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PROJECT HAZEL

AIRCRAFT DESIGN

REPORT NO. ZP-253

OCTOBER 1958

CONVAIR

A DIVISION OF GENERAL DYNAMICS CORPORATION

SAN DIEGO, CALIF.

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AIRCRAFT DESIGN

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FOREWORD

This report is presented as one of a set describing the Project "Hazel" study performed by the Convair San Diego Division of the General Dynamics Corporation. The entire set of reports, listed below, represents Convair's fulfillment of the publications obligation specified in Contract NOas-58-812 (SS-100) and Amendment #1, issued 14 August 1958 by the Bureau of Aeronautics.

ZP 252	Summary (Brochure of Charts with Text)
ZP 253	Aircraft Design
ZA 282	Aerodynamics
ZJ 026	Propulsion, Structure Heating, and Pressurization

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SUMMARY

This report summarizes the preliminary design work conducted under contract NO(as) 58-812 (SS-100) and Amendment #1, issued 14 August 1958 by BuAer for the time period June through September 20, 1958.

The following requirements were given for this study:

Total range	3,200 n.mi.
Recon. range	1,500 n.mi.
Altitude	125,000 ft.
Velocity	M - 3
Crew	1
Payload	800 lb.

The specific high altitude reconnaissance system under consideration is based on an airplane with an extremely low wing loading. This low wing loading was accomplished by an unconventional wing construction and extreme light weight engine designs.

The wing is an inflatable type, pressure stabilized design with a silicone impregnated fabric skin. Drop threads or ribbon trusses between upper and lower skin will keep the shape of the wing and provide, together with the pressure, the required structural stiffness.

Marquardt and Pratt & Whitney high altitude ramjet engines were considered for this vehicle.

The dimensions of the Marquardt ramjets were ratioed according to the required thrust. There was no specific upper limit defined for this method. The dimensions of the Pratt & Whitney engines were also ratioed for the required thrust; however, an upper limit was set for these engines due to the limitations of manufacturers' engine test facilities.

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The Marquardt engine is basically fabricated of a fiberglass laminated sandwich outer shell with a high temperature metal sandwich honeycomb inner shell. The P & W engine is all metal. (See Report ZJ-026 for detail engine data.)

Two basic configurations, a single engine and a twin engine configuration were considered which would best meet all requirements. These are shown in Figure 1, MC-10 and 2, PC-22. Both are delta wing airplanes with the following characteristics:

	MC-10	PC-22
Wing Area	1985 Ft^2	1985
Wing Leading Edge Sweep	60°	60°
Span	67.71 ft.	67.71 ft.
Overall Length	60.0 ft.	76.0 ft.
Weight at start of cruise	13,800 Lb.	9700 lbs.
Wing loading @ Mid Cruise	5.35 Lb/Ft^2	4.89 lb/ft^2
Cruising Speed	M = 3	M = 3
Cruising Altitude (average) . . .	131,400 Ft.	139,000 Ft.
Engine Type	Marquardt Ram Jet	P & W Ram Jet
Fuel Weight	6330 Lb.	2970 Lb.
Fuel	Pentaborane	SF-1

Typical mission profiles of these configurations are shown in Figure 3. The vehicle is lifted to a launch altitude of 45,000 ft. by either a conventional airplane (B-36 or B-52) or by a special air breathing booster to some higher altitude (80,000 ft.). After release from the carrier, the vehicle is boosted up to start of cruise altitude where it starts to cruise under its own power. It cruise climbs to the end of a 3000 N.Mi. cruise and then descends to sea level where it lands with a landing speed of 37 knots. The total range is 3200 N. Mi.

The higher cruise altitude of configuration PC-22 is due to the fact that by keeping the same wing area the wing loading is lower and, therefore, a higher altitude at start of cruise can be obtained.

The airplane is equipped with a pair of small penetration-type hydro skis mounted in tandem along the center line of the airplane. One is mounted near the nose and forward of the internal equipment while the other is positioned at the extreme stern. These two main skis in connection with small stabilizing skis near the wing tips will provide the required impact absorption and stability during landing.

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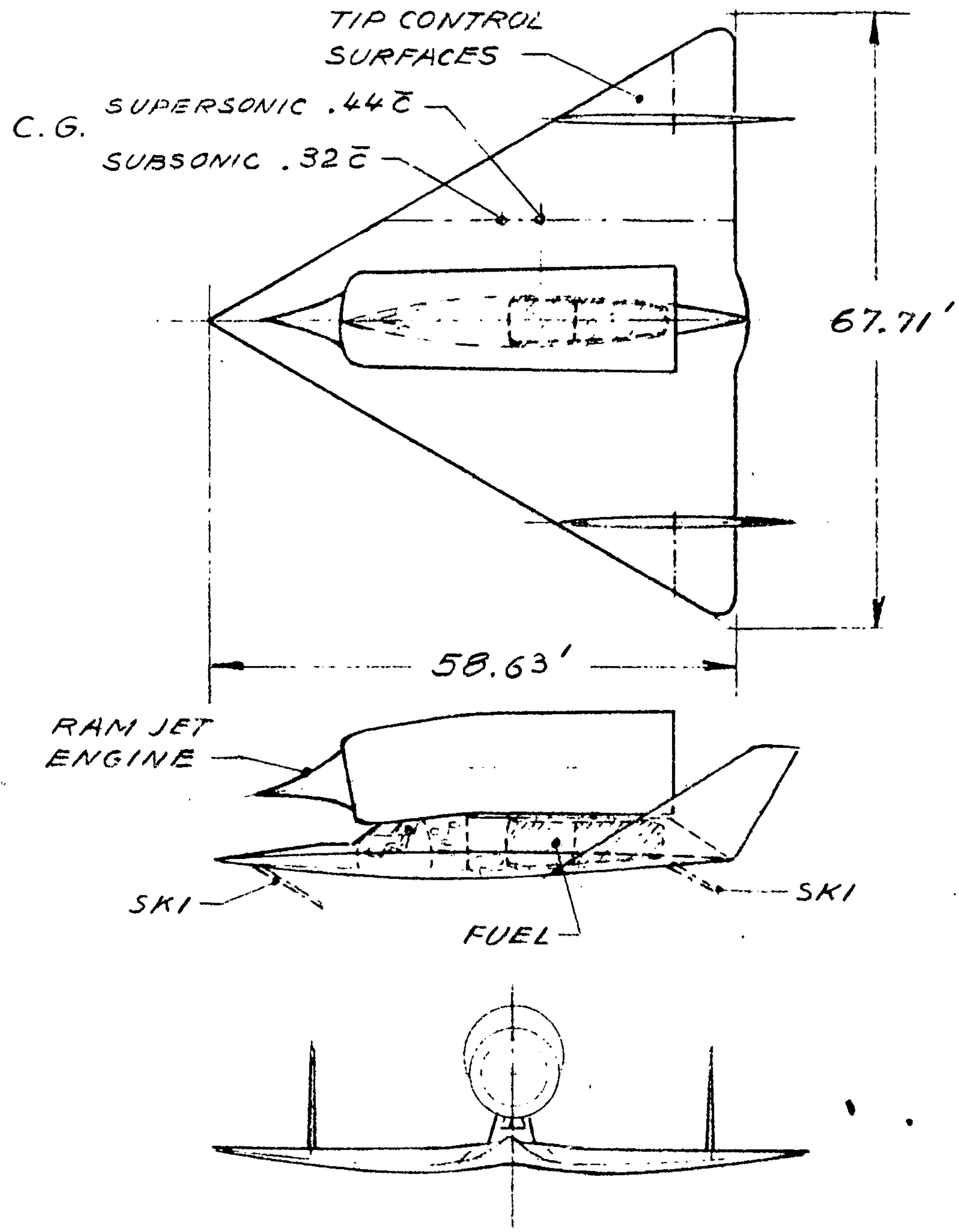

BASIC CONFIGURATION
MC-10
MARQUARDT ENGINE, PENTABORANE FUEL
FIG. 1

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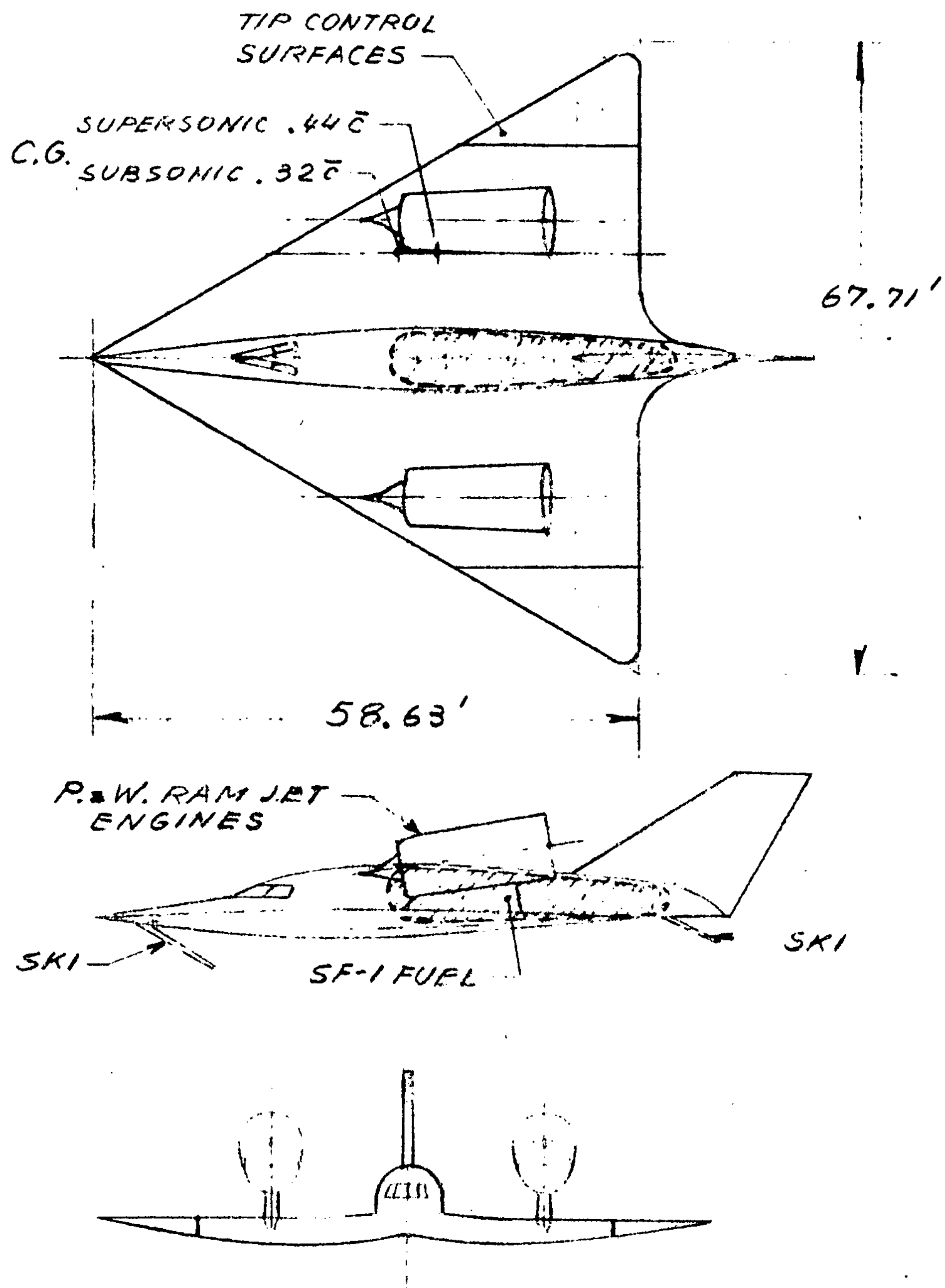

BASIC CONFIGURATION
PC-22
PRATT & WHITNEY ENGINES, SF-1 FUEL

FIG. 2

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FORM 1812-A-1

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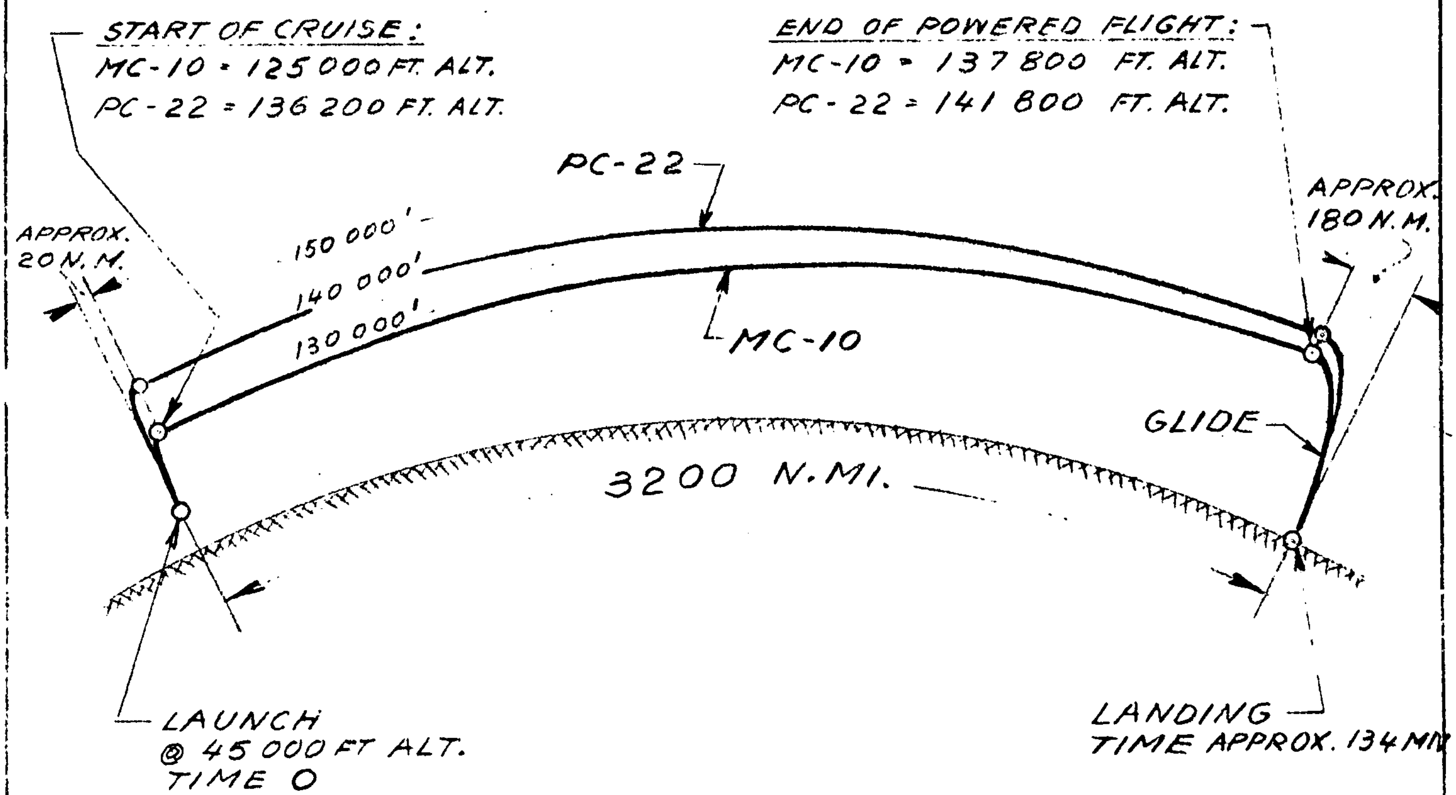

MISSION PROFILE

FIG. 3

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Several methods of launching the vehicle to the start of cruise were studied. These were:

1. Rocket Boost from ground.
2. Balloon lift and rocket-boost.
3. Airplane lift and rocket-boost.
4. Special air breathing booster and rocket-boost.

All these methods are feasible but more study will be required to properly evaluate the optimum system. At present, method 3 is favored.

Table II shows a summary of the basic structural data and Table III gives information on structural materials which are proposed for this vehicle.

The Hazel systems are shown schematically in Figure 30 (page 57) and consist of a liquid monopropellant auxiliary power system which supplies energy for the structural pressurization system, the aerodynamic control system, the electrical power system, air conditioning system and a fuel system. A separate booster control system is provided.

The crew capsule design is dependent on the configuration selected and the method of launching the vehicle to the required start of cruise altitude and speed.

A normal seated pilot has been chosen for the final configuration based on using airplane lift and rocket boost to start of cruise altitude. However, a prone position pilot was seriously considered because of the advantages of the prone position for this specific high altitude reconnaissance mission where the pilot's main vision has to be downward. For landing a special mirror device was considered in order to provide the pilot with forward vision. Also a combination of prone and seated position was investigated. This combination would provide a separate supine type seat for flight during launch at high accelerations. The normal flight position is prone and would provide maximum downward vision for the reconnaissance part of the mission.

Escape from the airplane in case of emergency is provided by a trap door on the bottom side of the airplane. The seat is mounted on this door. Deployment of a 30" stabilizing parachute snatches the pilot from the seat and provides a stabilized descent at a terminal velocity of about 100 MPH indicated airspeed. Automatic parachuting, actuated by an aneroid at 15,000 ft. altitude, would let the pilot safely to the ground. A "Global Survival Kit" is provided.

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TABLE I

Model

Page | Temp | Perm

Report No.

STRUCTURAL DESIGN CRITERIA

MC VEHICLES

(1) (2) CONDITION	(3) WEIGHT COND.	(4) V (B) KNOTS	(5) H (B) 10^3 FT	(6) q (B) PSF	(7) STRUCT. TEMP (B) 10 FT. L.E. DEG. F	(8) LOADING APPLICATION	(9) REACTION	(10) N_z LIMIT G	(11) N_y LIMIT G	(12) N_x LIMIT G
I HANDLING										
A TRANSPORT	EMPTY GLIDE	—	—	—	+70 −65	VEHICLE INERTIA	SUPPORT JIG	+2.0	±1.0	±1.0
B HOIST - JACK	EMPTY GLIDE	—	—	—	+70 −65	"	HOIST FITTINGS	+2.0	±1.0	±.5
II LAUNCH										
A PRIMARY A/C	SECONDARY LAUNCH	IT HAS BEEN ASSUMED THAT THE LAUNCH WILL BE ENGINEERED SUCH THAT THE VEHICLE CAPABILITIES OF WILL NOT BE EXCEEDED.								
B SECONDARY START	SECONDARY LAUNCH	480	45	52	250	GUST AIRLOAD	VEHICLE INERTIA	+.97*	* ASSUMES ZERO G TRAJECTORY	
C SECONDARY END	PRE STAGE	1865	125	52	305	THRUST	VEHICLE INERTIA	+.50	+.50	−2.0
III CRUISE										
A BEGIN. TURN (A)	START CRUISE	1865	125	52	308	AIRLOAD	VEHICLE INERTIA	+1.5	±.25	—
B MID. TURN	MID CRUISE	1880	132	41	"	"	"	+1.15	±.25	—
C END. TURN	GLIDE	1920	140	28	"	"	"	+3.0K_t	±.50K_t	—
								K_t = STRENGTH LOSS AT TEMP.		
IV GLIDE										
A MANEUVER (A)	GLIDE	1680	85	275		AIRLOAD	VEHICLE INERTIA	+2.0 −1.2	±.50	—
B GUST (A)	GLIDE	150	35	28		GUST AIRLOAD	"	"	±.50	—
V LANDING										
A MAX. SINK	GLIDE					SKI WATERLOAD	VEHICLE INERTIA			
B WORST ATTITUDE	GLIDE					"				
VI RECOVERY	EMPTY GLIDE	ASSUME SAME AS HANDLING								

(A) BASIC STRUCTURAL DESIGN CONDITIONS (B) TYPICAL VALUES FOR REF.

Form F-224

Date 10-28-58
Prepared By J.C.E.
Checked By
Revised Date

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TABLE II

Model

Page Temp Perm
Report No.

MATERIALS DATA SUMMARY

① ② COMPONENT	③ ④ ⑤ CONSTRUCTION	⑥ TEMP °F	⑦ TIME t_{ult} HR	$t_{...}$ HR	⑧ F_{tu} AT TEMP KSI	⑨ F_{tu} ROOM TEMP AFTER KSI	⑩ E_c KSI	⑪ F_{su} KSI	⑫ DENSITY #/IN³
NON - RIGID VEHICLES									
BODY NOSE	REPLACEABLE, RIGID	725	1.5	10	9.6	15.9	σ_{cx}	τ_{xx}	.065
	PLASTIC LAMINATE	915	.15	1	9.6	15.9			
BODY & PYLON	RIGID PLASTIC LAM.	305	1.5	1000	23.0	31.9	σ_{cc}	τ_{ce}	.065
	SANDWICH & SHAPES	425	.15	100	23.0	31.9			
LEAD. EDGES	REPLACEABLE, NON RIGID	630	1.5	100	6.2	14.2	0	$\sqrt{E_x E_y}$	.065
	DIAGONAL DOUBLED FABRIC	685	.15	10	9.2	14.2			
WING & FIN	NON RIGID, IMPREGNATED	305	1.5	1000	14.1	19.5	0	$\sqrt{E_x E_y}$	.065
	TAB. DOUBLED FABRIC	425	.15	100	14.1	19.5			
RIGID VEHICLES									
BODY NOSE	REPLACEABLE, RIGID	725	1.5	10	9.6	15.9	σ_{ce}	τ_{ce}	.065
	PLASTIC LAMINATE	915	.15	1	9.6	15.9			
BODY & PYLON	RIGID FIBERGLASS	305	1.5	1000	23.0	31.9	σ_{ce}	τ_{ce}	.065
	LAMINATE & SANDWICH	425	.15	100	23.0	31.9			
LEAD. EDGES	REPLACEABLE, [illegible]	630	1.5	100	10.5	23.1	[illegible]	τ_{ce}	.065
	PLASTIC LAMINATE	685	.15	10	19.3	23.1			
WING & FIN	RIGID FIBERGLASS	305	1.5	1000	23.0	31.9	σ_{ce}	τ_{ce}	.065
[illegible]	SANDWICH [illegible]	425	.15	100	23.0	31.9			
SUB. TRANSFER [illegible]	[illegible] FIBERGLASS	305	1.5	1000	23.0	31.1	[illegible]	τ_{ce}	.065
	[illegible]	425	.15	100	23.0	31.9			

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INTRODUCTION

This report has been prepared in compliance with the requirements of BuAer contract NOas 58-812 (SS-100) and Amendment #1, issued 14 August 1958.

In May 1958 Convair was asked by BuAer to perform a study of a high altitude Manned reconnaissance vehicle according to the following requirements:

Reconnaissance Altitude	150,000 to 200,000 ft. (100,000 ft minimum)
Cruise Speed (Boost Cruise Type)	M = 2 to 3
Glide Speed (Boost Glide Type)	As low as possible.
Range	3200 n.mi. desired (2500 n.mi. min.)
Reconnaissance Range at Optimum Altitude. .	1500 n.mi. desired (1000 n.mi. min.)
Payload	800 lb. desired (400 lb. min.)
Crew.	1
Seabased	
Wing Loading	<10 lb/ft^2
Engines for Cruise:	
Marquardt	Low "q" ramjet
Hughes	Slow burning liquid rocket (Low I.R. source type)

In order to achieve the desired low wing loadings of less than 10 lb/ft^2 a radically new design approach was taken. The wing of the vehicle was based on flexible materials such as impregnated fibreglass fabrics for the skin and a pressure stabilized construction using drop threads or ribbon trusses for holding the shape of the wing. For body and engines the skin was assumed flexible impregnated fiberglass fabric with protective insulation where required, the nose section and inlet lips were considered as rigid fiberglass laminates, the construction being also pressure stabilized.

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It was apparent early in the first phase of the studies, that, in order to achieve the required range, the speed of the boost-glide vehicles at the start of glide was much too high for a reconnaissance mission and that aerodynamic heating was too severe for the materials considered. Therefore, for further studies, only the boost-cruise type vehicles were considered.

A study of the rocket cruise vehicle showed that, in order to achieve an economical configuration, the cruise speed has to be increased to $M = 8$. The wing area consequently could be smaller. For the higher wing loading (at mid-cruise, 15 lb/ft^2) a rigid construction of the wing and body was chosen, material being metal or rigid plastic. Figure 4 shows this configuration. This configuration was later abandoned because of the high cruise speed of the vehicle which is not compatible with the reconnaissance requirements.

Figure 5 shows the first approach for a ramjet vehicle. It is a single engine, high wing, aerodynamically clean configuration with an average L/D of 5, a wing area of 1985 ft^2 and a mid-cruise wing loading of 3 lb/ft^2. This configuration fulfilled all the requirements except a requirement necessitating overwing engine locations. This resulted in a configuration as shown in Figure 6 which, in the process of refinements, was modified to the final basic configuration shown in Figure 1 (page 4). The cruise speed of these ramjet cruise vehicles was $M = 3$ and the cruise altitude 150,000 ft.

No information was available on the reconnaissance equipment. It was assumed, therefore, that an optical camera would be used with direct downward vision through a window in the lower skin of the vehicle. This assumed camera determined the approximate space requirement for the paylaod.

Consideration was given to the most favorable position of the pilot for this specific mission. Both the seated and prone position were considered. The prone position would be the ideal position for this mission because of the great value of visual search from high altitudes. The prone position would place the pilot's eyes and head in the most favorable position for this purpose. For landing, the pilot's forward vision is somewhat impaired. The seated position would assure better forward vision but downward vision would be greatly limited. A combination of prone position for the cruise and reconnaissance part of the missions and the seated position for launch and landing has certain advantages.

An early requirement for the vehicle was launching from and retrieving by a submarine. The inflatable type of construction is an advantage for this purpose. The vehicle can be deflated, packed in a container, launched by a

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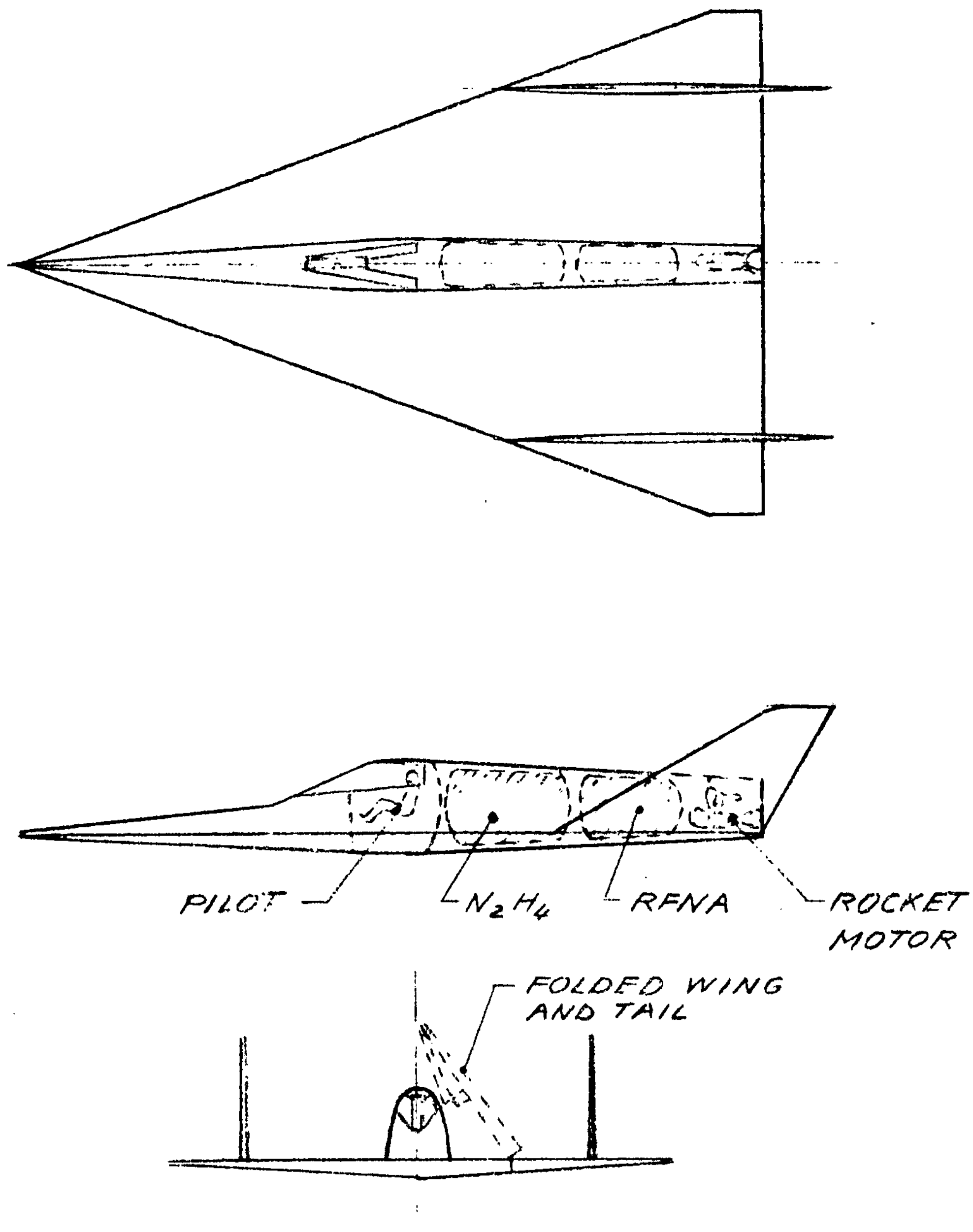

ROCKET CRUISE VEHICLE

FIG. 4

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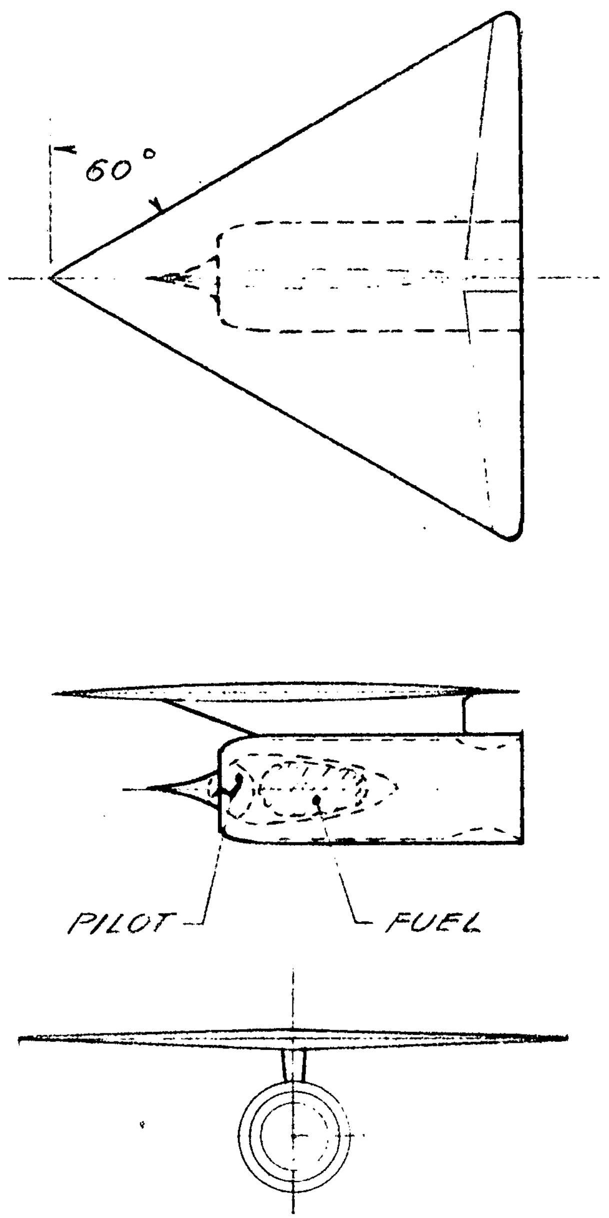

MC-2
RAM JET CRUISE VEHICLE
HIGH WING CONFIGURATION
FIG. 5

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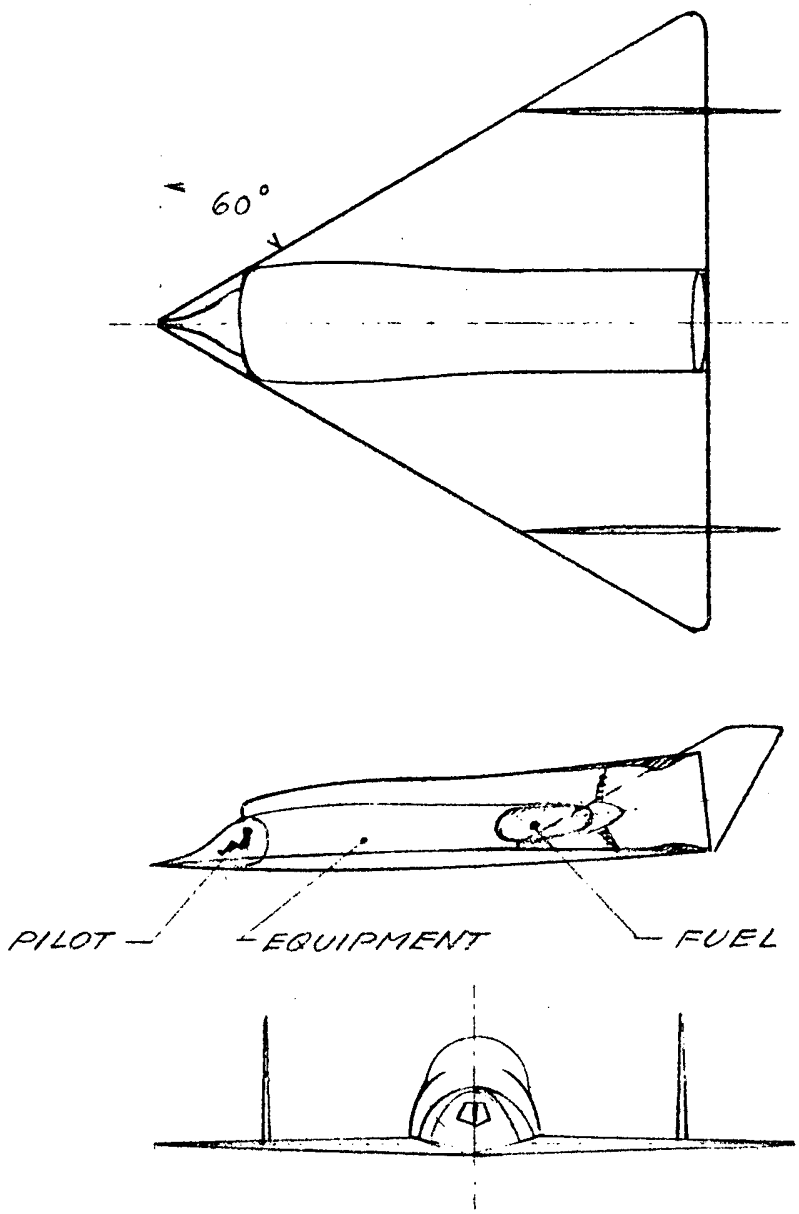

MC-4
RAM JET CRUISE VEHICLE
LOW WING CONFIGURATION

FIG 6

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separate rocket booster and inflated at the start of cruise. Figures 7 and 8 show two methods of a submarine launch. Figure 7 shows the folded ramjet cruise vehicle carried in a container and launched from a submarine and Figure 8 shows the same for the rocket cruise vehicle. Other methods of launching were emphasized later in the study.

