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R. DZ PI

Maj Haines

FC
MX-2276

RECONNAISSANCE AIRCRAFT WEAPON SYSTEM

SYSTEM DESIGN

CONTRACT NO. AF33(616)-2419

SUPPLEMENTAL AGREEMENT NO. 2

REPORT NO. D143-945-027

BELL *Aircraft* CORPORATION

1 DECEMBER 1955

MAR 28 1957

C6-15979

55RD2-8665-10

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BELL *Aircraft* CORPORATION
BUFFALO 5, · NEW YORK

TECHNICAL DATA

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REPORT NO.)143-945-027

MX-2276

RECONNAISSANCE AIRCRAFT WEAPON SYSTEM

SYSTEM DESIGN

DATE 1 Dec. 1955

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FOREWORD

During the period from 2 May 1955 to 1 December 1955, the Bell Aircraft Corporation conducted a study program for the Directorate of Systems Management, Deputy Commander for Weapon Systems, RDZ 1SB, Headquarters Air Research and Development Command under Supplemental Agreement 2 (56-284) of USAF contract AF33(616)2419 - RDO No. R441-47. The primary objective of this study is to conduct analytical investigations and design studies of a weapon system which adapts the MX-2276 concept to satisfy or exceed the criteria presented by Development Requirements, System No. 118P. A secondary objective is to supplement and advance studies previously made under this contract.

The work accomplished during this program is reported in the following series of reports:

D143-945-024	Aerodynamics
D143-945-025	Structures
D143-945-026	Navigation and Control
D143-945-027	System Design
D143-945-028	Photographic Subsystem
D143-945-029	Summary Report

In addition to these reports, separate powerplant proposals are submitted by the Bell Aircraft Corporation (Report No. 02-945-118) and North American Aviation, Inc. (Report No. PC-177P).

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I. ABSTRACT

A weapon system to satisfy the military requirements of System No. 118P has been investigated. The system consists of a rocket boosted airplane which achieves its range from a hypersonic glide. Preliminary analyses indicate that such a weapon system is a feasible means of attaining, and in some instances exceeding the desired military requirements. The weapon system is an adaptation of the MX-2276 basic concept.

Configuration studies of the airplane configuration and booster arrangement were made to attain a minimum take-off weight. The vehicle configuration presented may not be optimum but is sufficiently representative to show the performance potential and operation capability of such a weapon system.

Crew effectiveness and specific duties are outlined.

The target area covered by the weapon system is shown and the range required for complete coverage is indicated.

The specialized reconnaissance equipment is described and preliminary performance specifications listed.

Rocket propulsion systems are outlined for both the airplane and the booster. Preliminary performance specifications are presented in propulsion proposals.

A brief discussion of ground handling and operation procedures for the system is presented.

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II. INTRODUCTION

The initial studies of the MX-2276 advanced strategic weapon system were based upon a vehicle arrangement which was basically designed for bombardment missions. However, the bomb could be removed and specialized equipment installed for reconnaissance missions. In addition the various stages of the weapon system were designed to provide the proper distribution of energy among the three stages to achieve the lowest possible take-off weight.

Since one of the steps in the development of the basic bombardment system would be an airplane with one booster, BAC was awarded an extension of AF contract 33(616)-2419 (Supplemental Agreement 2 (56-284)) to determine the applicability of the MX-2276 concept with respect to Systems Requirement 118P. Both analytical investigations and design studies were to be made.

The philosophy adopted for this study consists of the investigation of a pure reconnaissance weapon system which includes an airplane and a booster. The vehicle arrangement provides the minimum take-off weight for a pure reconnaissance system. The maximum speed for the boost-glide flight path was dictated by an arbitrary range requirement of 5000 n. miles.

Reported herein are the results of the analytical investigations and design studies of a weapon system which has a capability of photographic, ferret and radar reconnaissance. This report contains a discussion of the weapon system concept, the results of the preliminary configurational design and human factor considerations. Studies of the reconnaissance equipment, target coverage, and the propulsion system are included along with a brief discussion of ground handling and pre-launch operation. The development program thru performance demonstration flight testing is also presented.

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III. WEAPON SYSTEM CONCEPT

A. Military Requirements.

The military requirements for a piloted special reconnaissance weapon system for use in tactical and strategic reconnaissance operations as outlined in Development Requirements System No. 118P are listed below.

1. Operational Features.

    a. Capabilities of daylight photography, high order ferret, and high resolution radar reconnaissance are required.

    b. Maximum survival in the air by high flight altitudes and low detectability is required.

    c. A high degree of weapon system reliability is required.

    d. Capability of take-off, penetration and recovery under poor weather conditions is required.

    e. Operational availability in the 1960-1965 time period is required.

2. Performance Objectives.

    a. Basic mission combat zone altitude is 100,000 feet, with 150,000 feet desired.

    b. Basic mission radius is 1500 nautical miles, with 2000 nautical miles desired. An alternate permissible mission is one having a total range of 3000 nautical miles, with 4000 nautical miles desired.

    c. Penetration speed will be the maximum possible. However, altitude and radius are of greater importance.

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B. General Description.

The very high altitude and very high speed resulting from the MX-2276 boost-glide concept provides the type of performance so essential for future reconnaissance operations.

The special design study reported herein shows that the MX-2276 concept can be adapted to satisfy and in some instances exceed the military requirements as noted above. The operational requirements of each type of reconnaissance are broadly defined in S.R. 118P.

