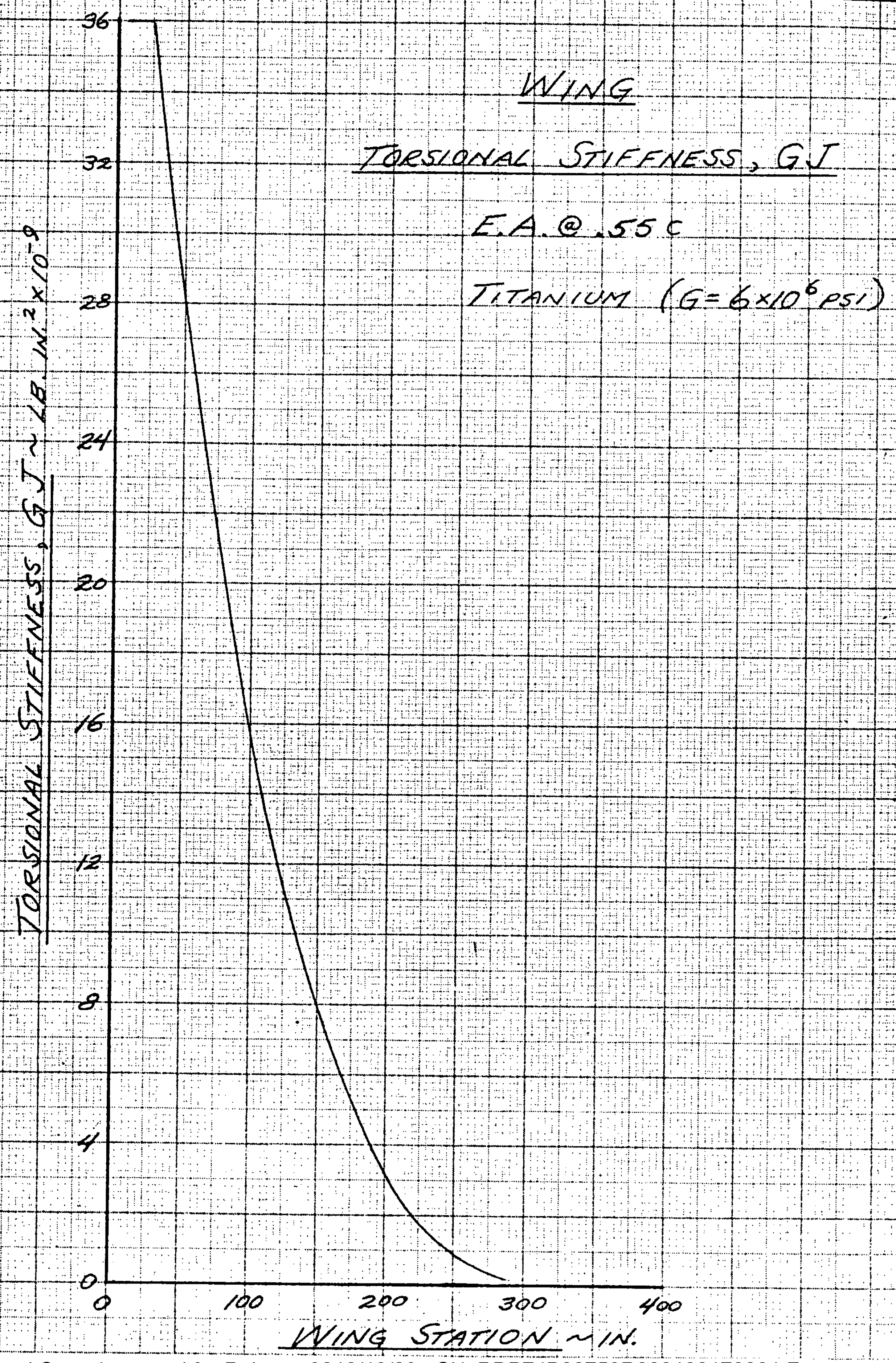
FIG. 12


WING
TORSIONAL STIFFNESS, GJ
E.A. @ .55C
TITANIUM ($G=6\times10^6$ PSI)
TORSIONAL STIFFNESS, GJ ~ LB. IN.^{2} × 10^{-9}
36
32
28
24
20
16
12
8
4
0
0
100
200
300
400
WING STATION ~ IN.

FUSELAGE

Description

The fuselage consists of three major assemblies; the forward, mid and aft sections as noted on the Inboard Profile. The construction of the fuselage in three sections will greatly facilitate fabrication of the structure and installation of the functional equipment required in each section. The provision of service joints on these fuselage sections permits rapid disassembly of the aircraft for transporting purposes.

The forward fuselage section contains the Flight Station, Military Equipment compartment, nose landing gear, air conditioning compartment and suitable compartments for the installation of electronic, navigation and communication equipment. The remainder of the forward section contains the forward fuel tanks.

The mid fuselage section provides for attachment of the wing box section and contains the main landing gear and mid section fuel tanks.

The aft fuselage section provides for attachment of the aft portion of the fin box section and contains the aft section fuel tanks and the landing chute.

The fuselage fuel tanks are of the integral type providing maximum fuel capacity for a minimum size structure.

FUSELAGE (Continued)

Description (Continued)

The fuselage structure is of semi-monocoque construction, consisting of skin, rings and four longerons. Since most of the fuselage structure adjacent to the skin is subjected to high temperatures for long periods of time, the material used is a titanium alloy (B-120VCA). For internal structure, where temperature is maximum at 300°F, 2024T6 or 2024T81 aluminum alloys will be used. The minimum skin gage is .016 at the nose, increasing to a maximum of .040 at the center section. Rings will be of gage comparable to the skin except the main frames in the center section. Rings (2.0 in. deep channel sections) will be spaced approximately 15.0 in. c.c., with two (1.0 in. deep) Z section intermediate rings spaced between, giving a panel spacing of 5.0 c.c. Four longerons, B.L. 14.0, left and right, resist up and down bending moments. Side bending is resisted by tension in the side skin and B.L. 14.0 upper and lower longerons on the opposite side. Longerons will be formed sheet metal channels, with inner and outer caps tying the channels together. The outer cap also acts as a splice plate for the skin and rings, and the inner cap as a splice plate for the inner flange of the 2.0 in. deep rings. Spot welding will be used extensively because of weight, low cost, reliability and strength.

The fuselage shell will be made in four parts, spliced longitudinally at the longeron points. This "quarter shell" breakdown permits spotwelding

FUSELAGE (Continued)

Description (Continued)

to be used extensively. The "quarter shells" will be spliced together by a maximum of two longitudinal rows of titanium (B-120VCA) rivets at each of the four longerons.

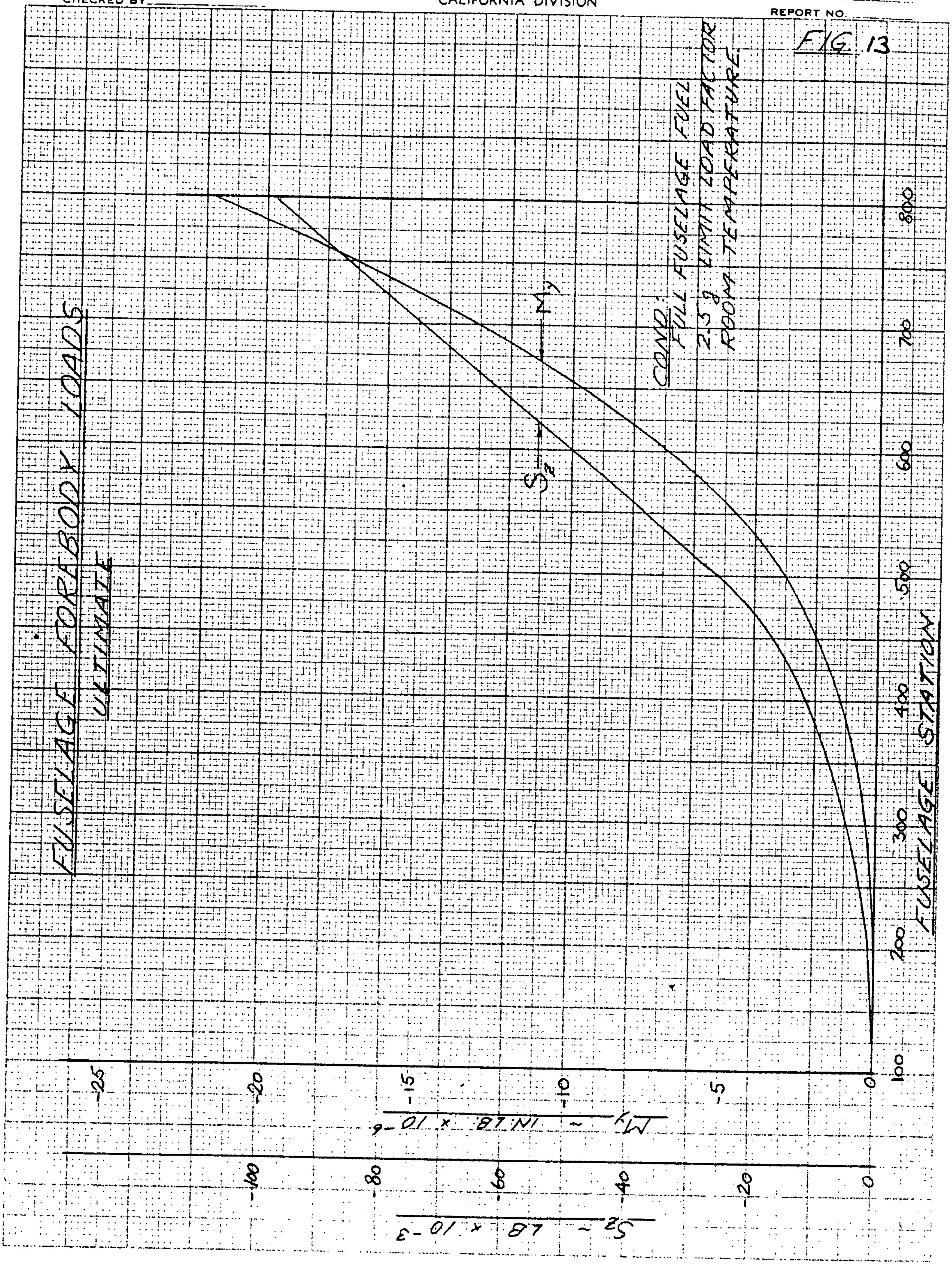
Figure 13 is a shear and moment curve, for the forward fuselage, critical for room temperature condition. The shear and moments for elevated temperature conditions are almost as critical. Figure 14 shows the longeron loads, areas, stresses, skin shear flows and skin thicknesses required. A detailed sketch, Figure 15, of typical lower longeron is shown. The upper longeron is similar but approximately half of the area of the lower longeron at any given fuselage station. Also a sketch, Figure 16, showing typical side shell construction and ring splice at longerons, is included.

The cockpit section is similar to the basic shell except that the upper longerons support the canopy and cockpit pressure loads. Pressure bulkheads in this area and other internal structure will be considered to be made of 2024ST aluminum alloy if temperatures are below 300°F.

Fuselage skin is also considered to carry internal pressure of 15.0 psi ult. due to fuel pressure in the fuselage fuel tank region. Surge bulkheads, where temperatures remain below 300°F will be made of 2024ST aluminum alloy.

 V-31

FIG. 13


FUSELAGE FOREBODY LOADS
ULTIMATE
COND:
FULL FUSELAGE FUEL
2.5 g LIMIT LOAD FACTOR
ROOM TEMPERATURE
M_y
S_z
FUSELAGE STATION
100
200
300
400
500
600
700
800
M_y ~ IN.LB. x 10^{-6}
-25
-20
-15
-10
-5
0
S_z ~ LB x 10^{-3}
-100
-80
-60
-40
-20
0