The launch problem was not completely resolved. Several methods were under consideration such as airplane launch (B-36, B-52) balloon launch or air breathing booster launch.

The vehicle will be equipped with a ski system which will enable the pilot to land the vehicle on water or snow fields as well as on land. The pilot and vehicle then can be retrieved by surface vessels or submarines from the water or by airplane or helicopter from a remote spot on land.

Because of the extreme performance requirements for this airplane and the studies made of the influence of weight increases especially on the altitude performance, all vehicles in the first phase were designed with practically no margins for structure and power; the performance quoted for these configurations, therefore, had to be considered as optimum designs.

The second phase of the study was then devoted to the redesign of the configurations according to reworked structural requirements and revised engineering data including margins for airspeed and structural design temperatures.

In order to stay within reasonable dimensions of the airplane and powerplants, the altitude requirement of 150,000 ft. was reduced to 125,000 ft. at start of cruise. The resulting configurations of this second phase of the Hazel study is the subject of this report. The new requirements for this study are as follows:

Total range	3200 n.mi.
Recon. range	1500 n.mi.
Altitude	125,000 ft.
Velocity	M = 3.0
Crew	1
Payload	800 lbs.

Engines: Marquardt & Pratt & Whitney low "q" ramjet engines
Fuels: Pentaborane (B_5H_9) Hydrogen (SF-1)

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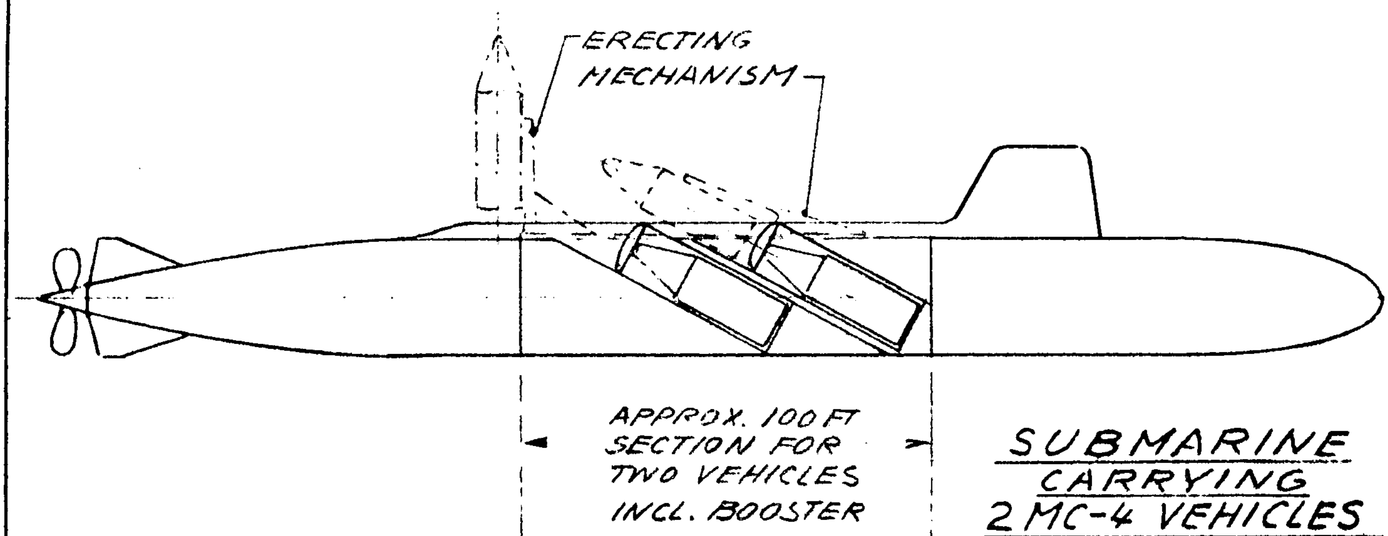
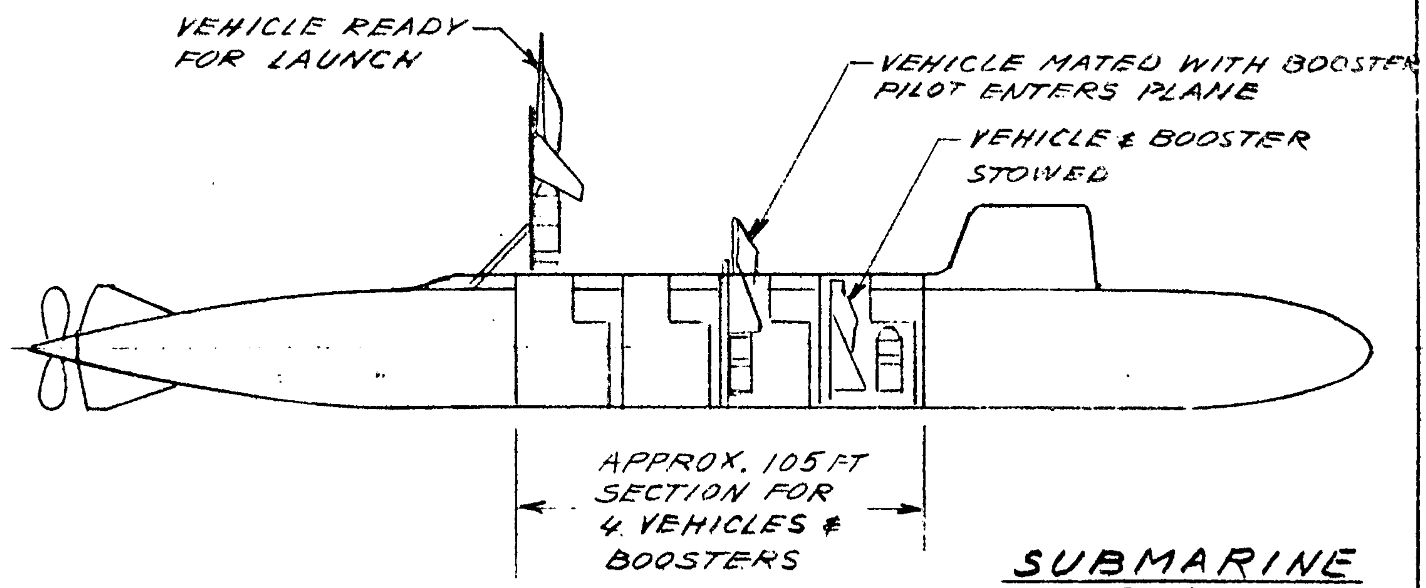


SUBMARINE
CARRYING
2 MC-4 VEHICLES

FIG. 7



SUBMARINE
CARRYING
4 ROCKET CRUISE
VEHICLES

FIG. 8

SUBMARINE LAUNCHING

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FORM 1812-A-1

SECTION I

CONFIGURATION STUDY

A. General

1. Configurations

A variety of different aircraft configurations were studied which included Marquardt and Pratt & Whitney engines, single and twin, and Pentaborane and SF-1 fuel.

The following Table III shows these configurations arranged according to number of engines and type of fuel.

The configurations have the following characteristics in common:

Wing area 1985 ft^2, 60° delta;
(except config. MC-4, wing area 3000 ft^2)
Lower wing surface smooth without any protruberances;
one man crew in normal seated position;
single engine versions have a twin tail arrangement;
engine inlet designed for average cruise angle of attack;
all movable wing tip control surfaces.

2. Mission Profile

Figure 9 shows a typical mission profile of the configuration MC-10. The launch altitude is assumed at 45,000 feet which could be a launch from an airplane such as a B-36 or B-52. The launch point is taken as time 0. The vehicle is boosted to cruise altitude by a rocket type droppable booster. Start of cruise is at 125,000 feet, the distance travelled 20 N.M. and the elapsed time 2 min. The vehicle cruise-climbs out to the end of powered flight which is reached at 137,800 feet altitude after 97.6 min. A glide of 180 N. M. will complete the 3200 N. M. mission at an elapsed time of 108.2 min. The glide speed is 91 knots and the landing speed is 37 knots.

The 4000 N. M. mission can be achieved only with a larger vehicle. The wing area for this vehicle is 3000 ft^2. The altitude at end of cruise will be 141,200 feet and the time 123.2 min. The total elapsed time for this mission will be 133.8 min.

The following table gives the weight and wing areas for the two vehicles:

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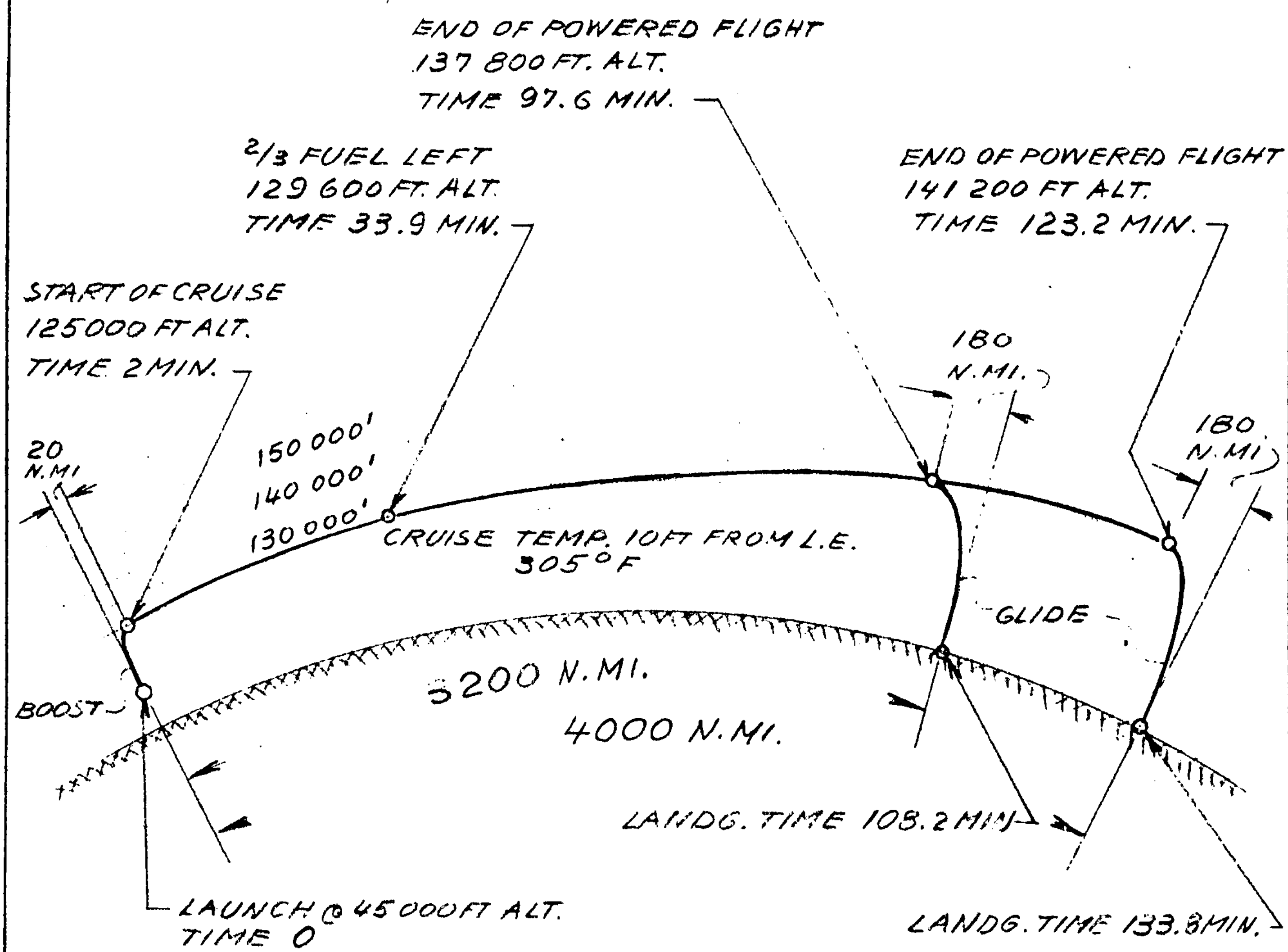

CRUISE SPEED M = 3
LANDING SPEED 37 KN.

MISSION PROFILE

3200 N.MI. AND 4000 N.MI. MC-10 VEHICLE

FIG. 9

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FORM 1812-A-1

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CONFIGURATION	ENGINE	FUEL	WEIGHT @ START OF CRUISE LBS	REMARKS
SINGLE ENGINE				
MC-10	MARQUARDT	B_5H_9	13 800	STABLE
MC-11	"	"	12 400	STABLE SUPERSONIC UNSTABLE SUBSONIC
MC-14	"	"	27 000	4000 N. MI. RANGE
MC-19	MARQUARDT	B_5H_9	18 310	RIGID FIBER-GLASS CONSTR.
TWIN ENGINE				
MC-20	MARQUARDT	B_5H_9	13 145	
MC-24	MARQUARDT	"	16 500	SAME AS MC20 SHORT ENGINES
PC-20	P & W	B_5H_9	14 350	2560 N. MI. RANGE
MC-22	MARQUARDT	SF-1	8 600	
PC-22	P & W	"	9 700	
PC-24	"	"	13 990	4000 N. MI. RANGE
PC-25	P & W	SF-1		70° DELTA WING

CONFIGURATIONS

TABLE III

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FORM 1812-A-1

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		MC-10	MC-14
Range	NM	3200	4000
Launch-wt	LB	30525	59700
Wt. at start of cruise	LB	13800	27000
Fuel Wt.	LB	6330	14700
Wing Area	FT^2	1985	3000

In order to simplify the design of the different configurations, a basic wing with an area of 1985 ft^2 has been applied to all configurations (with the exception of configuration MC-14). Due to the different weights, the wing loadings will be different for all configurations and therefore, a different start of cruise altitude could be obtained. However, 125,000 ft altitude was kept as a minimum.

For each configuration the cruise altitudes will be quoted.

3. Engines

The engines considered for this study are Marquardt and Pratt & Whitney ram jet engines. The engines are in the design stage and therefore could be scaled up or down, according to the thrust required. The scale factors used and the method of deriving of same is described in Report ZJ-026.

The Marquardt engines can be used for either Pentaborane or SF-1 fuel, the P & W engines are considered to use only SF-1 fuel.

The two engines are basically different in construction and outside dimensions. The Marquardt engine sized to 1200# thrust as shown in Figure 10 is constructed of plastic honeycomb throughout, except for injector grid, fuel system and combustion chamber liner, while the P & W engine sized to 1830# thrust as shown in Figure 11, is designed as an all metal engine of high temperature steel.

Based on a thrust of 1800 lbs. and a velocity of M = 3.0 at 150,000 ft., using Pentaborane fuel, the two engines compare as follows:

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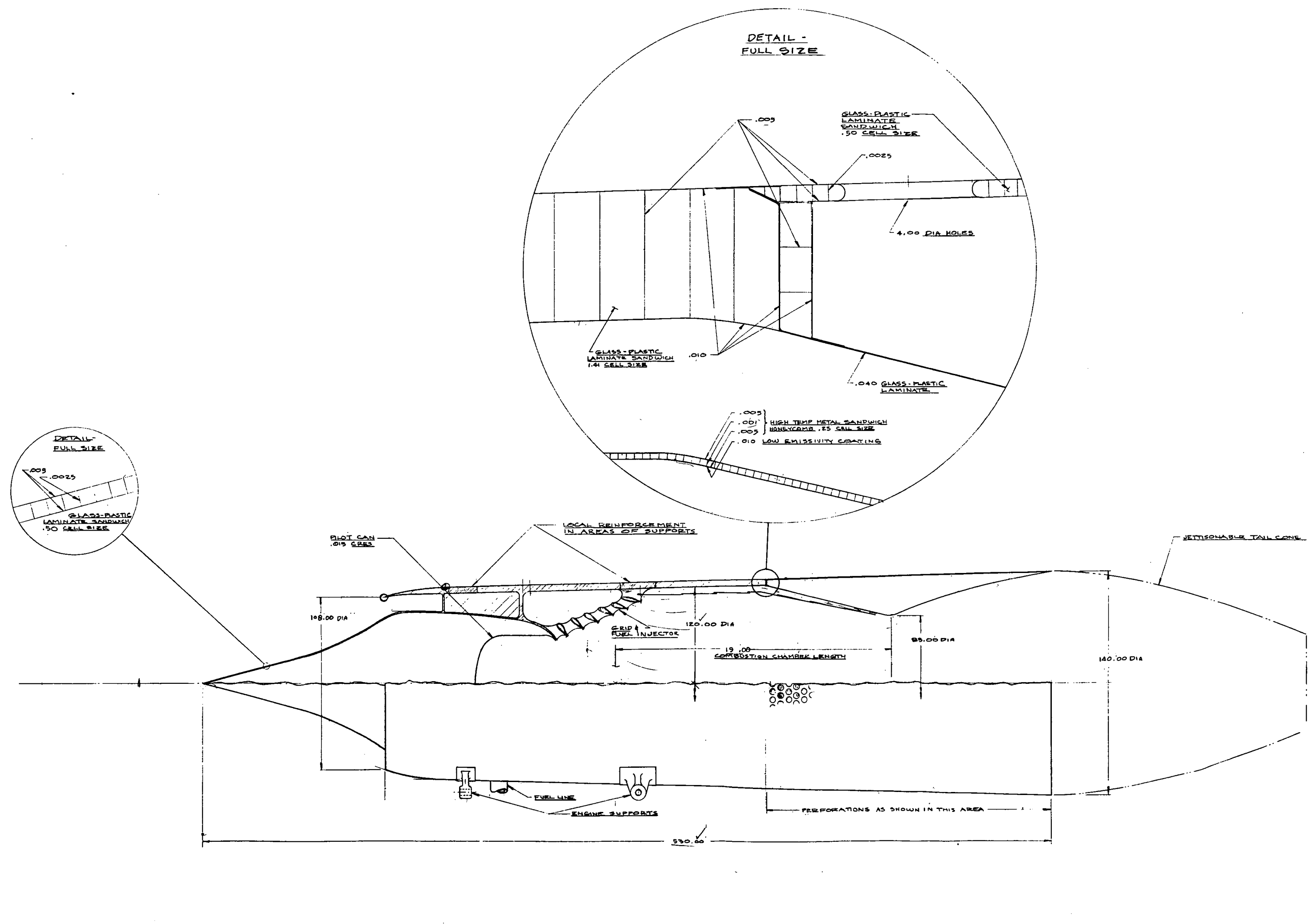

TYPICAL MARQUARDT RAM JET ENGINE

FIG. 10

Marquardt Ram Jet

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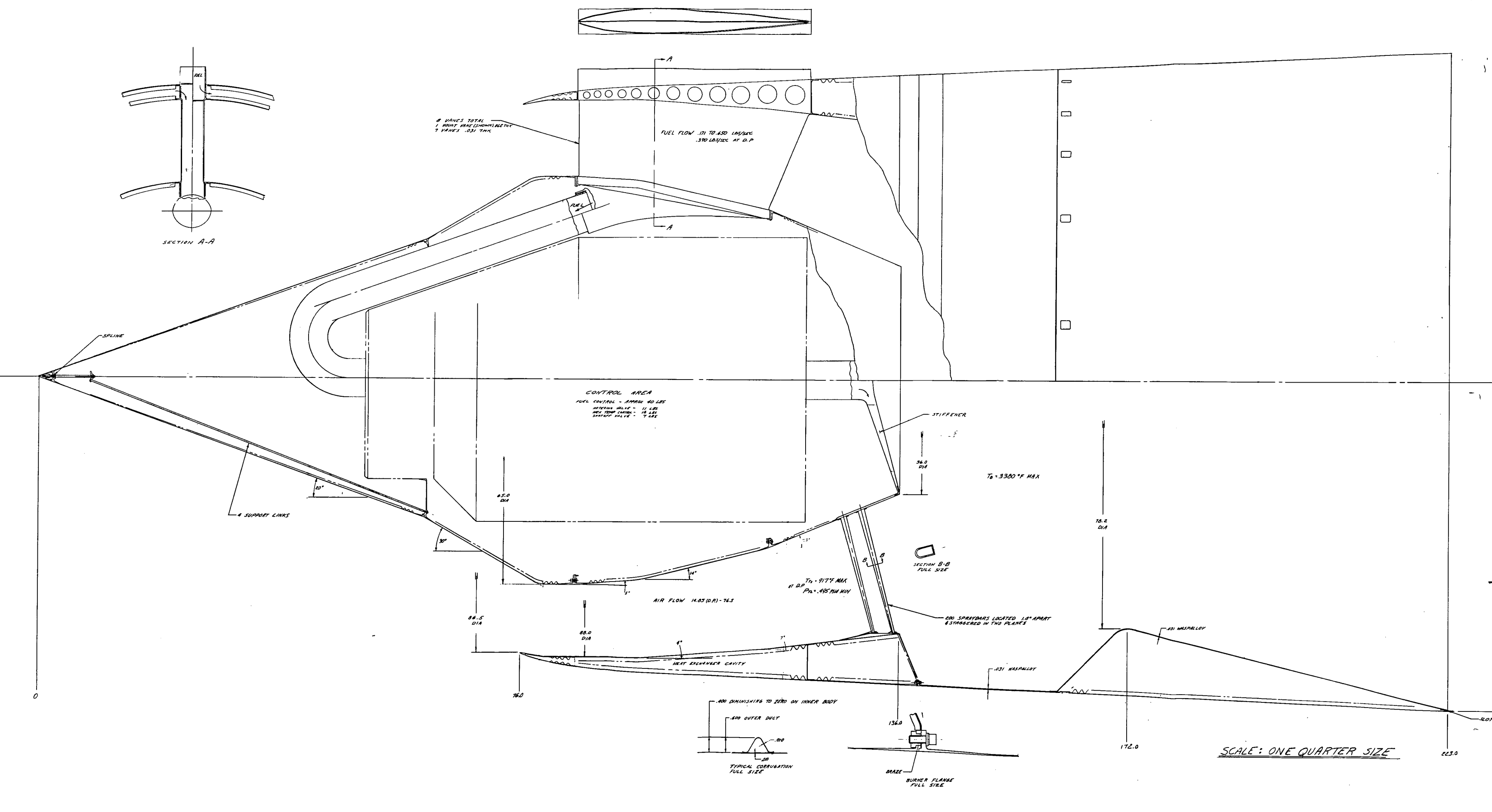

TYPICAL
PRATT & WHITNEY
RAM JET ENGINE

FIG. 11

L-37712

PRELIMINARY
LAYOUT

RAMJET LONG SECTION L-37712

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	Marquardt	Pratt & Whitney
Thrust installed	1800 lbs.	1800 lbs.
SFC	1.50 lb/thrust	2.10 lb/thrust
Weight	960 lbs.	1150 lbs.
External drag	270 lbs.	180 lbs.
Fuel/air ratio	.018	.040

4. Wing

Aerodynamic studies have shown that the induced drag of a wing can be reduced and lift to drag ratio improved by a proper choice of camber and twist. For this study, the camber and twist were optimized on the IBM 704 using the program described in Convair Report ZA-259, "Calculation of Optimum Supersonic Delta Wings". Further discussions are included in the aerodynamics report of this study, Report ZA-282.

Figure 13 shows the basic wing based on the above machine results.

The basic dimensions are:

Area	1985 $ft.^2$
Span	67.71 ft.
Root Chord	58.63 ft.
Leading Edge Sweep . . .	60°
Wing Thickness	4%
Aspect Ratio	2.3

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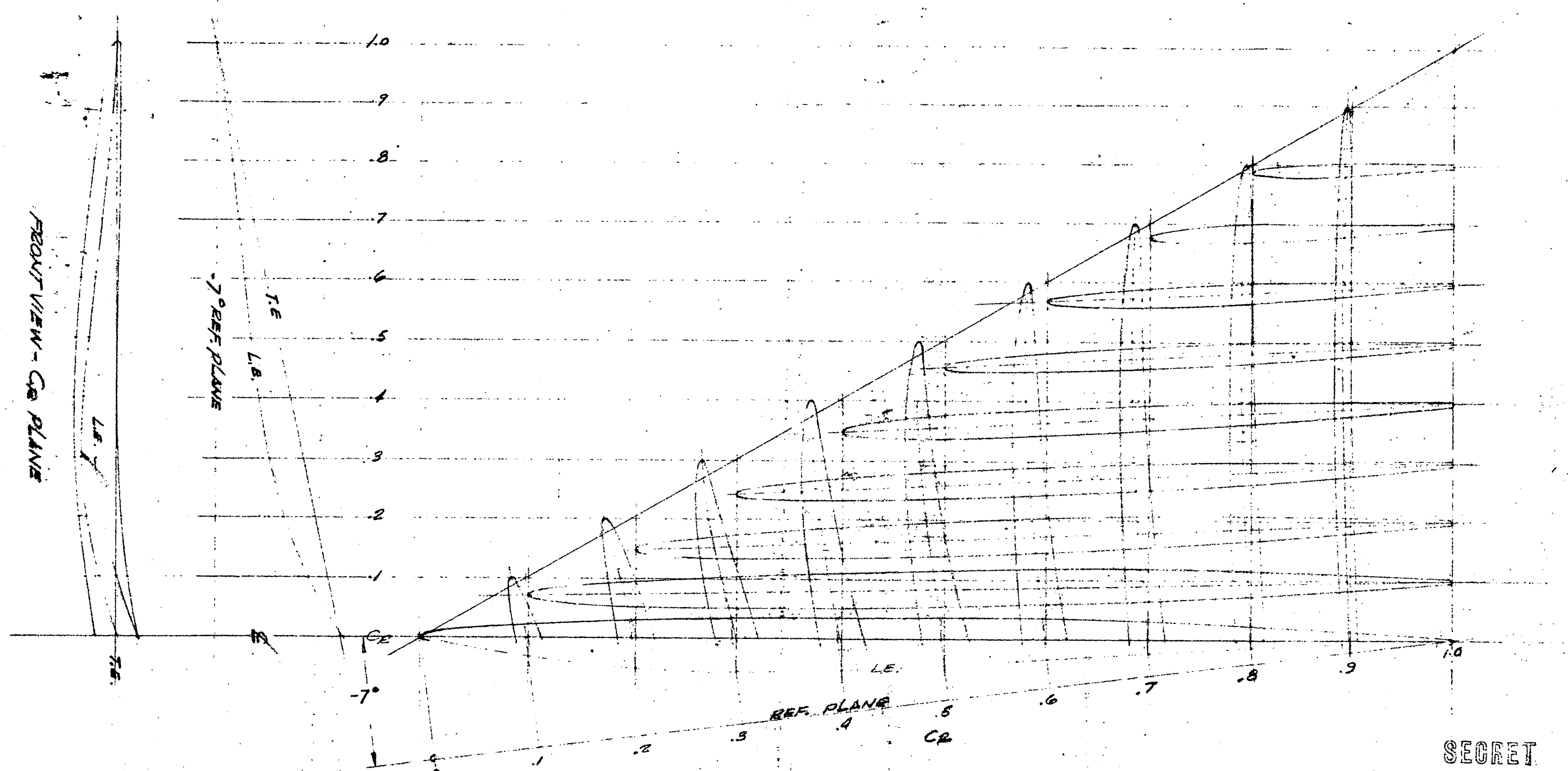

BASIC WING
AREA 1985 FT^2
SCALE 1" = 6'

FIG. 13

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5. C. G. Location and Stability

The L/D in supersonic flight can be improved appreciably by proper positioning of the c.g. It is desirable to keep the same location of the c.g. during the entire supersonic flight in order to obtain maximum L/D for the mission. The subsonic stable airplane, however, requires a further forward c.g. position. C.G. shift is accomplished in this case by fuel positioning. The fuel tank therefore has been so located and programmed that over a distance of 1/3 of the mission the c.g. remains most favorable for supersonic stable flight and gradually shifts forward towards the subsonic c.g. location as the fuel is burned. This results in a relatively high average L/D value for the total range which Figure 14 illustrates.

6. Hydrodynamic Considerations

The slow landing speed of the aircraft will facilitate landing on water.

Several aspects of the aircraft design and landing technique, however, rule out conventional alighting gear arrangements.

The design and operational aspects affecting the alighting gear arrangement are as follows:

1. Extreme landing trim attitude.
2. Light weight construction requiring protection for trailing edge control surfaces at touch down and for the entire wing during run-out.
3. Restrictions on alighting gear location due to structural considerations.
4. Critical weight restrictions and limitations on choice of material.
5. Open ocean surface conditions.

Figure 15 shows one promising alighting gear configuration. It consists of two small penetration type hydro skis of non-metallic construction mounted in tandem along the center line of the airplane. One ski is mounted near the nose and forward of the internal equipment, while the other ski is positioned at the extreme stern. These two main skis, in conjunction with small stabilizing skis near the wing tips, provide the required impact absorption and stability during landing. The rear ski contacts the water first, thereby decelerating the aircraft and protecting the control surfaces from impact forces. This, in turn, will cause the airplane to rotate until the nose ski assumes its position of the load. In this manner, the tandem

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skis should maintain a level attitude and maximum protection for the body to a very low forward speed. The forward ski is also expected to sense oncoming waves and raise the attitude of the airplane to minimize bow impact. By this procedure it is expected that the buoyancy of the wings can be maintained without damage to provide sustentation while the vehicle is being located.