The reconnaissance aircraft weapon system consists of an airplane and a booster in a tandem arrangement. The vehicle is launched vertically. The booster using a liquid rocket propulsion system accelerates the vehicle to a speed of 5400 feet per second at an altitude of 65,000 feet. The booster is dropped and the liquid rocket propulsion system in the airplane increases the speed to a maximum value of 16,600 feet per second at an altitude of 165,000 feet. At this point the airplane is in horizontal gliding flight and can achieve a ground to ground range of 4,680 nautical miles. This performance exceeds the minimum military requirements for the alternate mission.

The airplane system consists of a delta wing configuration with a landing wing loading of 25 lb/sq. ft. Conventional rudder and elevon controls provide the aerodynamics forces required to maintain the desired flight path. The basic sensing element which provides the position, rate, and acceleration information is a multi-axis inertial platform. Electronic computers and memory circuits compare the flight path information and provide the proper control thru the roll, pitch, or yaw autopilot systems.

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A side looking radar system with data presentation to the crew as well as automatic recording is provided for radar reconnaissance and as a secondary means of navigation in conjunction with pilotage techniques.

The crew compartment contains equipment which permits the crew to monitor the flight and navigation progress and the systems which would allow the crew to take over the control of the airplane or navigate to a landing site. Optical as well as electronic display systems will be provided.

The booster consists of propellant tanks and a liquid rocket engine system. The intelligence for the actuation of the gimballed rocket motor in the booster comes from the reference systems and programming units in the airplane.

C. Reconnaissance Philosophy

To better understand the weapon system proposed herein a brief discussion of reconnaissance philosophy is presented.

For the purposes of this discussion it is assumed that the critical intelligence needs in order of importance are:

1. Strategic Warning

2. Operational

3. Technical

The strategic warning need consists of obtaining indications that an attack by an enemy force is imminent. Such indications could include the concentration of forces and equipment as well as new transportation networks. The operational intelligence must provide data for the recognition, identification, and location of probable enemy operational bases or sites. The technical intelligence is required for the evaluation of the state-of-the-art of the equipment the enemy possesses for use in an agressive action.

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The objective of the target detail mission is to obtain high resolution photographic coverage of the target and the area surrounding the target. This type of intelligence is not only required for the intelligence analyst but is also an important input for the preparation of the air objective folder. From an altitude of 150,000 feet 9 foot objects can be detected and 25 foot objects recognized over the entire format of the picture.

2. Ferret.

Ferret surveillance from 30 mc to 40 kmc is required. Section VII describes the recommended approach for ferret reconnaissance equipment. The recommended equipment will cover the desired frequency range and will provide basic data for the determination of radar order of battle and technical intelligence.

3. Radar.

The over-all operational capability of this weapon system is to include radar surveillance using high resolution techniques. The definition of high resolution was not given, however, it was generally agreed that the radar resolution should at least be equal to that obtainable with low order photographic equipment. Section VII summarizes the results of the radar work. A side looking radar operating at K_u band is recommended. The spot size resolution varies from 500 x 535 ft. at the near range point to 1000 x 225 ft. at the far range point.

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IV. CONFIGURATIONS

A. Vehicle

1. Airplane Plus Booster.

General.

The configuration study has consisted of two phases. The first phase was a general exploratory and preliminary optimization study of two-stage configurations which satisfy the range and payload requirements. The second phase was concerned with the detail design of a realistic, but not necessarily optimum, two-stage configuration which appeared to represent the best over-all compromise.

This configuration is shown in Fig. 1. It consists of the manned reconnaissance airplane and a tandem, unmanned booster; the latter being either expendable or recoverable, in part or as a whole (e.g. by parachute).

There are a number of factors underlying the selection of a tandem, unmanned booster configuration. Although certain of these are described in Ref. 1, it is of value to state the more important factors here. First of all, the tandem arrangement provides for an inherently simpler separation of airplane and booster, as compared to configurations where the booster is along the side, top, or bottom of the airplane. Secondly, the drag of the airplane and booster is minimized in the tandem arrangement. In addition, the physical attachment of the airplane to the booster is accomplished readily. The large circular sections at the joint minimize the problem of carrying thrust and shear loads, as well as bending and torsional loads. Finally, the alignment of drag, mass, and thrust are generally superior in tandem arrangements, at the end of boost as well as at the start.