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BASIC FOREBODY STRUCTURE

FIG. 14

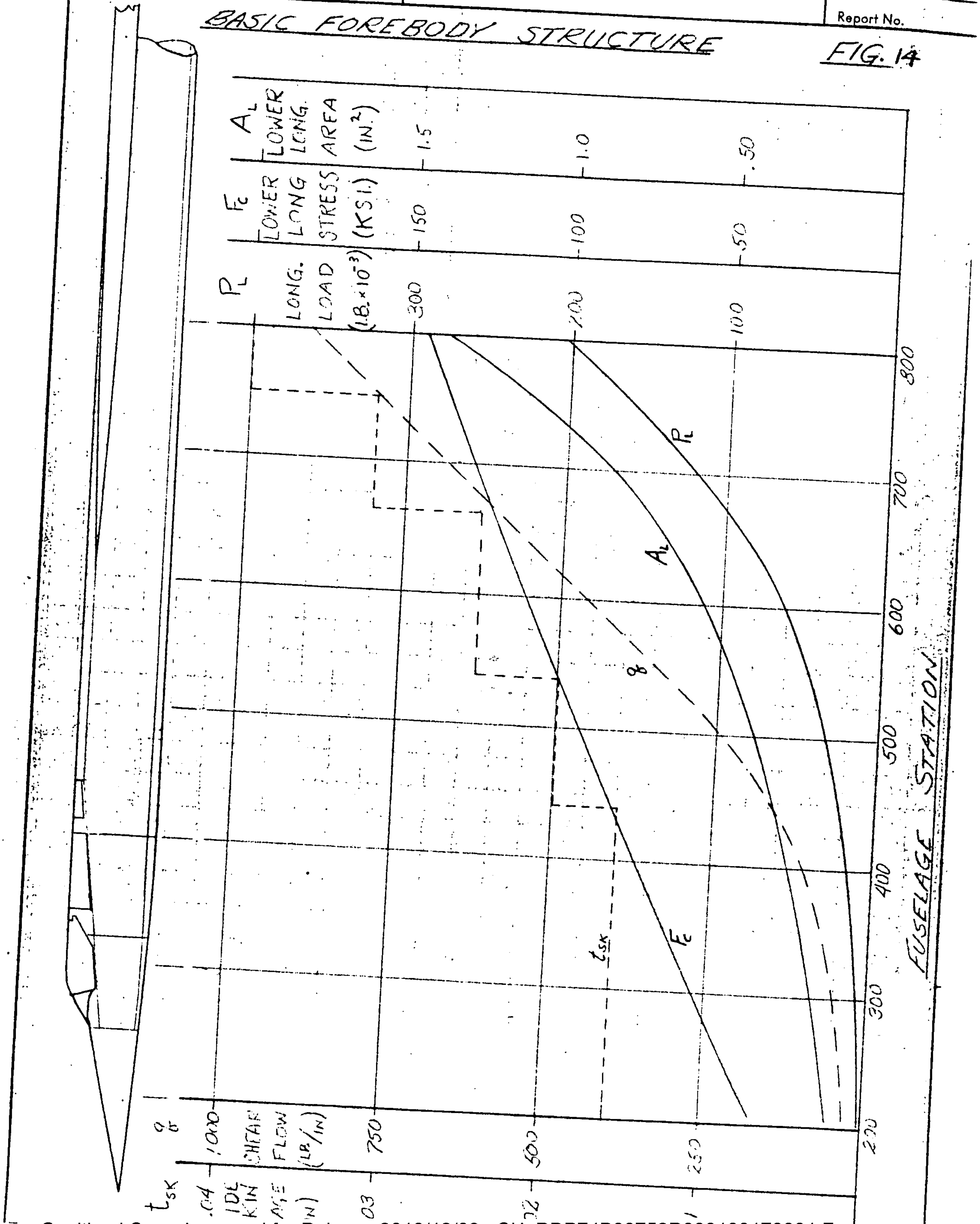
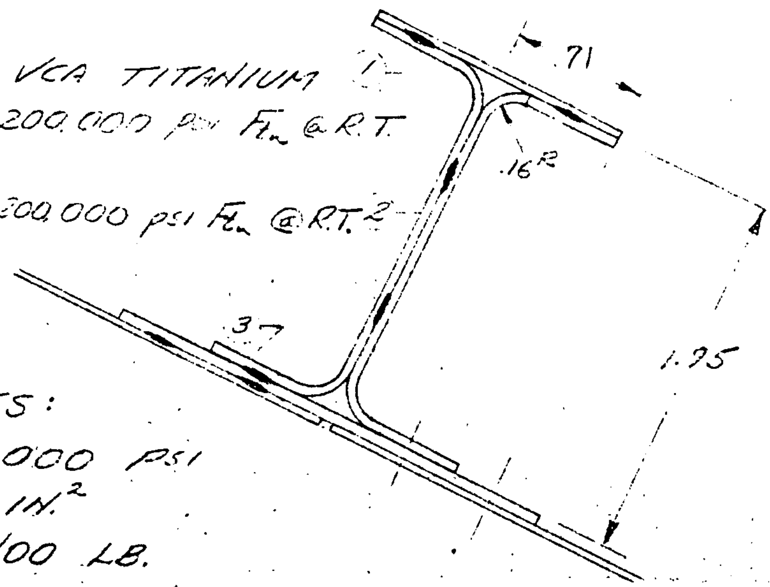


FIG. 15

TYPICAL LOWER LONGERON SECTIONS

F.S. 600

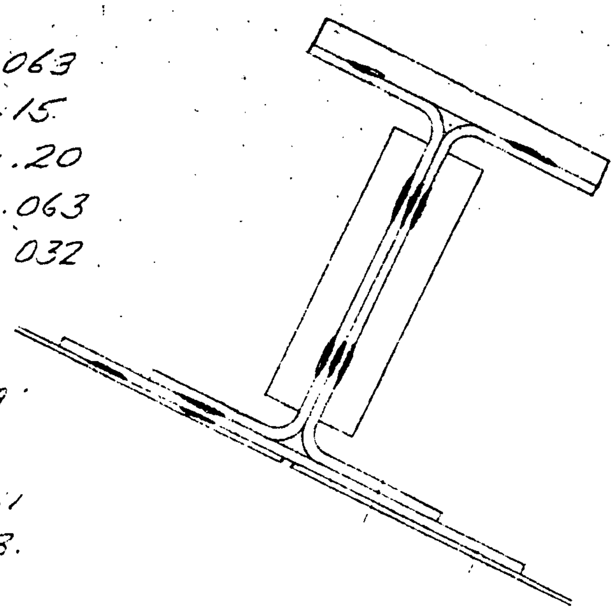
MAT'L: B 120 VCA TITANIUM
SKIN: .032, 200,000 psi F_{tu} @ R.T.
REMAINDER
.050, 200,000 psi F_{tu} @ R.T.



COMPRESSION
ALLOWABLES:
F_c = 110,000 psi
A_{eff} = .54 IN^2
P_{ALL} = 59,400 LB.

F.S. 800

MAT'L: B120 VCA TITANIUM
200,000 PSI F_{tu} @ ROOM TEMP
THICKNESSES:
CHANNELS ~ .063
TOP STRAP ~ .15
SIDE STRAPS ~ .20
SPLICE ~ .063
SKIN ~ .032



EFFECTIVE AREA
1.40 IN^2
F_c = 145,000 PSI
P_{ALL} = 203,000 LB.

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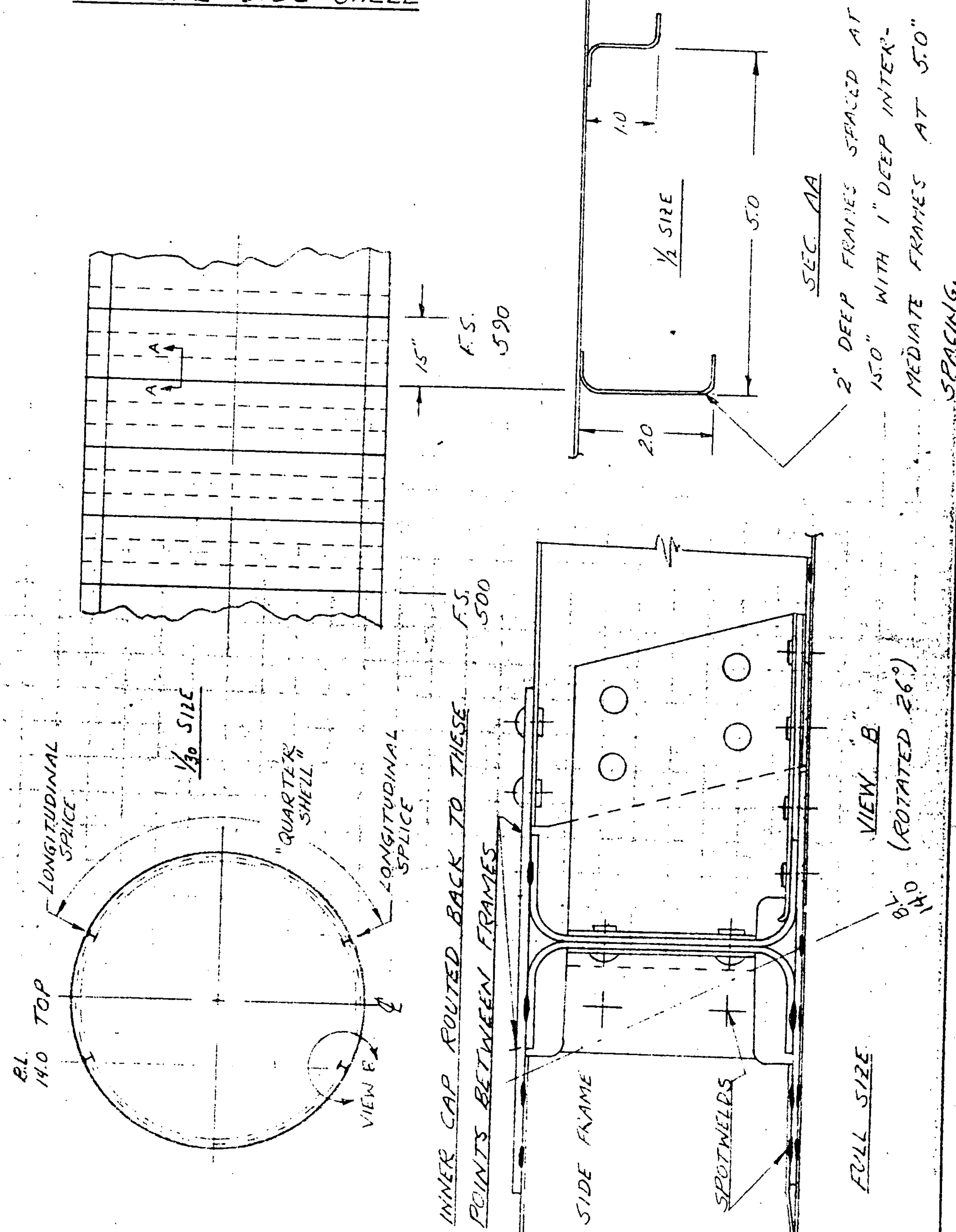
TYPICAL SIDE SHELL

FIG. 16



LANDING GEAR

The landing gear is of the conventional tricycle configuration with both main and nose gears retracting forward.

Static MG hub is at Sta. 915, WL 29.6. Static NG hub is at Sta. 477.5, WL 24.2.

The main gear stroke is 18½ in. and NG stroke is 15 in. With the ratio of 85,000 lbs. takeoff weight to 40,000 lbs. landing weight, and by virtue of the lengthy MG stroke, gear strength capabilities are determined mainly by ground handling conditions, (taxi, braking, and tow).

$$n_g \text{ (for landing)} = \frac{\left(\frac{V^2}{2g}\right)}{\eta_{STRUT}}$$

Assuming 7 ft./sec. sink speed and $\eta_{strut} = .9$,

$$n_g = \frac{49}{2 \times 32.2 \times 2 \times .9} = .84$$

and for the ground conditions, static loads, with CG at 25% MAC (Sta. 868) =

$$P_{V_{G_M}} = \frac{85,000}{2}\left(\frac{390.5}{437.5}\right) = 38,000 \text{ lbs.}$$

$$P_{V_{G_N}} = 85,000 - 76,000 = 9,000 \text{ lbs.}$$

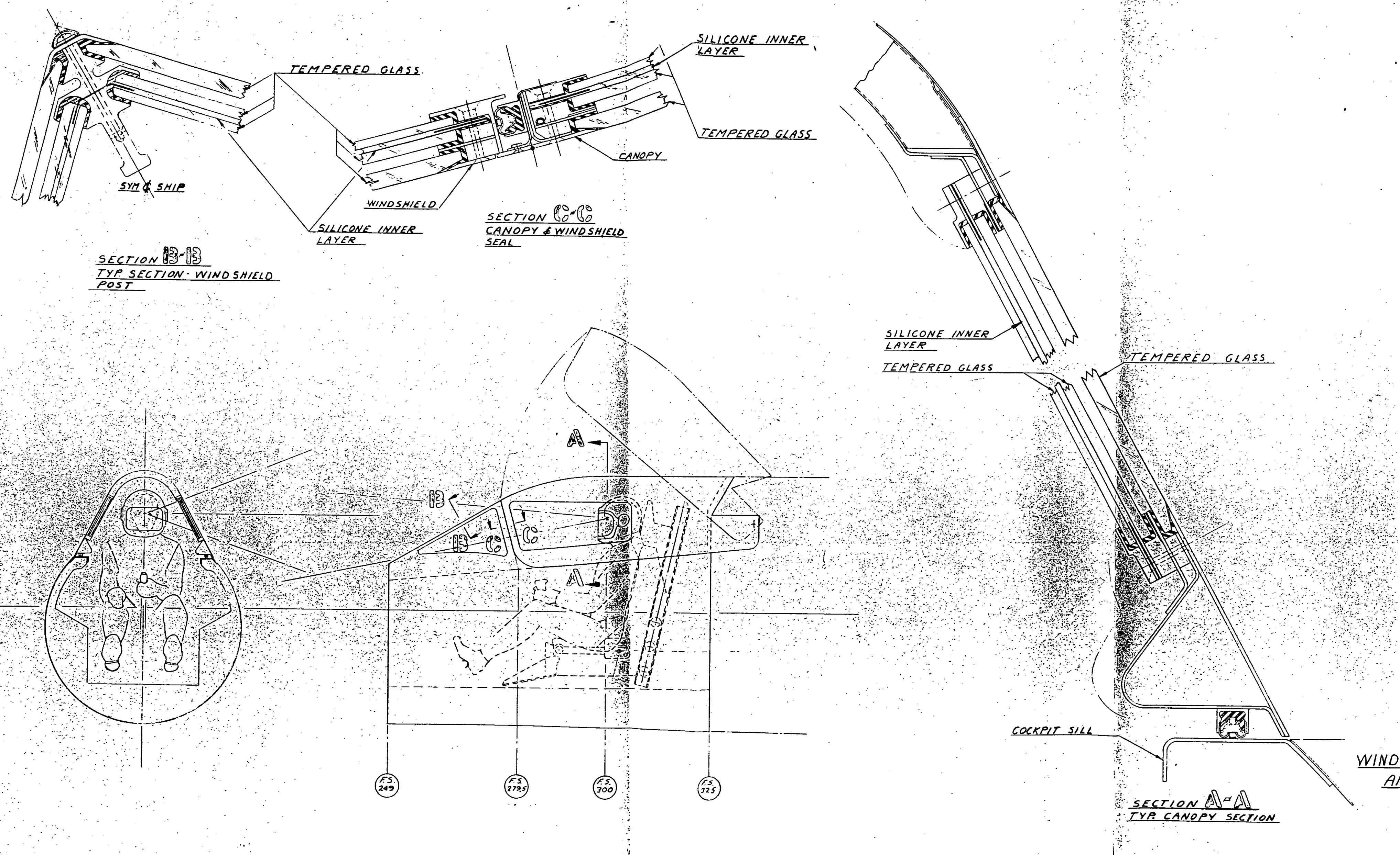
The main gear tires are 40 x 12 of 26 ply rating. Nose gear tire is 26 x 6.6 EHP, Type VII.