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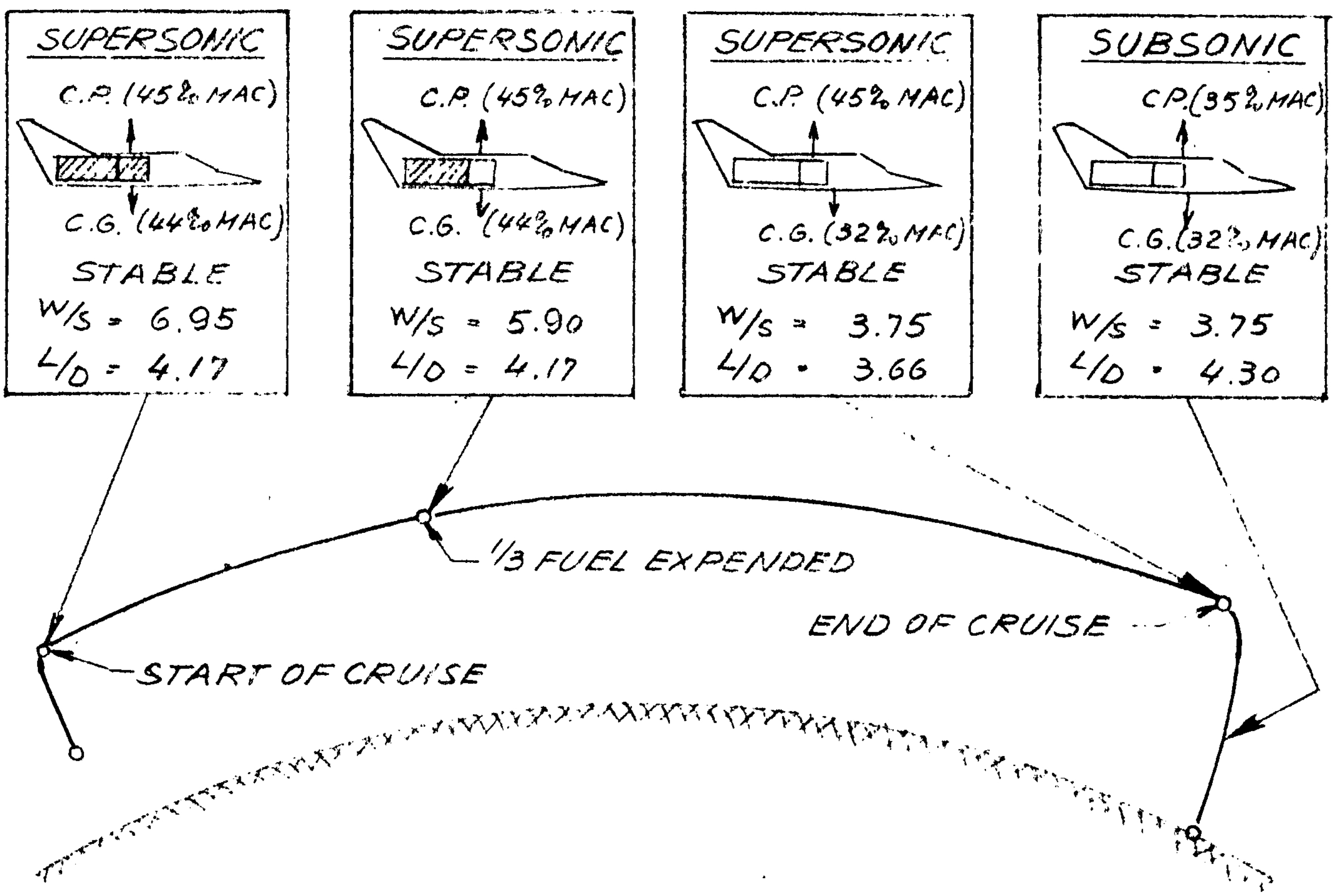

C. G. LOCATION AND STABILITY

FIG. 14

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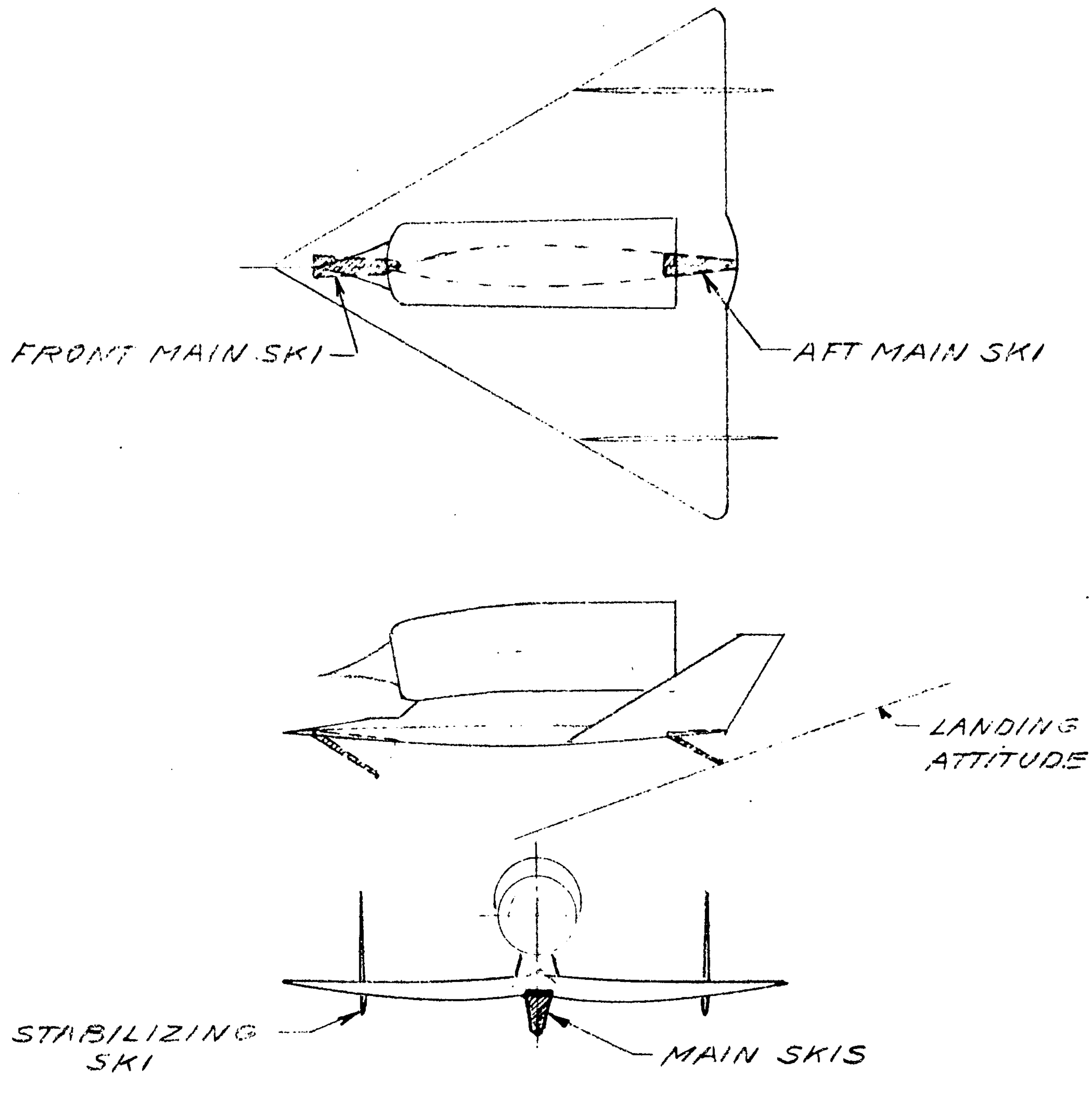

SKI ARRANGEMENT

FIG. 15

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B. Configurations

1. Single Engine Configuration

a. Marquardt Engines

The following single engine configurations have been studied:

Configuration	Wt. at start of Cruise	Wing Area Ft^2	Fuel	Construction	Remarks
MC-10	13,800	1985	B_5H_9	inflatable	stable subsonic & supersonic.
MC-11	12,400	1985	B_5H_9	inflatable	stable supersonic, unstable subsonic.
MC-14	27,000	3000	B_5H_9	inflatable	4000 n.mi. range
MC-19	18,310	1985	B_5H_9	rigid fiberglass	3200 n.mi. range

Configuration MC-10

Figure 16 shows a general arrangement drawing of the vehicle. Configuration MC-10 has the basic 1985 ft^2 wing as described in Section I - 4.

The Marquardt ramjet engine is mounted in the center on top of the wing.

From the front to the rear the fuselage houses the ski and parts of the electronics equipment, the pilot's capsule, guidance, navigation and electronics compartment, reconnaissance (payload) compartment, the fuel tank and the aft ski.

The two vertical surfaces are located near the wing tip. They separate the wing main surface from the wing tip control surfaces.

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CONFIGURATION MC-10

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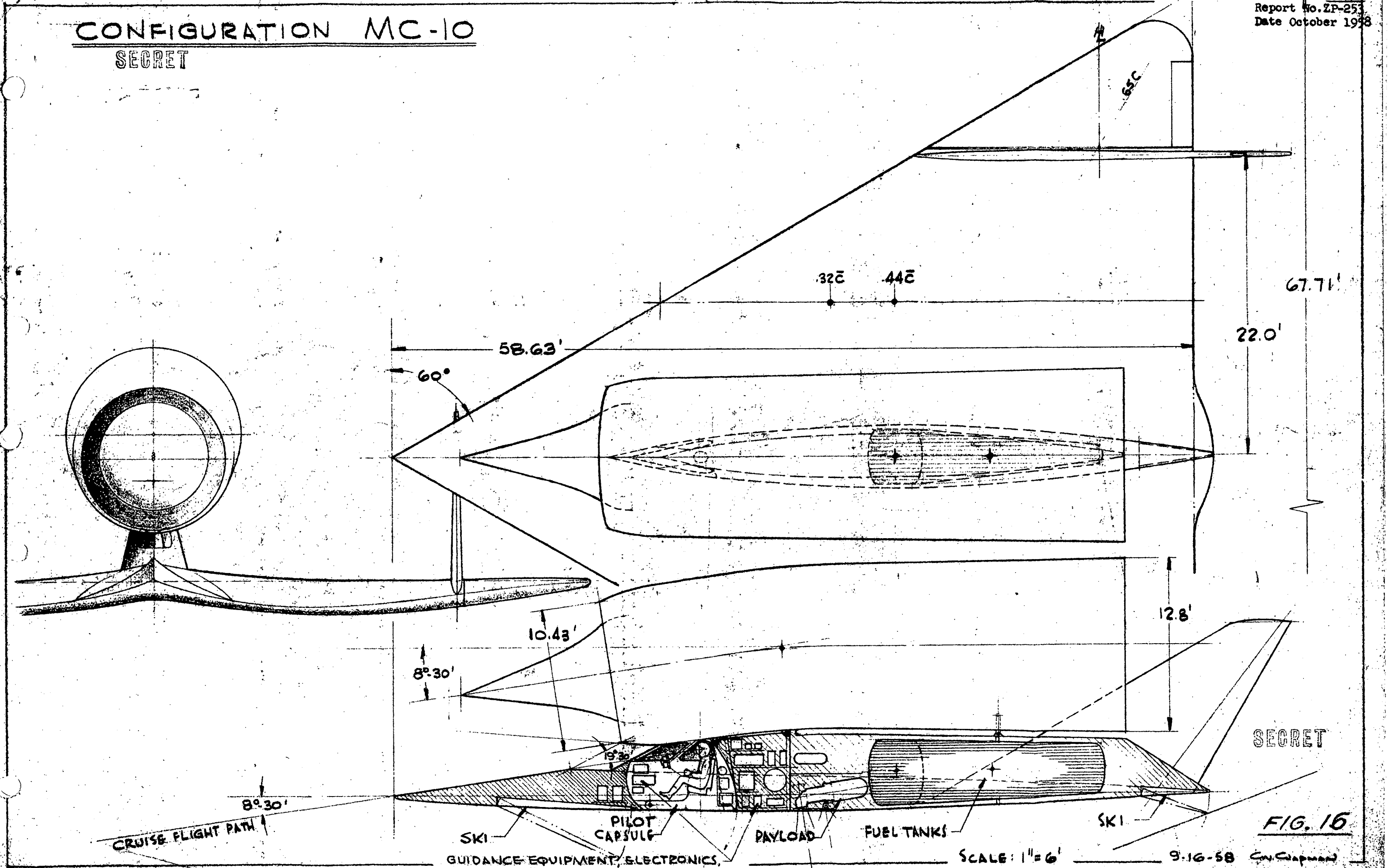

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FIG. 16

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The main characteristics are:

Wing area	1985 ft.^{2}
Span	67.71 ft.
Leading edge sweep.	60°
Overall length.	65.7 ft.
Overall height.	13.5 ft.
Weight at start of cruise . . .	13800 lbs.
Wing loading @ start of cruise.	6.95 lb/sq.ft.
Cruise altitudes: start	125000 ft
end	137800 ft

Configuration MC-10 is designed as a stable airplane with c.g. locations according to Section I - 6 and Figure 14.

The treatment of the engine inlet in combination with the wing leading edge is explained in detail in Report ZJ-026, Propulsion, Structure Heating and Pressurization.

Configuration MC-11

Figure 17 shows a general arrangement drawing of this configuration.

The main difference between MC-10 and 11 is that MC-11 is designed as an airplane which is stable supersonically but unstable subsonically. For the subsonic condition artificial means of stabilization have to be provided.

The weight saving of MC-11 over MC-10 is 1400 lbs.

This weight difference is due to the lower trim drag of the unstable airplane which improves the lift to drag ratio, therefore requiring a smaller engine. The lower engine weight and decrease in fuel weight results in a lower total weight.

Weight at start of cruise	12400 lbs
Wing loading at start of cruise . .	6.25 lb./ft.^{2}
Cruise altitudes: start	128000 ft.
end	143400 ft.

Configuration MC-14

This configuration has not been drawn. It is basically a larger version of the MC-10 configuration.

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CONFIGURATION MC-11

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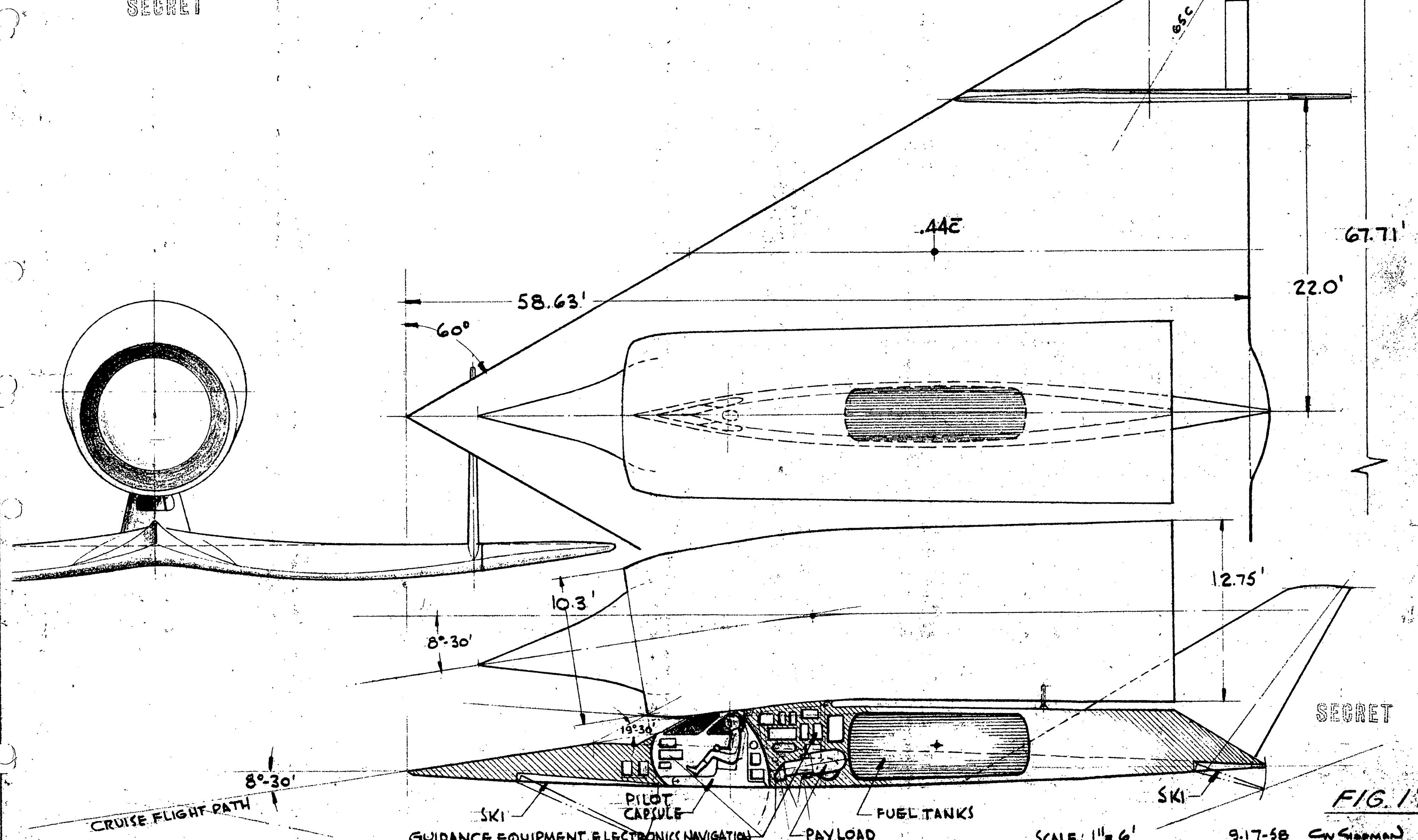

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FIG. 17

The larger amount of fuel for the 4000 N. M. range and, therefore, higher gross weight of the vehicle required a larger engine and, consequently, a larger airplane.

The main characteristics are:

Wing area	3000 ft.^{2}
Span	83.1 ft.
Root chord	72.2 ft.
Leading edge sweep	60°
Weight at start of cruise	27000 lbs.
Wing loading at start of cruise .	8.65 lb/sq.ft.
Cruise altitude: start	125000 ft.
end	141200 ft.

Configuration MC-19

In order to determine the weight difference between the proposed inflatable type construction of MC-10 and a rigid version MC-19, using the same wing planform, has been studied.

Wing, tail and fuselage pylon are fabricated of fiberglass laminate substructure (spars, ribs, frames) and covered with fiberglass laminate sandwich skins with honeycomb or corrugated core.

This configuration is also a stable airplane. The interior arrangement is the same as MC-10.

The main characteristics are:

Wing area	1985 ft.^{2}
Span	67.7 ft.
Leading edge sweep.	60°
Overall length.	65.7 ft.
Weight at start of cruise	18310 lbs.
Wing loading at start of cruise .	9.22 lb/sq.ft.
Cruise altitude: start	125000 ft.
end	137400 ft.

Configuration MC-19 is shown in Figure 18.

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CONFIGURATION MC-19

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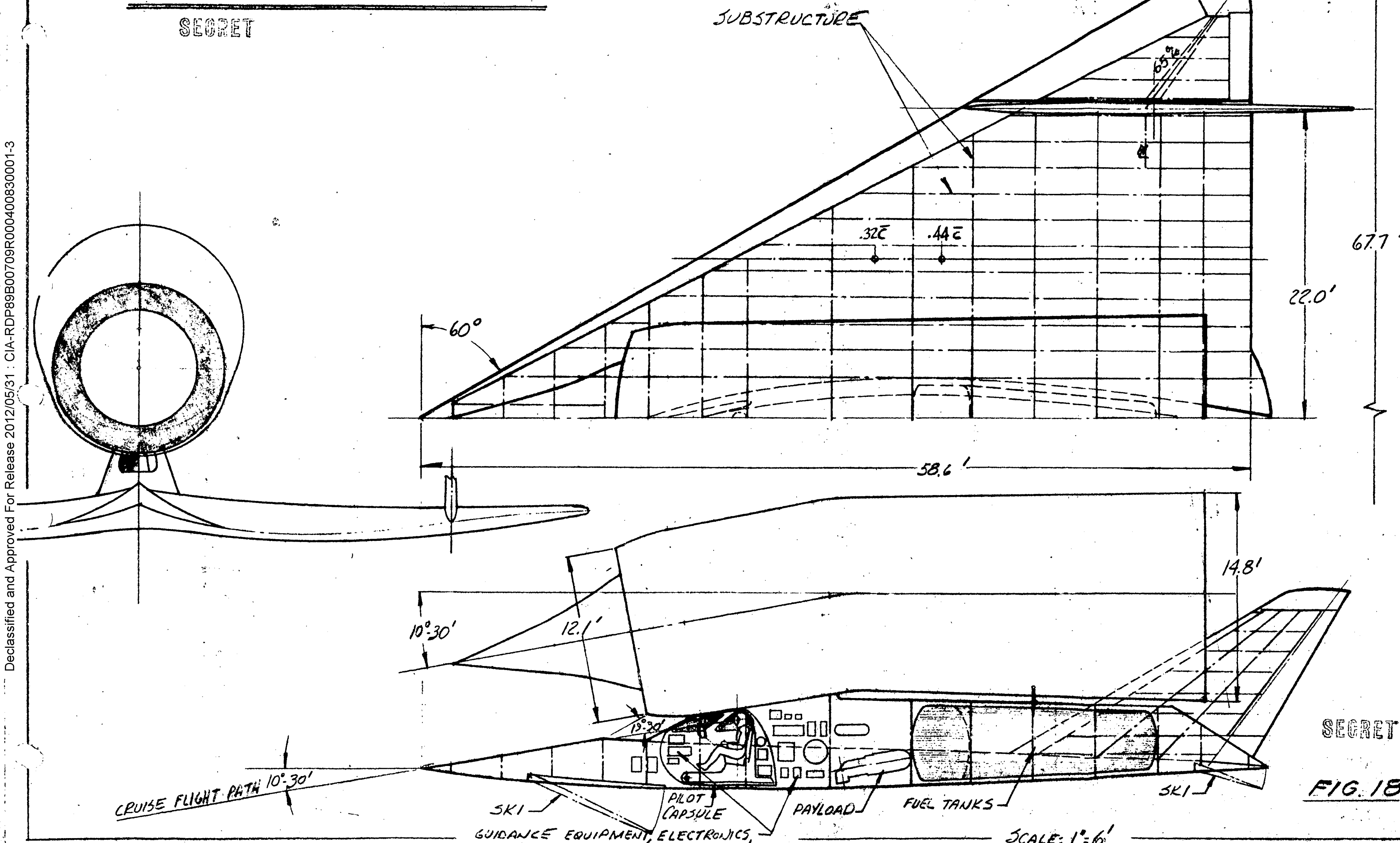

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FIG. 18

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2. Twin Engine Configurations

The following twin engine configurations have been studied.

Configuration	Wt. at start of cruise	Fuel	Engine Manufacturer	Remarks
MC-20	13,145	B_5H_9	Marquardt	60° Delta
MC-22	8,600	SF-1	Marquardt	60° Delta
MC-24	16,500	B_5H_9	Marquardt	60° Delta
PC-20	14,350	B_5H_9	P & W	60° Delta
PC-22	9,700	SF-1	P & W	60° Delta
PC-24	13,990	SF-1	P & W	60° Delta, 4000 n.mi.
PC-25		SF-1	P & W	70° Delta

Note: Wing area constant = 1985 ft^2

a. Marquardt Engines

Configuration MC-20

The twin engine configuration MC-20 is shown in Figure 19.

The engines are mounted approximately 13 feet outboard of the centerline of the airplane. Pilot, electronics, guidance equipment, payload and fuel is housed in a center fuselage in about the same manner as in configuration MC-10.

For lowest specific fuel consumption the Marquardt engine requires a burner length of approximately 16 ft. and the size of the engines for this configuration is based on this length.

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CONFIGURATION MC-20

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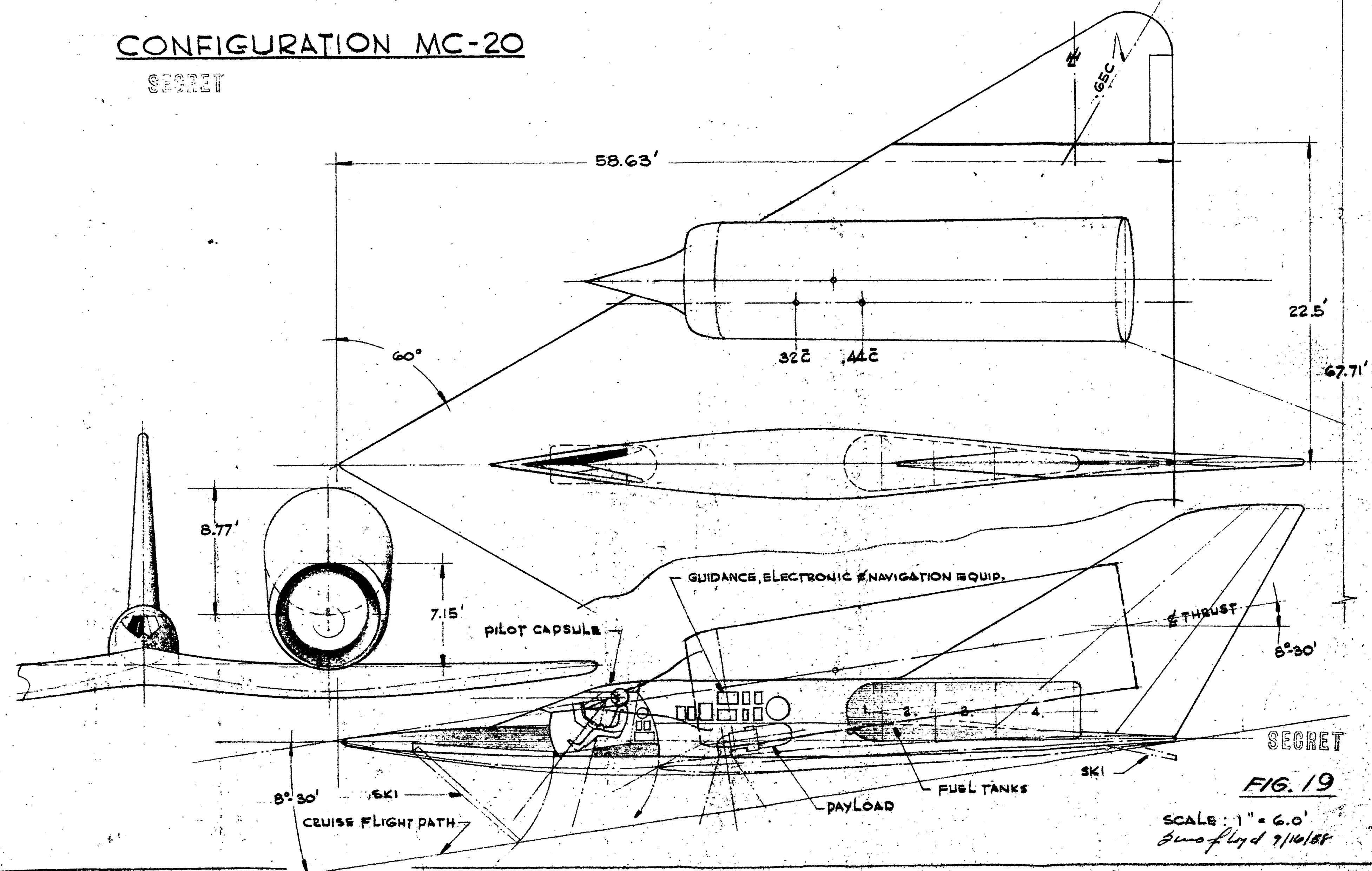

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FIG. 19

SCALE: 1" = 6.0'

The main characteristics are:

Wing area	1985	$ft.^2$
Span	67.71	ft.
Leading edge sweep	60°	
Overall length	67.7	ft.
Overall height	18.7	ft.
Wing loading at start of cruise . .	6.67	lb/sq.ft.
Cruise altitude: start	125,000	ft.
end	137,200	ft.

Configuration MC-22

Configuration MC-22 is configuration MC-20 using SF-1 fuel and is shown in Figure 20.

Better specifics for the SF-1 fuel reduce the fuel weight; however, the extreme low density of this fuel requires a large tank volume. The aft fuselage section is therefore determined by the size of the fuel tank.

Due to the low fuel weight of this configuration, the weight at start of cruise and, therefore, the wing loading is low. The table shows the effect in cruise altitude.

Weight at start of cruise	8600	lb.
Wing loading at start of cruise . .	4.34	lb/sq.ft.
Cruise altitude: start	139,000	ft.
end	146,000	ft.

Configuration MC-24

A study was made to determine the effect of shorter engines on the weight of the vehicle.

Figure 21 shows the configuration. Due to the decreased engine performance, as a result of the shorter burner length and the increase of propellant weight due to the higher specifics, the engine diameter had to grow accordingly. This, in turn, increased the total vehicle weight.

The penalty in weight over Configuration MC-20 is 3355 lbs. The main dimensions are the same as for Configuration MC-20.

Weight at start of cruise	16,500	lb.
Wing loading at start of cruise .	8.3	lb/sq.ft.
Cruise altitude: start	125,000	ft.
end	140,000	ft.

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CONFIGURATION MC-22

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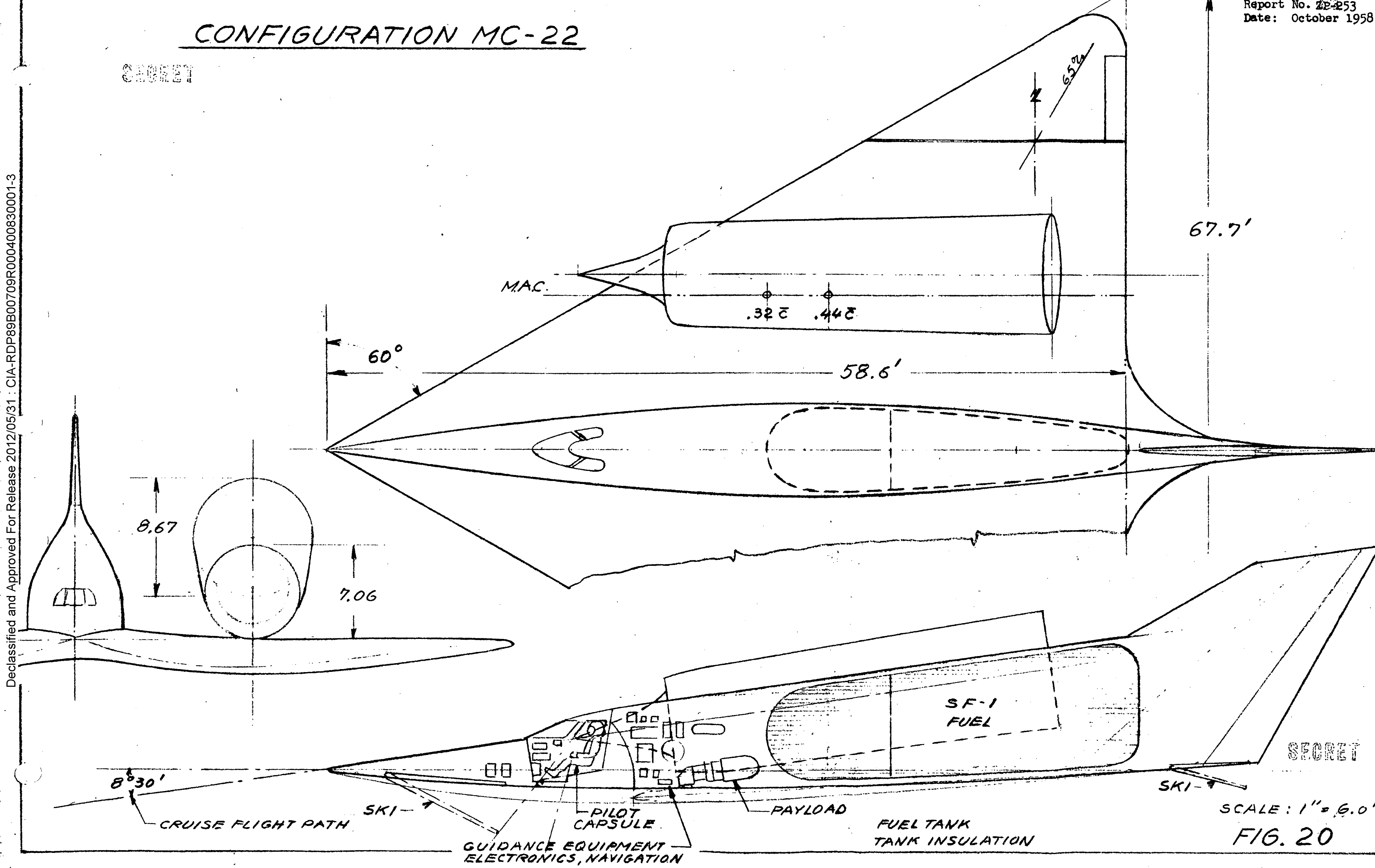

FIG. 20

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CONFIGURATION MC-24

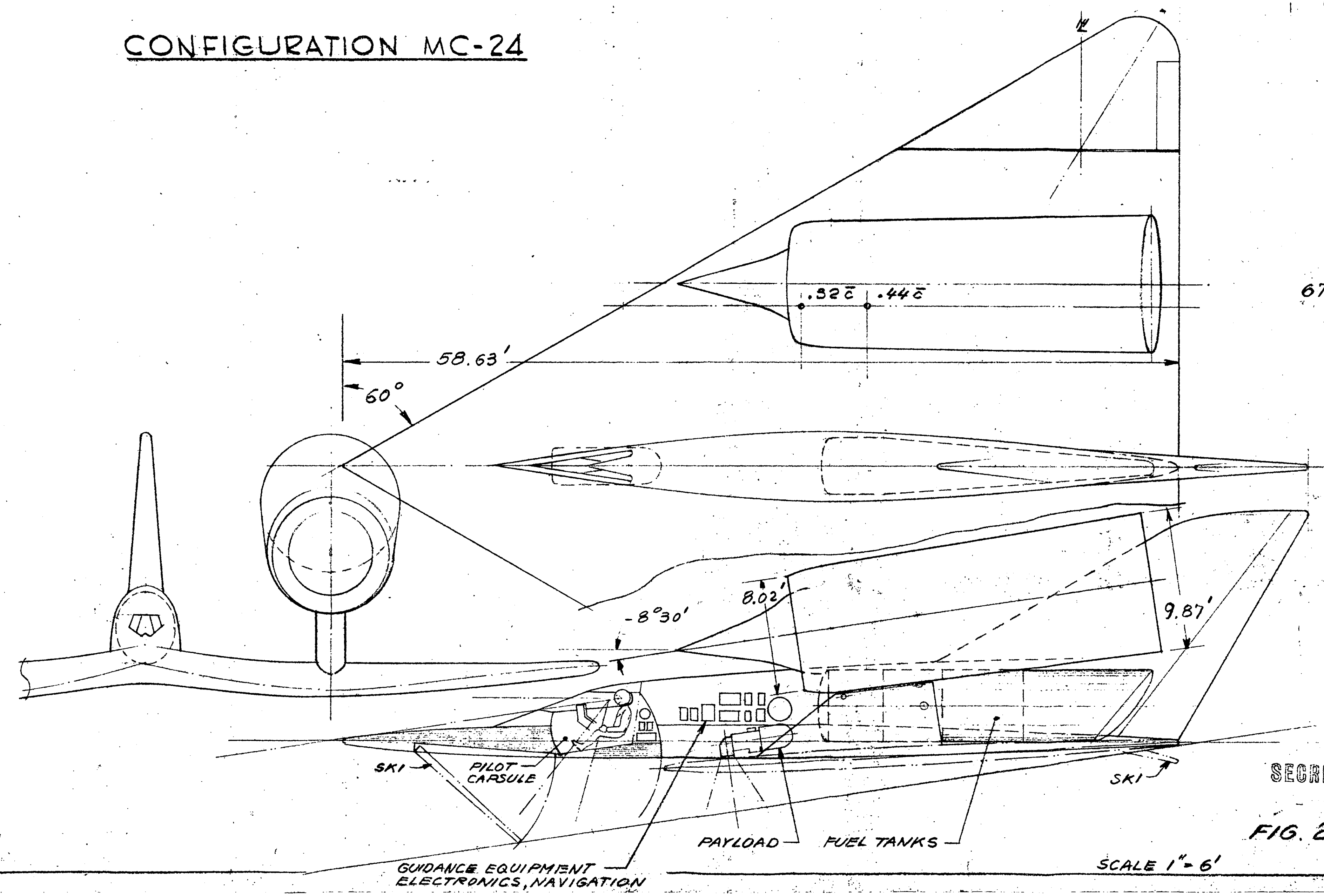

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FIG. 21

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b. Pratt & Whitney Engines

Configuration PC-20

A general arrangement drawing of Configuration PC-20 is shown in Figure 22. The configuration is similar to the twin engine Marquardt configurations with the basic wing of 1985 ft^2 area.

The center fuselage carries the same interior arrangement as all the other vehicles, the front ski, pilot's capsule, guidance, electronics and navigation equipment, payload, the fuel tank and the aft ski.

The two engines are mounted on top of the wing approximately 11.5 ft. from the center line of the airplane.

The main characteristics are:

Wing area	1985 $ft.^2$
Span	67.7 ft.
Leading edge sweep	60°
Overall length	67.5 ft.
Overall height	18.5 ft.
Weight at start of cruise	14350 lbs.
Wing loading at start of cruise. .	7.2 lb/sq.ft.
Cruise altitude: start	125000 ft.
end	136400 ft.

Configuration PC-22

Figure 23 shows the general arrangement drawing of this configuration. The basic 1985 $ft.^2$ wing has been used.

Generally it is similar to Configuration PC-20.