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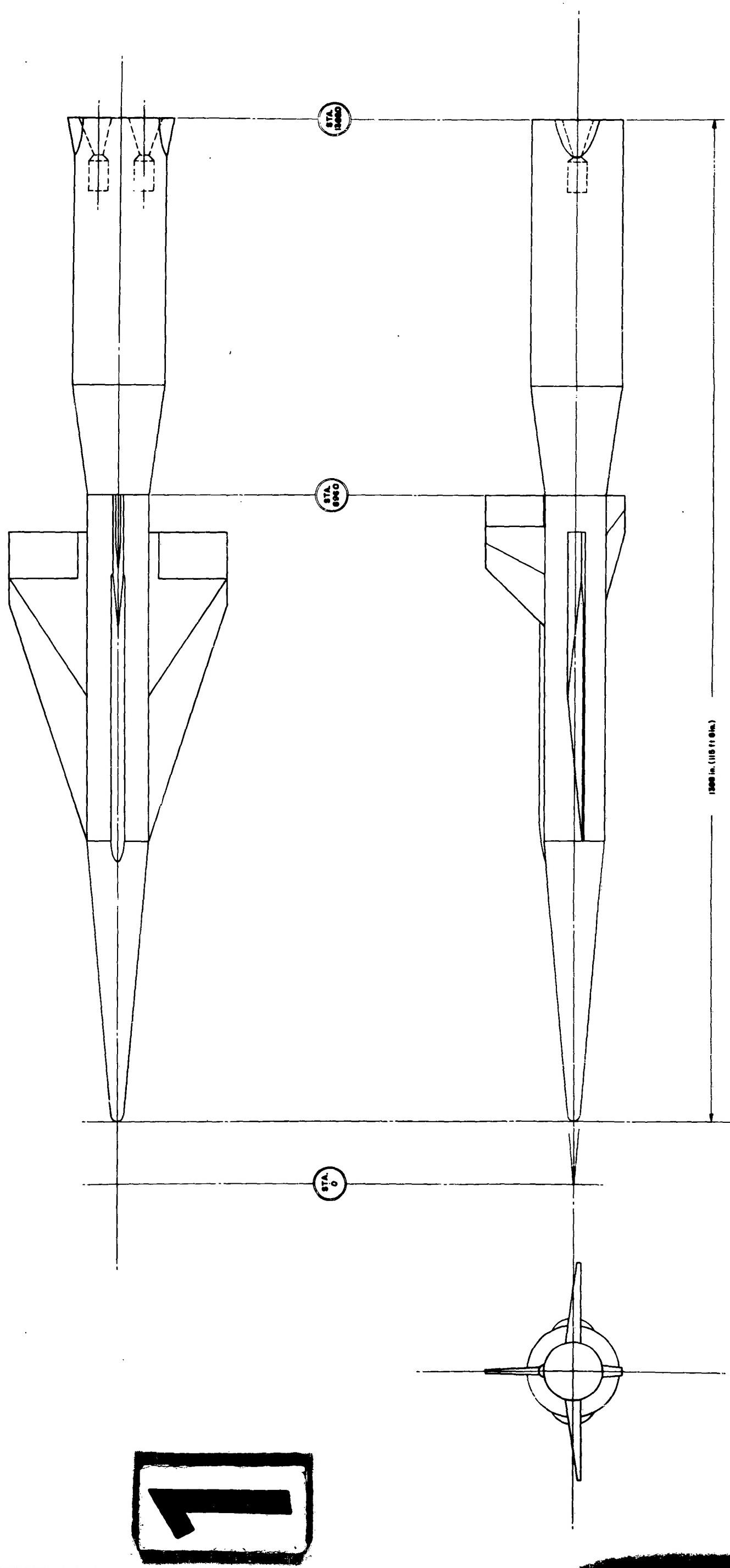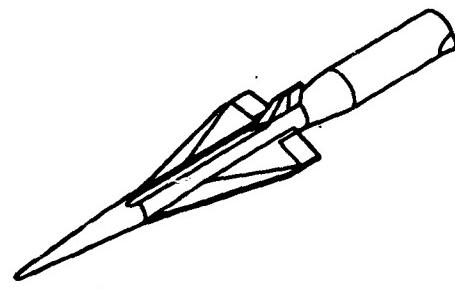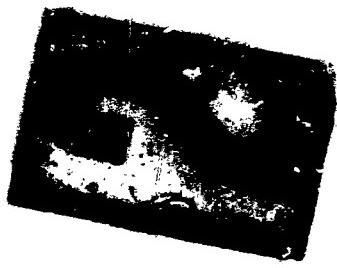
PROJECT MX-2276

U.S. AIR FORCE
PHOTOGRAPHIC RECONNAISSANCE
VEHICLE
two-stage

BELL *Aircraft* Corporation

TAKE-OFF GROSS WEIGHT

TOTAL PROPELLANTS	201,050 lb
DESIGN THRUST AT TAKE-OFF	166,350 lb
	300,000 lb

BOOSTER (GROSS WEIGHT)

PROPELLANTS O_2 AND JP-4	142,450 lb
	124,450 lb

RECONNAISSANCE AIRPLANE (GROSS WEIGHT)