COCKPIT ENVIRONMENT

The general arrangement of the cockpit, windshield and canopy is as shown in Figure 1.

This is the simplest and lightest configuration which we believe adequate to provide the required comfort and safety for the pilot in the flight regime which this airplane will encounter.

In this section the problems of air conditioning, emergency escape and personal equipment are given separate consideration.


TEMPERED GLASS
SYM ℄ SHIP
SILICONE INNER LAYER
SECTION B-B
TYP SECTION · WINDSHIELD POST
WINDSHIELD
SILICONE INNER LAYER
TEMPERED GLASS
CANOPY
SECTION C-C
CANOPY & WINDSHIELD SEAL
SILICONE INNER LAYER
TEMPERED GLASS
TEMPERED GLASS
COCKPIT SILL
SECTION A-A
TYP CANOPY SECTION
F.S. 249
F.S. 279.5
F.S. 300
F.S. 325
WIND.
AF

AIR CONDITIONING

Pressurization

The gaseous nitrogen released from liquid storage for cooling purposes, as discussed below, serves also to pressurize the cabin. Various cabin pressurization schedules were investigated, each selected for study on the basis of its particular effect on the following "critical" criteria:

Nitrogen required for pressurization alone.

Cabin differential pressure.

Pilot comfort in descent.

Isobaric Schedule - Unpressurized ram operation from sea-level until 26,275 feet cabin altitude is reached, then cabin altitude remains isobaric at this 26,275 feet at all higher airplane altitudes. Cabin differential increases during climb, reaching 5 psi as the airplane arrives at maximum altitude. This system requires the minimum nitrogen for pressurization of any possible schedule, using 161 pounds on a mission flight consisting of 400 knot climb, 123 minute cruise, and 200 knot descent. (This compares to 202 pounds for the same flight using the constant rate of climb system below). During descent at 200 knots the maximum cabin rate of descent is only 3930 fpm. On a 400 knot descent the cabin rate of descent goes as high as 33,500 fpm; however, the more important rate of absolute pressure increase associated with this at the altitude concerned is exactly the same as the maximum encountered with the military-type schedule below, and involves less sustained time at high rate.

AIR CONDITIONING (Continued)

Pressurization (Continued)

Constant Cabin Rate of Climb Schedule - The cabin altitude changes at constant rates set by the pilot to carry the cabin from initial to final altitude in the exact time spans of the airplane's climb and descent. A cabin differential of 5 psi is reached at initial cruise altitude, and remains constant during cruise, resulting again in a 26,275 ft. cabin at maximum altitude. However, this system has the inherent peculiarity of requiring cabin differential to reach a high of 6 psi on the way to or from cruise altitude. It was selected for study as requiring the maximum pressurization nitrogen of any practical schedule, using 202 pounds for the mission flight noted above. Pilot comfort during descent, on the other hand, is by far the best of any possible system, as indicated by the cabin pressure rate changes of 1460 fpm during 200 knot descent and only 8070 fpm at 400 knots. The latter rate compares to the above noted 33,500 fpm maximum for the isobaric system, and to 21,200 fpm reached with the military-type system below.

Military-Type Schedule - Unpressurized ram operation from sea-level until 5,000 ft. cabin altitude is reached, then isobaric pressure is held at 5,000 ft. until cabin differential has built up to 5 psi, with constant 5 psi differential at all higher airplane altitudes (above 18,365 ft). Since the first two systems above spanned the mission flight nitrogen requirement from minimum to maximum, this more normal system's nitrogen usage

AIR CONDITIONING (Continued)

Pressurization (Continued)

Military-Type Schedule (Continued)

was investigated for descent only. Here the 200 knot descent nitrogen amounted to 33 pounds, compared to 12 pounds and 40 pounds on the isobaric and constant rate systems, respectively, for the same descent. 400 knot descent nitrogen was also calculated for this system, amounting to only 9 pounds. The main use made of this particular schedule, however, was in investigating the relative pilot comfort between it and the isobaric schedule, during the 400 knot descent ($3\frac{1}{4}$ minutes). On the military-type schedule, after leaving maximum cruise altitude at time zero, the pilot spends the first 125 seconds subjected to cabin descent rates varying from 0.85 up to 15.5 inches of mercury/minute, whereas, with the isobaric schedule no cabin change whatsoever occurs during this time span. For the remaining 70 seconds both of these schedules would follow the same rate curve (approaching 21 inches of mercury/minute at sea-level), except that at 20 seconds the pilot with the military-type system starts a half minute of reprieve at zero rate in his 5,000 ft. isobaric cabin, then returning to the high rate for the final 20 seconds. This comparison shows both systems to be relatively severe on the pilot, such that he should not attempt such a rapid descent unless blessed with exceptional ear and nasal passages, or in an emergency. Study of the exact rate curves vs. elapsed time would seem to give the isobaric system

AIR CONDITIONING (Continued)

Pressurization (Continued)

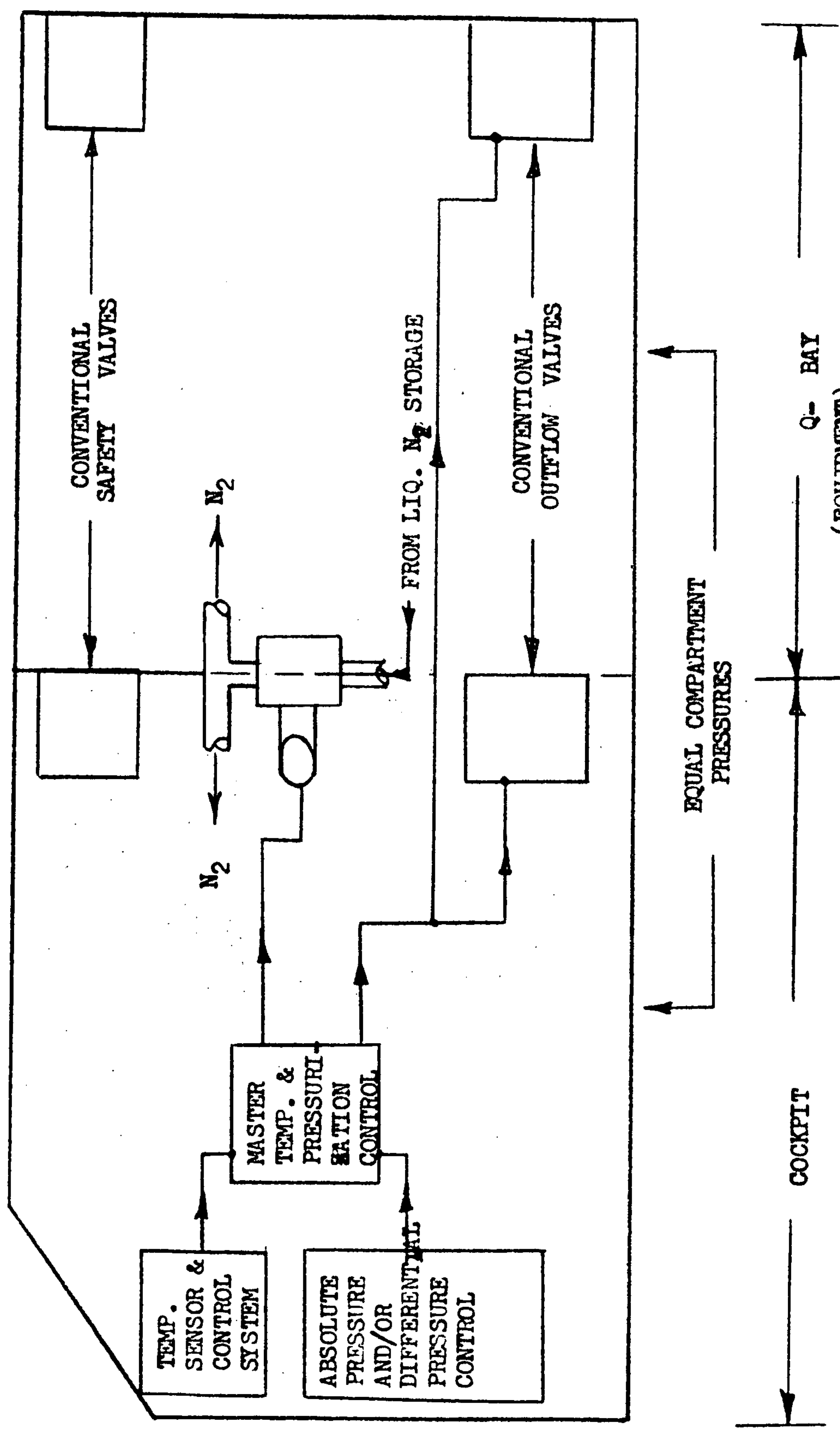
Military-Type Schedule (Continued)

a slight edge over the military-type, on the basis of sustained times at high rate of change; this could probably be a point of argument between any two given pilots, however.

For the present, no attempt is being made to finalize the pressurization schedule, but merely to have at hand the information on which the discussions above were based. This is necessarily the case because of the inter-relationship between the nitrogen required for pressurization, and that required for cooling. It is felt that until final decision on the cooling system is reached, it will not be possible to give proper consideration to all factors for both systems.

Thus, no physical concept of the actual control components for pressurization can be stated at present; however, Figure 1a shows schematically a simplified general concept covering the inter-related controlling that must be accomplished. The master controller, whatever its form, must accept signals from both the temperature and the pressure sensors, and then influence both the outflow and the nitrogen flow valves accordingly. For example, with an increasing cooling requirement the master control must simultaneously increase the flow of cooling nitrogen, while opening the outflow valve to prevent over-pressure. Note that this example by itself

FIGURE 1a



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AIR CONDITIONING (Continued)

Pressurization (Continued)

Military-Type Schedule (Continued)

does not point up the necessity for such an inter-relating master control, since it is obviously quite a normal function of a pressure sensor to directly control its outflow valve towards open for such a case. However, reversing the example, assuming sufficient decrease in cooling-nitrogen inflow to drive a direct-controlled outflow valve fully closed, would result in depressurization. (The outflow valve by itself cannot "pump up" the cabin, being capable only of controlling a higher pressure generated at or beyond the point of cabin inflow. Note the dissimilarity between the more normal case of an "infinite" bleed air source available to a cabin, and the present "release it only as you need it" source). Now the need for the master becomes more evident, since it must recognize that even though temperature-wise the nitrogen flow can be reduced, it must still signal for nitrogen as an inflowing pressure source. (The temperature controller at this time will function only to position the recirculation bypass valve or valves).

For the latter condition of pressure-nitrogen requirement exceeding that for cooling, the outflow valve will close completely so that the only nitrogen flow will be that required for leakage make-up (plus or minus that involved in maintaining the contained weight of cabin atmosphere

AIR CONDITIONING (Continued)

Pressurization (Continued)

Military-Type Schedule (Continued)

during climb and descent). The "pressurization alone" values for nitrogen, quoted under the various schedules above, were calculated on this basis.

Note in Figure (14) that the series arrangement of outflow and safety valves gives double protection to the pilot against cockpit depressurization. For example, if the equipment bay were to depressurize for any reason the cockpit remains fully pressurized. As an alternate example, if the cockpit's outflow valve became stuck in the open position, the cockpit would again remain fully pressurized by riding on the equipment bay's valve. The probability of simultaneous-open failures is very low.

With the ram operation proposed for the unpressurized portions of the above described schedules, and the variable pressure source available during pressurization, it is considered that no negative pressure differential problem can normally exist, even during the 400 knot descent. In this regard calculations were made to determine the required variation between nitrogen flow rates for the 400 knot and 200 knot descents, to maintain full pressurization on the military-type schedule. This had been considered a possible problem on the fast descent from the standpoint of that nitrogen portion required just to increase the contained weight of cabin atmosphere. The results show, however, that even though the 400 knot descent time was faster by a factor of almost 6, its nitrogen

AIR CONDITIONING (Continued)

Pressurization (Continued)

Military-Type Schedule (Continued)

discharge rate (including leakage make-up) merely doubled. Further in regard to negative differentials, for the case of a nitrogen system failure during rapid descent, the outflow and safety valves will all be of the vacuum relief type and so sized as to prevent excessive structural loading.

Cooling

In the early stages of investigation, a look was taken at air-cycle ram cooling, with several variations of machinery and water boilers. As might be expected, the size and weight of the required equipment, plus material development problems due to the temperatures involved, eliminated this as a possible solution.