The main characteristics are:

Wing area	1985 $ft.^2$
Span	67.71 ft.

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CONFIGURATION PC-20

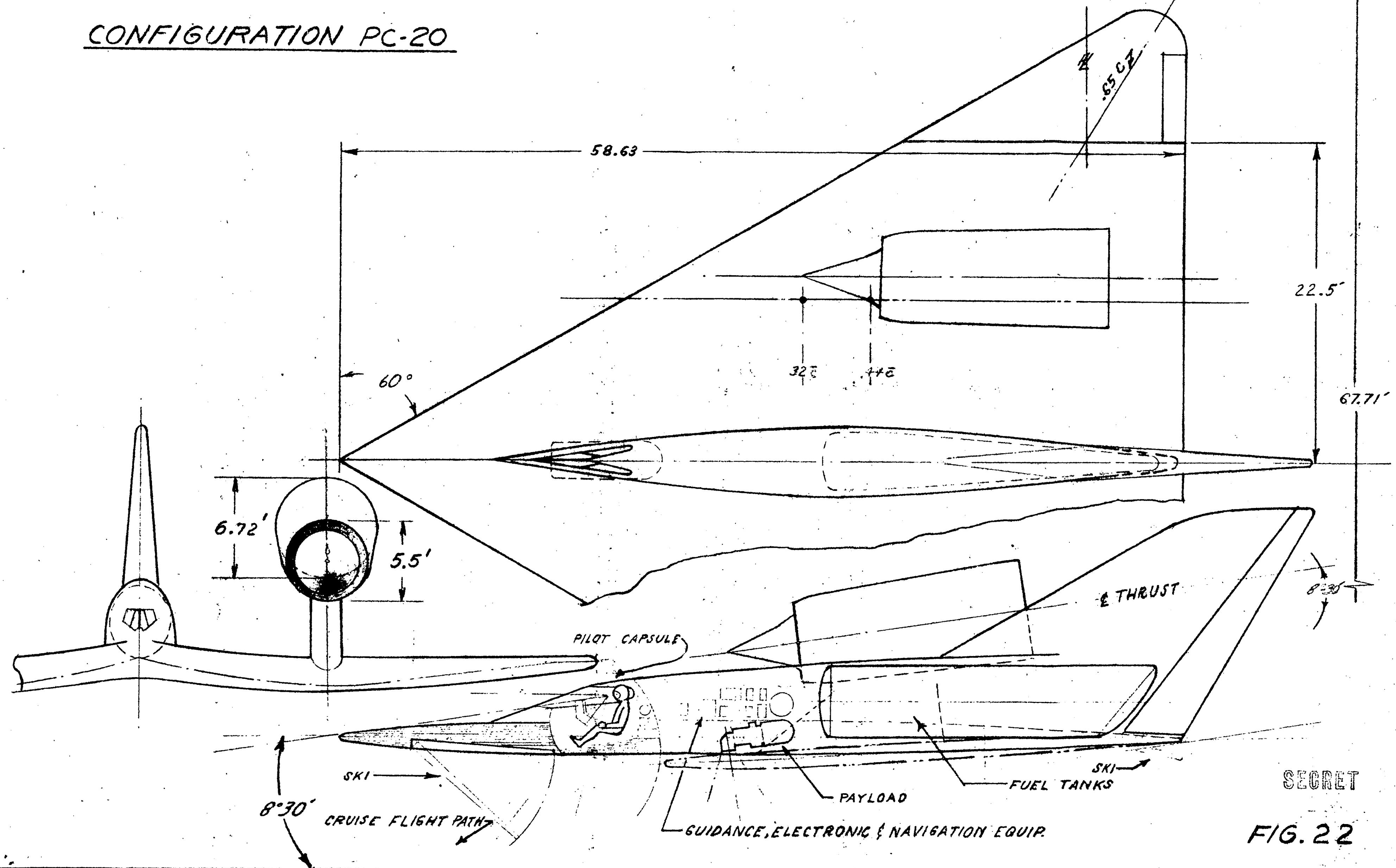

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FIG. 22

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CONFIGURATION PC-22

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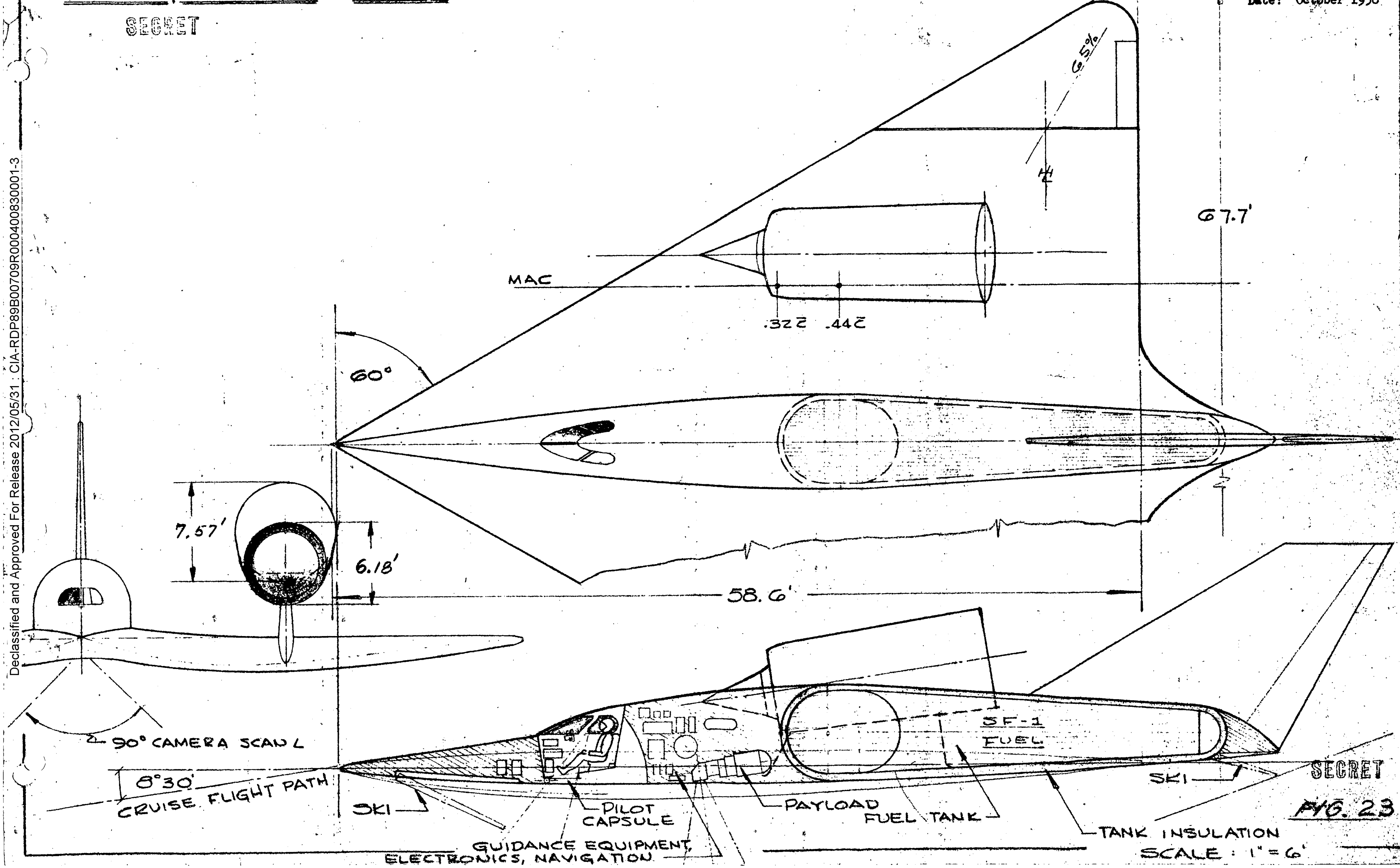

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FIG. 23

SCALE : 1" = 6'

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Leading edge sweep	60°
Overall length	76 ft.
Overall height	18 ft.
Weight at start of cruise . . .	10167 lbs.
Wing loading @ start of cruise .	4.89 lb.sq. ft.
Cruise altitude: start	136200 ft.
end	141800 ft.

Configuration PC-24

Configuration PC-24 is a 4000 N. M. version of the PC-22. A general arrangement drawing is shown in Figure 24.

The large volume required for the SF-1 fuel necessitated a special treatment of the aft end of the fuselage and part of the vertical tail as a fuel tank. The fuselage had to be extended about 12 ft. in order to provide for the necessary fuel tank space. The wing fairing is extended aft as a flat lower wing surface.

Due to the limitations of their test facilities Pratt & Whitney established a maximum engine size with the following data:

Inlet diameter	86.6 inches
Exit diameter	104.0 inches

This configuration required this maximum engine.

The main characteristics are:

Wing area	1985 $ft.^2$
Span	67.71 ft.
Leading edge sweep	60°
Overall length.	81 ft.
Overall height	18.5 ft.
Weight at start of cruise	13990 lbs.
Wing loading @ start of cruise .	7.04 lb./sq.ft.
Cruise altitude: start	125000 ft.
end	135000 ft.

Configuration PC-25

The long fuselage extension of configuration PC-25 which was required by the necessary fuel volume and the location of the fuel tank, lead to the design of Configuration PC-25.

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CONFIGURATION PC-24

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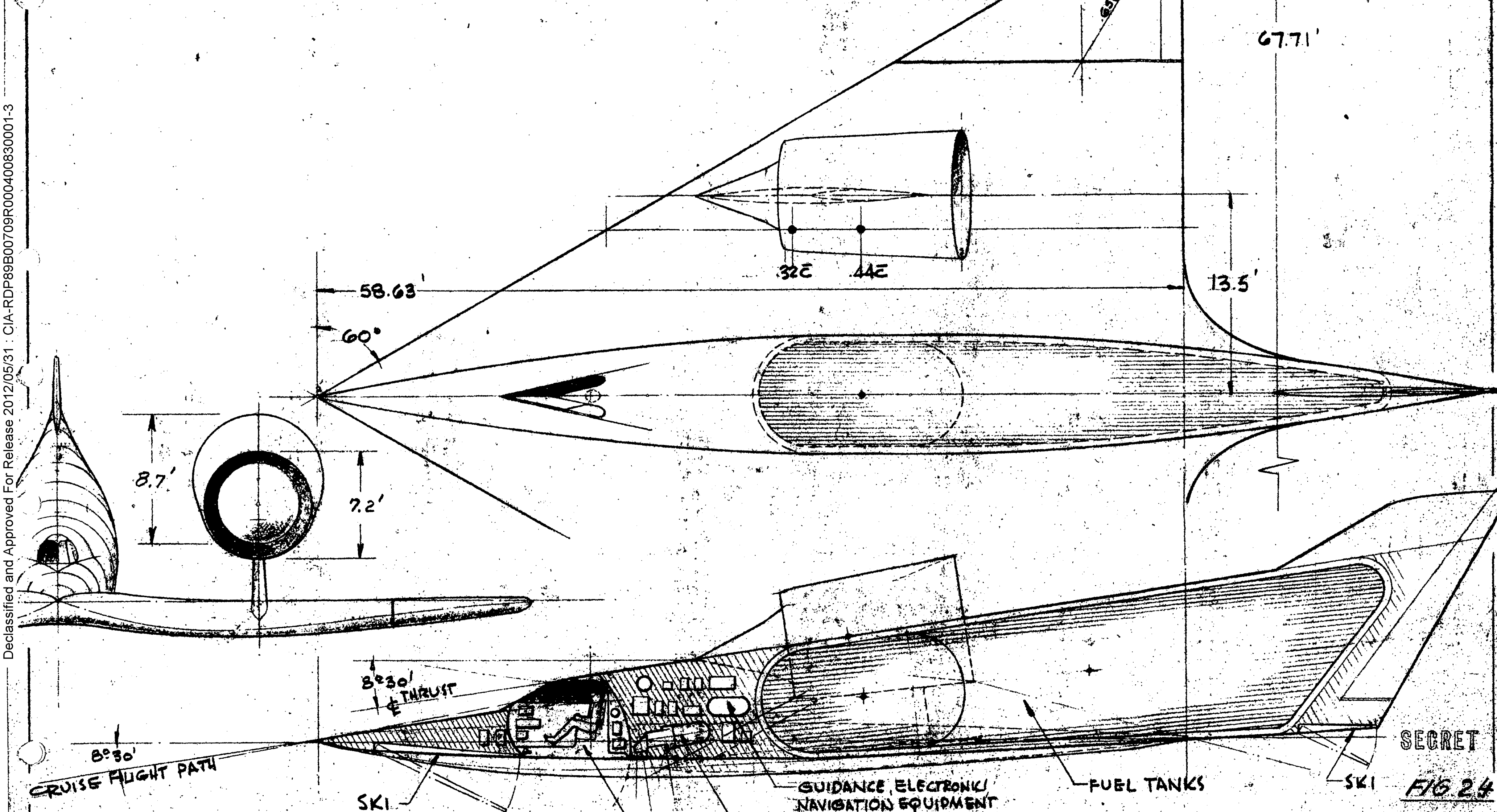

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FIG. 24

SCALE: 1" = 6'

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CONFIGURATION PC-25

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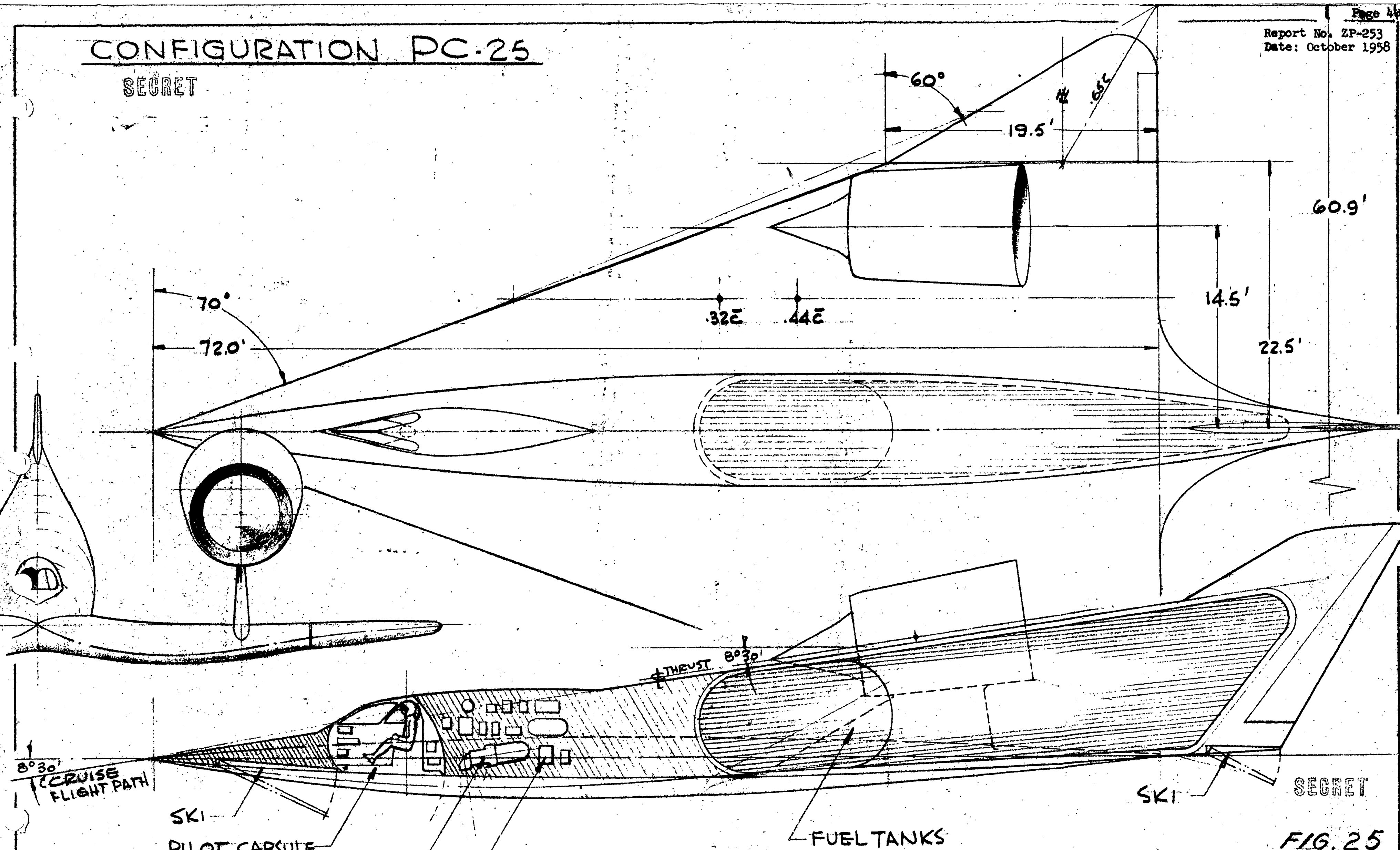

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FIG. 25

SCALE: 1" = 6'

9-23-58 CWChapman

The leading edge sweep was increased to 70° to provide a longer root chord of 72 ft. which shortened the overhang of the fuselage.

This configuration was not explored fully and no performance data was generated because of lack of time.

C. Weights

For the weight estimation it has been assumed that fixed equipment, crew and payload will be constant for all configurations. The fixed equipment includes the APU system, instruments, electrical equipment guidance and control, furnishings and environmental control. The fuel system, and inflation system, however, will vary with the type of fuel and the size of the wing. The basic engine weights were supplied by the engine manufacturer and has been ratioed according to size and thrust for each configuration. The wing weights were based on the wing weight curve Figure 45, page 114.

A typical weight breakdown for Configuration MC-10 is shown below and a weights comparison of all configurations on Table VII.

Item	Weight
Wing	1750 LB
Fins	145 "
Fuselage	207 "
Skis	150 "
Engine	1460 "
Miscellaneous	68 "
Fuel System	633 "
APU System	431 "
Instruments	50 "
Electrical	15 "
Guidance Equipment	615 "
Furnishings	200 "
Environmental Control	190 "
Inflation System	173 "
Contingency	383 "
Weight Empty	6470 LB
Crew	200 "
Payload	800 "
Wt. at End of Cruise	7470 LB
Propellant	6330 "
Wt. at Start of Cruise	13800

WEIGHT COMPARISON

CONFIGURATION	MC-10	MC-11	MC-14	MC-19	MC-20	MC-22	MC-24	PC-20	PC-22	PC-24
STRUCTURE	2 635	2 397	5 062	4 316	2 508	1 623	2 978	2 393	1 796	2 320
MISCELLANEOUS	68	65	103	68	100	100	100	100	100	100
ENGINE	1 460	1 200	2 960	1 800	1 340	920	1 338	1 470	1 710	2 198
FIXED EQUIPMT.	2 307	2 225	3 175	2 376	2 272	2 107	2 534	2 379	2 124	2 552
CREW	200	200	200	200	200	200	200	200	200	200
PAYLOAD	800	800	800	800	800	800	800	800	800	800
GLIDE WEIGHT	7 470	6 887	12 300	9 560	7220	5 750	7950	7 342	6 730	8 170
FUEL	6 330	5 513	14 700	8 750	5925	2 850	8 550	7 008	2 970	5820
WT. @ START OF CRUISE	13 800	12 400	27 000	18 310	13 145	8 600	16 500	14 350	9 700	13 990
BOOSTER	16725	15 030	32 700	22 190	15 920	10 400	19 950	17 380	11 700	16 960
LAUNCH WEIGHT	30 525	27 430	59 700	40 500	29 065	19 000	36 450	31 730	21 400	30 950

ALL WEIGHTS IN LBS

CONFIGURATION	MC-10	MC-11	MC-14	MC-19	MC-20	MC-22	MC-24	PC-20	PC-22	PC-24
NUMBER OF ENG.	1	1	1	1	2	2	2	2	2	2
FUEL	B_5H_9	B_5H_9	B_5H_9	B_5H_9	B_5H_9	SF-1	B_5H_9	B_5H_9	SF-1	SF-1
RANGE N. MI.	3200	3200	4000	3200	3200	3200	3200	3200	3200	4000

TABLE IV

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D. Launching Methods

The ram jet engine, designed for M = 3 at extreme high altitudes, as the main power plant requires a supplementary power system to launch the vehicle and boost it to M = 3 at cruise altitude. Methods possible to accomplish this are as follows:

1. Rocket launch, from sea level to cruise altitude;
2. Launch from balloon at approximately 80,000 ft. and boost to cruise speed and altitude.
3. Launch from conventional airplane (B-36, B-52) at approximately 54,000 feet altitude and boost to cruise speed and altitude.
4. Launch from air breathing booster (special vehicle) at approximately 80,000 ft. altitude and boost to cruise speed and altitude.
   a. using rockets
   b. using vehicle ram jet

Method 1, the straight rocket boost from sea level, was studied but due to severe difficulties foreseen in inflating the folded vehicle at 125,000 ft. altitude at a velocity of M = 3, the method was not further considered.

Methods 2, 3, and 4 are schematically shown in Figures 26, 27, and 28.

These methods seem feasible, however, a detailed study of each oneof them is necessary in order to solve all problems involved.

The main difficulty with methods 3 and 4 is that the vehicle should not exceed the design g limits during the boost phase. Therefore the boost acceleration at lower altitudes must be controlled and speeds gradually increased to M = 3 at cruise altitude.

The size and power of the final boost system will be established by the selected launching system.

For the MC-10 vehicle with a weight of 13800 lbs. lifted by the B-36 to 45,000 ft. altitude the boost to M = 3 and 125,000 ft. altitude can be accomplished by the system shown in Figure 29. The boost system consists of three existing liquid propellant rockets, Bell Aircraft Corp. Model LR-81-BA-1. This rocket uses JP-4 as a fuel and Inhibited Red Fuming Nitric Acid as an oxidizer. The three (3) rockets would be mounted in a horizontal plane across the aft of the vehicle with the center rocket in line with the vehicle center of gravity. All three (3) rockets would be fired for the first stage with only the outer two (2) rockets being used for the second stage. At completion of the second stage, all three rockets would be jettisoned in a cluster from the vehicle.

The clustered mounting scheme for the boost rocket appears to be a straight forward method of securing the thrust component in the desired direction at a minimum of weight. Also, there are no vehicle structural penalties as a result of the booster rockets after these have been fired.

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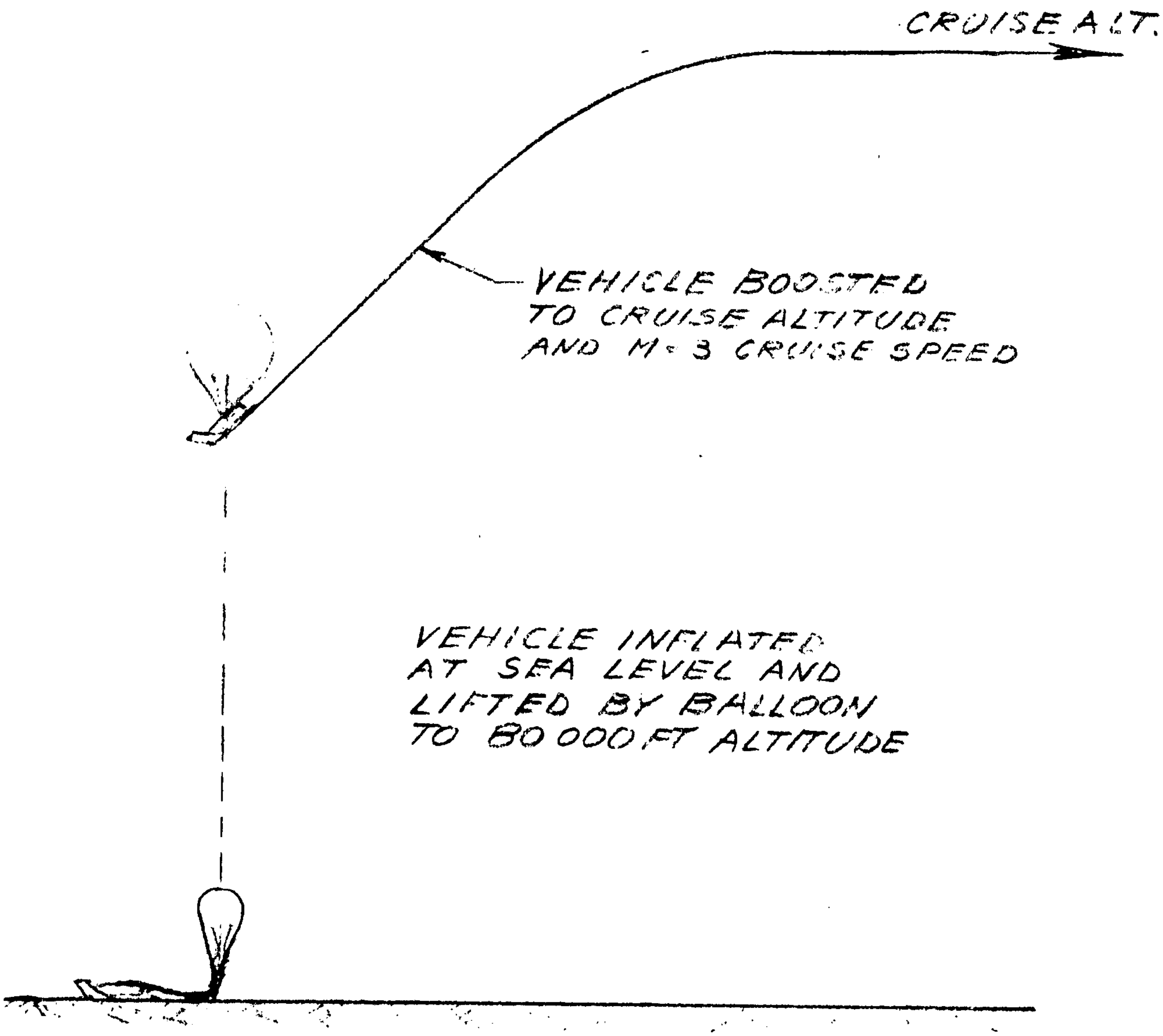

BALLOON LAUNCH
FIG. 26

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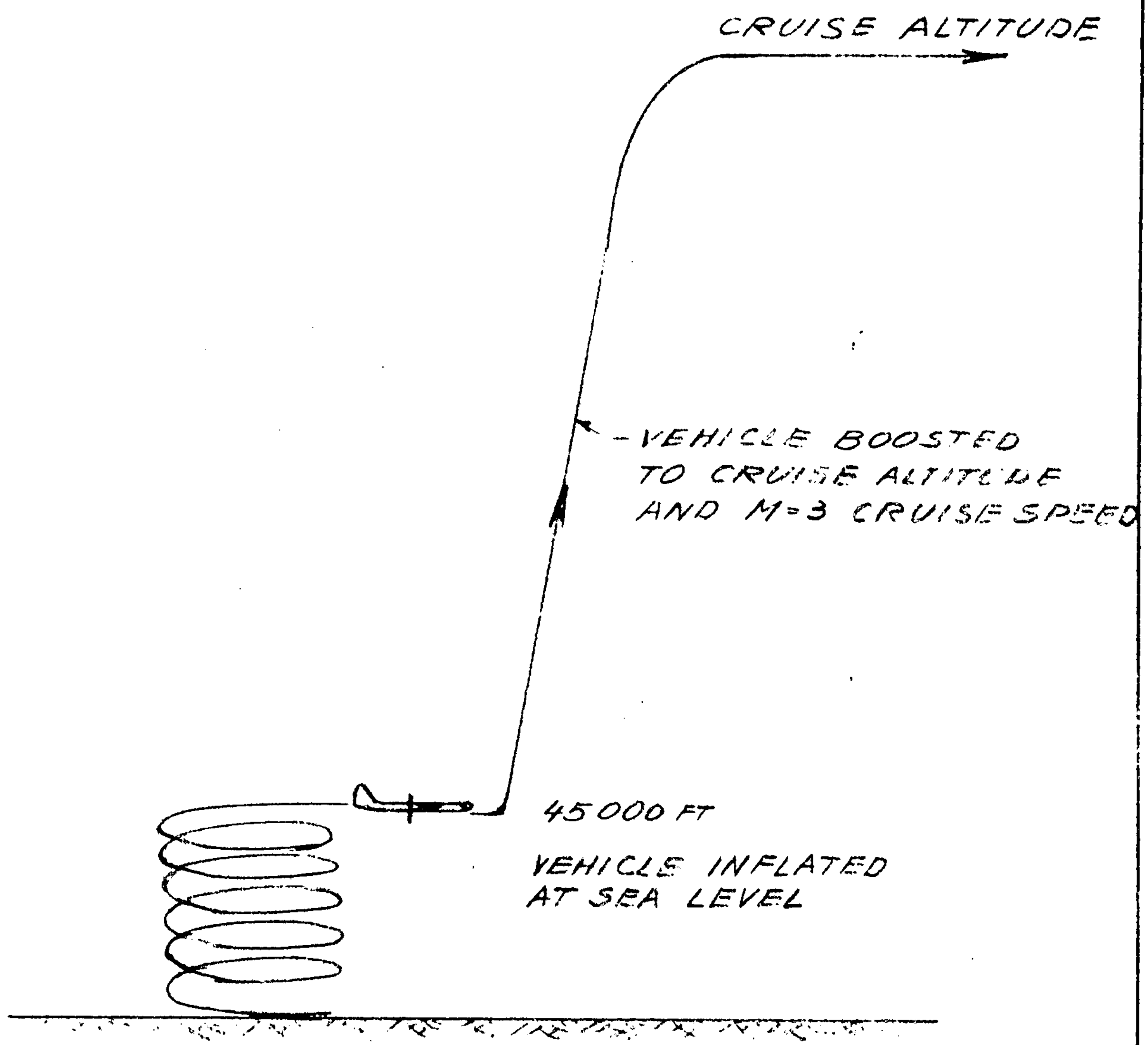

LAUNCH WITH B-36
FIG. 27

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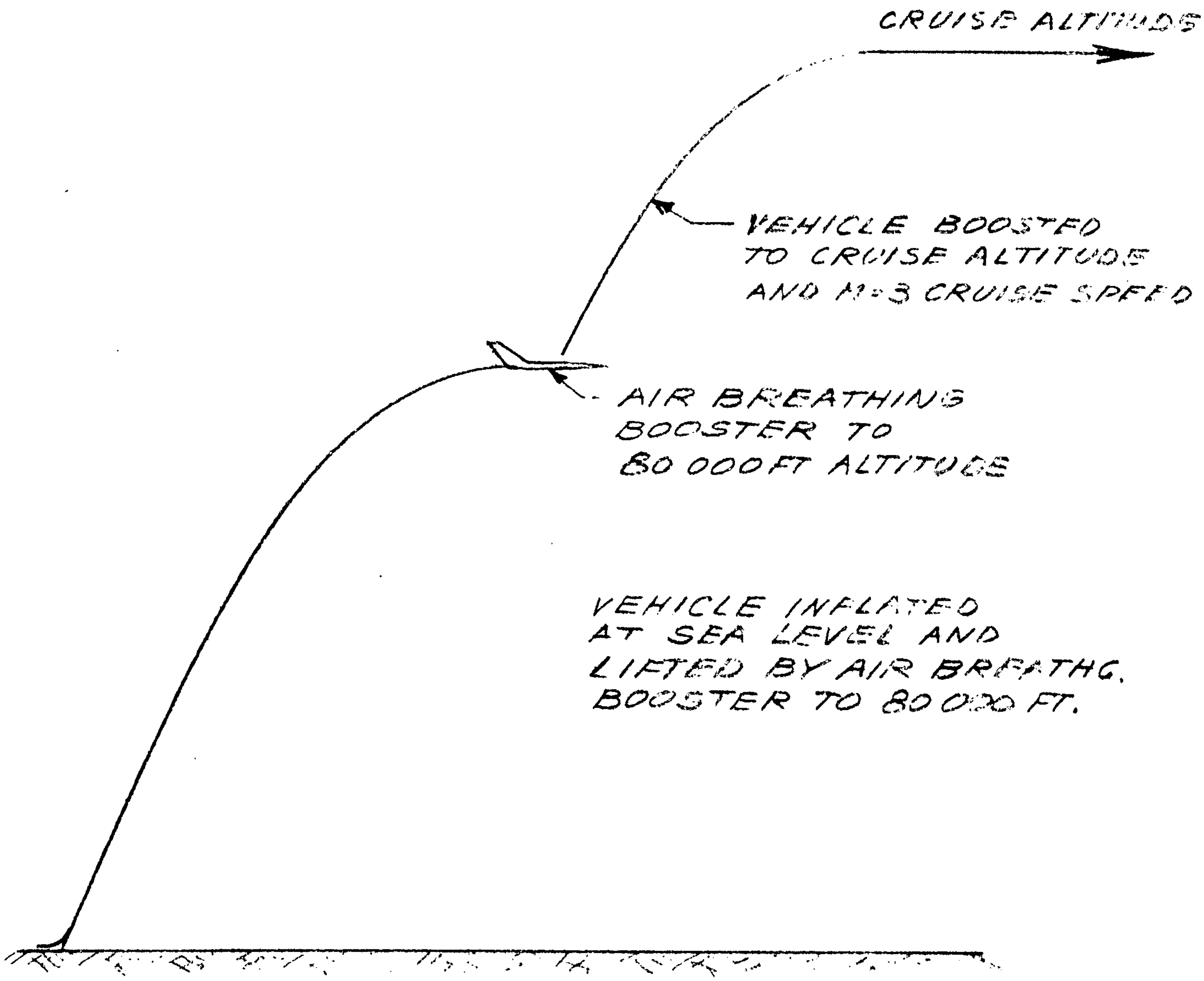

LAUNCH WITH AIR BREATHING BOOSTER

FIG. 28

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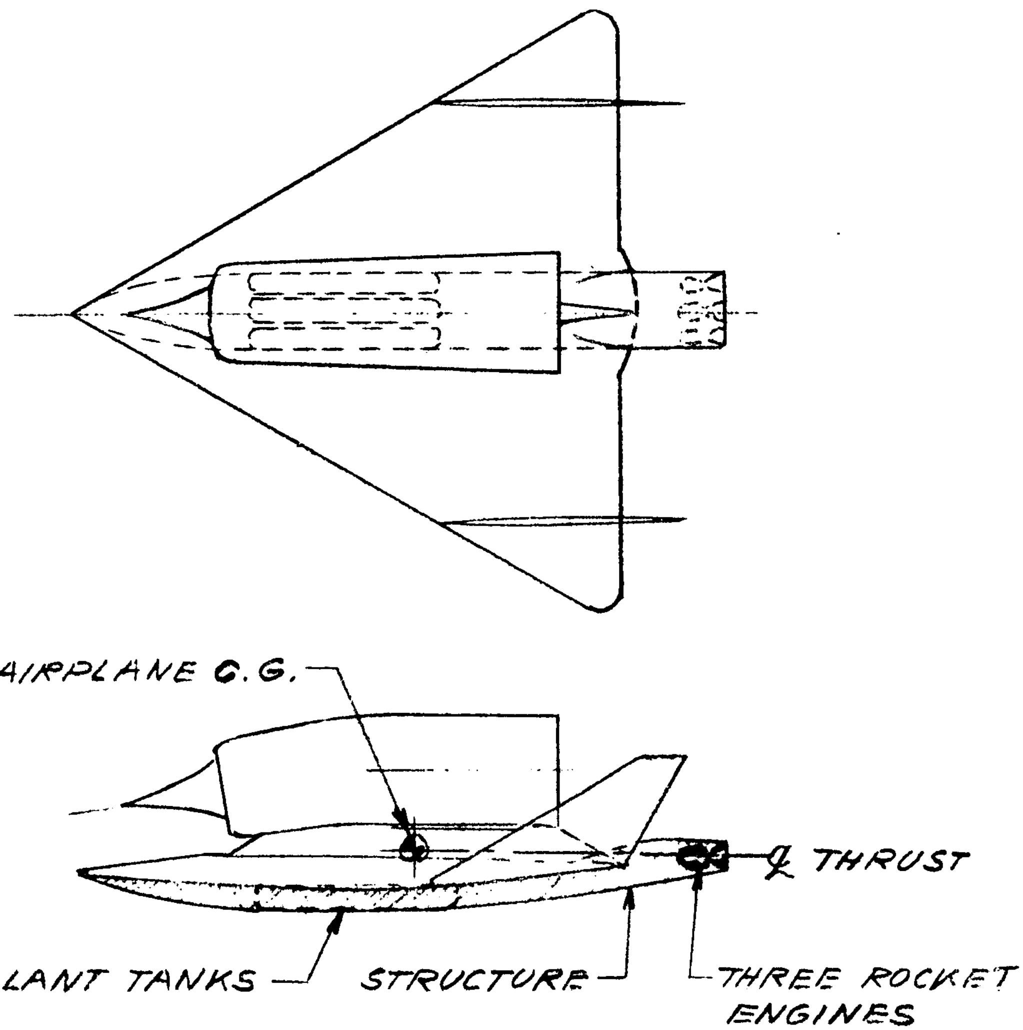

BOOSTER PACKAGE
JETTISONABLE

1 1/2 STAGE ROCKET BOOSTER USING
3 BELL MODEL 117 ROCKET ENGINES
(STORABLE LIQUID SYSTEM)

BOOST SYSTEM
FIG. 29

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SECTION II

SYSTEMS

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The Hazel Systems are shown schematically in Figure 30. The following subsystems are summarized in this section:

1. Auxiliary Power
2. Thrust Vector Control
3. Control Actuation
4. Boost Rocket Separation
5. Structure Pressurization
6. Air Conditioning
7. Electrical Power Generation
8. Fuel System

1. Auxiliary Power

Estimated auxiliary power requirements can be satisfied with the following two (2) auxiliary power systems:

a) Vehicle Liquid Monopropellant Auxiliary Power System
b) Boost Rocket Pneumatic Servo System

a) Vehicle Liquid Monopropellant Auxiliary Power System

This system consists of a liquid monopropellant, such as hydrozine, decomposed in a catalyst gas generator. The propellant is supplied to the gas generator by pressure secured from the pressurized structure helium supply. Flow is varied on demand by sensing gas generator pressure. Systems receiving energy are:

1) Structural Pressurization
2) Aerodynamic Surface Control
3) Electrical Power

b) Boost Rocket Pneumatic Servo System

This system consists of a gas source which supplies energy to an actuator through a pneumatic servo. A more detailed analysis relative to state-of-art at time of detail design will select either a hot gas, bottled gas or other energy source. The system receiving energy is the Boost Rocket Vector Control.