PROPELLANTS $F_2 + O_2$ AND JP-4	58,600 lb
	38,900 lb

STA 0.000

STA 0.000

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BELL *Aircraft* CORPORATION

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Missile
Airplane _____ Report _D143-945-027_
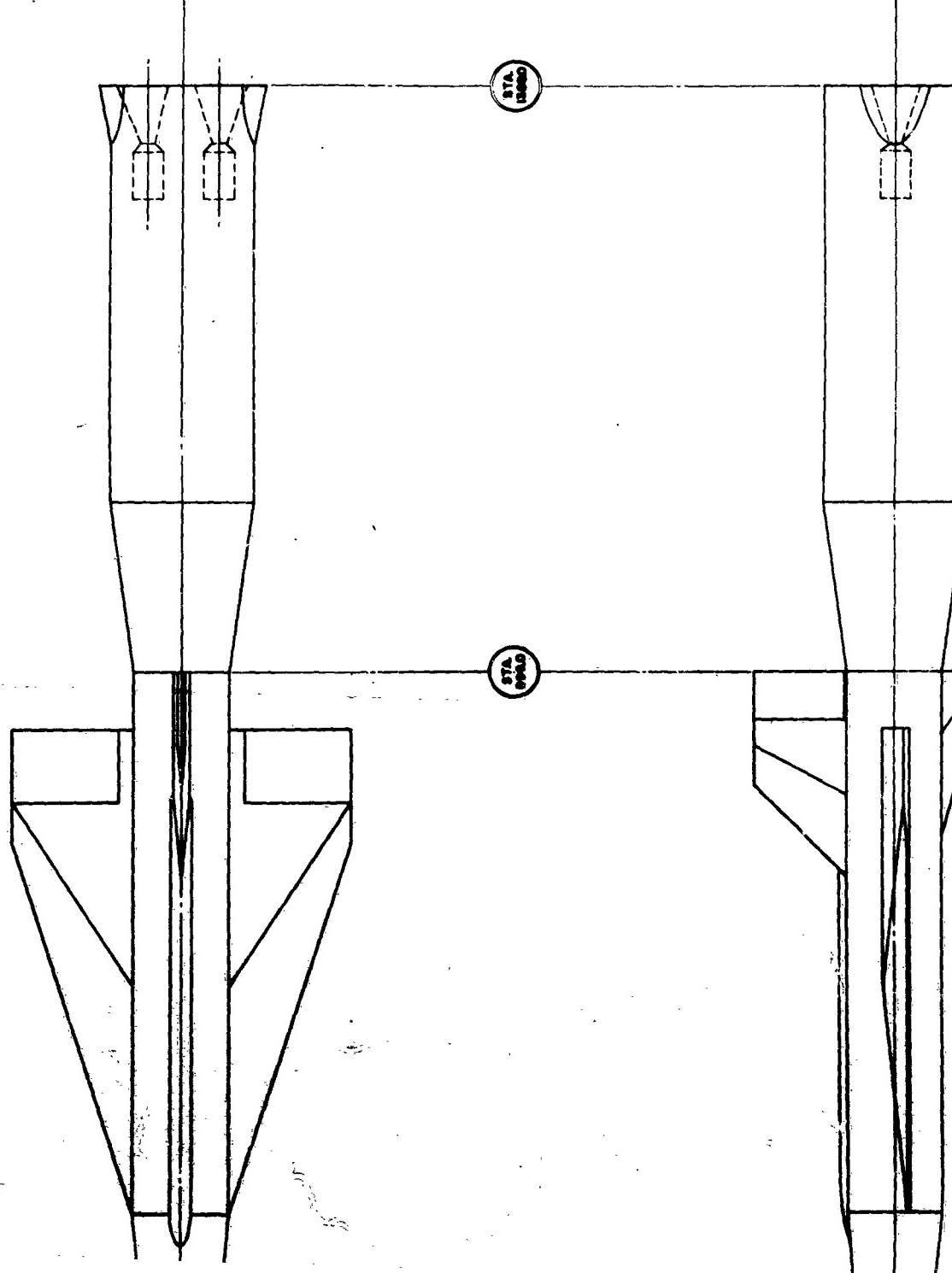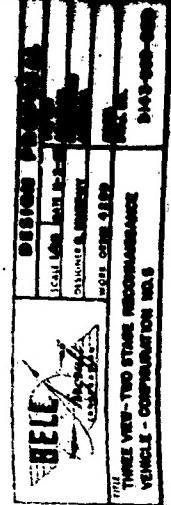
The fact that the booster is a simple tank-motor configuration should reduce the design and development problems associated with the booster stage as compared to the airplane stage.

The airplane would contain all the basic sensing equipment required for the entire flight. Since the airplane is recovered and only the relatively simple booster is discarded the proposed system will be developed sooner and should not result in appreciably higher operational costs compared to a completely recoverable booster. In this respect, it should be noted that parachute recovery of the rocket motors of the booster is a possibility. The size and weight of the motors are such that helicopters could be used to pick up and transport these motors back to the launch base.

The reconnaissance airplane which evolved from the preliminary studies is considerably larger than previously shown in MX-2276 studies. This occurs primarily because the optimum propellant loading of the final stage is substantially larger in the subject two-stage system than in the final stage of three stage systems, even when the latter is designed to provide greater range.

Although the reconnaissance airplane has grown in size as a result of the optimization studies described in Ref. 2, the booster has shrunk at a faster rate. Consequently, the take-off weight shown in Fig. 1 is substantially less than previously shown for two-stage reconnaissance systems of equivalent range. In addition to the lower take-off weight obtained in the manner just described, there are two other inherent improvements. First, more hardware is placed in the airplane which is intended always to be recovered and less in the booster. Thus it becomes of lesser importance

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whether or not the booster is recovered. Secondly, the over-all mass dis-
tribution is improved in that the center of gravity of the two-stage vehicle
lies much further forward than in less efficient configurations where the
airplane is smaller and the booster is larger. The latter has unfavorable
effects both with regard to aerodynamic stability and control effectiveness.

An alternate type of flight, boost-glide-boost-glide, was considered
in an attempt to reduce the maximum speed of the airplane. If a substantial
reduction in speed were possible, the development time would be shortened,
aerodynamic heating would be reduced and the structural design would be
simplified. To achieve this type of flight, the two boost phases are
separated by a period of power off glide. Figure 1a shows the reduction in
maximum speed and the resultant increase in take-off weight. For example,
if one half the total range of 5000 nautical miles is attained after the
second boost, the maximum velocity decreases from 16,600 fps to 12,600.
The corresponding take-off weight would be approximately 6 times that re-
quired for the pure boost-glide path. On the basis of these results it
was felt that the relatively small decrease in maximum speed was outweighed
by the large increase in take-off weight and attendant problems of delayed
motor operation.

Stability and Control.

It will be seen that the two-stage vehicle as shown, does not
employ fins on the booster. The reasons underlying this tentative configur-
ational arrangement are described in Ref. 1, but it will be of value to re-
view them here. First of all, previous experience with large vehicles
taking off in similar manner indicates that powerful and fast responding

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FIGURE 1a

Weight and Velocity Requirements of a Boost-Glide - Boost-Glide System Capable of 5000 n. mi.