Engine bleed air was peremptorily eliminated for cabin use due to the airplane performance losses associated with bleed at our altitude. Note, however, that it is planned to use limited amounts of bleed air for windshield defogging and for ram air heating, if required, during the unpressurized portion of the pressurization schedules discussed above. The latter usage would become especially important were in-flight refueling to be considered, as here the normal time of ram operation would be far exceeded.

The most recent investigations have been aimed at accomplishing certain assumed design directives as follows:

AIR CONDITIONING (Continued)

Cooling (Continued)

> 100°F space temperature in the cockpit and equipment bay.
>
> 135°F maximum touch temperature of the trim in areas not directly over conductive structure, with minimum possible touch temperatures in all other areas.
>
> Cooling to be accomplished entirely by liquid media stored aboard the airplane (nitrogen, and possibly water).
>
> Cooling medium to double as cabin pressure source, as discussed above under pressurization.

The above temperatures take into account the fact that the pilot's comfort will be at the much more suitable level associated with direct suit ventilation by nitrogen gas, as on the X-15. This will include pilot-selected temperature controls.

One of the most attractive features of having aboard liquid nitrogen is the ease with which spot cooling of critical areas or equipment components can be accomplished. Thus such local areas are considered to be no problem.

A recirculation system was investigated on the basis of the above temperatures, wherein cabin atmosphere was cooled in two stages: first by passage through the air side of a water boiler, and then by injecting into it liquid nitrogen which topped-off the required cooling. This system was considered unattractive weight-wise at the time.

AIR CONDITIONING (Continued)

Cooling (Continued)

The most recent work, just completed, was a detailed study made of a water-panel system, for handling the major cooling load in those portions of the cabin wall between structural rings. Until the final stages of this investigation were reached, and the results could be integrated, this system looked extremely attractive. Regrettably, the final system weight has turned out to far exceed that of much less elaborate systems, even though as expected the amount of water expended was very small.

The studies made to date serve to indicate that the cooling problem, while severe, is not so extreme but that it is completely feasible to accomplish a practical system within the weight allowance set forth elsewhere in this report. This would be so even for a "nitrogen alone" system, and note in this regard that nitrogen's heat of vaporization amounts only to approximately a tenth that of water, at the pressures involved.

In ensuing investigations it is intended to exploit still further the advantages of using water's high heat of vaporization, in combination with top-off cooling by the "double-duty" pressurization nitrogen.

EMERGENCY ESCAPE

The A-11 will be provided with both low altitude and high altitude escape capability. Zero velocity escape systems will be carefully evaluated and used if feasible.

The pilot's seat will eject upward and will be provided with a rocket catapult.

This catapult assures adequate ejection clearance to the tail of the aircraft during all operating regimes. The long nose of the aircraft, which places the pilot far forward of the tail, and the low aspect ratio of the tail allow the man and the seat in an emergency ejection to clear the airplane tail with a considerable margin of safety.

The seat accommodates a suitable survival kit and bail-out oxygen system, and has a special shoulder harness and lap belt automatic release system.

Emergency ejection at the high altitudes and high speed at which the mission of the aircraft will be performed introduces new aspects to be considered in the problem of escape. The opening shock of the parachute, chute oscillation and the rotation of the pilot's body during free-fall must be retained within tolerable limits. The heat generated during ejection at cruise Mach number and deceleration of the man have been considered to insure that they are within body tolerances.

EMERGENCY ESCAPE (Continued)

Wind blast is a consideration during emergency ejection. The cruise Mach number, although high, corresponds to only 275 knots EAS. Maximum EAS to be used will be approximately 400 knots which occurs less than 15% of the flight time.

Wind blast effects and deceleration at these velocities are relatively low and safe ejections can be made within airspeeds with which we have had experience starting with the first jet fighters.

Heating of the man during ejection is not a serious problem. At 100,000 feet altitude, Mach 3.2 with the man in the seat, deceleration time is approximately 52 seconds. During this period the man and seat decelerate from Mach 3.2 to terminal falling velocity.

The initial ten seconds of this deceleration is the critical period so far as aerodynamic heating is concerned. After this ten seconds, the ejected pilot has slowed to approximately Mach 2.3 with a corresponding reduction in temperature. Some improvement of the temperature resistance of face pieces and pressure suit details is probably required to accommodate this transient condition.

It is proposed to use automatic seat belt and shoulder harness release of the type used in the F-104 to permit the pilot to remain in the seat until the initial deceleration is complete. This will reduce the hazard of pilot injury due to spin and tumbling, and prevent premature deployment of the primary chute.

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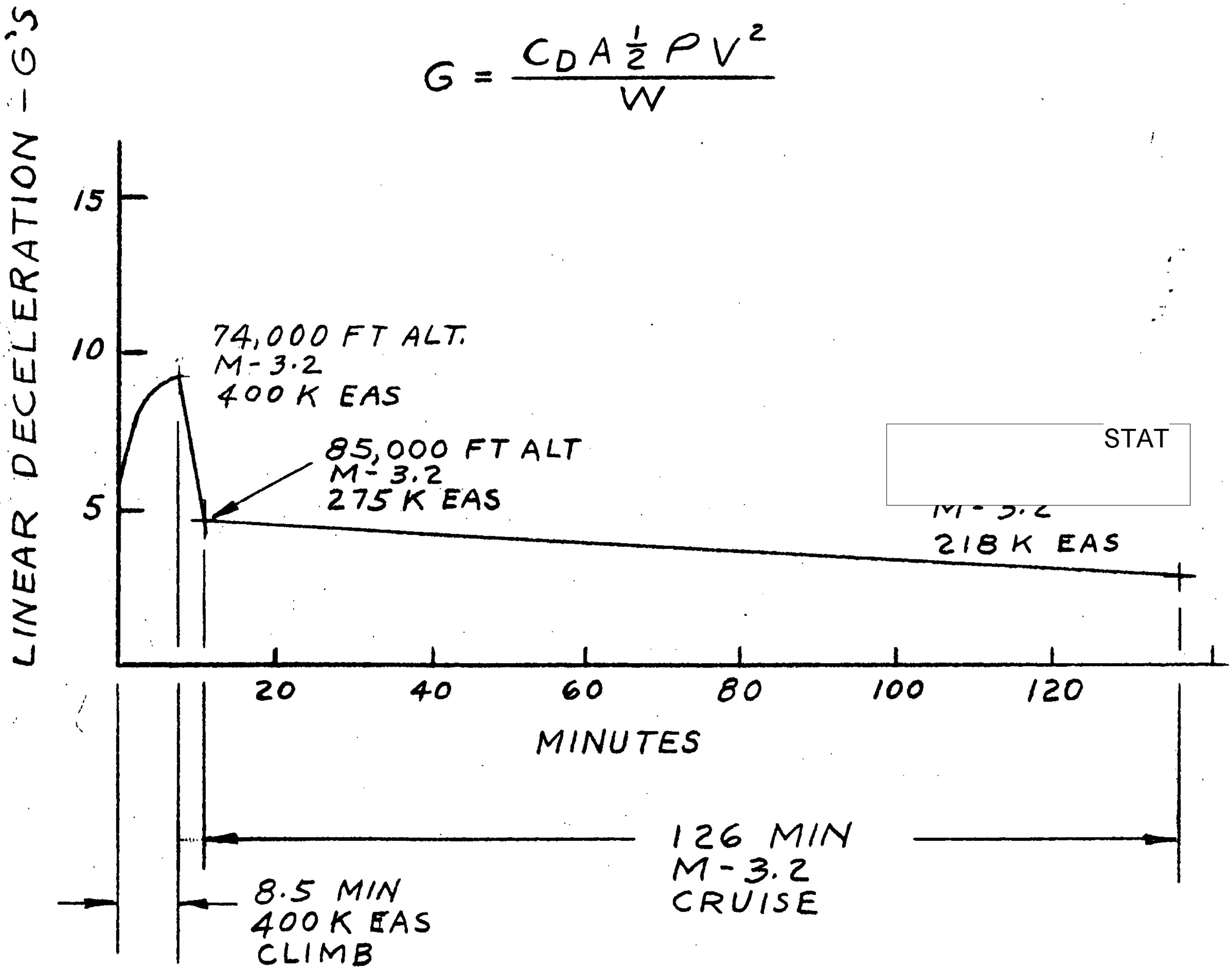
EMERGENCY ESCAPE

MAN & SEAT DECELERATION
STABILIZED CONVENTIONAL SEAT

$$G = \frac{C_D A \frac{1}{2} \rho V^2}{W}$$



FORM 5767A 1

PERSONAL EQUIPMENT

In order to be compatible with the airplane and emergencies including escape, it is necessary to initiate a personal equipment development program for this particular airplane. The general nature of this program is to adapt and modify presently proven equipment and to develop a minimum of new equipment as required. A partial-pressure suit, such as used so successfully in the U-2 program is proposed as the basis for the pilot's wear.

The cockpit temperatures of 100°F require that the pilot be provided with some sort of a cooling and ventilation garment.

Lockheed understands that pressure suit development is being done at Edwards Air Force Base and proposes to make use of this program to provide the A-11 with the latest advance in full pressure or partial-pressure suits that the state of development permits.

A liquid oxygen system will be installed to provide oxygen for breathing and emergency suit pressure for the pilot.

To reduce the chances of system failure and interruption of oxygen flow to the pilot a dual system will be installed. Oxygen pressure failure, malfunction of system components, and loss of oxygen due to leaks will not abort the mission or endanger the pilot because the operative system can be isolated from the malfunctioning system.

PERSONAL EQUIPMENT (Continued)

The dual system design concept is included in the survival kit which contains the dual oxygen regulators. The dual oxygen regulators are joined together by a manifold valve which automatically accepts oxygen from the operative regulator when a regulator malfunctions, and blocks loss of pressure through the inoperative regulator.

The pilot also has a control valve on the survival kit to isolate the malfunctioning regulator. A pressure gauge is included to check the maintenance of oxygen pressure in the emergency bail-out bottle within the survival kit.

GENERAL

To obtain the maximum performance of the A-11 airplane, one of the available boron fuels will be used in the engines' afterburners. The superior heating value of boron fuels over hydrocarbon (26,500 Btu/lb. vs 18,400 Btu/lb.) will have a net result of approximately 12% overall increase in airplane range. A much greater increase of performance can be realized when the boron fuels can be burned in the main burners of the turbojet; however, since there has been only a limited amount of testing of boron fuels in turbojets, this will not be considered at this date. On the other hand, there have been highly successful runs made with boron fuels (HEF-2 and Hi Cal-3) on both afterburners and ramjets on test stands and with ramjets in flight on the Lockheed X-7 vehicle.

The A-11 airplane will carry approximately 31,000 pounds of boron fuel and 17,000 pounds of hydrocarbon fuel in separate tanks. This does not appreciably complicate the tankage problem as multiple tanks are required so an optimum airplane c.g. can be maintained during the mission by proper fuel scheduling from the various tanks. (See Figure 1 of "Structural Description" Section for c.g. location vs. time curve.)

It should be noted that the tendency of HEF fuels to cause high-altitude vapor trails is unknown at this time. Ground testing of this material results in large volumes of white vapor, but the dilution occuring in flight may reduce this to invisibility. A high speed, high altitude test in the X-7 would answer this question.

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AIRPLANE FUEL SYSTEM

The A-11 airplane will carry 18,000 pounds of fuel in the wing tanks and 30,000 pounds of fuel in the fuselage. The bulk of the wing fuel will be JP-150; the small amount of HEF-3 in the wing will be burned before aerodynamic heating of the fuel becomes a problem.

The main HEF-3 tanks in the fuselage will be pressurized to approximately 8 psig. Of this, 2 psi will be used for fuel transfer and the remaining pressure will be used to prevent the fuel from boiling due to aerodynamic heating. This pressure will be maintained by dry nitrogen gas to insure an inert blanket over the HEF.. The nitrogen used will be carried aboard in liquid form to save the weight of high pressure bottles and also to serve as a heat sink for functional components.

Center of gravity control will be accomplished by scheduling the fuel from the various tanks as shown in Figure 1 of "Structural Description" section.

All tanks, fuselage and wing, will be integral with the structure. This saves considerable weight and gives maximum utilization of volume for fuel. The tanks will not be insulated; however, if aerodynamic heating of the fuel becomes a problem, it may be necessary to refrigerate the fuel prior to takeoff to insure adequate heat absorbing capacity.

FORM 8767A-1

BORON FUELS

There are several boron fuels currently being produced in limited quantities. By the end of 1959, Olin-Mathieson's and Callery's large plants will be "on stream", which will permit accelerated engine development and flight testing which has been hampered to date by the shortage of material. Olin-Mathieson Chemical Corporation will produce HEF-3, which is ethyldecaborane ($C_2 H_5 B_{10} H_{13}$). Callery Chemical will produce Hi Cal-III which is similar to ethyldecaborane $(C_2 H_4)_x B_{10} H_{14}$. Both of these fuels are relatively easy to handle and store as compared to the earlier boron fuels HEF-1 (pentaborane) and HEF-2 (propylpentaborane) which had many "nasty" characteristics and were difficult and dangerous to handle. The new fuels HEF-3 and Hi Cal-III are less toxic, have lower vapor pressure, are not pyrophoric, and are compatible with JP-150. Their thermal stability has been improved and their heating values are slightly higher. There is an increase in viscosity which will require somewhat more power to run pumps, but this is no major problem. Listed below are some of the properties of HEF-2 and HEF-3.