2. Thrust Vector Control

Thrust vector control is required on the Boost Rocket to correct deviations from the proposed flight path. The choice of method to obtain jet deflection cannot be made without further study. Present indications are that the final choice will be between swivel (gimbaled) nozzles and jetevators.

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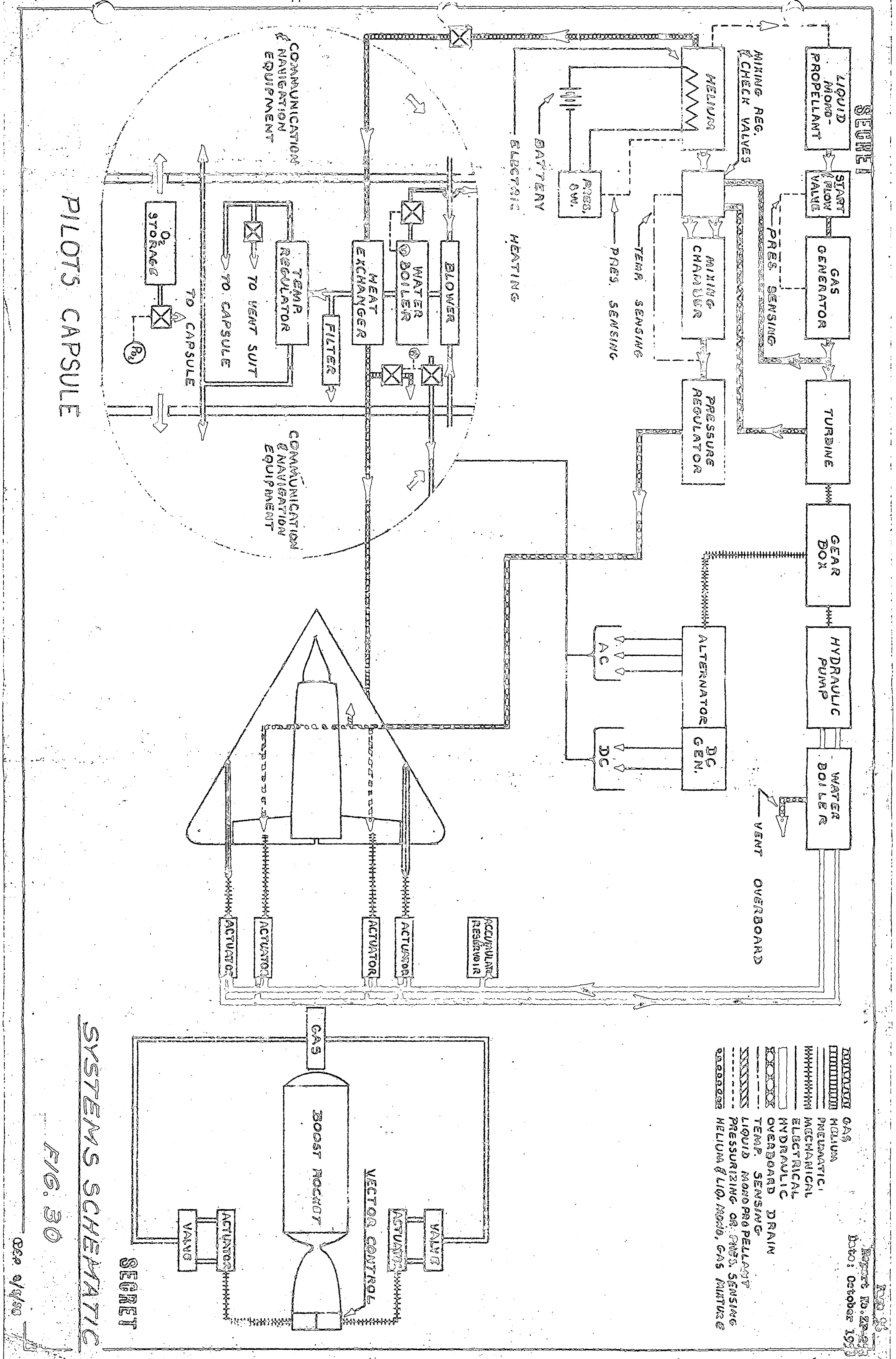

SYSTEMS SCHEMATIC

FIG. 30

However, all feasible means should be evaluated. The accepted method will be the one that gives the lightest weight, is most reliable, and is capable of production for the time period of the Hazel Program.

3. Control Actuation Systems

Progress in the development of hot gas servo systems justifies their use at low power levels and for short durations. Extending their application to long durations and high power levels is desirable. However, this will require significant development advances before there is an advantage relative to the combination of gas turbine and hydraulic system.

Hot gas servos are used for the boost rocket vector controls. They offer minimum weight for the low power and short duration requirements of the Boost Rocket Vector Controls.

Aerodynamic control surfaces are actuated by hydraulic servos driven by the Vehicle Liquid Monopropellant Auxiliary Power System. A small water boiler is used for hydraulic fluid cooling. The system is sized for the average load, with an accumulator supplying energy at the peak loads. A parallel development of a hot gas servo system is recommended with the objective of decreasing weight and increasing reliability.

4. Boost Rocket Separation System

The Boost Rocket System will be jettisoned at the end of the boost. The selection of an optimum separation scheme can only be made after a more detailed final configuration analysis.

5. Structure Pressurization System

Initial pressurization of wing and tail surfaces is supplied on the ground prior to take-off. Means must be provided for the controlled escape of a portion of this gas with 1) decreasing ambient pressure as the vehicle is projected from a static condition at sea level to M = 3 at 125,000 feet and 2) increased internal temperature due to aerodynamic heating during cruise. Following this loss and stabilization at cruise conditions, gas must be added to off-set leakage and maintain the given 15 PSIG pressure differential as increasing ambient pressures are encountered during let-down from altitude. The inlet gas must be injected at such temperatures as to preclude thermal damage to the structure and to minimize total system weight. The pressurization medium chosen must remain a gas over the temperature and pressure range encountered within the structure.

Minimum system weight for structural pressurization is afforded by a system using helium gas stored in liquid form and heated to the desired temperature by direct mixing with hot exhaust gases from the Vehicle Monopropellant Auxiliary Power Gas Generator. Total weight of the Proposed system is 142 pounds. Report ZJ-026, "Propulsion, Structural Heating and Pressurization" evaluates various systems for supplying the pressurization and outlines those found most promising.

6. Air Conditioning

Minimum total system weight is achieved by a completely sealed cabin (zero leakage) utilizing circulating air at sea level pressure with water boiling at reduced pressure (and temperature) as the ultimate heat sink. Carbon dioxide, odors and water vapor are removed by a multi-purpose filter while make-up oxygen is supplied from a high pressure gaseous storage bottle. Cooling, during flight altitudes at which the water boiler is no longer effective, is supplied by the helium gas used for pressurization during this phase.

7. Electrical Power Generation

The output from the vehicle liquid monopropellant auxiliary power system gas generator, through a turbine and gear box, powers an electric alternator. The alternator provides alternating current directly and direct current through a direct current generator to systems requiring electric current.

8. Fuel System

Figure 31 shows a Pentaborane, vapor feed fuel system schematic.

It is assumed that the heat exchanger is to be integral with the engine. For a 4000 pound fuel system, approximately 75 square feet of heat exchanger area is required if it is part of the engine, while approximately 1500 square feet would be required if it were part of the airframe wing area. The heat exchanger weighs approximately one (1) pound per square foot. The weight saving aspect is obvious.

The pumping rate through the heat exchanger provides simple control of vapor boil off.

Decomposition within the heat exchanger will be within operational limits to avoid heavy deposits for single flights. The system should be cleaned after each flight. It is estimated that five hours operation should be the maximum without cleaning.

Fuel pressure must be at design maximum pressure at engine ignition, and fuel temperature must be within 3°F of the boiling point at ignition to promote vapor for starting.

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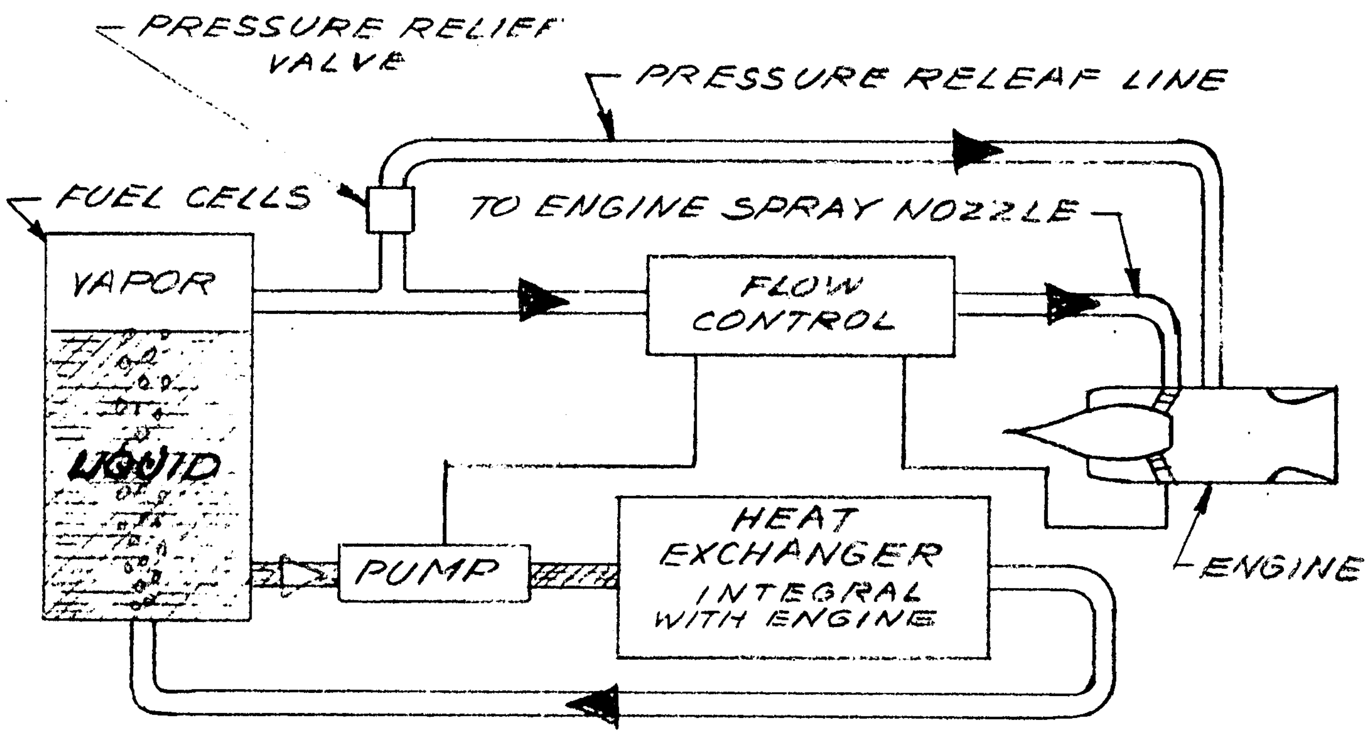

FUEL SYSTEM SCHEMATIC
PENTABORANE VAPOR FEED
FIG. 31

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SECTION III

CREW CAPSULE AND ESCAPE METHODS

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Both the seated and the prone position for the pilot were considered. The prone position would be the ideal position for the mission because of the prime requirement of this vehicle, namely reconnaissance from high altitudes. The prone position, however, is debatable for approach and landing.

Both types are shown.

Figure 32 represents a normal seated cockpit with standard type controls and console arrangement. Access would be through side door. Escape from the aircraft would be by hinging the seat downward. Deployment of a 30" diameter stabilizing parachute snatches the pilot from the seat and provides a stabilized descent at a terminal velocity of approximately 100 M.P.H. indicated air speed. Automatic parachuting, actuated by an aneroid, would occur at 15,000 ft. altitude. A back type parachute and a seat type "Global Survival Kit" are provided.

Figure 33 shows a prone installation with the pilot close to the lower surface of the aircraft. The prone support is integral with a hinged door. Extension of this door provides an escape chute. A chest type survival pack and a back type parachute is provided. This system also incorporates a 30" diameter stabilizing parachute for stabilized descent from high altitude. The prone position is not considered satisfactory for high longitudinal accelerations, such as launching, unless further support is provided.

Figures 35 and 36 show means for supporting the pilot when subjected to high and prolonged accelerations.

The two position cockpit (Figure 34) provides a separate supine type seat for flight during launch. The seat also provides emergency escape during launch. Normal flight position is prone and provides maximum downward or reconnaissance vision. An escape chute provides for escape when the pilot is in the prone position. A chest type survival pack forms part of the prone support and is secured to the pilots harness. The prone support is an integral part of a lower surface door (hinged on the forward edge). Extension of this door provides an escape chute. Three seconds after leaving the aircraft a 30" diameter stabilizing parachute is deployed and provides the pilot a terminal (falling) velocity of approximately 100 M.P.H. indicated air speed. An aneroid set for 15,000 ft. releases the main parachute automatically.

Figure 35 shows a prone installation with the pilot suspended in water. A close fitting rigid container froms the pilots compartment and is completely filled with water around the pilot. Water is circulated through a conditioner and is pressurized back to the compartment. The system is designed to withstand 80 G's. and is an integral part of a section of the aircraft capable of

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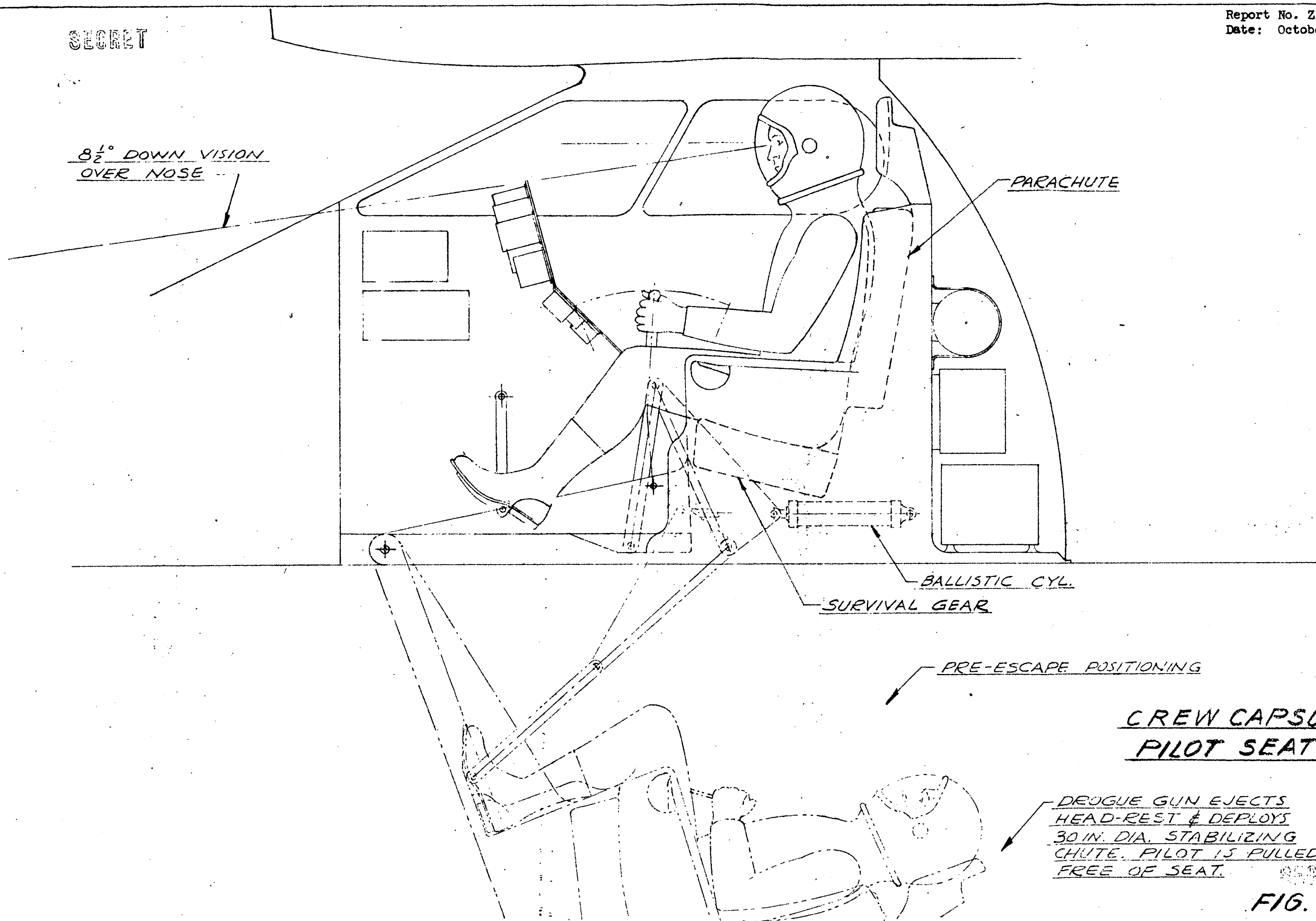

CREW CAPSULE
PILOT SEATED

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FIG. 32

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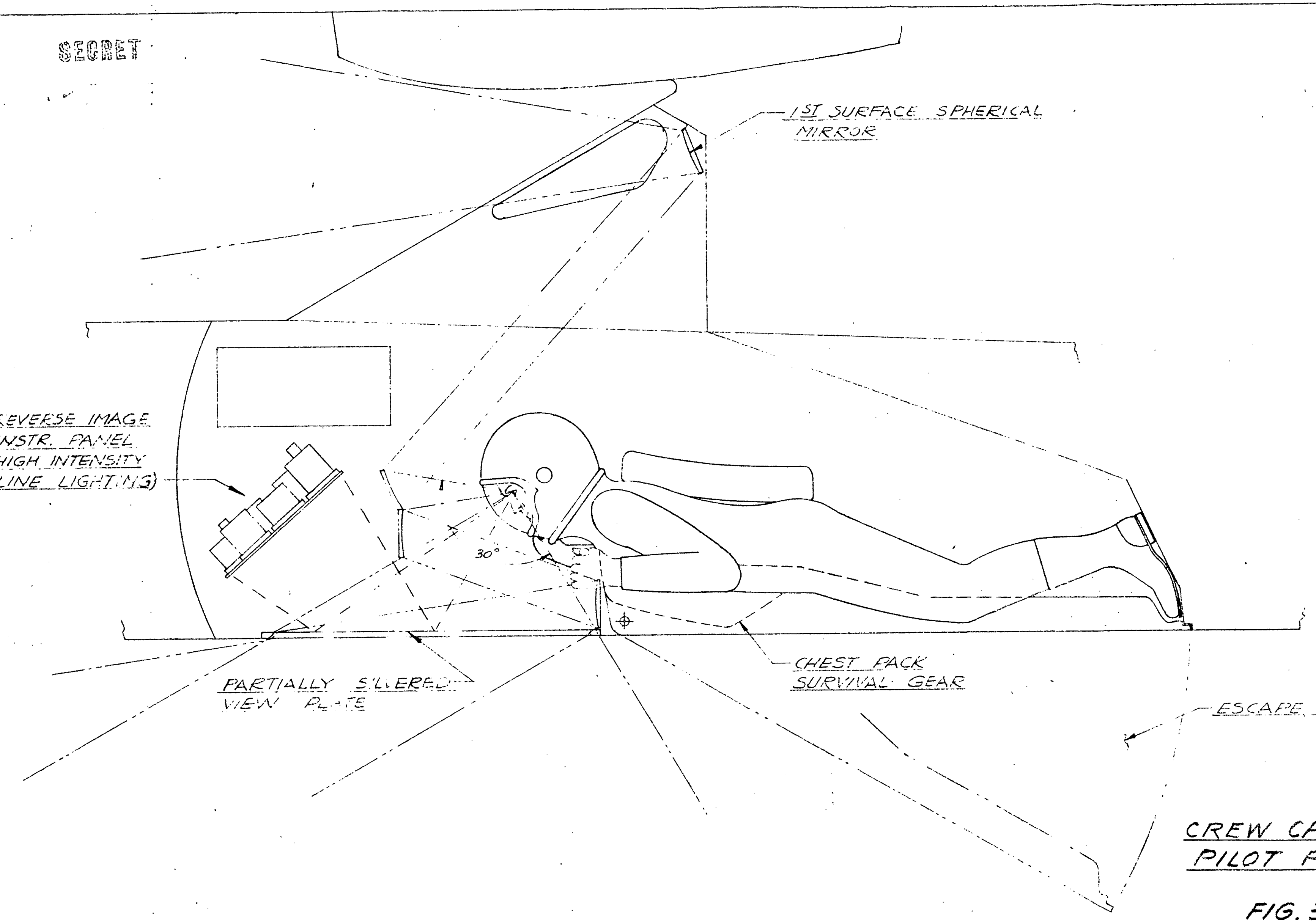

CREW CAPSULE
PILOT PRONE

FIG. 33

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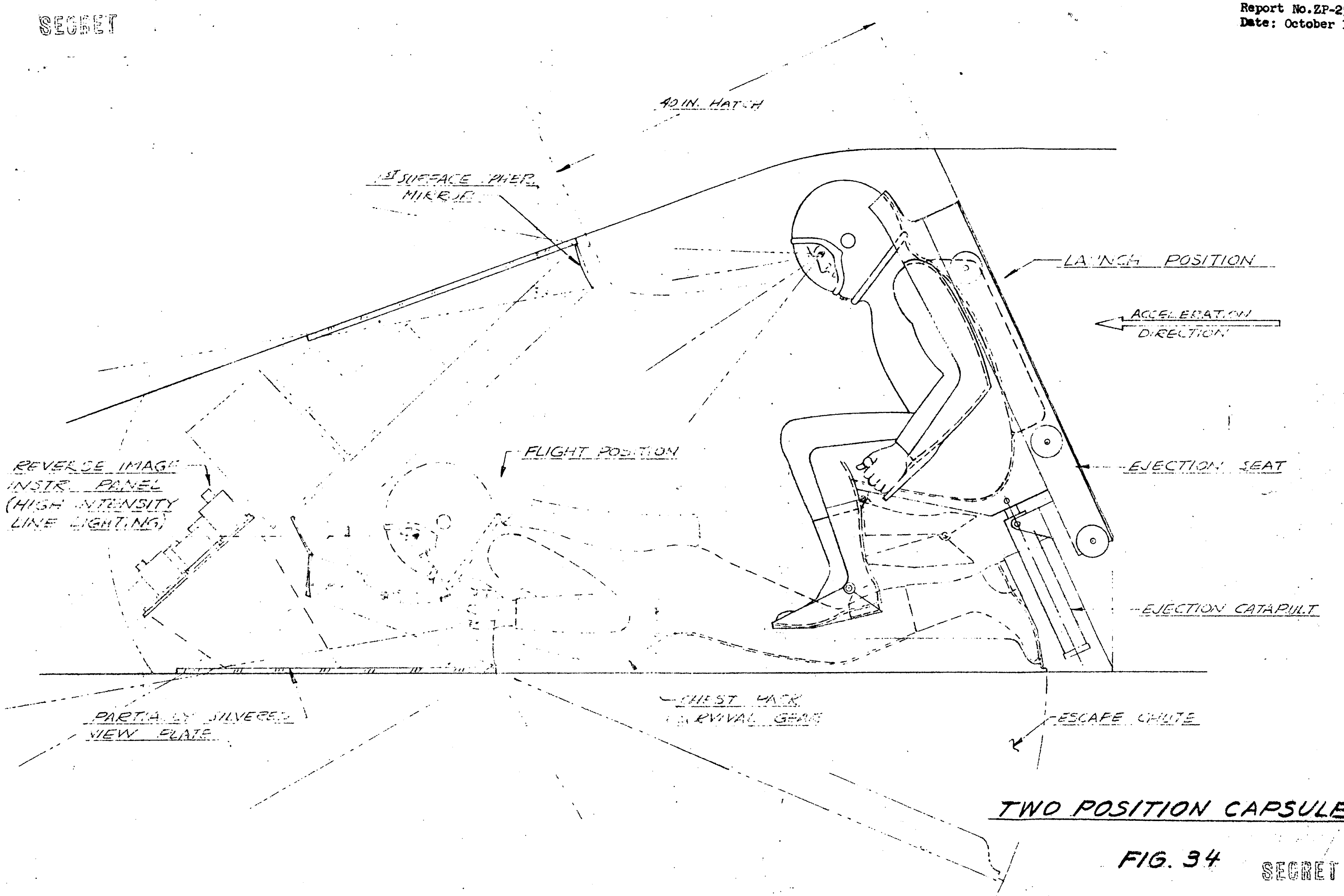

TWO POSITION CAPSULE

FIG. 34

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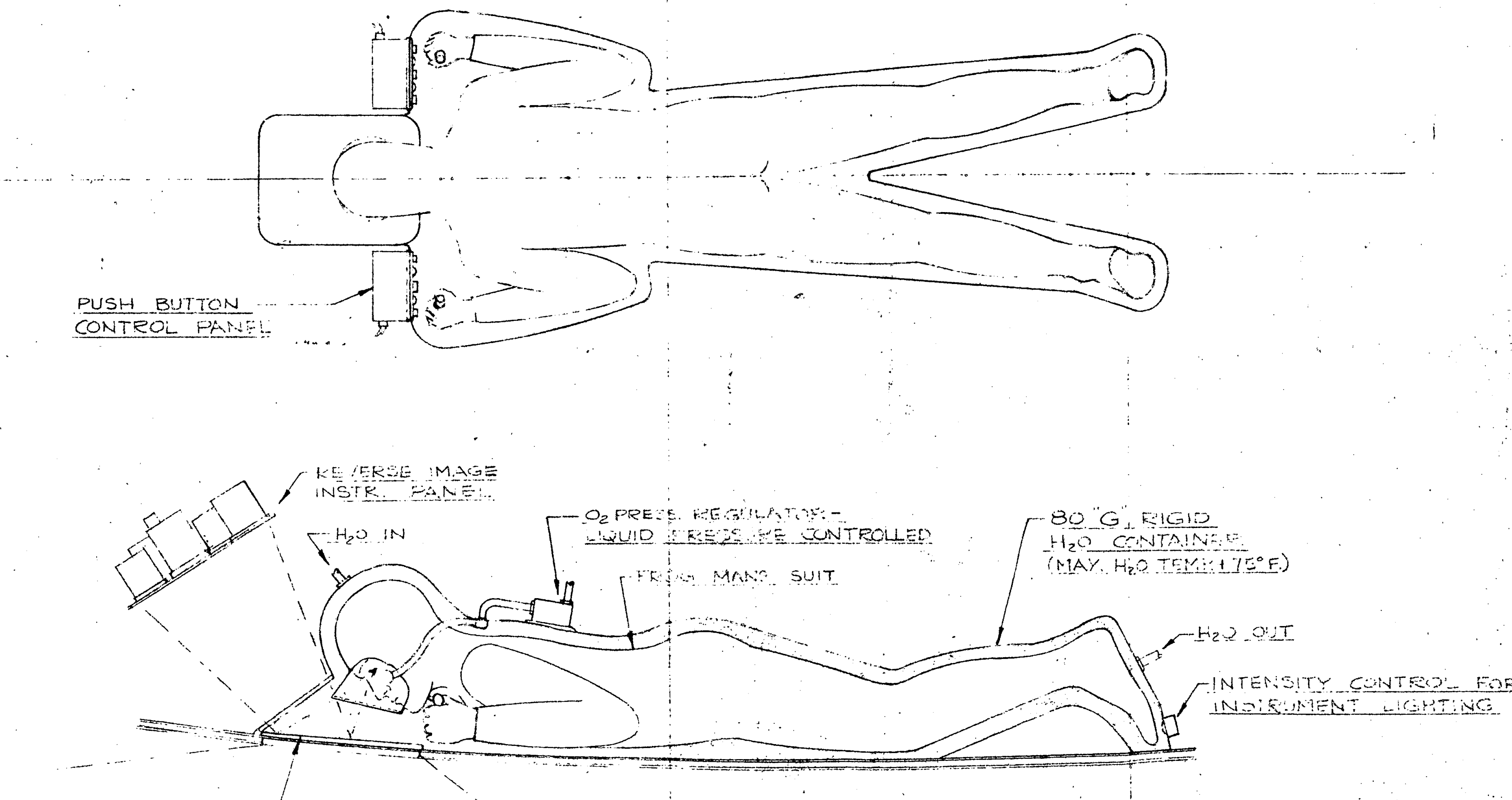

FIG. 35

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PRONE PILOT
WATER SUSPENSION SYSTEM

E. KOP. 7-11-58

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remaining intact up to 30 or 40 G's. The object is to insure a low density of the inhabited vehicle even in the event of aircraft failure and subsequent break-up. Terminal velocity of this low density section would be less than 40 ft. per sec. indicated.

This permits an entirely new concept in pilot survival. A man suspended in water in a rigid container can withstand extremely high accelerations. Accelerations are manifested as pressures if similar densities prevail. Since the human body is not of constant density (the lung cavity is a veritable bubble in the system), some human acceleration limitation exists even when suspended in water. The limit appears to be in excess of 50 G's. Escape from the aircraft is not necessary if impact with the ground or water can be held below about 60 G's. The low density structure described above can absorb the impact energy of the falling body by crumpling and/or imbedding at ground contact. An energy absorbing stroke of 5 ft. will permit survival.

The water suspended pilot is clothed in a "frog-mans" suit and breathes as diagrammed by Figure 36. All controls are accessable from inside the rigid pilot's compartment. Entrance to the compartment is via a full length lower surface door.

The cockpit environment will be controlled to temperatures, pressures, and humidity as in present day fighters. Use of a partial pressure suit and helmet and a standard ventilating suit is recommended.