$(W_P/W_0)_{Initial} = 90$

$I_{Initial} = 250$ sec $I_{Final} = 350$ sec
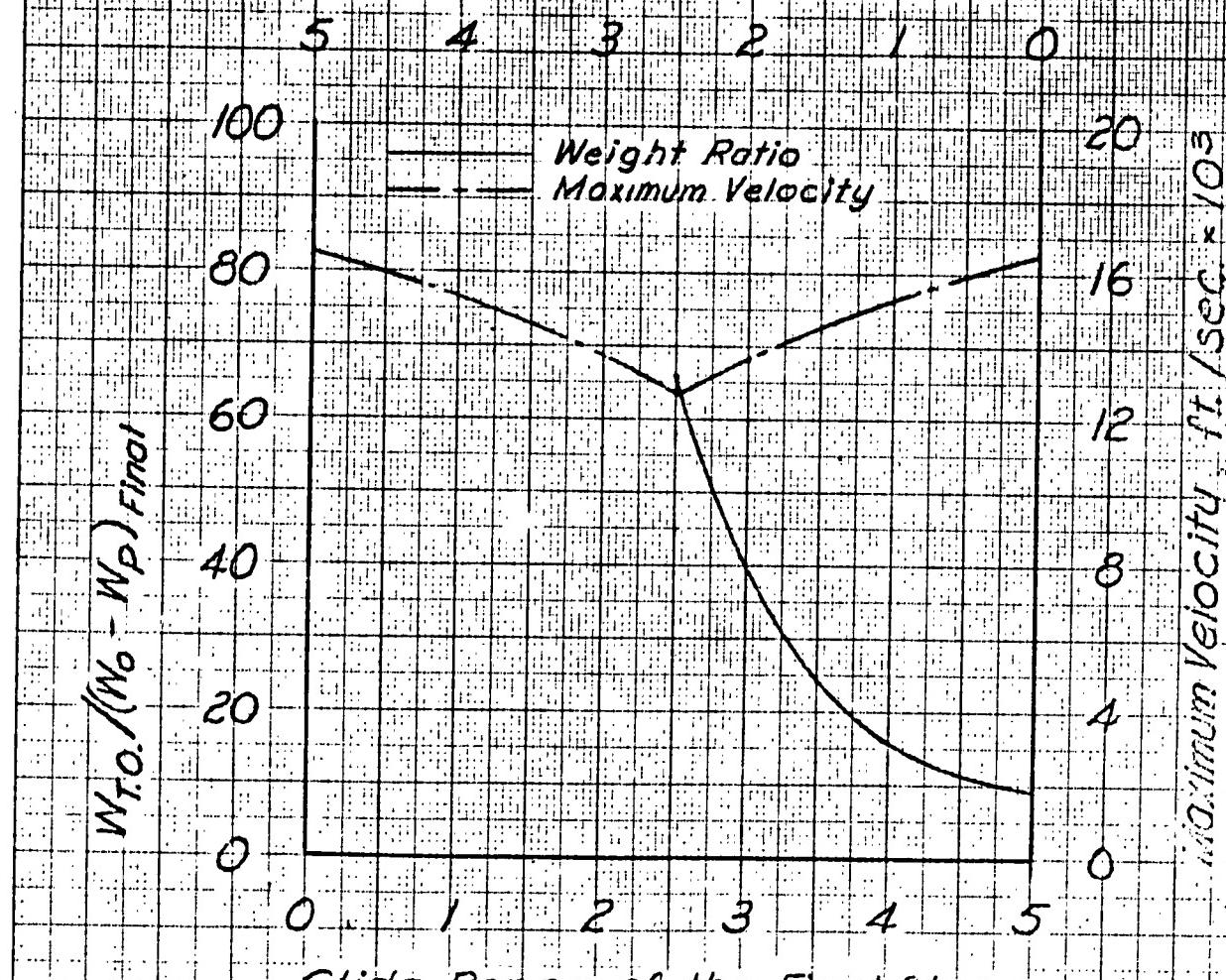
control means must be provided. Thi is particularly important for the first few seconds after take-off where the airspeed is still very low and winds or gusts could incline the vehicle to a point where the thrust could not support the weight. At this critical condition, high weathercock stability is less desirable than none at all since the more stable the vehicle, the more the swing into the relative wind. Thus it appears quite likely that the rocket motors will be called upon to provide a substantial contribution to the stability in the ascent phase.

To provide this stability both booster motors must be fully gimballed to provide pitch, yaw, and roll control, regardless of the aerodynamic configuration of the two-stage vehicle.

Therefore, since the means for stabilization are already present, it has seemed reasonable, at this time, to eliminate booster fins and use the rocket motor force for stabilization during the ascent phase.

Results of a preliminary investigation of stability and control characteristics at the critical take-off condition are contained in Ref. 3. This analysis demonstrates that satisfactory characteristics can be obtained readily with the present configuration and thus supports the qualitative analysis leading to the over-all configurational concept.

On the other hand, the configuration is not felt to be frozen with regard to the features described in the preceding text; in fact, it is felt that further work should be planned on this very important subject. Instead, the main conclusion that should be drawn at this time is that the configuration appears practical and that preliminary analyses show it to be feasible. It does not represent the result of a design optimization study, and further study may indicate some changes to be desirable.

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Propellant Selection.

The propellants selected for the reconnaissance airplane are JP-type as fuel and a 70% - 30% mixture ratio of liquid fluorine and liquid oxygen as oxidizer. The propellant combination and its advantages to the system have been described in previous reports. The booster, on the other hand, uses liquid oxygen and JP-type as propellants, mainly because the large rocket engines currently under development would be directly applicable. In fact, the booster shown herein is designed to use without modification, a two-chamber, 300,000 pound thrust engine now under development and test.

It should be noted that the use of liquid oxygen and JP-type fuel was considered for the final stage as well as the booster. Using identical propellants in both vehicles would be of advantage, logistically. Results were unfavorable, however, in that the take-off weight increased appreciably (approximately 50,000 lb.) thus tending to increase the fabrication cost as well as the cost per sortie.

On the other hand, a favorable outcome was that the optimum final stage size is about the same whether or not the oxidizer is fluorine enriched, even considering the lower oxidizer density for the all oxygen system. Thus, if it became necessary to change to all oxygen, for any reason, it would be necessary to change only the booster, which is the simpler vehicle of the two vehicles by far. This feature adds a certain amount of insensitiveness to propellant selection, which is certainly desirable at this time.