	HEF-2	HEF-3 Until 12-31-61
Thermal Stability		
a. Decomposition and solid formation	To be investigated.	Less than 1% decomposition by weight as evidenced by gas evolution and zero solid formation at the conditions listed below.

Time	Temp.
2 hrs.	350°F
1 hr.	390°F
30 min.	400°F
1 min.	500°F
15 sec.	570°F

FORM 3767A 1

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BORON FUELS (CONT.)

	HEF-2	HEF-3 Until 12-31-61
b. Increase in viscosity		After being held for 30 min. at 400°F, the viscosity shall not have increased more than 2 cs at 77°F.
Heating Value (77°F) Refined as the reaction yielding Amorphous B_2O_3 and H_2 vapor	24,100 Btu/lb nom. 23,600 Btu/lb min.	25,800 Btu/lb. nominal 25,500 Btu/lb min. 26,200 Btu/lb max.
Specific Gravity (77°F)	0.70 nominal 0.65 minimum	0.82 nominal 0.80 minimum
Viscosity at 77°F	1.5 cs nominal 3.0 cs maximum	7 cs nominal 9 cs maximum
at -40°F	To be investigated.	150 cs maximum
Vapor Pressure 77°F	1.2 psia nominal 3.0 psia maximum	0.01 psia nominal maximum to be investigated.
Freezing Point	-76°F	-76°F
Spontaneous Ignition Temperature	Pyrophoric	260°F nominal 250°F minimum
Compatibility with JP-6 Fuel	Incompatible	Compatible under N_2 atmosphere
Storage Stability	Zero solids formed after 3 months storage in an inert atmosphere at temperatures in the range of -65°F to +160°F.	Zero solids formed after 6 months storage in an inert atmosphere at temperatures in the range of -65°F to +160°F.
Flash Point	Pyrophoric	160°F minimum
Boiling Point		468°F to 510°F

FORM 5767A 1

BORON FUELS (CONT.)

Toxicity

The products of complete combustion of HEF-3 or Hi Cal-III are water and boric acid, which is a very mild acid (eye wash). Little, if any, trouble is expected from this source with the possible exception of some damage to vegetation around ground run-up areas.

The unburned fuel is highly toxic when inhaled or absorbed through the skin or swallowed. Since these fuels have a strong distinct odor, there is no excuse for trained personnel to inhale sufficient quantities to cause a health hazard. Absorption through the skin comes in much the same category as inhalation as it can be easily detected and can be washed off before any damage is done. Swallowing several c.c. of HEF-3 is entirely possible, but extremely improbable by trained personnel.

The Callery Company, producers of Hi Cal, have had lost-time accidents due to boron exposure, but in all cases these were not caused by the final product (Hi Cal-III) but by one or more of the more active agents used in processing such as diborane, boronhydrides, etc.

Material Compatibility

There has been enough work done on the material compatibility program to prove that the most of the materials proposed for the A-11 airplane are uneffected by HEF-3 or Hi Cal-III. The few questionable ones are non-structural so there will be no weight or performance compromises to be made on the airplane if substitutes are made.

BORON FUELS

Material Compatibility (Cont.)

As the A-11 airplane structure is predominately titanium, which is inert to boron fuels, this will reduce the amount of testing to a few metals.

There is a mild reaction between boron fuels and aluminum which may or may not be tolerable for airplane components; future test will verify this. The amount of aluminum which could be used on the A-11 airplane fuel system is limited to tank baffles and valve bodies for which there are several substitutes.

Steel, copper, stainless and most other structural metals are satisfactory for use in boron fuels. Rubber and most plastic materials must be avoided as they deteriorate rapidly in HEF. In most cases, these materials must be replaced with Teflon, Kel. F., Vitron, or one of the other fluorinated materials.

There are several currently active programs for development of sealant materials being carried out for WADC. The results are promising; however, it is anticipated that further testing and development will be required.

Boron fuels may be used with hydrocarbons such as the JP fuels and lubricating oils, provided the latter are free of water which hydrolizes

BORON FUELS

Material Compatibility (Cont.)

the fuel and forms a boric acid deposit which could clog fuel lines, etc.

Chlorinated material such as carbon tetrachlorid must not be used around or with HEF as the reaction compounds are explosive and shock sensitive. Some of the other halogen compounds must likewise be avoided for the same reason.

Fuel Handling

The new boron fuels HEF-3 and Hi Cal-III can be handled safely if a few safety precautions are taken. The fuel should be kept under an inert atmosphere such as nitrogen gas. Spills should be avoided even though the fuel is not pyrophoric. If fuel is spilled, it should be washed away with water or burned to avoid the vapors from being inhaled. If it is impractical to burn the fuel, it may be hydrolized with a water-methanol mixture (50-50).

Personnel handling quantities of boron fuels should wear protective clothing such as rubber gloves and sleeves, gas masks and safety goggles. The most important thing, however, is adequate training and good common sense.

Before a container is fueled, it should be thoroughly cleaned and dried and purged with a dry inert gas such as nitrogen. The purging can

BORON FUELS

Fuel Handling (Cont.)

best be accomplished by pressure purging; i.e., filling the container with dry N_2 several times to its permissible operating pressure and bleeding to ambient. If venting is required when fueling the container, the vapor should be disposed of by burning or bubbling through a water-methanol trap.

Lockheed AIRCRAFT CORPORATION
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THERMODYNAMICS

A. Power Plant System

I. Engine Selection

Early in the Archangel series of airplanes, a comparison of the Pratt and Whitney J-58 engine and the General Electric J-93 engine was made. The performance comparison on a specific weight and specific thrust basis were almost identical up to about 75,000' altitude. Above that height the J-58 engine was better than the J-93.

Since maximum altitude was a major criterion, the engine having the best thrust/weight ratio could achieve the highest altitude. A preliminary analysis showed that approximately 3,000 feet greater altitude could be achieved with the J-58 engine. In addition the hardware development of the J-58 engine is approximately a year ahead of the J-93 engine. Consequently, the J-58 engine was selected as the A-11 airplane power plant.

II. Engine Performance

The installed J-58 engine thrust and fuel flows at maximum power are presented in Figure 1. The data are based on the use of JP-150 in the primary burner and HEF in the afterburner. The performances are based on the inlet recoveries shown in Figure 4. The data are for a climb speed of 400 knots E.A.S. up to 74,000 feet and at M = 3.2 above 74,000 feet. Figure 2 shows the variation of S.F.C.

FORM 3787A 1

THERMODYNAMICS (Cont.)

A. Power Plant System (Cont.)

II. Engine Performance (Cont.)

with afterburner power at M = 3.2. The effect of using HEF in the afterburner only results in a decrease in overall SFC of 12% over an all JP-150 system. This factor is substantially less than 28% theoretical gain and is due primarily to the present uncertainty of the condensation of boron-oxide in the nozzle and secondarily to the difference of the molecular weights of the combustion products. It is believed that with development, a greater factor will be achieved.

An engine weight of 5,950 lbs. was used to allow for the large ejector diameter, for the structure required to use the engine up to M = 3.2, 100,000 feet, as well as the dual fuel compatibility of the afterburner.

III. Induction System Performance

A two-dimensional external-internal compression inlet was selected for the induction system of the A-11 airplane. A schematic diagram of this inlet type is shown in Figure 3. This inlet was chosen since its layout fits the general layout of the airplane without sacrificing performance. A three dimensional inlet would provide slightly greater recovery, however, it would be harder to control. The final selection of the inlet would be made after an intensive wind tunnel program.

THERMODYNAMICS (Cont.)

A. Power Plant System (Cont.)

III. Induction System Performance (Cont.)

NASA has recently shown that at Mach numbers greater than 2.5, the inlet recovery is almost independent of inlet type. The pressure recovery within ± 3% is dependent on the amount of compression surface boundary layer removed. Table 1 summarizes the effect of boundary layer removal.

TABLE 1

Total Head Recovery, %	B.L. Bleed Req'd., % of Inlet Captured Flow
90	30
85	20
80	10
75	5

Although the high pressure recovery is attractive, the utilization and/or efficient disposal of above 10% of the inlet captured flow at high Mach number is extremely difficult without incurring large quantities of drag. For example, dumping 20% of flow in engine secondary nozzle results in an increase of approximately 8% in SFC.

Figure 4 shows the inlet pressure recovery assumed in the engine analysis. Also, shown are test data obtained in wind tunnel tests conducted both by this contractor in its supersonic transport studies and by NASA at M = 3.0. Also shown for comparison is the new proposed A.I.A. standard recovery curve.

FORM 3767A 1

THERMODYNAMICS (Cont.)

A. Power Plant System (Cont.)

III. Induction System Performance (Cont.)

Another factor in inlet selection is inlet drag. This drag is composed of the external drag (cowl pressure and friction drag) and the inlet spillage drag. With the common external compression inlet, the cowl pressure drag that accompanies the large compression surface angles, as well as the spillage drag at off-design conditions make the simple inlet impractical at high Mach numbers. With a mixed variable inlet, a low angle external cowl surface is possible minimizing the cowl pressure drag, and the spillage drag is supersonic which is considerably lower than subsonic spillage drag associated with spillage behind a normal shock.

The engine location was determined by the airplane c.g. requirements. An under-wing inlet was selected since this type of inlet would be insensitive to angle of attack. It was then necessary to determine inlet location, that is, ahead of or behind the wing shock.

Locating the inlet ahead of the wing shock has the following advantages:

a) Requires no boundary layer diverter.

b) Utilizes the top surface of the inlet for additional wing area.

c) Allow for a long subsonic diffuser thereby improving pressure distribution.

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THERMODYNAMICS (Cont.)

A. Power Plant System (Cont.)

III. Induction System Performance (Cont.)

On the other hand locating the inlet behind the shock:

a) Lowers the inlet Mach number thereby making it possible to achieve a higher recovery and lowering the inlet capture area.

b) Reduces the inlet momentum or ram drag of the engine thereby increasing net thrust, since the momentum decrement has been charged to the wing pressure drag. (No allowance was made for this in the performance).

c) Reduces the inlet capture area thus reducing the inlet-engine matching problem and associated spillage drag.

The principal disadvantage of this latter location is the requirement of a boundary layer diverter and its attendant drag. This contractor is currently running a full scale diverter wind tunnel test at the NASA Lewis Research Center to verify the drag data previously obtained with scale model diverters. Boundary layer diverter drag has been included in the performance.

In addition to the aerodynamic factors the latter location is shorter and therefore lighter. Consequently, the behind-the-wing shock under-wing inlet was initially selected subject to the wind tunnel program.

THERMODYNAMICS (Cont.)

A. Power Plant System (Cont.)

VI. Exhaust System

The Pratt and Whitney rectangular ejector is proposed for the A-11 exhaust system. The engine manufacturer's test data show that there is no difference in performance between a rectangular and circular ejector nozzle. The use of a rectangular ejector minimizes the base drag problem and allows greater ground clearance angle. The inlet compression surface boundary layer air, approximately 7% at cruise, will be ducted aft to act as the secondary airflow required by the ejector. At off-design conditions, a portion of the by-pass air required to minimize spillage drag will be dumped in the secondary nozzle. The quantity will have to be determined in wind tunnel tests.

THERMODYNAMICS (Cont.)

B. Aerodynamic Heat Transfer

I. Structural Temperatures

A steady state heat transfer analysis was made of a typical wing chord. Figure 6 shows the temperature distribution on the upper and lower surface of the wing. The data show that the average upper surface will be approximately 425° F and the average lower surface temperature will be approximately 475° F. The difference in temperature between the upper and lower surface is due to the compression resulting from the wing operating at 7° angle of attack. The heat transfer analysis was based on the Van Driest method which has been checked experimentally by NASA in free flight tests.

The temperature on the fuselage will vary from 400° F to 500° F. The minimum temperature will occur on the bottom surface with maximum occurring at the 10 and 2 o'clock positions due to effects of boundary layer crossflow at angle of attack. A temperature of 475° F will occur at the wing-fuselage intersection as well as at the intersection of any protuberances such as antenna masts. The protuberance temperature itself can be decreased by sweep. These data are based on calculation methods checked experimentally on the X-15 configuration.

The external windshield surface temperatures will vary from 770° F at the stagnation point to an average of 450° F over most of the

THERMODYNAMICS (Cont.)

B. Aerodynamic Heat Transfer (Cont.)

I. Structural Temperatures (Cont.)

canopy surface.

II. Fuel System

The engine manufacturer requirements as well as the fuel temperature never exceed 300° F for either JP-150 or HEF fuel.

The results of a preliminary investigation show that through judicious scheduling of fuel to regions which are less susceptible to rpaid transient temperature rise, an uninsulated fuel tank system can be used for the A-11 airplane.

Integral fuel tanks, both for the fuselage and wing, have been designed for the airplane. At this time, a detailed analysis has been made for the wing integral tanks and is presented below. No analysis, as yet, has been made for fuselage integral tanks; however, preliminary spot checks as well as conclusions drawn from a bag type fuselage tank system design, discussed below, show that the fuel temperature limitation can be adequately met. The bag type fuselage tank design study made represents a feasible tank design configuration in the event the sealing material problem associated with integral tanks fails to be resolved.

A preliminary heat transfer was made on typical fuel system

THERMODYNAMICS (Cont.)

B. Aerodynamic Heat Transfer (Cont.)

II. Fuel System (Cont.)

configuration. The study was divided into wing fuel tanks and fuselage fuel tanks. The wing fuel tanks were assumed to be integral and that the wing fuel would be used either during climb or the initial portion of the cruise. The fuselage fuel tanks were assumed to be in bags and to be used only during the cruise portion of flight. No insulation was used in either the wing or fuselage tanks. It is assumed that the wing tanks will be kept at 5 psi differential and fuselage tanks at 8 psi differential.