Vision from the various cockpits shown has been studied and is presented as direct vision and/or reflected vision. It is considered that periscope vision and television type vision are unsatisfactory for use in the subject aircraft. The normal seated cockpit version (Figure 32) is limited in down vision and is somewhat inferior to present fighter type vision due to the nose high flight attitude. Vision is ideal from the prone position (Figure 33) with the exception of upward side vision. The proximity of the pilots head to the window permits a very large viewing angle.

Instruments would be installed normally for the seated type cockpit and would be special for the prone type cockpit. This special instrument installation is composed of reverse type instruments mounted such that reflection from the window permits normal viewing to the pilot. The section of the window appearing as the instrument panel would be partially silvered for improved reflection of the high intensity line lit instruments. Intensity of instrument lighting is controllable.

Figure 37 is presented to show the escape capabilities as compared to the aircraft performance limits. Normal cruise for the aircraft is at a "q"

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of about 35 P.S.F. and is considerably below the aircraft capabilities. At flight conditions approaching the heat limit, escape is also feasible. High temperatures encountered (600°F stagnation) during such an escape would prevail initially and would quickly decrease as the man is slowed by the stabilizing chute. The pilot's flight clothing is standard and will withstand the short time heating.

For all configurations in this report the normal seated position of the pilot according to Figure 32 is recommended. The pilot has escape provisions through the bottom by means of a trap door and parachute.

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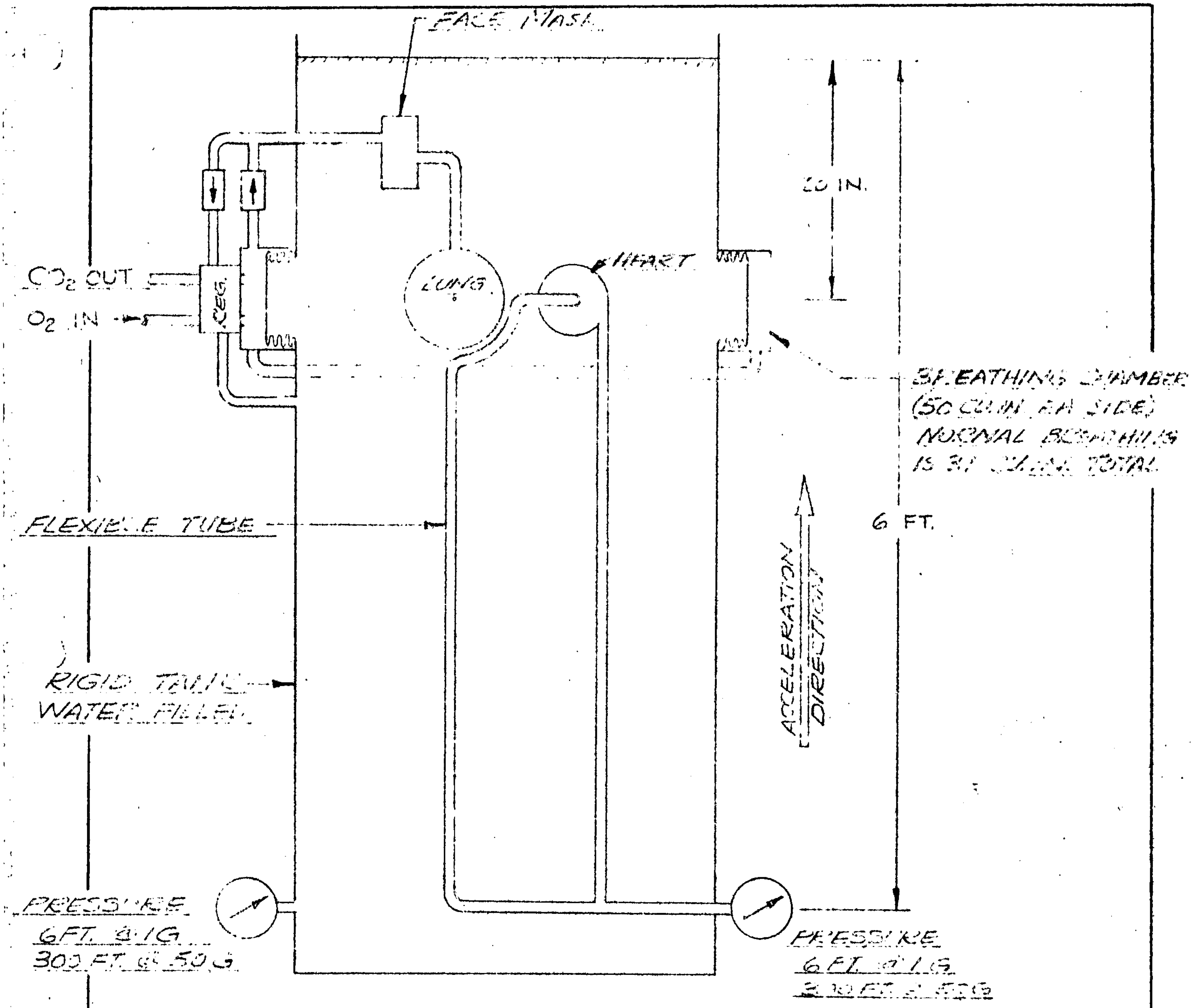

NOTES:

1. HEART PUMPING REQUIREMENT IS THAT REQUIRED TO CIRCULATE THE LIQUID IN THE TUBES ONLY (FRICTION LOSSES).
2. BLOOD HAS A DENSITY OF 1.055, THEREFORE TO SIMULATE THE BLOOD PRESSURES OF MAN — A H_2O SUSPENDED MAN WOULD REQUIRE

$$\frac{1}{1.055-1.00} = \frac{1}{.055} = 18 \text{ G's}$$

"BLOOD-POOLING" OF A H_2O SUSPENDED MAN @ 18 G's CORRESPONDS TO A MAN UNSUPPORTED @ 1.0 G.

SYSTEMS SCHEMATIC FOR WATER SUSPENSION SYSTEM

FIG. 36

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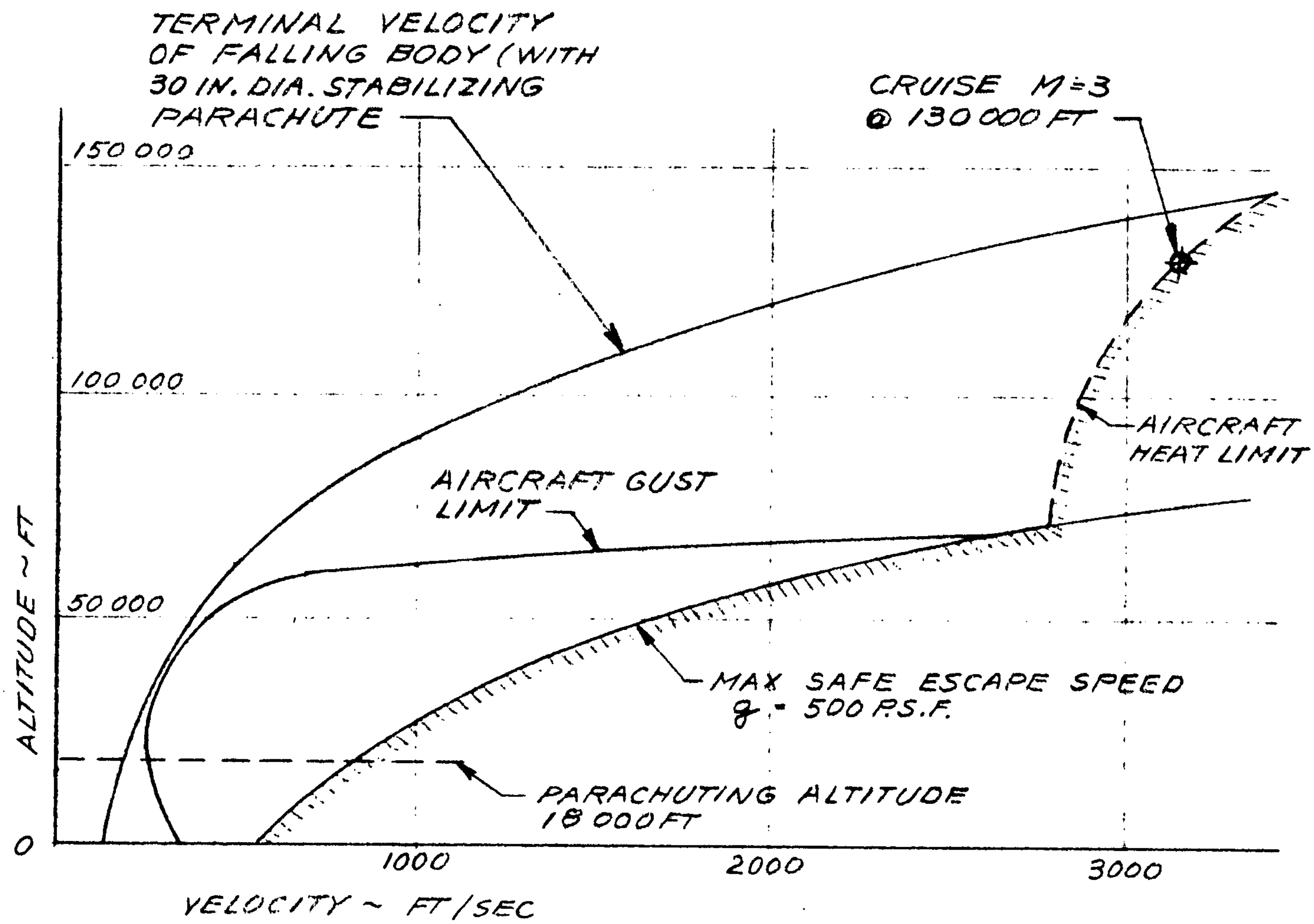

ESCAPE POTENTIAL

FIG. 37

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SECTION IV

STRUCTURAL CONSIDERATIONS

A. Structural Design Criteria

The vehicles discussed in this report are all predicated on certain ground rules. These have channeled the structural design concepts into a small area which demands somewhat radical methods. A summary of the rules which affect structural design are as follows:

1. Mission - Reconnaissance with minimum maneuvering requirements.
2. High Altitude Cruise - Light wing loading.
3. Staged Launch - The vehicle should not be penalized appreciably by the launch conditions.
4. Water Landing - Minimum landing requirements.
5. Reasonable vehicle Life

Item 1, demanding small maneuvering load factors combined with item 3 which would restrict launching methods, allows first the consideration of a minimum load factor at start of cruise. This load factor was chosen as 1.5 limit. As the weight of the vehicle decreases due to fuel burn-off, reasonable turns can be made during cruise. As the second design point, consideration was given to gust loads during the glide after the end of cruise. Rudimentary checks indicated that a 3G limit gust or maneuver factor at glide weight would be adequate for glide path variation or landing area control. These two conditions were then chosen as the primary design condition, with the assumption that all other requirements would be restricted within this capability.

Item 2 demanded very efficient, relatively lightly loaded structures with aerodynamically efficient shape. These combined requirements then make the non-rigid inflated airframe look quite attractive, and, therefore, has been given primary consideration.

In order to utilize the potential of the non-rigid construction more efficiently the use of a lower factor of safety than that used for rigid metal manned aircraft is proposed. The proposal is outlined on pages 81 and 82 and has been incorporated in the strength and weight estimations.

Item 3 has proven to be a more difficult problem than originally anticipated. From the structural design standpoint, the only practical

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methods of accomplishing this requirement are:

a. Balloon launch from ground to above 70 to 80,000 feet, then rocket boost to start of cruise.
b. Launch by modified existing aircraft with airload protection for the vehicle, to 45 to 50,000 feet, then a zero G trajectory launch to start of cruise. Aircraft vehicle separation should occur at relatively low airspeeds. (q = 50 psf)
c. Launch by an air breathing booster, possibly by towing at low speeds or with airload protection at high speed to above 70,000 ft. and M = 2.0, then minimum rocket boost or ramjet climb to start of cruise.

Method a. perhaps practical for testing, has been assumed impractical for tactical application and has therefore not been thoroughly investigated.

Method b. has been evaluated for a modified B-36. The vehicle is assumed to be supported with a structural platform to which the vehicle is attached by vacuum. This would transfer all vehicle airloads directly from the top skin through the substructure to the platform. This allows the vehicle to be carried at speeds up to the q allowable for the mother vehicle.

Method c. has been investigated; however, indications are that the air breathing booster aircraft will be a radical and complex vehicle in itself and should be the subject for a separate study.

Due to low wing loading and therefore, relatively slow landing speed, a water ski system controls landing loads within the previously determined capability.

Item 5, the design life of the vehicle is important from two standpoints, namely, fatigue of the structure and thermal degradation of the airframe materials. An estimated life has been chosen on a thermal degradation basis, with fatigue assumed to be of secondary importance. In detail design, however, it has to be considered.

Thermally, two temperature time limits are involved. The first, based on the cruise design point, assumes a vehicle life of 1,000 hours (667 flights at 1-1/2 hours per flight) at the cruise speed and altitude. Secondly, to

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allow for launch considerations and momentary overshoots or operation off the cruise design point, a short time exposure is considered at a higher temperature, based on 100 hours total or .15 hours (9 min.) per flight.

The temperature increase is assumed as that occuring with a 40% increase in the structural design temperature (10 feet aft of leading edge.).

Elevated temperature structural data on impregnated fabrics is very inadequate at this time and the cumulative effect of many exposures on strength after exposure on both impregnated fabric and rigid plastic lamination are scarce for the newer developed materials. It is felt, therefore, that in the highest temperature regions i.e., nose and leading edge, some components will have to be made replacable and shorter lives at these highest temperatures be accepted for these components.

For the basic airframe the assumption is made, however, that the 1000 hours at design cruise point plus the 100 hours over design cruise point constitute the thermal life criteria for the vehicles.

Table V summarizes the structural design criteria for the vehicle. Figure 38 illustrates a typical launch, Figure 39 shows a typical glide envelope based on gust velocities and formula as shown on page 76.

Figure 40 illustrates typical temperature distribution and thermal design envelopes.

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CONVAIR

A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

Date 10-23-58
Prepared By J.C.E
Checked By
Revised Date

Model TABLE V

Page Temp Perm

Report No. ZP-253
Date: October 1958

STRUCTURAL DESIGN CRITERIA

MC VEHICLES

①	②	③	④	⑤	⑥	⑦	⑧	⑨	⑩	⑪	⑫
CONDITION		WEIGHT COND.	V [B] KNOTS	H [B] 10^3 FT	q [B] PSF	STRUCT. TEMP [B] 10 FT. L.E. DEG. F	LOADING APPLICATION	REACTION	N_z LIMIT G	N_y LIMIT G	N_x LIMIT G
I HANDLING											
A TRANSPORT		EMPTY GLIDE	—	—	—	+70 −65	VEHICLE INERTIA	SUPPORT JIG	+ 2.0	± 1.0	± 1.0
B HOIST – JACK		EMPTY GLIDE	—	—	—	+70 −65	"	HOIST FITTINGS	+ 2.0	± 1.0	± .5
II LAUNCH											
A PRIMARY A/C		SECONDARY LAUNCH	IT HAS BEEN ASSUMED THAT THE LAUNCH WILL BE ENGINEERED SUCH THAT THE VEHICLE CAPABILITIES OF [A] WILL NOT BE EXCEEDED.								
B SECONDARY START		SECONDARY LAUNCH	280	45	52	250	GUST AIRLOAD	VEHICLE INERTIA	+ .97*	* ASSUMES ZERO G TRAJECTORY	
C SECONDARY END		PRE-STAGE	1865	125	52	305	THRUST	VEHICLE INERTIA	+ .50	+ .50	− 2.0
III CRUISE											
A BEGIN, TURN [A]		START CRUISE	1865	125	52	305	AIRLOAD	VEHICLE INERTIA	+ 1.5	± .25	—
B MID, TURN		MID CRUISE	1880	132	41	"	"	"	+ 1.95	± .25	—
C END, TURN		GLIDE	1920	140	28	"	"	"	+ $3.0K_t$	$\pm .50K_t$	—
									K_t = STRENGTH LOSS AT TEMP.		
IV GLIDE											
A MANEUVER [A]		GLIDE	1680	85	275		AIRLOAD	VEHICLE INERTIA	+ 3.0 − 1.5	± .50	—
B GUST [A]		GLIDE	150	35	28		GUST AIRLOAD	"	"	± .50	—
V LANDING											
A MAX. SINK		GLIDE					SKI WATERLOAD	VEHICLE INERTIA			
B WORST ATTITUDE		GLIDE					"	"			
VI RECOVERY		EMPTY GLIDE	ASSUME SAME AS HANDLING								

[A] BASIC STRUCTURAL DESIGN CONDITIONS [B] TYPICAL VALUES FOR REF.

Form 1-???

GUST LOAD CRITERIA

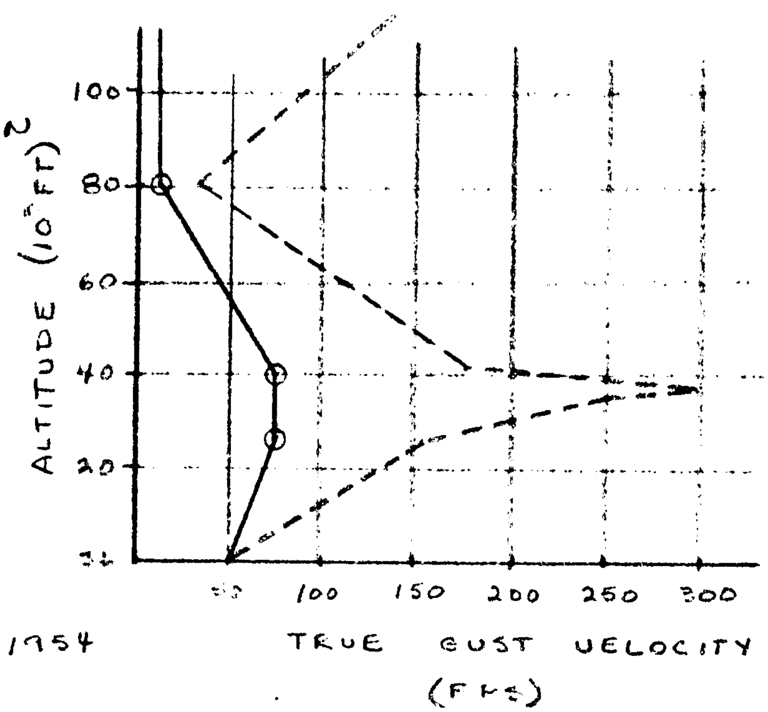
GUST VELOCITIES

ASSUME AS SHOWN

○—○ GUST VELOCITY

S.L. – 25000 — $50/\sqrt{\sigma}$ FPS
25000 – 40000 — 75 FPS
40000 – 80000 — 75 – 15.5 FPS
> 80000 — 15.5 CONST.

– – – – 1% PROBABILITY WIND PROFILE SISSENWINE ARDC 1954



GUST LOAD FORMULA

$$\text{AIRLOAD} = \frac{m K U S \rho V}{2}$$

REF: MIL-A-8629

$$\text{LOAD FACTOR} = 1 \pm \frac{m K U \rho V}{2\, W/S}$$

WHERE:

m = SLOPE OF LIFT CURVE $\approx \dfrac{2\pi AR}{2 + AR}$ (RADIANS) $\quad AR = \dfrac{b^2}{S}$

K = ALLEVIATION FACTOR $= \dfrac{.88\,\mu}{5.3 + \mu}$

$\mu = \dfrac{2\, W/S}{g\, c\, m\, \rho}$

W = VEHICLE WGT #
S = WING AREA FT^2
g = GRAVITY 32.2
c = MEAN AERO CHORD (FT)
ρ = DENSITY

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CRITERIA

GUST LOAD FORMULA (CONT)

U = TRUE GUST VELOCITY (FPS)
V = VEHICLE TRUE AIRSPEED (FPS)

FOR OPERATIONAL ENVELOPES, THE GUST FORMULAE ARE REARRANGED AS FOLLOWS

$$V_{MAX} = \frac{2\left(\frac{W N_z}{K_t} - \text{STEADY STATE AIRLOAD}\right)}{m K U S \rho}$$

OR

$$V_{MAX} = N_z - N_{\substack{\text{STEADY} \\ \text{STATE}}} \left(\frac{2\, W/S}{m K U \rho} \right)$$

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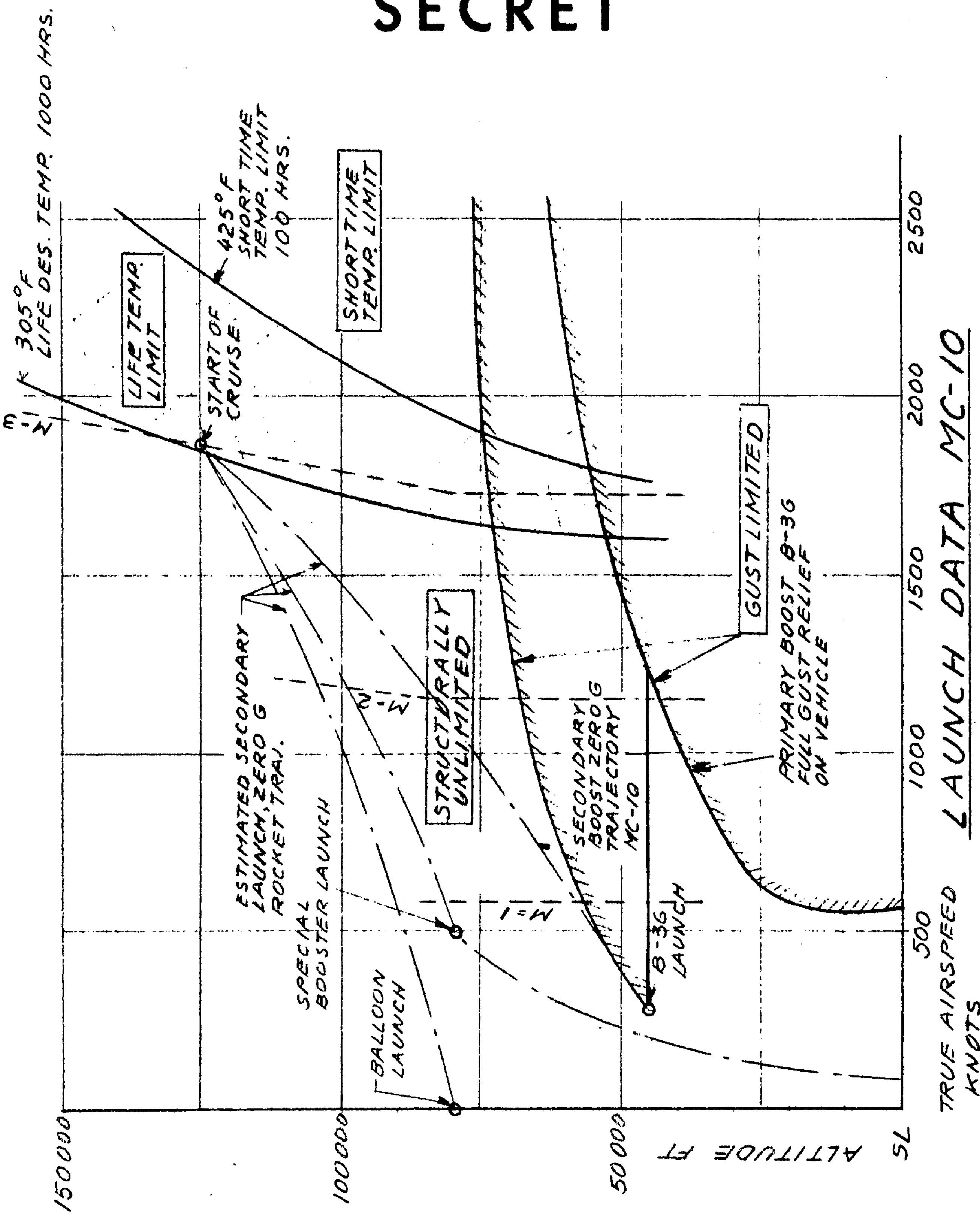

LAUNCH DATA MC-10
FIG. 38

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FORM 1812-A-1

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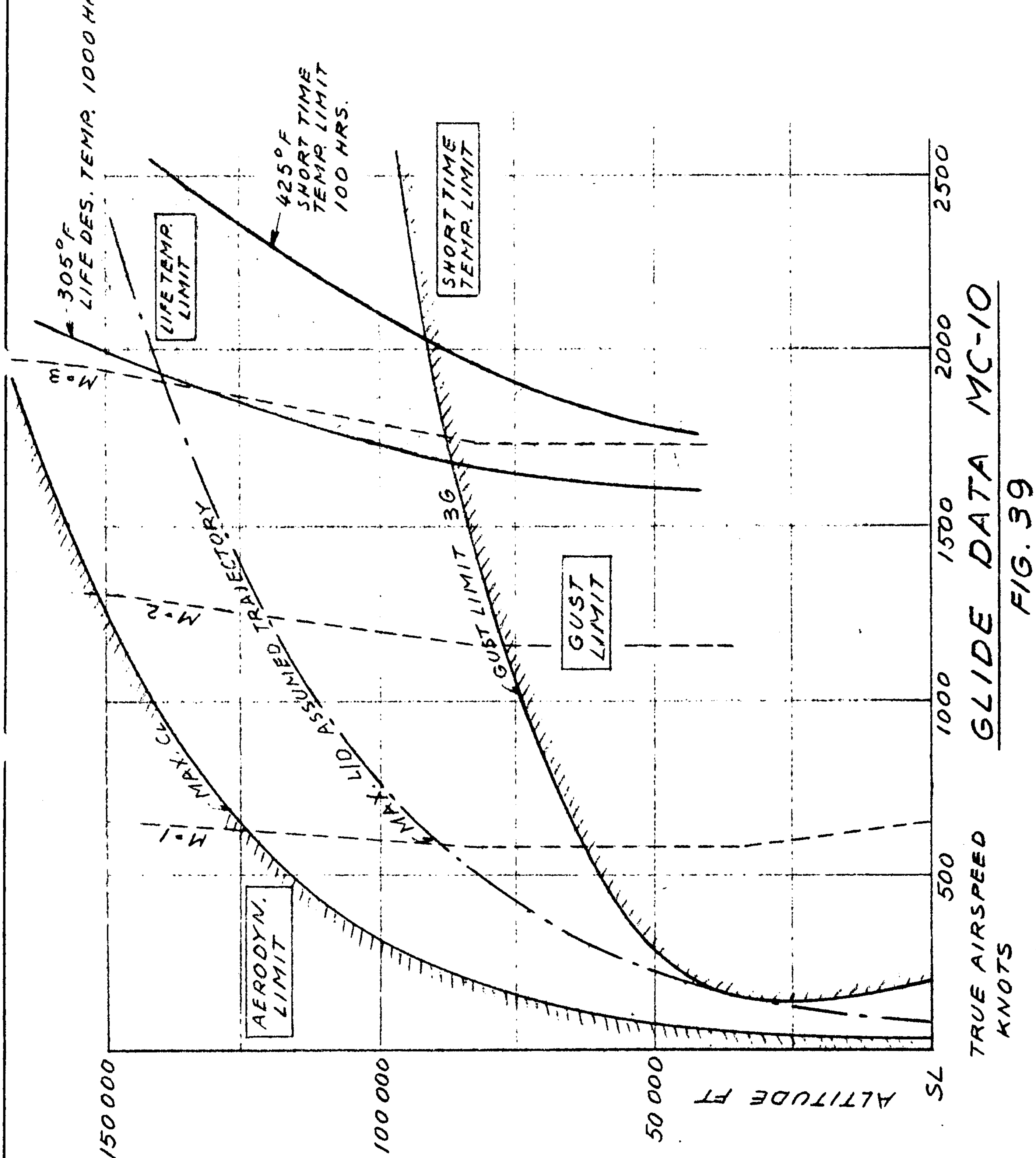

GLIDE DATA MC-10
FIG. 39

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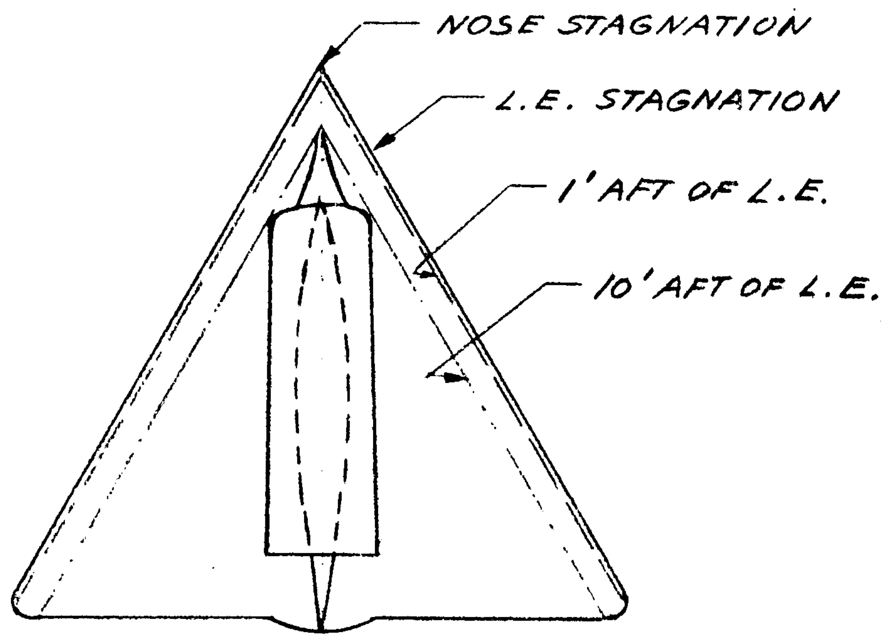
	LIFE DESIGN			SHORT TIME LIMIT		
	SPEED	ALTITUDE	TEMPERATURE	SPEED[1]	ALTI-TUDE[1]	TEMPERATURE
NOSE STAGNATION	M = 3	135 000 FT	725°F	M = 3.9	135 000 FT	815°F
L.E. STAGNATION	|	|	630	|	|	685
1' AFT OF L.E.	|	|	400	|	|	550
10' AFT OF L.E. UPPER S.	|	|	305	|	|	425
LOWER S.	|	|	291	|	|	—

[1] CHOSEN AS EXAMPLE



TYPICAL TEMPERATURE DISTRIBUTION

CONFIG. MC-10; $\alpha = 10°$; $\varepsilon = .8$; DAYLIGHT

FIG. 40

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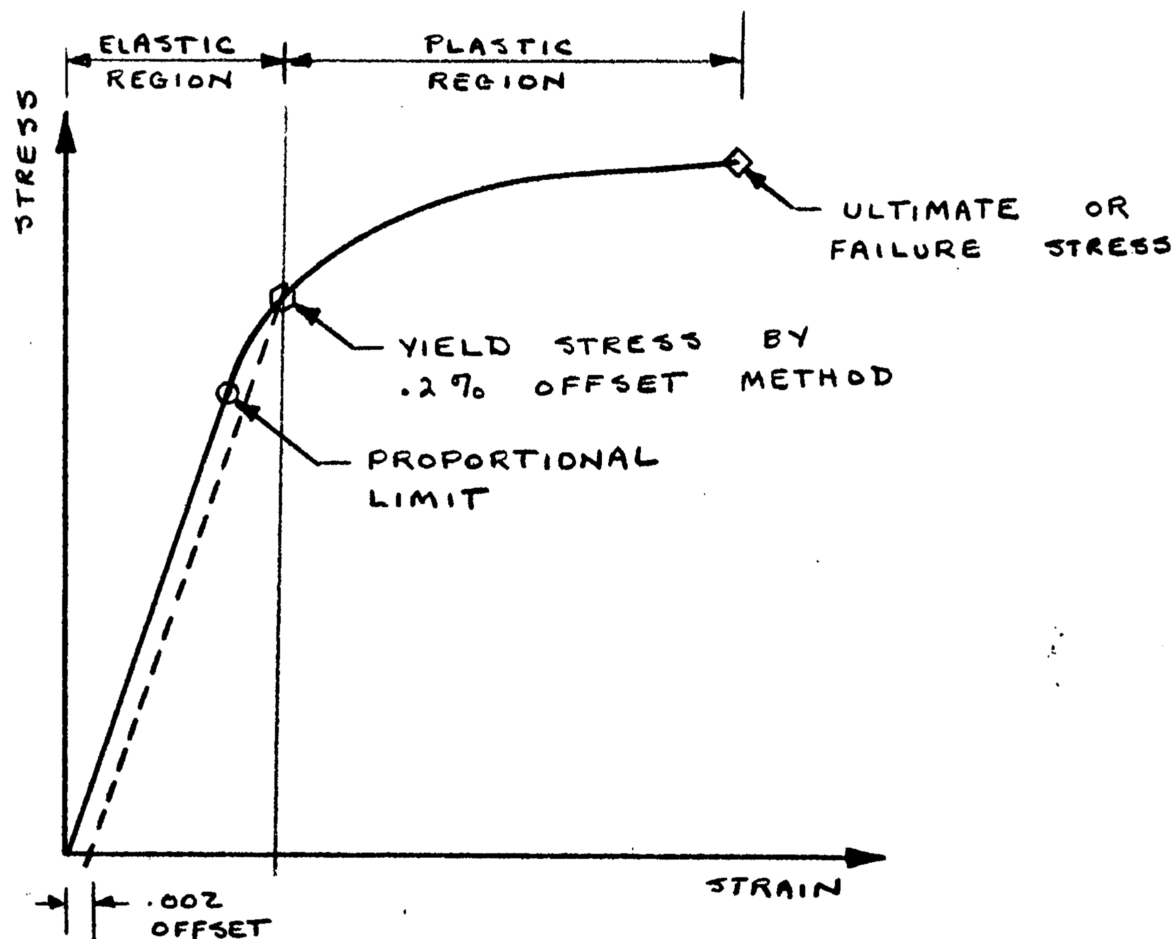
FORM 1812-A-1

STRUCTURAL DESIGN PROPOSAL

PRESENT CRITERIA RIGID METAL AIRCRAFT:

Factors of Safety - Vehicle shall be capable of sustaining limit load (maximum expected load) multiplied by a yield factor of safety of 1.15 without excessive permanent deformation, and ultimate load which is limit load multiplied by an ultimate factor of safety of 1.50 without failure.
Ref: MIL-A-8629 "Airplane Strength & Rigidity"

Note: The primary basis for this dual criteria is the fact that most common aircraft metals have a yield stress which is considerably below the failure or ultimate strength, which allows large permanent deformation below ultimate load, as shown below:



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TYPICAL DUCTILE METAL STRESS - STRAIN CURVE

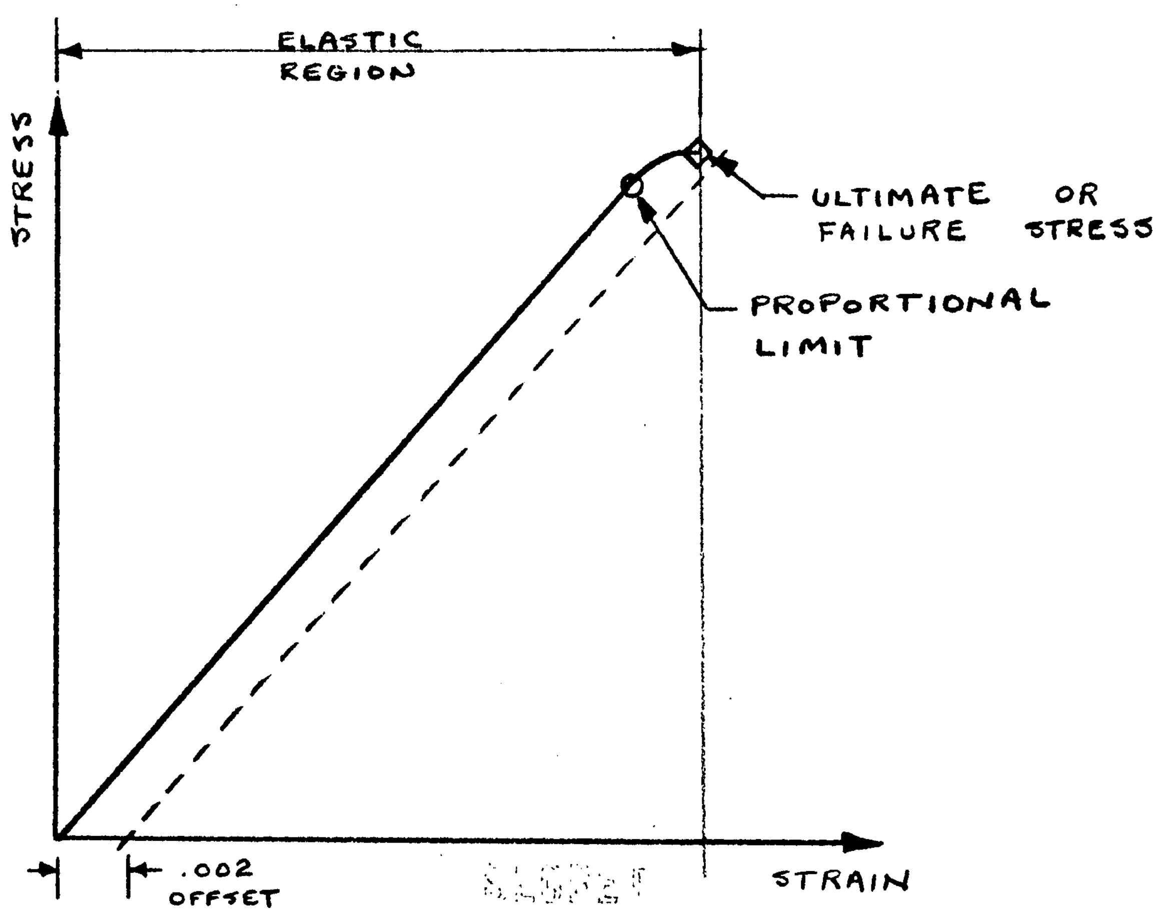
STRUCTURAL DESIGN PROPOSAL

PROPOSED CRITERIA NON RIGID PRESSURIZED PLASTIC A/C:

Factor of Safety - Vehicle shall be capable of sustaining limit load without excessive permanent deformation and ultimate load, which is limit load multiplied by a factor of safety of 1.15, without failure, provided the following conditions are met:

1. The structure is relief valve pressurized to limit load and will deflect freely with loads greater than limit load, without adversely affecting the aerodynamic characteristics, mechanical operation of any part or strength at or below limit load, by virtue of the non-rigid pressure stabilized construction.