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2. Airplane.

General.

The airplane configuration resulting from this supplemental study is shown in Fig. 2. The vehicle represents a good compromise between the major aerodynamic and structural parameters and is considered to be a substantial improvement over configurations shown in previous studies.

In general, the configuration was obtained by incorporating the best hypersonic design characteristics possible, since these are of major importance. However, it was remembered that the vehicle must have satisfactory flying qualities at all speeds and must also have acceptable landing characteristics. This aerodynamic design is discussed in detail in Ref. 2; the structural design is discussed in Ref. 17.

A main feature of this new configuration is the low aspect ratio, highly swept wing. The combination of sweepback and aspect ratio chosen was based on many considerations including gust loads in ascent (which tend to be minimized) as well as the factors previously mentioned. In addition, the very favorable structural weight afforded with these planforms was taken into account.

The wing has a flat bottom, with typical section being a wedge to the 39% chord station at the root and 35% chord station at the tip, followed by parallel upper and lower surfaces, and ending with a full blunt trailing edge. Thickness is a constant 4% over the span. The leading edge is rounded to ease the heating problem, the radius being 1/2" in a plane perpendicular to the leading edge.

484.7

78°

⊕ STA. 484.7

7.8 SPHERICAL RADIUS

GROUND LINE

194.0

898.0 in. (74 ft 9 in.)

1818.5
FIVE PLY TIRE

STA.
84.8 ⊕

STA.
0 ⊕

8°   10°

4.85°

69.8

296.5 in. (25 ft 10 in.)
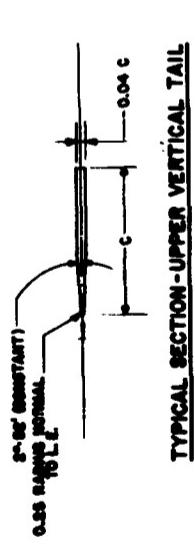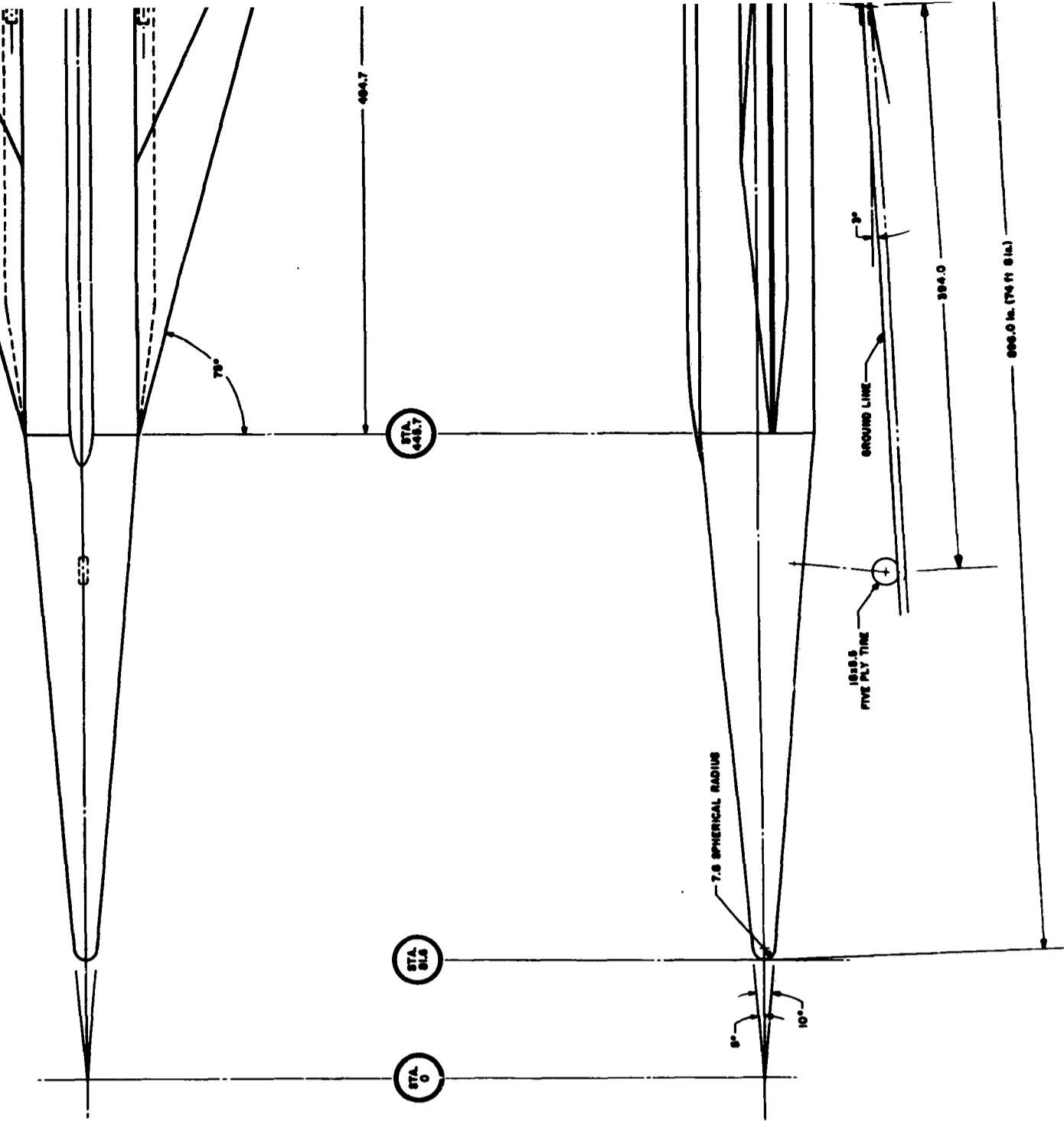
0.04 C
C
8°-55' (CONSTANT)
0.85 RADIUS NORMAL
TO L.E.
TYPICAL SECTION-UPPER VERTICAL TAIL
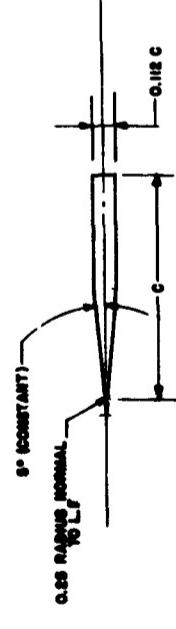
0.112 C
C
8° (CONSTANT)
0.85 RADIUS NORMAL
TO L.E.
TYPICAL SECTION-LOWER VERTICAL TAIL
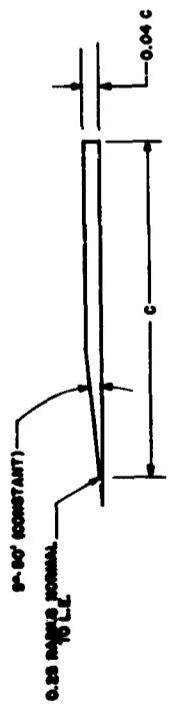
0.04 C
C
9°-00' (CONSTANT)
0.85 RADIUS NORMAL
TO L.E.
TYPICAL SECTION-WING