Figure 6 shows the time-temperature history of the wing integral tanks. The curve for use of wing fuel for climb shows that at the beginning of cruise the fuel temperature has risen 120° F. The temperature reaches the 300° F limit, assuming an initial 60° F temperature, after 25 minutes or 20% of the total flight. The rapid temperature rise experienced in the tanks are due principally to the tanks being empty. Also, shown in Figure 6 is condition if wing fuel is not used until beginning of cruise. Results show that 40 minutes are available before 300° F is reached assuming initial fuel temperature is 60° F. This time is more than adequate to use up the wing fuel.

Figure 7 presents the time-temperature history of the fuselage tanks. It should be noted that maximum fuel temperature rise is 52° F

THERMODYNAMICS (Cont.)

B. Aerodynamic Heat Transfer (Cont.)

II. Fuel System (Cont.)

where the vapor at top of tank rises 300° F.

It is emphasized that this preliminary study should be used to give an order of magnitude since results can vary if fuel quantities and schedules or tank configuration are different from those assumed.

The study also indicates that fuel can be routed from the fuselage to the wing tanks, during the entire mission if necessary, without exceeding the limit temperature. It is intended at a later date to use IBM 704 Thermal Analyzer to obtain a more accurate analysis and thereby determine the optimum fuel routing compatible with c.g. requirements as well as the actual tank configuration.

Another fuel problem other than material compatibility is the temperature effects on the residual fuel. It is presently planned to purge the tanks completely upon emptying the tanks so as to insure no residual fuel, since any residual fuel will result in tank coking.

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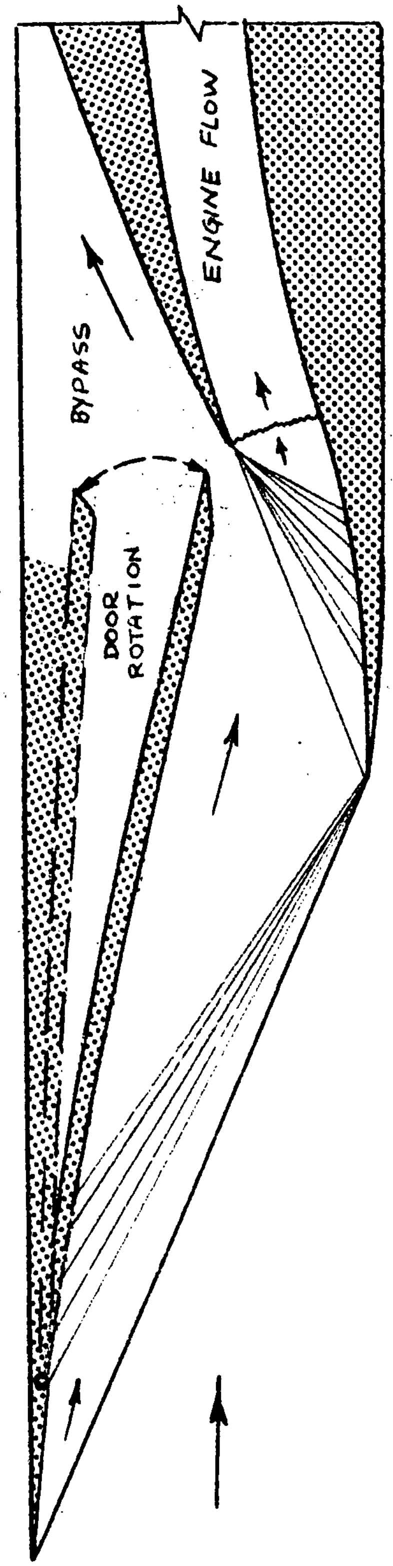
FIG. 3



SCHEMATIC OF 2-DIM. EXTERNAL-INTERNAL COMPRESSION INLET

FORM 3787A-1

 FIG. 3

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A-11

ESTIMATED INLET PERFORMANCE

FIG. 4

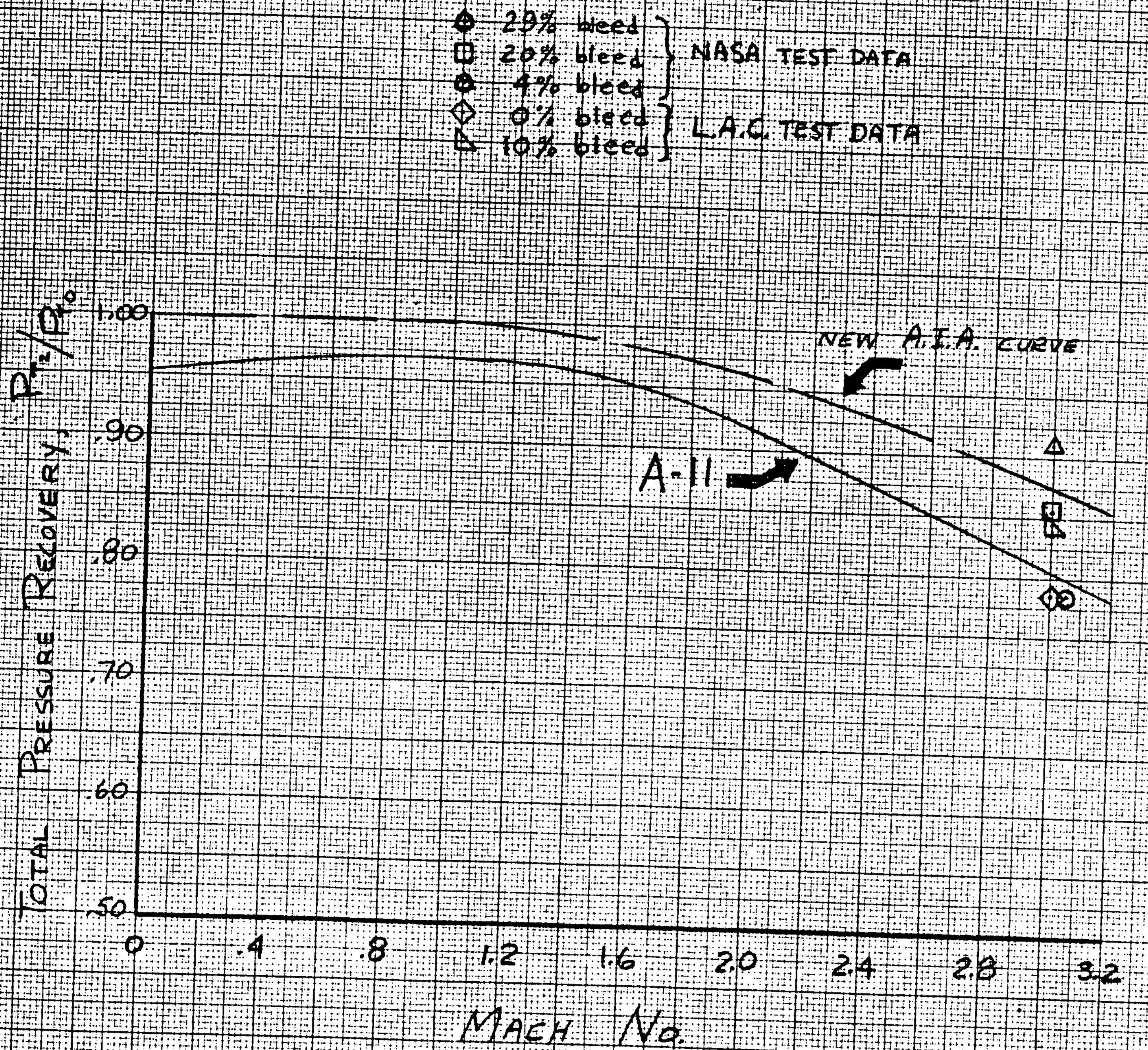

FIG. 4

FORM 527B A

REPORT NO.