2. Material properties are such that a yield stress, as ordinarily defined, does not exist, as shown below, and that the material and fabrication specification are adequate to insure less than 15% deviation.



TYPICAL STRUCTURAL PLASTIC STRESS-STRAIN CURVE

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B. Structural Configurations

The primary configuration studied is the non-rigid pressure stabilized wing with minimum body structure, configuration MC-10, Figure 16. The secondary configuration is the rigid vehicle, which is identical geometrically to MC-10, namely MC-19 Figure 18. Engine construction is not discussed.

For MC-10, the wing is assumed to be a full cantilever, pressure stabilized, truss rigged fabric structure. The wing center section, body-pylon combination is assumed to be of rigid fiberglass laminate construction. The bending loads are carried directly by the pre-tensioned skins, the shears by the pretensioned trusses formed by the angled tension ties to give in effect a multi web full effective skin wing structure. Some diagonal trussing would be required in a chordwise direction for redistribution of airload, with wing torsional stiffness provided by both the differential bending of the "multi-spars" and the total wing enclosed area as a torque box, utilizing the rigidity in shear of the diagonally doubled fabric in biaxial tension.

As discussed in the criteria section proposal for factor of safety, the non-rigid structure, if properly designed, can have the ability to limit developed airloads by wing bending deflections due to exceeding of wing pressurization with a resultant upper skin folding, without loss of design load capability.

The rigid vehicle (Configuration MC-19, Figure 18) is envisioned as also a full contilever, multi spar wing with minimum body structure. The wing covering was assumed as honeycomb or corrugated core sandwich panels of sufficient thickness to yield reasonable buckling allowables. Corrugated webs will provide stabilized shear paths with good extensional deflection characteristics to reduce thermal stresses from rapid heat applications.

The fins would be similar to the wing structure, with proper detail arrangements at the fin to wing junction for either configuration.

Static aeroelastic problems have been assumed as controllable due to the all tension design, delta wing planform and low wing loading. However, cognizance should be given to the fact that most fabrics are very flexible until pre-stressed, and can be quite flexible in comparison to conventional aluminum structures even when pressurized. As discussed in the materials section, this is an area which needs further investigation for proper evaluation The rigid design would be conventionally analyzed for its aeroelastic stiffness requirements.

Thermoelastic problems are reduced by the non-rigid pressure stabilized construction which utilizes thin skins, minimum substructure and excellent

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extensional differential capability for unequal surface heat inputs. The rigid construction utilizes corrogated webs in the substructure to reduce thermal stress.

Dynamic aeroelastic problems or flutter difficulties are reduced with the pre-stressed fabric design. The large hysteresis of fabrics coupled with the ability of the pressurized structure to deflect freely with greater than pressurization loads will not allow stored energy for divergance to destruction as in a rigid metal structure. Panel flutter or "Flag waving" phenomena should be precluded by the tensioning of the skins due to pressurization.

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C. Materials and Fabrication

Material and fabrication considerations are important parameters of an airframe structural design. These qualities are even more important because of the requirements for a high efficiency non metallic structure which has to operate at elevated temperatures for relatively long periods of time. Adequate data on the mechanical properties of rigid plastic laminates at normal temperatures are available. At elevated temperatures, some questions are answered inadequately for detail design, but adequate for preliminary design purposes. Fabrication techniques likewise are well developed for normal temperatures, but less than adequate for the temperature limits of the rigid laminates themselves.

Structural fabric and fabrication data on suitable structural fabrics are lacking for normal temperature and particularly for elevated temperatures. This is due to the relatively small use made of fabric structures rather than any inherent mechanical difficulty in obtaining the necessary data.

For the rigid portions of the vehicles or all-rigid vehicles, epoxy, phenolic or expoy-phenolic resin and fiberglass laminates are assumed to be the most efficient materials. Various and complex shapes, corrugations and sandwich panels can be fabricated from these materials to give good strength and weight properties. Table IX summarizes material properties assumed. These data were taken from the typical data presented in pages 88 to 96.

The non-rigid structures present many development problems which have not been adequately solved to date. A survey of available materials indicates than an impregnated fiberglass offers the best strength and weight characteristics for the Hazel criteria. Pages 88 to 96 substantiate an approximation of typical fiberglass strength and weight as shown in Table VI.

It is assumed that 1) the impregnate contributes nothing structurally, 2) the design uses only tension and shear strength (thru biaxially stressing diagonally doubled fabrics), and 3) the tension properties of the fabric vary similiarly to that of the rigid resin impregnated fabrics at elevated temperatures.

Fabrication of the fabrics has been assumed to be either by sewing or glueing or both at splices with efficiencies as assumed on page 93. The shape holder ties which are angled to serve as trusses, are considered woven through the fabrics, by hand or machine, such that adequate tie strength is realized and proper sealing maintained. The impregnate was considered as either applied prior to, after or both before and after fabrication.

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Coatings or impregnates for adequate sealing to minimize the structural pressurization gas leakage may also require some development or modification. Typical data indicate that Silicone, Acrylic or Flouro rubber compounds have adequate life at the design temperatures for all except perhaps the leading edge stagnation point where insulation or other protection should be applied.

The deflection characteristics of fabric structures, particularly of the delta wing configuration with airloads, is virtually unknown. Before suitable approximation can be attempted, basic deflection data must be obtained on fabrics as discussed above, with model testing to substantiate the data. Since lack of material study funds has precluded tests by Convair, it has been assumed that the fabrics will offer sufficient rigidity when properly used.

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TABLE VI

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MATERIALS DATA SUMMARY

① COMPONENT	③④⑤ CONSTRUCTION	⑥ TEMP DEG. F	⑦ TIME Exp. @ Temp HRS	⑦ TIME Life Total HRS	⑧ F_{tu} AT TEMP. KSI	⑨ F_{tu} ROOM TEMP AFTER KSI	⑩ F_c KSI	⑪ F_{su} KSI	⑫ DENSITY #/IN³
NON - RIGID VEHICLES									
BODY NOSE	REPLACABLE, RIGID	725	1.5	10	9.6	15.9	σ_{cr}	τ_{cr}	.065
	PLASTIC LAMINATE	815	.15	1	9.6	15.9			
BODY & PYLON	RIGID PLASTIC LAM.	305	1.5	1000	23.0	31.9	σ_{cr}	τ_{cr}	.065
	SANDWICH & SHAPES	425	.15	100	23.0	31.9			
LEAD. EDGES	REPLACABLE, NON RIGID	630	1.5	100	6.2	14.2	0	$\sqrt{f_x f_y}$	.065
	DIAGONAL EMBEDDED FABRIC	685	.15	10	8.2	14.2			
WING & FIN	NON RIGID, IMPREGNATED	305	1.5	1000	14.1	19.5	0	$\sqrt{f_x f_y}$	.065
	DIAG. DOUBLED FABRIC	425	.15	100	14.1	19.5			
RIGID VEHICLES									
BODY NOSE	REPLACABLE, RIGID	725	1.5	10	9.6	15.9	σ_{cr}	τ_{cr}	.065
	PLASTIC LAMINATE	815	.15	1	9.6	15.9			
BODY & PYLON	RIGID FIBERGLASS	305	1.5	1000	23.0	31.9	σ_{cr}	τ_{cr}	.065
	LAMINATE & SANDWICH	425	.15	100	23.0	31.9			
LEAD. EDGES	REPLACABLE, RIGID	630	1.5	100	10.5	23.9	σ_{cr}	τ_{cr}	.065
	PLASTIC LAMINATE	685	.15	10	14.3	23.9			
WING & FIN	RIGID FIBERGLASS	305	1.5	1000	23.0	31.9	σ_{cr}	τ_{cr}	.065
SKINS	SANDWICH PANELS	425	.15	100	23.0	31.9			
SUBSTRUCTURE	RIGID FIBERGLASS	305	1.5	1000	23.0	31.9	σ_{cr}	τ_{cr}	.065
	CORRUGATED WEBS	425	.15	100	23.0	31.9			

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TABLE VII — MATERIALS

UNIMPREGNATED FIBERGLAS MECHANICAL PROPERTIES

ROOM TEMPERATURE

FABRIC STYLE REF.	THICKNESS REF. IN	WEIGHT REF. OZ/YD²	DENSITY WGT/THICK. #/IN³	BREAKING STRENGTH WARP REF #/IN	BREAKING STRENGTH FILL REF. #/IN	BREAKING STRESS WARP STR/THICK #/IN²	BREAKING STRESS FILL STR/THICK #/IN²
ECC 106	.0015	.85	.0274	46	52	30700	34700
ECC 125	.005	3.93	.0380	160	150	32000	30000
ECC 181	.0085	8.90	.0506	340	330	40000	38800

REF: OWENS CORNING FIBERGLAS CORP. "STANDARD CLOTH CONSTRUCTIONS"

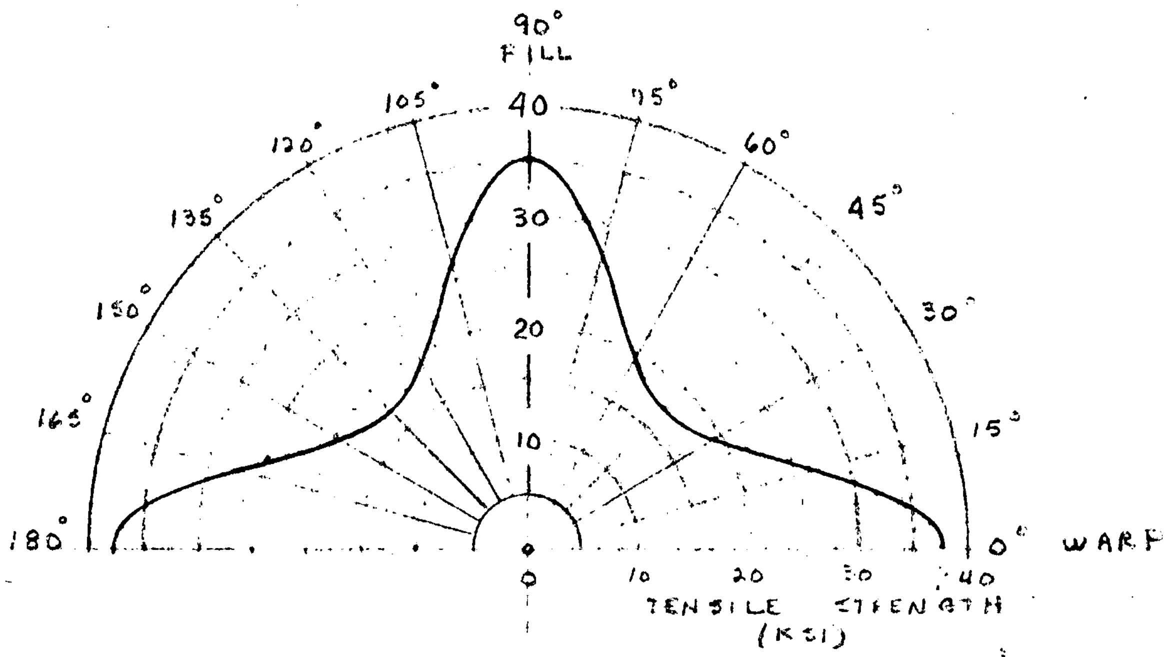

FIGURE 41

VARIATION OF TENSILE STRENGTH IN RELATION TO WEAVE *

REF: ANC-17 PLASTICS FOR AIRCRAFT PART I P66

* PARALLEL LAMINATED 181 FABRIC - POLYESTER RESIN .125 RIGID PANEL AT ROOM TEMP

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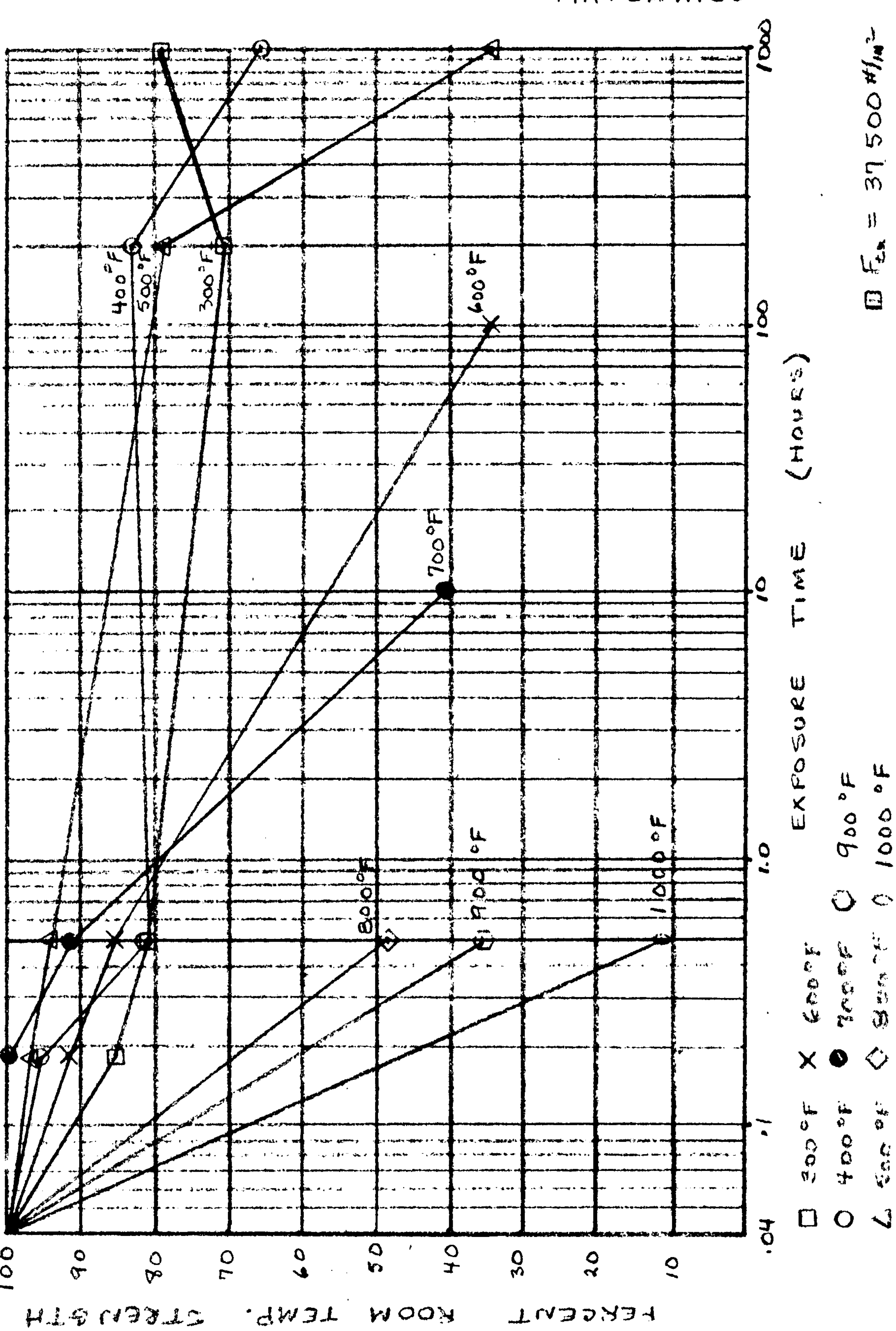
FIGURE 42

TENSILE STRENGTH VERSUS EXPOSURE TIME AT ELEVATED TEMPERATURE

EPOXY - PHENOLIC - ECC 181 GLASS CLOTH RIGID LAMINATE

REF: WADC TR 57-37



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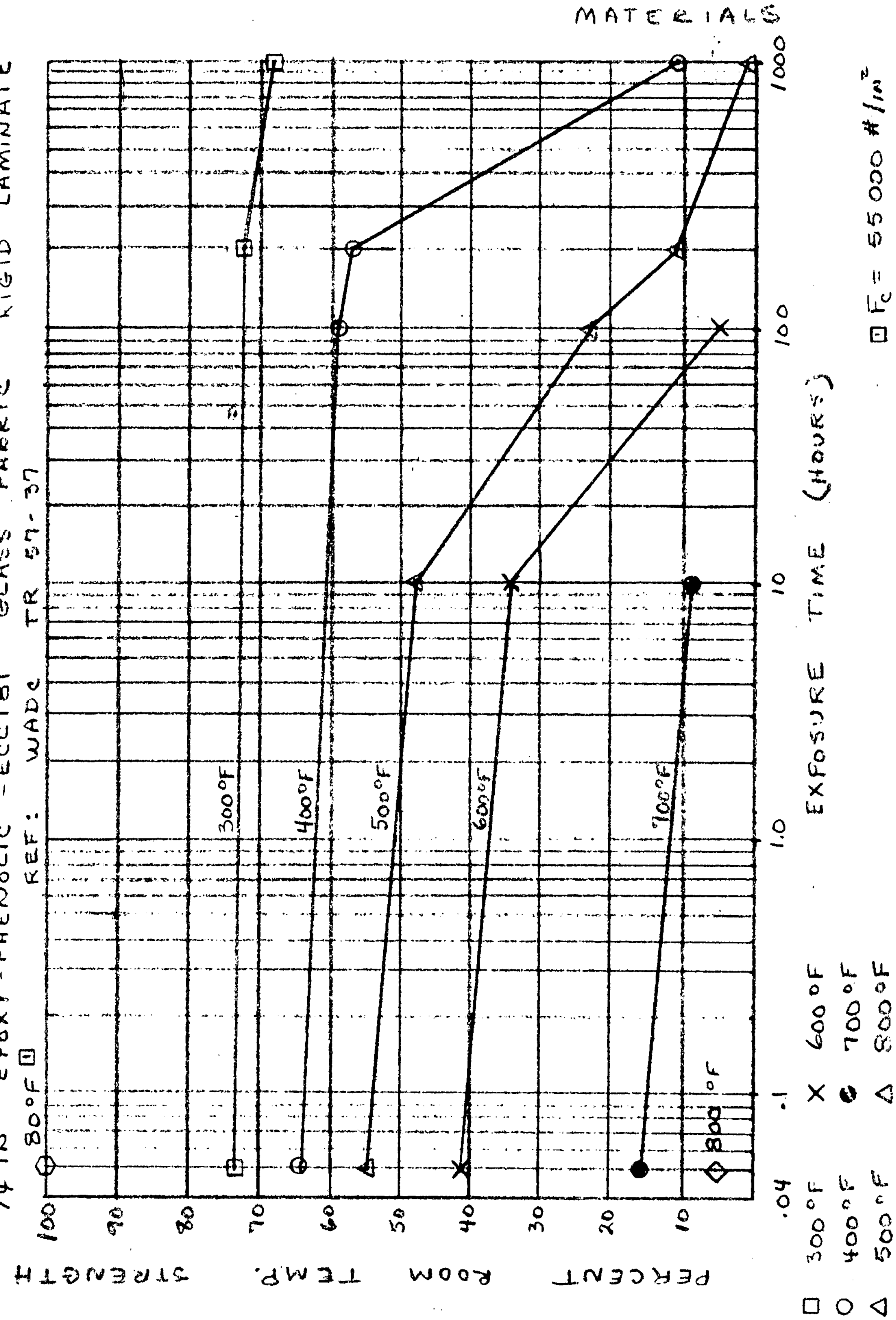

FIGURE 43
COMPRESSION STRENGTH VERSUS EXPOSURE TIME AT ELEVATED TEMPERATURE
1/4 IN EPOXY-PHENOLIC - ECC181 GLASS FABRIC RIGID LAMINATE
REF: WADC TR 57-37

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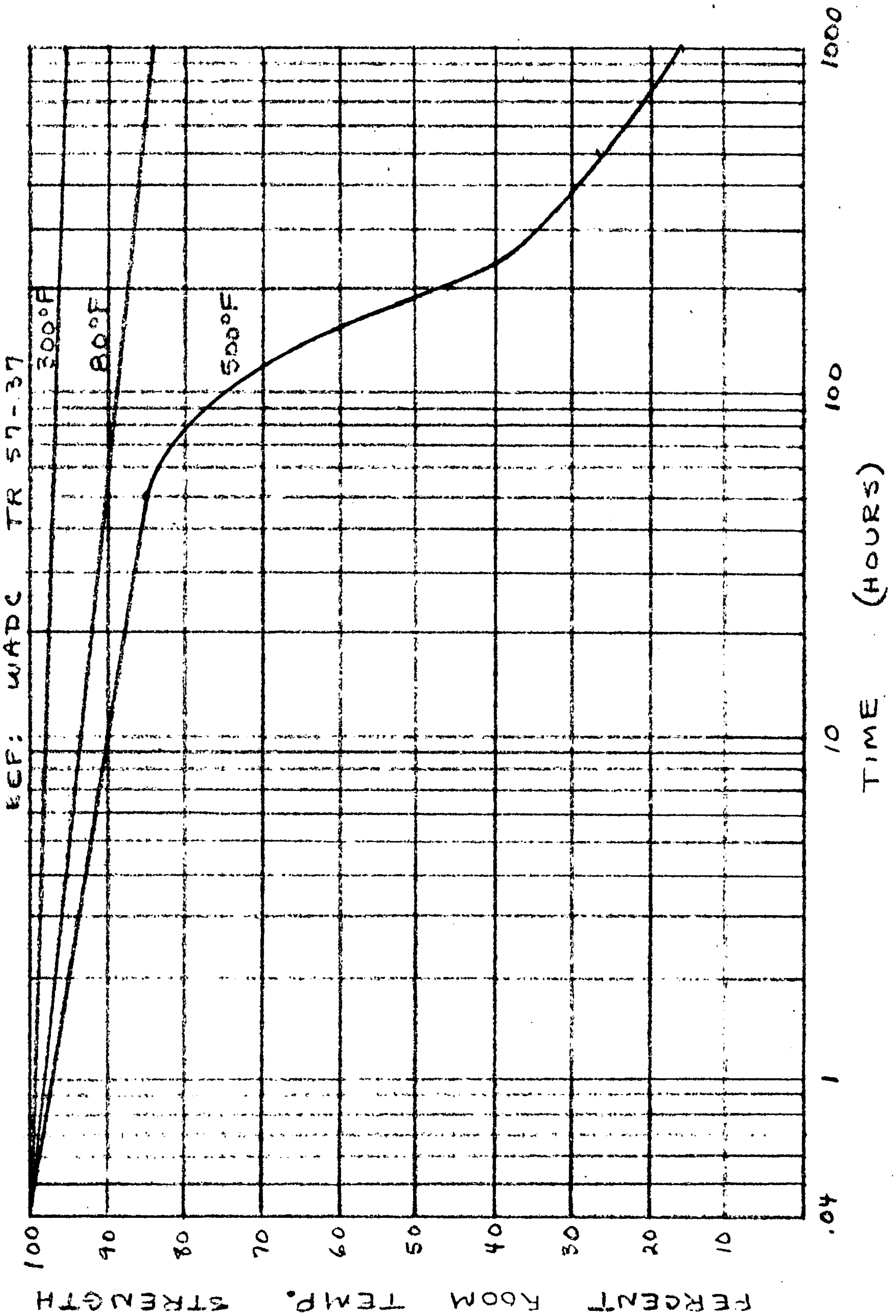
FIGURE 44

STRESS TO RUPTURE IN TIME INDICATED

1/8 IN. EPOXY - PHENOLIC - ECC 181 GLASS CLOTH RIGID LAMINATE

REF: WADC TR 57-37



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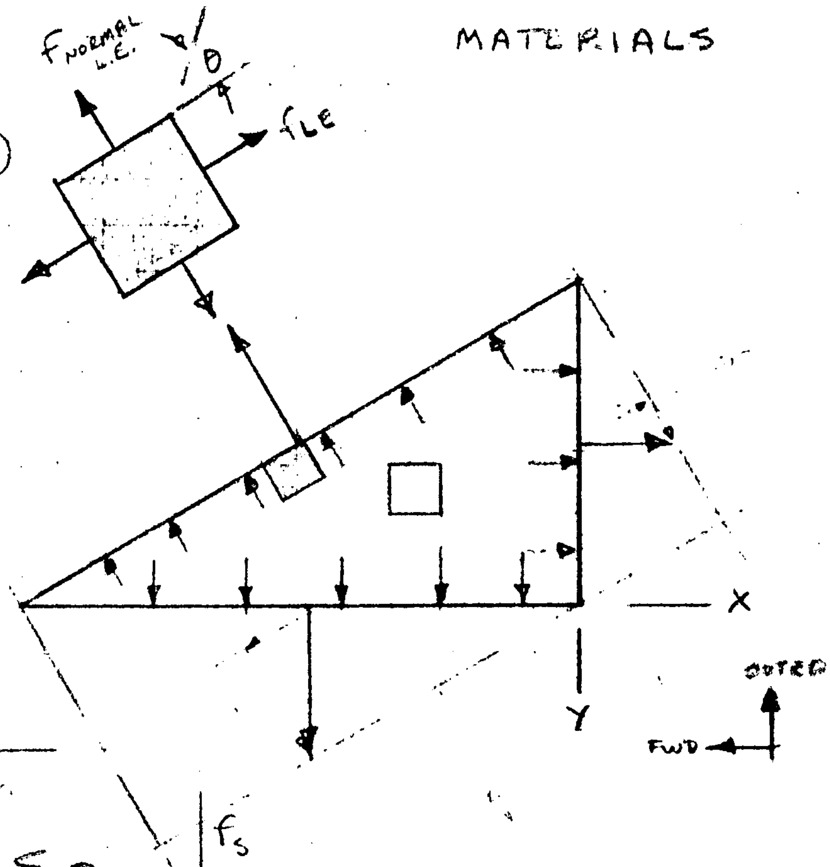
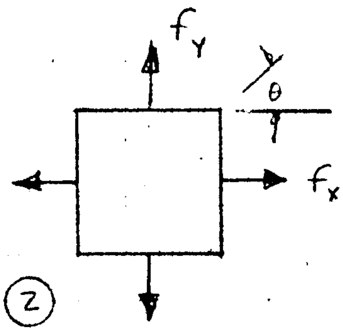
SKIN STRESSES
NON RIGID WING





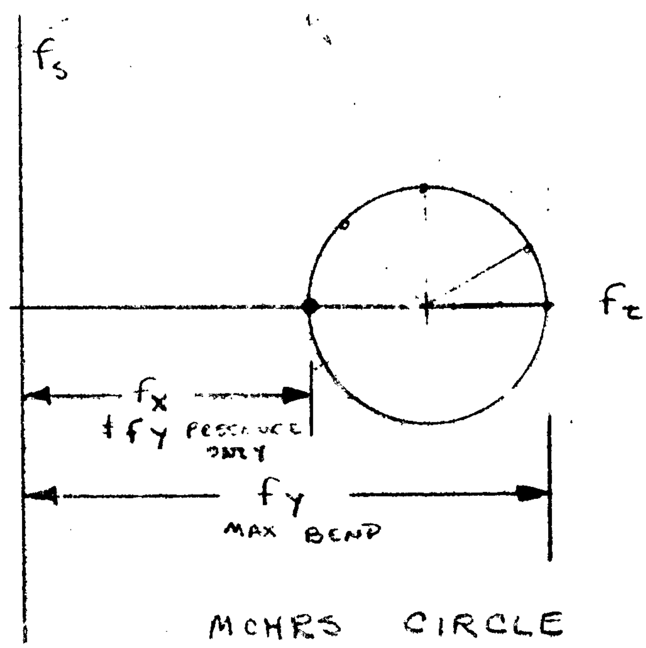
① PRESSURE ONLY

$$f_X = f_Y$$

$$f_{MAX} = f_x = f_Y \quad \text{FOR } \Sigma\theta$$

② FOR MAXIMUM BENDING PLUS PRESSURE

$$f_X = \frac{f_Y}{2}$$



$$f_{MAX} = f_Y = 2f_X \quad @ \quad \theta = 0$$

$$f_{MIN} = f_X = \frac{f_Y}{2} \quad @ \quad \theta = 180^\circ$$

$$f_{\theta=90^\circ} = 3/2\, f_X = 3/4\, f_Y$$

$$f_{\theta=45^\circ} = f_x + \frac{f_x}{2} + .707\,\frac{f_X}{2} = 1.853\, f_X$$

$$= .93\, f_Y$$

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NON RIGID DESIGN ALLOWABLES

FABRIC

FROM TABLE I

$$F_{tu_{AVE}} = \frac{30.7 + 20.0 + 38.3}{3} = 33.2 \text{ KSI IN FILL DIRECTION}$$

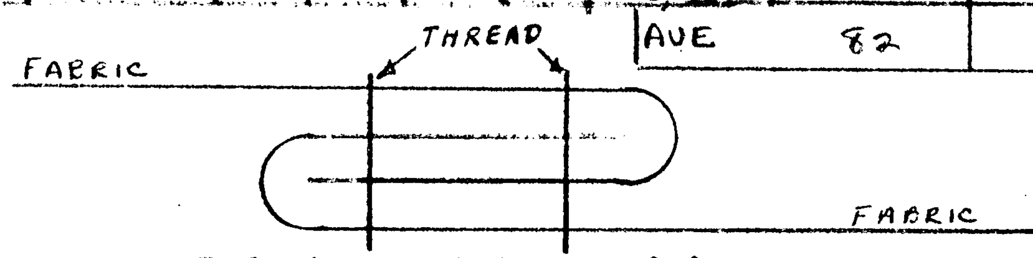
FOR 2 PLYS OF FABRIC, ONE PARALLEL TO MAXIMUM AXIAL LOAD & ONE DIAGONAL, WITH DIAGONAL STRESS EQUAL TO .73 MAXIMUM (SEE PRECEDING PAGE), FROM FIG. 41 ESTIMATE:

$$F_{tu_{R.T.}} = \frac{17.5}{36(.73)}(33.2) + 33.2 = 25.2 \text{ KSI}$$

FROM WADC TR 56-313 PART I "A STUDY OF PARACHUTE SEAM DESIGN CRITERIA" FOR AN LS_c SEAM, 301 STITCH, E THREAD THE FOLLOWING SEAM EFFICIENCIES WERE REALIZED.