2
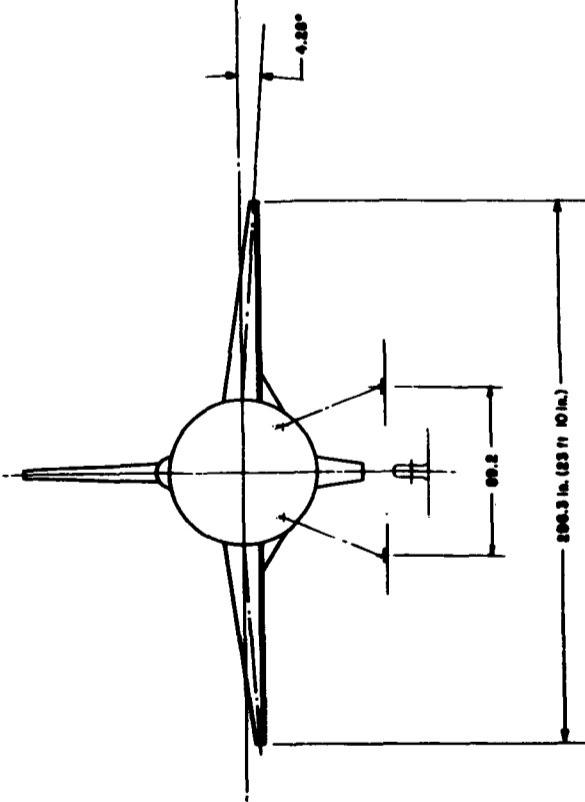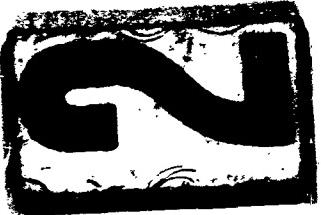
PROJECT MX ...

U.S. AIR FORCE
PHOTOGRAPHIC RECONNAISSANCE
AIRCRAFT
final stage

BELL*Aircraft* corporation

DIMENSIONS

WING SPAN (OVER-ALL)	286.3 in.
AIRCRAFT LENGTH (OVER-ALL)	896.0 in.
AIRCRAFT HEIGHT (OVER-ALL)	177.0 in.
LANDING GEAR TREAD	89.6 in.

WEIGHTS

	PHOTO RECONN.	FERRET
EMPTY WEIGHT	17,876	16,926
USEFUL LOAD	40,724	40,724
DESIGN GROSS	58,600	57,650
LANDING WEIGHT	17,433	16,483
PROPELLANTS $F_2 + O_2$ AND JP-4	39,900	39,900