FIG. 5

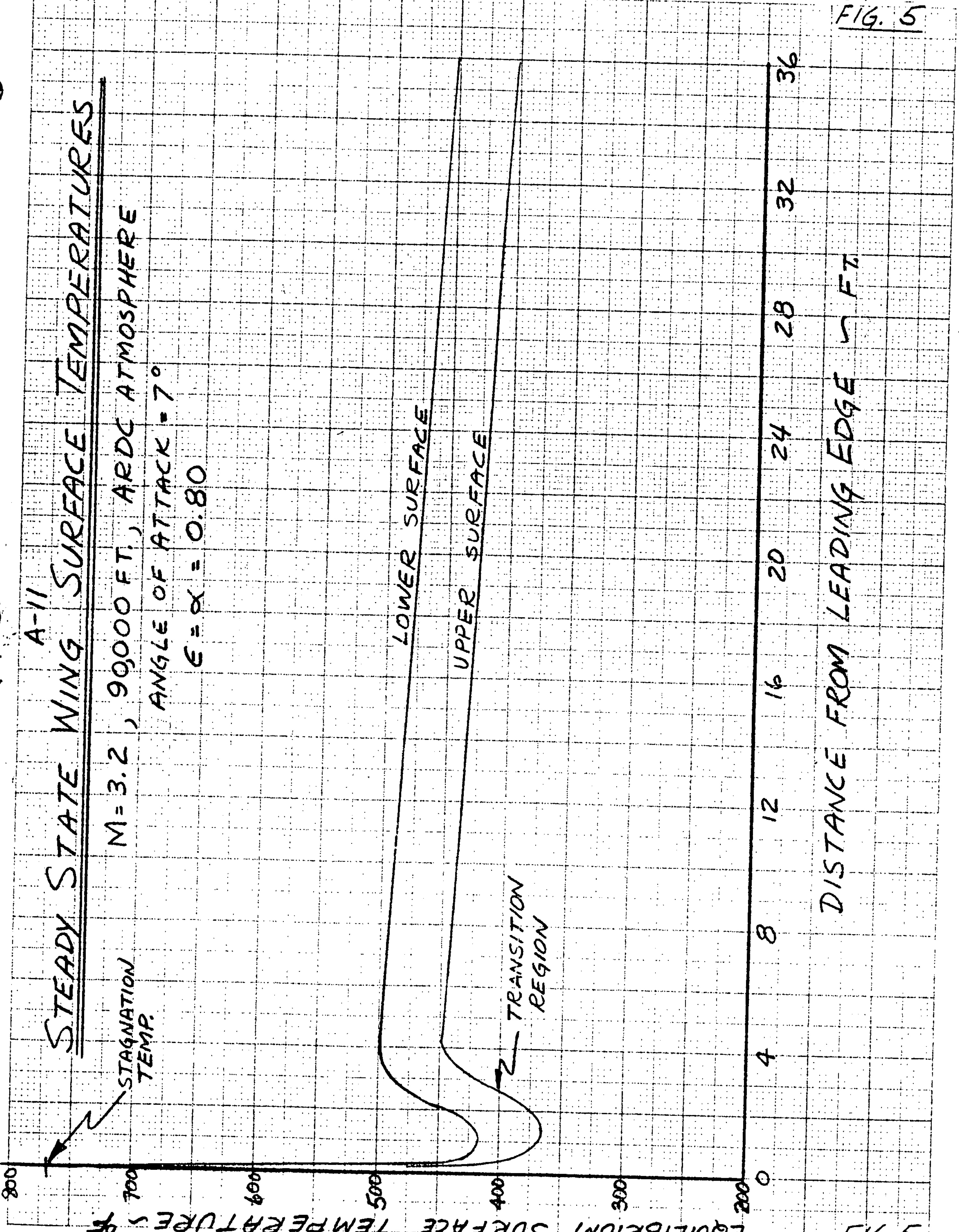

FIG. 5

FORM 527B

A-11

FIG. 6

FUEL AND WING SKIN TEMPERATURES
INTEGRAL TANKS

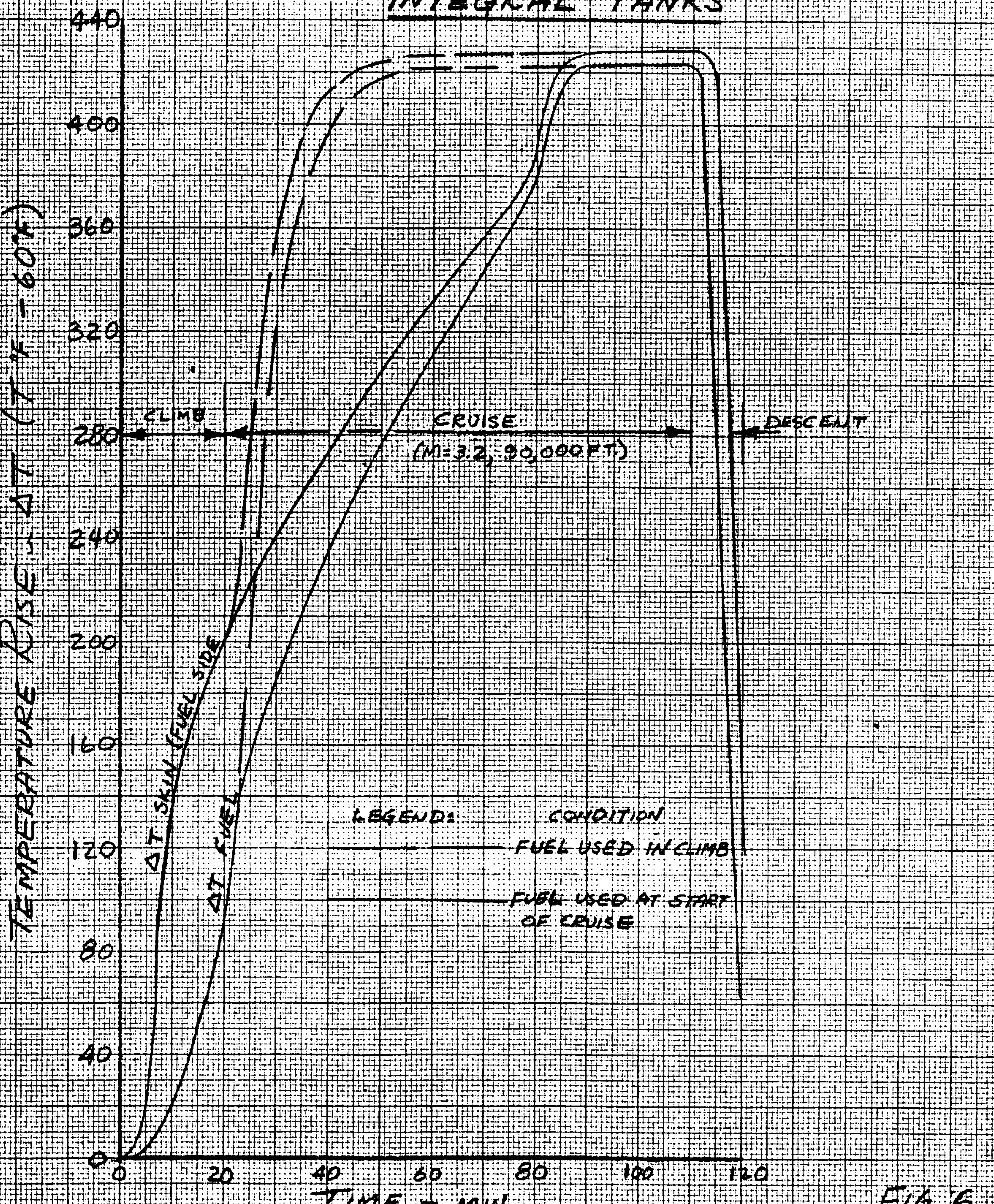

FIG. 6

FORM 527 B A

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MODEL

REPORT NO.

FIG. 7

ESTIMATED FUSELAGE FUEL AND WALL TEMP.

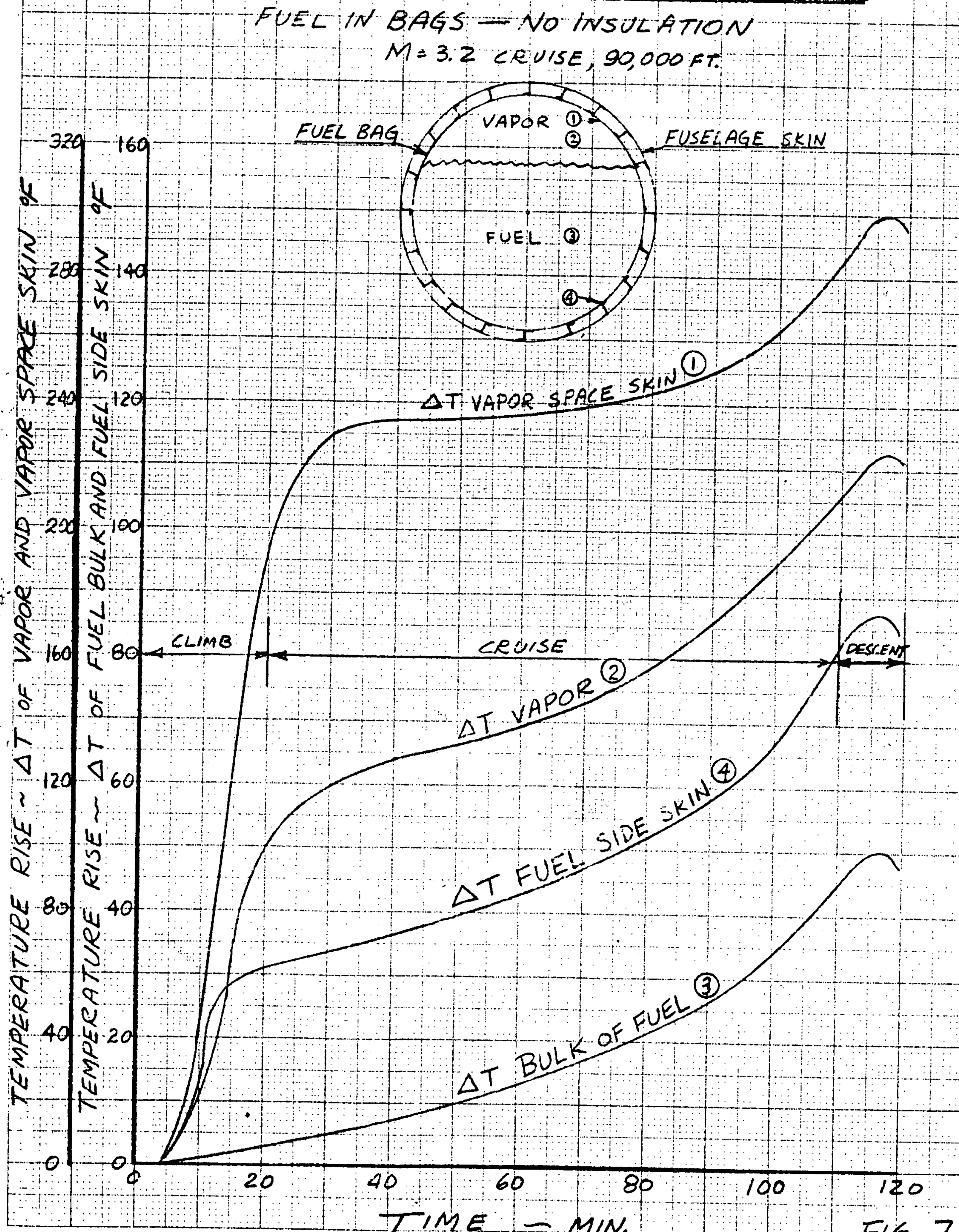

FIG. 7

FORM 5278

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NAVIGATION SYSTEM

Considering the high speed capability of this aircraft, approximately 30 nautical miles per minute, an automatic navigation system becomes almost mandatory. It is impracticable to expect the pilot to determine his position by conventional methods of sun sights and drift readings and perform his normal duties and properly fulfill the mission.

To this end we propose to install an inertial reference guidance system. At this time, several manufacturers such as Nortronics Company, Kearfott Company, Minneapolis Honeywell, and the Autronics Company are designing and developing components and complete navigational systems of the inertial reference and stellar inertial types.

The stellar inertial systems are all basically the same with a daylight star tracking telescope mounted on a gyro stabilized platform which is constantly aligned to the local mass attraction vertical. This type of instrument constantly corrects the gyro stabilized platform drift by a program of star sights; thus, the position error will never exceed 1 nautical mile C.E.P. See Figure 1.

The weight of such a stellar inertial system would be approximately 250 pounds.

The simpler type system herein proposed utilizes a stable platform with a three gimbal-three axis assembly aligned to the local mass

FORM 8787A 1

NAVIGATION SYSTEM (CONT.)

attraction vertical. If we utilize the stabilized platform as a reference for determining position, and since this platform will be uncorrected during the mission flight, it will build up a position error at the end of a 2.5 hour flight of 2.9 nautical miles C.E.P.

This error is derived from a position error drift rate of 1 nautical mile per hour and an assumed initial datum error of 2,500 feet. See Figure 1.

As the MK III driftsight will be used for final pin-pointing of a target, a position error of 3 nautical miles will be more than satisfactory.

The pilot will have a display panel showing distance traveled and present position in latitude and longitude; he will also have an instrument showing true heading.

The weight of this type of system complete, including electronics and computer, will be approximately 150 pounds.

This automatic navigational system will be supplemented by the following components:

1. MK III Driftsight.
2. MA-1 Compass System.
3. ARN-44 Radio Compass.
4. AN 5766-74 Standby Compass.

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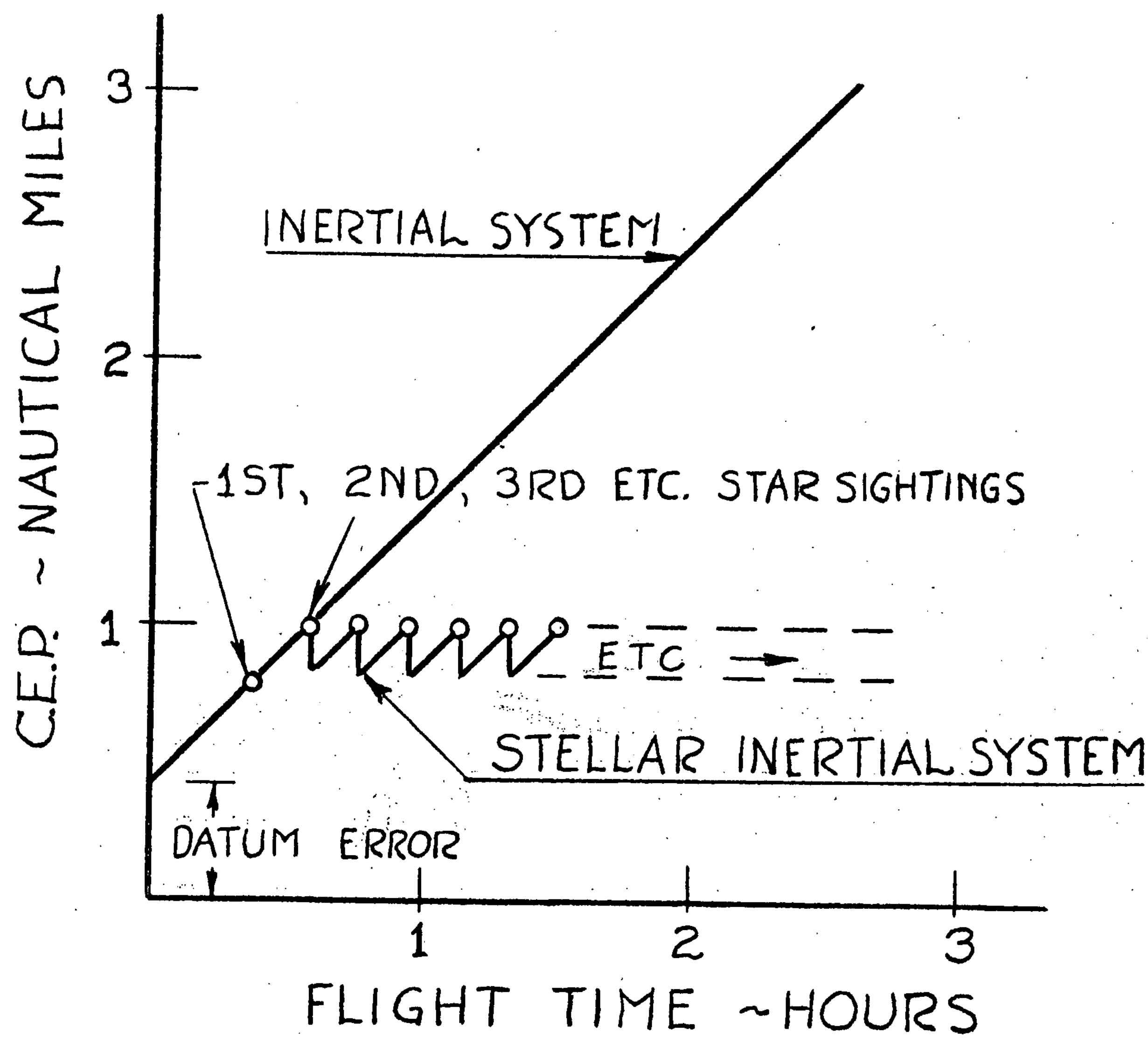
FIGURE 1

POSITION ACCURACY VERSUS TIME



FORM 5787A-1

CONTROLS

Cockpit

The flight controls in the cockpit are of the conventional rudder pedal and control stick arrangement. The rudder pedals are also used in the normal manner to apply brakes. Movements of the control stick are mixed mechanically in the cockpit for elevon control. After the control signal is mixed, the control system for each elevon is separate and independent of the other elevon permitting pitch and/or roll control, depending on stick position.

Cable Systems

The pilot forces are transmitted from the cockpit to the boosters by control cables. The rudder system has single cables for each direction of movement. Each elevon has two (2) control cables for each direction of movement; either of the two (2) cables in these dual systems can carry the full pilot load.

These cable systems include tension regulators to maintain a nearly constant rigging tension regardless of changes of the airframe due to variations in temperature.