TABLE VIII

FABRIC	MATERIAL	STITCH	THREAD	SEAM	STITCHES IN	EFF.	PROT. EFF.
MIL-C-7020 I	NYLON	301	E	LS_c 2	8, 11	86	83
" II						89	85
MIL-C-7350 I						79	71
" II						77	72
MIL-C-8021 I	NYLON	301	E	LS_c 2	8, 11	78	76
					AVE	82	77.5



TYPICAL LS_c SEAM

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FABRIC ALLOWABLE (CONT)

FROM TABLE XI ASSUME EFF = 77.5 %

$$F_{tu_{RT}} = 25.2 (.775) = 19.5 \text{ KSI}$$

CORD ALLOWABLE

ASSUME LOOP STRENGTH & SKIN TIE CONSIDERATIONS GIVE CORD STRENGTH SIMILIAR TO FABRIC

TEMPERATURE REDUCTIONS

CONDITION		TEMP °F	TIME HRS	K_t REF: FIG. 42,43,44	F_{tu} KSI
BEFORE EXPOSURE		80	—	1.00	19.5
AT TEMP	10 FT L.E	305	1000	.72	14.1
AFTER		80	—	1.00	19.5
		425	100	.72	14.1
		80	—	1.00	19.5
	1 FT L.E.	400	1000	.66	12.8
		80	—	1.00	19.5
		550	100	.70	13.6
		80	—	1.00	19.5
	L. EDGE STAGNATION	630	100	.32	6.2
		80	—	.75	14.6
AT TEMP.		685	10	.42	8.2
AFTER		80	—	.75	14.6

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RIGID DESIGN ALLOWABLES

FROM FIG. 42 & 43

$F_{tu_{RT}} = 37.5$ KSI
$F_{c_{RT}} = 55.0$ KSI

} TYPICAL RIGID FIBERGLASS LAMINATE PROPERTIES

ASSUME BONDED JOINTS WITH EFFICIENCY OF 85 %

$F_{tu_{RT}} = 37.5(.85) = 31.9$ KSI
$F_c = 55.0$ KSI

TEMPERATURE REDUCTIONS

CONDITION	TEMP DEG F	TIME HRS	K_t REF: FIG. 41 —	F_t KSI	K_{t_c} REF: FIG 42 —	F_c KSI
BEFORE EXP	80	—	1.00	31.9	1.00	55.0
AT TEMP	305	1000	.72	23.0	.68	37.5
AFTER	80	—	1.00	31.9	1.00	55.0
AT TEMP.	425	100	.72	23.0	.55	30.0
AFTER	80	—	1.00	31.9	1.00	55.0
AT TEMP	630	100	.33	10.5	.05	2.7
AFTER	80	—	.75	23.9	.75	41.5
AT TEMP	685	10	.45	14.3	.12	6.6
AFTER	80	—	.75	23.9	.75	41.5
AT TEMP	725	10	.30	9.6	.08	4.4
AFTER	80	—	.50	15.9	.50	27.5
AT TEMP	815	1	.30	9.6	0	0
AFTER	80	—	.50	15.9	.50	27.5

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IMPREGNATED FABRIC DENSITY ESTIMATION

MATERIAL	SPECIFIC GRAVITY	DENSITY	IMPREGNATED FABRICS		
			EC 106	ECC 125	ECC 181
	—	#/in³	#/in³	#/in³	#/in³
GLASS FIBER	2.54	.092	.0274	.0380	.0506
$1 - w_{FAB}/w_{GLASS}$			.702	.587	.450
POLYESTER RESIN	1.20	.043	.0574	.0717	.0699
PHENOLIC RESIN	1.30	.047	.0604	.0734	.0717
EPOXY RESIN	1.20	.043	.0574	.0717	.0699
		AVE →	.0583	.0723	.0805
SILICONE RUBBER	1.90	.069	.0757	.0823	.0816
ACRYLIC RUBBER	1.09	.039	.0548	.0701	.0682
FLOUROELASTOMERS	1.20	.043	.0574	.0717	.0699
		AVE →	.0626	.0748	.0800

$$w_{\substack{IMPREG \\ FABRIC}} = \left(1 - \frac{w_{FABRIC}}{w_{GLASS}}\right) w_{IMPREG} + w_{FABRIC}$$

CLOTH	RIGID AVE w	NON RIGID AVE w
106	.0583	.0626
125	.0723	.0748
181	.0805	.0800
AVE FOR 3 FABRICS	.0700	.0725

ANC-17 P. 83
INDICATE RIGID LAMINATES RANGE IN SPECIFIC GRAVITY FROM 1.4 TO 1.8.

1.7 (.0361) = .0614 #/in³
1.8 (.0361) = .0650 #/in³

ASSUME FOR IMPREGNATED FABRIC AN AVERAGE DENSITY* OF $\frac{.0632}{.0700}(.0725) = .065$ #/in³

* BASED ON FABRIC THICKNESS

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D. Structural Feasibility

On the following pages approximate formulae have been developed for the determination of structural pressure and minimum gages or areas of material for the non-rigid wing. Assumptions made and detailed in the calculations are:

1. Uniform airload distribution.
2. No wing inertia relief.
3. Root at ℄ is typical critical section.
4. 2/3 of chord is uniformly loaded by the design bending moment of the wing outboard of section.
5. Section requirements are constant from root to tip.

The rigid structure would be analyzed by well established methods and are not detailed here.

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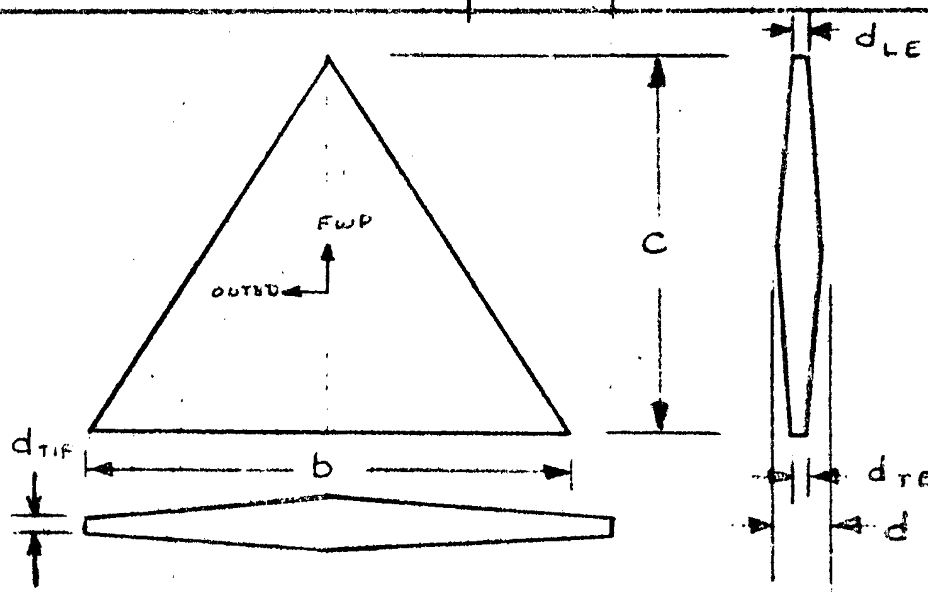
GENERAL WING PARAMETERS

PARAMETER	SYMBOL	REMARKS
MATERIAL STRENGTH	F	TENSION, COMPRESSION OR SHEAR
MATERIAL DENSITY	w	WEIGHTS OF STRUCTURAL MATERIAL
STRUCTURAL TEMPERATURE	K_t	EFFECT OF TEMP. &
THERMAL OR LOADING LIFE		TIME AT TEMP. ON F
PRODUCIBILITY LIMITS		MINIMUM THICKNESS, ETC
VEHICLE GROSS WEIGHT	W_v	WEIGHT OF VEHICLE FOR CONDITION
ULTIMATE LOAD FACTOR	N_z	DESIGN FACTOR FOR CONDITION
AIRLOAD DISTRIBUTION	—	C.P. LOCATION, LOCAL PROBLEMS
MASS DISTRIBUTION	—	INERTIA RELIEF, SHEAR PATHS
AERO ELASTIC RQMTS.	—	TORSIONAL STIFFNESS
SMOOTHNESS RQMTS.	—	TYPE OF CONSTRUCTION
WING PLANFORM	K_c	SHAPE OR $K_c = c/b$
SPAN	b	TIP TO TIP
ROOT CHORD	c	APEX TO APEX
MAXIMUM DEPTH	d, K_d	$K_d = d/b$
SPAN TAPER	K_s	$K_s = d_{TIP} + d_E / 2d_E$
CHORD TAPER	K_{CH}	$K_{CH} = d_{LE} + d_E / 2d_E$



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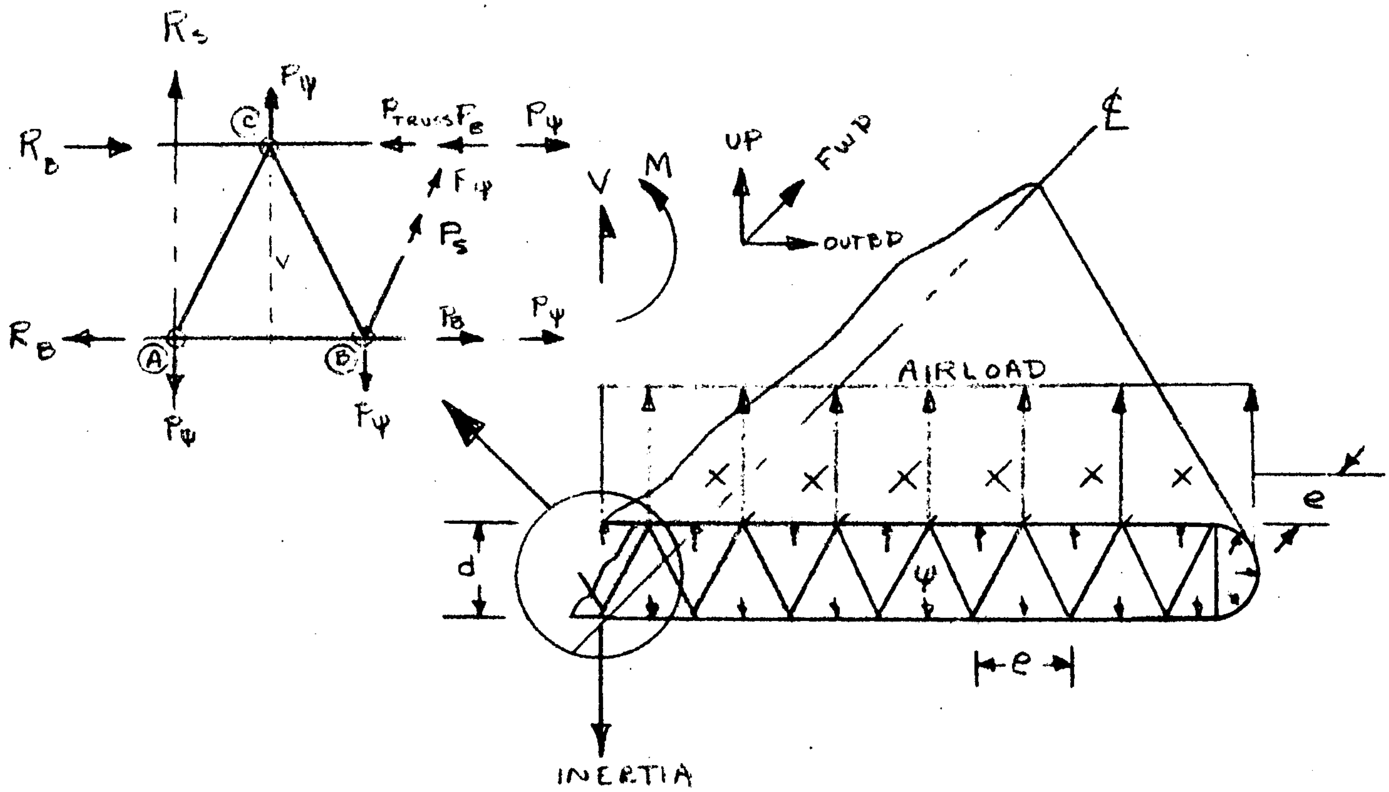
NON RIGID WING



MAXIMUM SKIN LOAD, (UNIT WIDTH) ELEMENT A-B

$$P_{SK} = P_\psi + F_B + F'_{TR}$$

$$F_\psi = \frac{\psi\, d_{EFF}}{2}$$

ψ = INTERNAL PRESSURE #/IN²
$d_{EFF} = K_{CH}\, d$ (IN)

$$P_\psi = \frac{\psi K_{CH} d}{2}$$

$$P_B = \frac{M'}{d_{EFF}} = \frac{M'}{K_{CH} d}$$

M' = BENDING MOMENT APPLIED PER UNIT WIDTH (IN-#/IN)

$$P'_{TR} = \frac{V'e}{2 d_{EFF}} = \frac{V'e}{2 K_{CH} d}$$

V' = SHEAR LOAD APPLIED PER UNIT WIDTH (#/IN)
e = TRUSS SPACING (IN)

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NON RIGID WING

MAXIMUM SKIN LOAD (CONT.)

$$P_{SK_{MAX}} = \frac{\psi K_{CH} d}{2} + \frac{M'}{K_{CH} d} + \frac{V'e}{2 K_{CH} d} \quad (1) \quad \#/IN$$

MINIMUM STRUCTURAL PRESSURIZATION

FOR ZERO COMPRESSION CAPABILITY, THE PRE TENSION OF PRESSURIZING MUST EQUAL THE MAXIMUM COMPRESSION EXPECTED OR:

$$P_{\psi} = P_{B} + P_{TR}$$

$$\psi \frac{K_{CH} d}{2} = \frac{M'}{K_{CH} d} + \frac{V'e}{2 K_{CH} d}$$

$$\psi_{MIN} = \frac{2}{(K_{CH} d)^2} \left(M' + \frac{V'e}{2} \right) \quad (2) \quad PSIG$$

REWRITING 1 MAXIMUM SKIN LOAD

$$P_{SK_{MAX}} = \frac{K_{CH} d}{2} \left[\frac{2}{(K_{CH} d)^2} \left(M' + \frac{V'e}{2} \right) \right] + \frac{M'}{K_{CH} d} + \frac{V'e}{2 K_{CH} d}$$

$$P_{SK_{MAX}} = \frac{2M' + V'e}{K_{CH} d} \quad (3) \quad \#/IN$$

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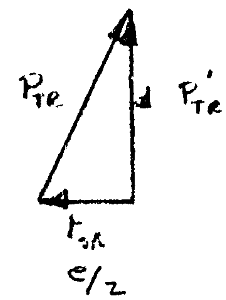
NON RIGID WING

MAXIMUM TRUSS LOAD

$$P_{TR} = P_\psi + P_S$$

$$P'_\psi = \frac{\psi e^2}{2} \qquad P_\psi = \frac{\sqrt{d^2 + e^2/4}}{d}\left(\frac{\psi e^2}{2}\right)$$



$$P'_S = V' e \qquad P_S = \frac{\sqrt{d^2 + e^2/4}}{d}(V'e)$$

$$P'_{TR} = \frac{\psi e^2}{2} + V'e$$

$$P_{TR} = \frac{\sqrt{d^2 + e^2/4}}{d}\left(\frac{\psi e^2}{2} + V'e\right)$$

FOR ZERO COMPRESSION CAPABILITY

$$P_\psi = P_S$$

$$\frac{\psi e^2}{2} = V'e$$

$$\boxed{\psi_{MIN} = \frac{2V'}{e}}$$ (4) PSIG

$$\boxed{P_{TR_{MAX}} = 2V'e}$$ (5) #

OR IF ψ_{MIN} BENDING IS LARGER

$$\boxed{P_{TR_{MAX}} = \frac{\sqrt{d^2 + e^2/4}}{d}\left[\frac{\psi e^2}{2} + V'e\right]}$$ (5)A #

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NON RIGID WING

EVALUATING M' & V'

ASSUME FOR SIMPLICITY

1. UNIFORM AIRLOAD DISTRIBUTION
2. NO INERTIA RELIEF
3. TYPICAL CRITICAL SECTION IS ROOT CHORD
4. 2/3 OF CHORD IS EFFECTIVE IN RESISTING THE LOADS OUTBOARD OF THAT SECTION APPLIED UNIFORMLY
5. SECTION REQUIREMENTS ARE CONSTANT, ROOT TO TIP

$$M = \frac{WN}{2}\left(\frac{b}{6}\right) = \frac{WNb}{12}$$

$$M' = \frac{M}{C'} = \frac{M}{2/3\,C} = \frac{WNb}{8C} \quad (6) \quad \frac{\text{IN-\#}}{\text{IN}}$$

$$V = \frac{WN}{2}$$

$$V' = \frac{V}{C'} = \frac{3WN}{4C} \quad (7) \quad \#/\text{IN}$$

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NON RIGID WING

SKIN THICKNESS

$$f_{SK} = \frac{P_{SK}}{t_{SK}} \qquad t_{SK} = \frac{P_{SK}}{F'_{tu}}$$

$$P_{SK_{MAX}} = \frac{2M' + V'e}{K_{cu} d} \qquad M' = \frac{WNb}{8c}$$

$$F'_{tu} = K_t F_{tu} \qquad V' = \frac{3WN}{4c}$$

$$t_{SK_{ROOT}} = \frac{WN}{4c K_{cu} d K_t F_{tu}} (b + 3e) \quad (8) \quad \text{IN}$$

STRUCTURAL PRESSURIZATION

$$\psi_{MIN} = \frac{2}{(K_{cu} d)^2} \left(M' + \frac{V'e}{2} \right) \quad \text{OR} \quad \frac{2V'}{c}$$

WHICHEVER IS LARGER

OR

$$\psi_{MIN} = \frac{WN}{4c(K_{cu}d)^2} (b + 3e) \quad \text{OR} \quad \frac{3WN}{2ce} \quad (9) \quad \text{PSIG}$$

TRUSS AREA

$$f_{TR} = \frac{P_{TR}}{A_{TR}} \qquad A_{TR} = \frac{P_{TR}}{F'_{tu}}$$

$$F'_{tu} = K_t F_{tu}$$

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TRUSS AREA (CONT)

$$P_{TK} = \frac{\sqrt{d^2 + e^2/4}}{d}\left[\frac{\psi e^2}{2} + V'e\right] \text{ OR } 2V'e$$

ORDINARILY, ψ BENDING IS LARGER THAN ψ TRUSS, SO

$$A_{TR} = \frac{\sqrt{d^2 + e^2/4}}{d\,K_t F_{tu}}\left[\frac{\psi_B e^2}{2} + V'e\right]$$

$$\psi_B = \frac{WN}{4c(K_{cH}d)^2}(t + 3c) \qquad V' = \frac{3WN}{4c}$$

$$\boxed{A_{TR} = \frac{WN\sqrt{d^2 + e^2/4}}{4cd\,K_t F_{tu}}\left[\frac{(t+3e)e^2 + 6e(K_{cH}d)^2}{2(K_{cH}d)^2}\right]} \quad (10) \text{ IN}^2$$

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E. Weight Estimates

A method of determining an inflated wing structure weight is developed on page 106 to 111 based on the data of Section IV D. It assumes uniform skin thickness on wing plus a 10% factor for local problems. The actual wing weight calculation for the MC-10 configuration are shown on page 108. Pages 110 to 112 develop an approximate formula for evaluating the various wing weight parameters. Figure 45 is a curve of wing structure weight versus parameters for configuration evaluation.

In the weight breakdown, Table IV, page 49, the fin weight has been chosen as being around 50% of the wing unit weight, with fin area approximately 14% of the wing area. Body-Pylon structure weights have been approximated by extrapolation of existing weights data.

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"NON RIGID WING"

WEIGHT ESTIMATION

$$W_w = \text{SKIN WGT} + \text{TRUSS WGT} + \text{GAS WGT} + \text{MISCLEANOUS}$$

SKIN WGT

$$W_{SK} = A_{SK}\, t_{SK_{AVE}}\, w_{SK}$$

$$A_{SK} = 2\,\frac{bc}{2} = bc$$

$$t_{SK_{ROOT}} = \frac{WN}{4c\,K_{CH}\,d\,K_t\,F_{tu}}\,(b+3e)$$

ASSUME CONSTANT OVER TOTAL WING

$$\boxed{W_{SK} = \frac{WN\,b\,w_{SK}}{4\,K_{CH}\,d\,K_t\,F_{tu}}\,(b+3e)} \quad (11)\ \#$$

TRUSS WGT

$$W_{TR} = A_{TR_{AVE}}\,\Sigma L_{TR}\,w_{TR}$$

$$A_{TR_{ROOT}} = \frac{WN\sqrt{d^2+e^2/4}}{4\,C\,d\,K_t\,F_{tu}}\left[\frac{(b+3e)e^2 + 6e(K_{CH}d)^2}{2(K_{CH}d)^2}\right]$$

$$A_{TR_{AVE}} = \frac{WN\sqrt{d_{AVE}^2 + e^2/4}}{4\,c\,d_{AVE}\,K_t\,F_{tu}}\left[\frac{(b+3e)e^2 + 6e(K_{CH}d)^2}{2(K_{CH}d)^2}\right]$$

WHERE

$$d_{AVE} = \frac{K_{CH}\,d + K_S\,d}{4}$$

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NON RIGID WING

TRUSS WGT. (CONT)

$$\Sigma L_{TR} = \frac{c}{e}\left(\frac{b}{2}\right)\sqrt{d_A^2 + e^2/4}$$

$$W_{TR} = \frac{W N b w_{TR}}{8 d_A e K_t F_{tu}}\left(d_A^2 + e^2/4\right)\left[\frac{(t+2e)e^2 + 6e(K_{tH} d)}{2(K_{tH} d)^2}\right]$$

(12) #

GAS WGT

$$W_G = Vol\left(w_{GAS} - w_{AIR}\right)$$

$$Vol = \frac{t c d_A}{2}$$

$$W_G = \frac{t c d_A}{2}\left(w_G - w_A\right)$$

(13) #

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NON RIGID WING

MC-10 TYPICAL CALCULATION

$W = 13800\#$ $N_{Z_{ULT}} = 1.5(1.15) = 1.72$ G ULT HOT

$K_t F_{tu} = 14100 \ \#/in^2$ REF: TABLE IX } SKINS & TRUSS

$w = .065$

$w_{GAS} = .00000646 \ \#/in^3$ @ 15 PSIA, HELIUM

$w_{AIR} = .0042(.0000443) = .000000186$ @ 125000 FT

$b = 67.71(12) = 813$ in $C_R = 58.63(12) = 703$ in

$d = .04 C_R = .04(703) = 28.1$

$K_S = K_{CH} = \dfrac{2 + 28.1}{2(28.1)} = .537$ $K_{CH} d = 15.1$

$d_{AVE} = 2(.537)(28.1)/4 = 7.65$ $K_d = \dfrac{28.1}{813} = .0347$

$e = 1.0$

$$\Psi_{MIN} = \frac{W N}{4c\,(K_{CH} d)^2}\,(b + 3e)$$

$$= \frac{13800(1.72)}{4(703)[.537(28.1)]^2}\,[813 + 3(1)] = 29.4 \text{ PSIG}$$

(.0361)

$$\text{OR} \quad \frac{3(13800)(1.72)}{2(703)} = 5.07 \text{ PSIG}$$

$$W_{SK} = \frac{W N b\, w_{SK}}{4 K_{CH} d\, K_t F_{tu}}\,(b + 3e)$$

(1.035)

$$= \frac{13800(1.72)(813)(.065)}{4(.537)(28.1)(14100)}\,[813 + 3(1)] = 1205 \#$$

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NON RIGID WING

MC-10 (CONT)

$$W_{TR} = \frac{\omega N L w_{TR}}{8 d_A e K_t F_{tu}} \left(d_A^2 + e^2/4 \right) \left[\frac{(t+3e)e^2 + 6e(K_{t4}d)^2}{2(t_{+4}d)^2} \right]$$

$$\underline{\underline{W_{TR}}} = \frac{13800(1.72)(813)(.065)}{8(7.65)(14100)} \left[(7.65)^2 + .25 \right] \left[\frac{(813+3) + 6(15.1)^2}{2(15.1)^2} \right]$$

$$= \underline{\underline{400}}$$

$$W_{GAS} = \frac{b c d_A}{2} \left(w_G - w_A \right)$$

$$\underline{\underline{W_{GAS}}} = \frac{813(703)(7.65)}{2} \left(6.46 \times 10^{-6} - .19 \times 10^{-6} \right) = \underline{\underline{14 \#}}$$

Σ WEIGHT

SKINS	1205
TRUSS	400
GAS	14
	1619 #

ASSUME 10% FOR LOCAL PROBLEMS, ATTACHMENTS, ETC. — 162

1781 # TOTAL

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APPROX WING WEIGHT — NON RIGID WING

$$W_w = W_{SK} + W_{TR} + W_{GAS} + W_{MISCL.}$$

SIMPLIFY AS:

$$W_w = K_o \left(W_{SK} + W_{TR} \right)$$ SINCE SKIN & TRUSS ARE MAJOR ITEMS

ASSUME e IS SMALL < 1.0

$$W_w = K_o \left[\frac{W N L^2 w_{SK}}{4 K_{CH} d K_t F_{tu}} + \frac{W N L\, w_{TR}\, d_A}{8 d_A K_t F_{tu}} \left(\frac{b + 6 K_{CH}^2 d^2}{2 K_{CH}^2 d^2} \right) \right]^{*}$$

* FROM EQ 11 & 12

FOR SHARP L.E, T.E & TIP $R \leq 1.0$

$K_{CH} = .50 \qquad d_A = d/4$

ALSO ASSUME

$K_t F_{tu} w_{SK} = K_t F_{tu} w_{TR}$

THEN

$$W_w = K_o \left[\frac{W N b w}{2 K_t F_{tu}} \left(\frac{1}{K_d} + \frac{1}{8 K_d} + \frac{K_d b}{10} \right) \right]$$

FOR COMPLETE GENERALITY OF WING PLANFORM & SHAPE, ASSUME:

$K_s = \dfrac{d}{c}$ = AERODYNAMIC WING DEPTH RQMT.

$K_c = \dfrac{c}{b}$ = PLANFORM SHAPE PARAMETER

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APPROX. WING WEIGHT (CONT) NON RIGID WING

$$K_d' = \frac{d}{b} \quad \text{BY PRIOR DEFINITION}$$

OR

$$K_d = \frac{K_s c}{b} \quad \& \quad c = K_c b$$

SO

$$K_d = \frac{K_s K_c b}{b} = K_s K_c$$

NOW:

$$W_w = K_0 \left[\frac{W N b w}{2 K_t F_{tu}} \left(\frac{1}{K_s K_c} + \frac{1}{8 K_s K_c} + \frac{K_s K_c b}{10} \right) \right]$$

$$W_w = K_0 \left[\frac{W N b w}{160 K_s K_c K_t F_{tu}} \left(90 + 8 K_s^2 K_c^2 b \right) \right]$$

EVALUATE K_0 FROM MC-10 CALC. COMP.

$$K_0 = 1791 \Big/ \frac{12800 (1.92)(81.3)(.065)}{160(.04)(.464)(14100)} \left[90 + 8 (.04)^2 (.864)^2 (81.3) \right]$$

$$= 1.13$$

$$W_{w \text{ APPROX}} = .00707 \frac{W N b w}{K_s K_c K_t F_{tu}} \left(90 + 8 K_s^2 K_c^2 b \right) \quad (14)\ \#$$

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APPROXIMATE WING WEIGHT, NON RIGID

① W	② S	③ W/S	④ b	⑤ N ult	w	⑥ K_s	K_c	⑦ $K_t F_{tu}$	⑧ $\frac{W N b w}{K_s K_c K_t F_{tu}}$ 224×10^{-6} Wb	⑨ $K_s^2 K_c^2 b$ .00119 b	⑩ $.0566 K_s^2 K_c^2 b$	⑪ $.636 + .0566 K_s^2 K_c^2 b$	⑫ W_w
#	FT²		IN	G	#/IN³	—	—	#/IN²					
4000	1000	4	577	1.72	.065	.04	.864	14100	527	.687	.0387	.675	356
	2000	2	812						744	.967	.0547	.691	514
	4000	1	1162						1065	1.395	.0784	.714	761
	6000	.67	1410						1290	1.680	.0952	.731	943
8000	1000	8	577						1055			.675	712
	2000	4	812						1489			.691	1030
	4000	2	1162						2130			.714	1520
	6000	1.21	1410						2585			.731	1890
12000	1000	12	577						1585			.675	1070
	2000	6	812						2230			.691	1540
	4000	3	1162						3190			.714	2280
	6000	2	1410						3720			.731	2830
16000	1000	16	577						2110			.675	1430
	2000	8	812						2780			.691	2060
	4000	4	1162						4260			.714	3040
	6000	2.66	1410						5170			.731	3790

$$W_w = .00707 \frac{W N b w}{K_s K_c K_t F_{tu}} \left(90 + 8 K_s^2 K_c^2 b\right) = \frac{W N b w}{K_s K_c K_t F_{tu}} \left(.636 + .0566 K_s^2 K_c^2 b\right)$$

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NON RIGID WING

PRODUCIBILITY LIMITS

$$t_{SK} = \frac{W N}{4 c K_{CH} d K_t F_{tu}} (b + 3e)$$

FOR $e \leqq 1.0$ $K_{CH} \approx .50$ $c = K_c b$

$$d = K_d b = \frac{K_s c}{b} b = K_s c$$

$$t_{SK} = \frac{W N}{2 b K_c^2 K_s K_t F_{tu}}$$

ASSUME MINIMUM PRACTICAL GAGE OF DIAGONALLY LOADED FABRIC IS .015 THEN:

$$\frac{W N}{2 b K_c^2 K_s K_t F_{tu}} = .015$$

$$\boxed{\frac{W}{b} \geqq .030 K_c^2 K_s K_t F_{tu} N} \quad (15)$$

Form

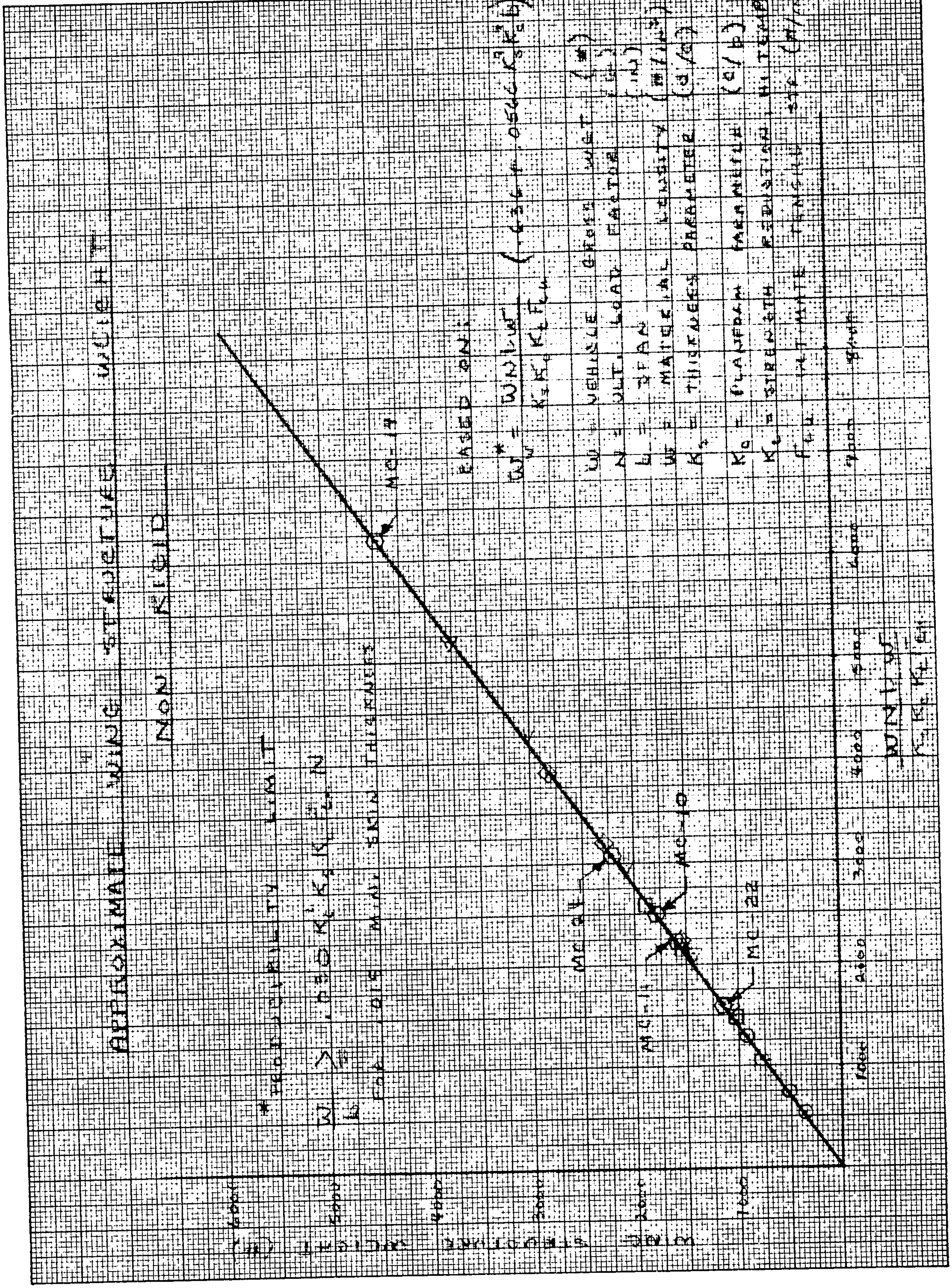
APPROXIMATE WING STRUCTURE WEIGHT
NON - RIGID
* PRODUCIBILITY LIMIT
FOR .015 MIN. SKIN THICKNESS
MC-14
MC-24
MC-11
MC-10
MC-22
BASED ON:
W = VEHICLE GROSS WGT
N = ULT. LOAD FACTOR
L = SPAN
K₁ = THICKNESS PARAMETER
K₀ = PLANFORM PARAMETER
1000
2000
3000
4000
5000
6000

FIG. 45

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