Boosters

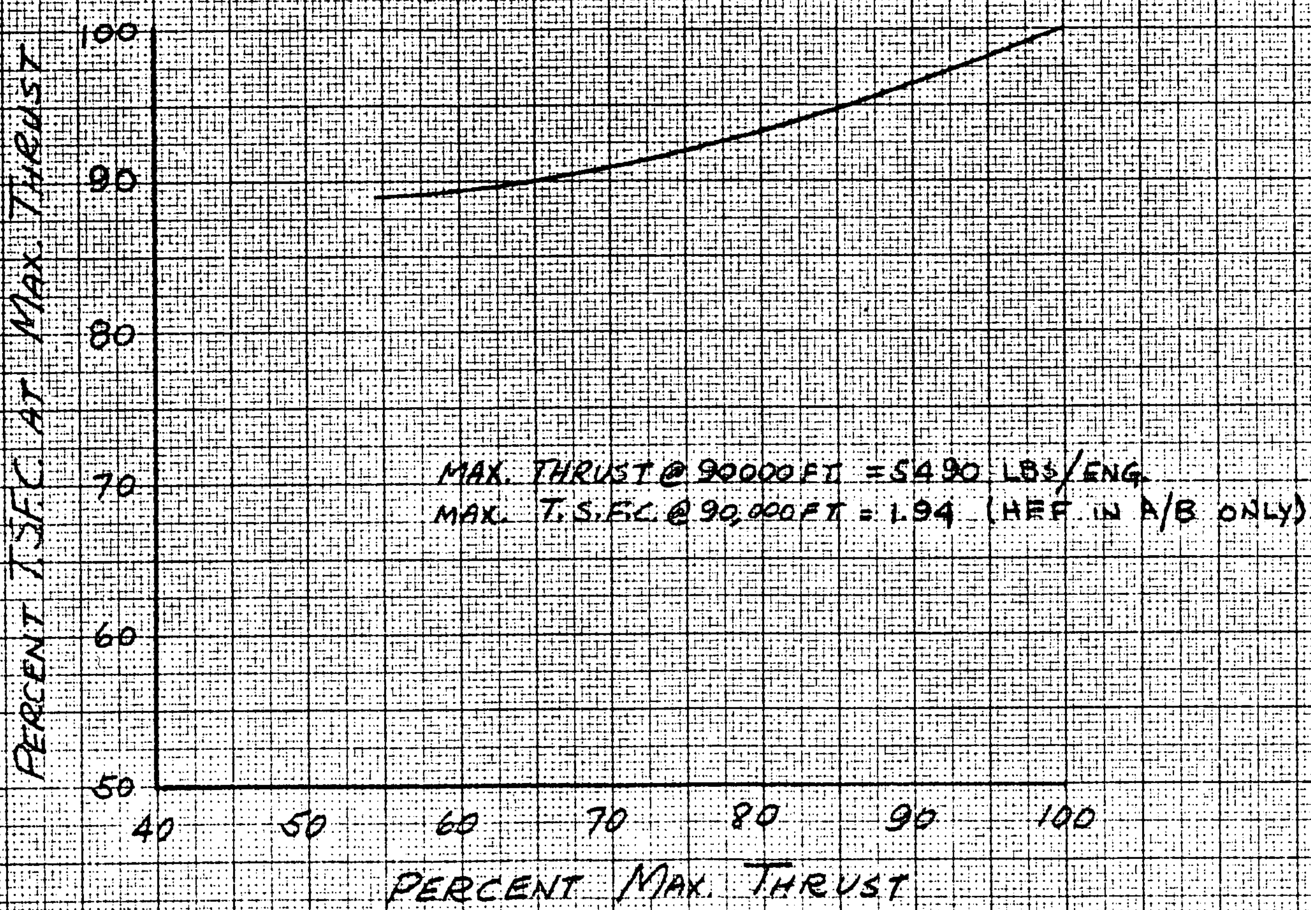
The hydraulic boosters are located at the control surfaces, one at each elevon and one at the rudder. The boosters are all irreversible

A-11

ESTIMATED PARTIAL AFTERBURNING PERFORMANCE

J58 ENGINE

M = 3.20



FORM 527B A

CONTROLS (CONT.)

with pilot feel being supplied artificially. Each booster has dual control valves, dual cylinders and is supplied hydraulic pressure from two (2) independent systems. The failure of one (1) control valve, cylinder or hydraulic system will not prevent operation of the control surface with the remaining system.

Control surface trimming is accomplished by actuators at each control surface booster which change the relationship between the zero artificial feel position and the control surface position.

HYDRAULICS

The hydraulic system design shall conform to the requirements of MIL-H-5440B, except the system operating pressure will be 4000 psi. A fuel oil cooler will be provided to maintain a maximum system temperature of +400°F. Low temperature operation will be -65°F.

Monsanto OS-45-1 hydraulic fluid is expected to be used to meet the high temperature (+400°F) requirements. Considerable experience has been gained with this fluid and it is compatible with present hydraulic system components with a minimum of system modification.

The design of the hydraulic system without a cooler will be considered and investigations will be made to determine feasibility of using a +700°F system. Fluids under consideration will be General Electric Versilube f-50 and turbine engine oil.

The hydraulic pumps will be engine driven and variable delivery type.

Dual systems will be provided with each system supplying one half the power requirement. System No. I will operate the landing gears, nose wheel steering, two (2) fuel pumps and one half the required hinge moment for the rudder and elevon. System No. II will supply power to two (2) fuel pumps and one half the required hinge moment for the rudder and elevon control surfaces.

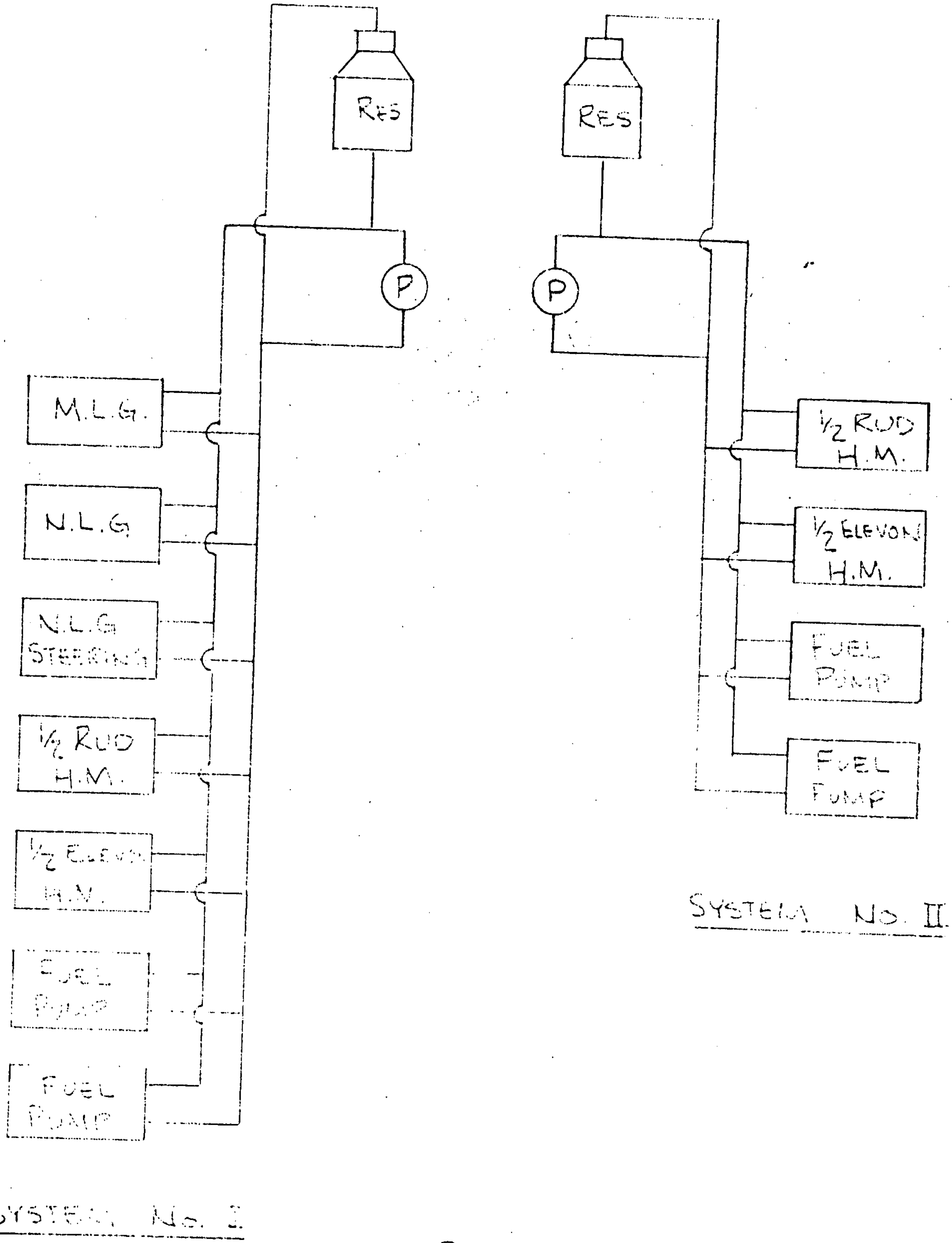
HYDRAULICS (CONT.)

The fluid reservoir shall be the airless type. The return system will be closed with returning oil being directed to the pump inlet and the reservoir acting as a low pressure accumulator.

Pressure lines will be 304-1/8 stainless with steel fittings. Line connections will be flareless type fittings in accordance with the MS standard.

Lockheed AIRCRAFT CORPORATION

CALIFORNIA DIVISION



BLOCK DIAGRAM - HYDRAULIC SYS.

FORM 5787A-1

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ELECTRIC SYSTEM

Introduction

Special consideration is given to the design problems of the electric system which exist due to the high operating altitudes and the supersonic flight conditions of the aircraft.

High temperature is the basic electrical problem associated with supersonic speed. It causes physical and/or electrical changes in the materials and equipment used in the system. Wire resistance increases, the volume-resistivity of insulation materials is lowered, and the magnetic characteristics of electrical irons and steels change as the temperature increases.

Since uncooled, high-temperature operation electrical systems are not available, all possible electrical and electronics equipment is installed in pressurized and cooled compartments.

Where required, high-temperature components such as the following will be used:

1. Nickel-clad copper wire and lugs.
2. Teflon fiberglas clamps.
3. High-temperature, environmental type connectors. A special HR series using ceramic inserts and crimped silver-alloy contacts is satisfactory at 1000°F.

FORM 5767A 1

ELECTRIC SYSTEM

Introduction (Cont.)

4. MIL-R-25018-Cl.C relays which are miniaturized, hermetically sealed, and rated for continuous operation at 392°F.
5. MIL-R-6106-Type I - Cl. D10 relays which are rated for continuous operation at 482°F and 100,000'.
6. Special high-temperature and hermetically sealed switches.

Another problem associated with supersonic flight which has an effect on the electric system and component design, is that of the so-called "white noise" - the noise level which is estimated at 150db. The basic effect is unusually high induced vibration loads which are minimized by adequate acoustical vibration and insulation techniques.

High altitude operation presents many electric system problems and Lockheed has had considerable experience with high altitude aircraft. Corona has deleterious effects on wire insulations and it increases the hazard of arc-over or voltage breakdown at altitude. Another undesirable side effect of corona is the radio noise problem created. Any damaging effects of ozone concentrations on the materials will be evaluated; however, the associated high stagnation temperatures will considerably reduce this problem. Also, lower ionization potentials are required at altitude.

AC and DC generators are available which are brushless and oil cooled with integral oil pumps. These generators are lightweight and permit high

ELECTRIC SYSTEM

Introduction (Cont.)

temperature and high altitude operation. A 28V DC system has been selected for this aircraft based on known and expected aircraft and military equipment loads. The corona and ionization problems at altitude are considerably reduced by using low voltage. Also, a DC system is simpler than an AC system, since there are no frequency or phasing problems; therefore, it is inherently more reliable.

DC System

Two engine mounted 150A oil cooled, brushless DC generators supply power to the monitored and essential DC busses. Either generator can supply the total electric load. Should both generators fail, a 20 A-H silvercel battery will provide essential DC power for approximately 30 minutes.

COMMUNICATION SYSTEM

AN/ARC-62 Command Communications Set (Cont.)

An improved control box is used in conjunction with the AN/ARC-62 receiver-transmitter to provide channel selection versatility. Twenty channels can be preset on a memory drum in a matter of seconds and the pilot may choose channels either from the preset number or by setting the five digits of the 3500 possible manual channels. The automatic tuning system operates so rapidly that the pilot is on the air with voice communications completely established in less than 4 seconds after making his selection.

AN/AIC-10 headset, microphone, and interphone control components are installed and used with the UHF communication set.

JP-150 MISSION

It is of interest to determine the effect upon airplane performance of using only hydrocarbon fuel. Flight testing of airframe, engine and equipment and crew training as well as some tactical missions can be conducted on a more economical basis with the less exotic fuel.

To accomplish the identical mission radius of the HEF equipped airplane requires a fuel load of 55,330 pounds with a take-off weight of 92,130 pounds. These numbers are 7,330 pounds greater than the HEF equipped airplane. However, the basic airframe will accommodate the greater weight of fuel at the lesser average density because sufficient fuselage diameter and length have already been established by payload and balance considerations.

The increased take-off weight results in a take-off ground run of 2,900 ft. The landing weight is not affected so that the landing distance remains 2,700 feet. The initial penetration altitude is reduced 1,700 feet and the target altitude is reduced 800 feet, also by virtue of the increased flight weight. The performance is otherwise unaffected by the sole use of JP-150 fuel.

It is noted at this point that the use of JP-150 exclusively does not show up to be as much of a disadvantage as might at first be expected. This comes about because the fuselage size and length required by payload and balance requirements can hold more fuel than is compatible with attaining the highest possible altitude at a 2,000 n. mi. radius using the HEF fuel

JP-150 MISSION (CONT.)

combination. It therefore appears that the basic airplane (Ref. Figure 1 in "Performance Section") could be overloaded with an HEF fuel combination of 55,330 lbs. With this overload of fuel the mission radius will improve to approximately 2,250 n. mi. with about the same altitude profile as attained with JP-150 fuel alone.