CENTER OF GRAVITY

	STA.	STA.
DESIGN GROSS WEIGHT	597.4	593.3
EXTREME FORWARD	597.4	593.3
EXTREME AFT	680.6	671.0
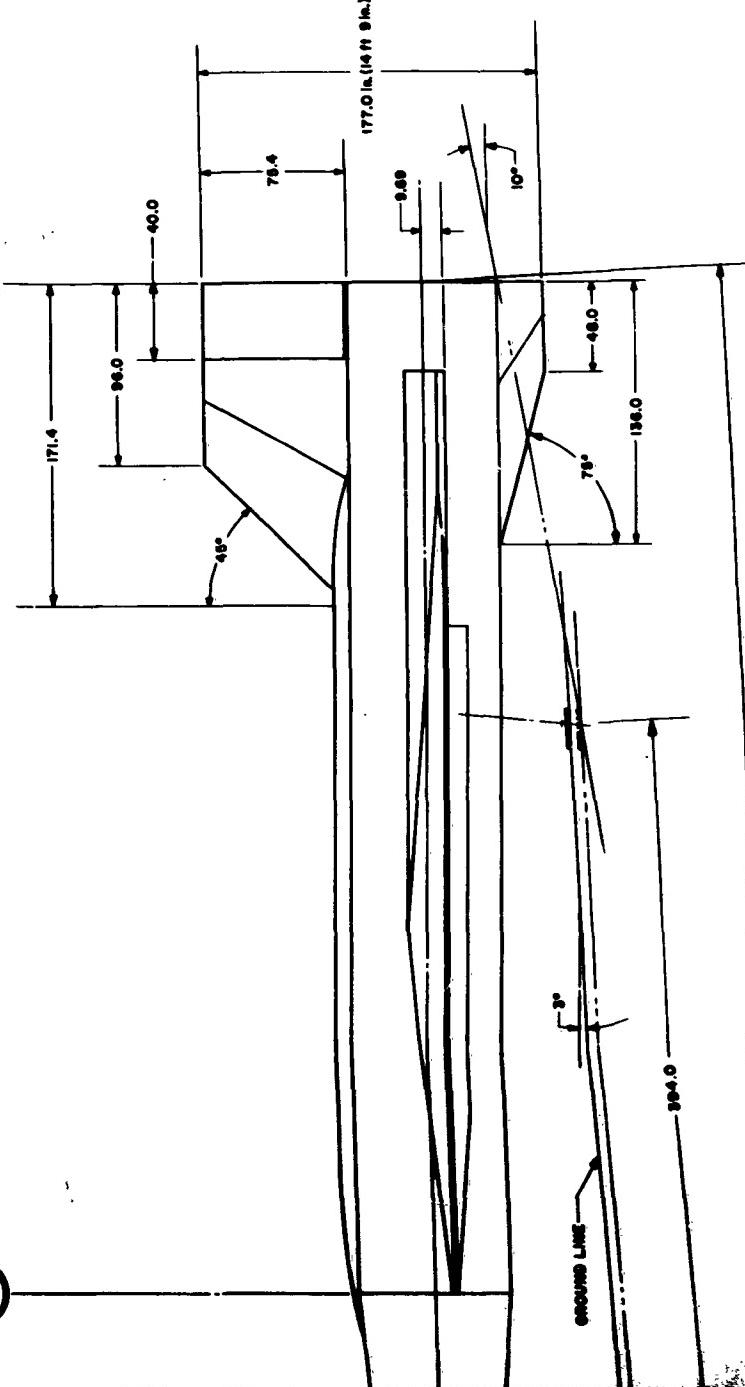
POWER PLANT

2 30,000-lb THRUST CHAMBERS $A_e/A_t = 25$
PROPELLANT F_2(70)$+O_2$(30) AND JP-4
GAS DRIVEN TURBINE PUMPS

WING

AREA (EXPOSED)	420 sq ft
AIRFOIL SECTION	MODIFIED WEDGE
INCIDENCE	0
SWEEPBACK (L.E.)	75°
ASPECT RATIO (SUBSONIC)	0.834
CATHEDRAL	4.25°

TAIL

FIN AND RUDDER	
AREA	70 sq ft
AIRFOIL SECTION	MODIFIED DOUBLE WEDGE
SWEEPBACK (L.E.)	45°
VENTRAL FIN	
AREA	15 sq ft
AIRFOIL SECTION	MODIFIED DOUBLE WEDGE
SWEEPBACK (L.E.)	75°

The fuselage is a simple cone-cylinder, with the nose blunted to ease the heating problem, but not so blunted as to increase the drag. The cone is very slender, to provide for low heat inputs as well as low drag. As discussed in Ref. 2, the nose is symmetrical to provide for a higher value of trimmed L/D and for reduced temperatures over the body in general, compared to a nose configuration drooped to provide a straight bottom line. The temperature along the bottom line of the cone is decreased, however, and it is possible that further investigation may show some degree of nose droop to be desirable.

The vertical tail configuration consists of an upper vertical surface which is completely adequate for stability and control at the lower end of the speed regime. A lower vertical surface is added to augment directional stability at hypersonic speeds, especially where much of the upper surface becomes ineffective at high angles of attack. This lower surface is jettisoned prior to landing.

Equipment.

The interior arrangements of the airplane when used for typical photographic reconnaissance or ferret reconnaissance are shown in Fig. 3a and Fig. 3b, respectively. Included also is Fig. 3c which shows that the equipment compartment has sufficient volume to accommodate a bomb containing a special warhead weighing 3000#. (A complete description of the bomb is contained in Ref. 1.) It will be seen from Figs. 3a and 3b that the two vehicles are alike except for installation of basic reconnaissance equipment. In fact, with the layout shown, a minimum amount of change is required to convert to a bomb carrying vehicle.

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1

SECTION A-A

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SECTION E-E

SECTION D-D

VERTICAL FIN

ROCKET MOTORS

ROCKET MOTORS

RADAR ANTENNA

HELIUM TANK (ARO4 TANK DOME)

LOWER FIN

BALL FILL JUNCTION PORT (ACROSS END GAP)

VERTICAL FIN

ROCKET ENGINE

LOWER FIN

EXHAUST ELECTRONIC EQUIPMENT

EXHAUST ANTENNAS

DIAGONAL FRAMES

MAIN LANDING GEAR

RADIO FACSIMILE TRANSMITTER
AUTOMATIC CONTROL & PERFORMANCE
COMPUTER
STABLE PLATFORM
RADAR ALTIMETER (FLUSH)

NO. 1 BOTTLE (AIR BORNE UNIT ONLY)
NO. 2 BOTTLE (AIR BORNE UNIT ONLY)
COMM EXTENSION BOTTLE
MAGNETIC TRANSMISSION
RADAR UNIT

SECTION A-A

INBOARD PROFILE - 2nd STAGE

D433000

VERTICAL FIN

ROCKET ENGINE

LOWER FIN

EBMET ELECTRONIC EQUIPMENT

EBMET ANTENNAS

PACKAGE FRAME

ROCKET MOTORS

DUAL HYDROTHERMAL FLIGHT CONTROL

STA

STA

STA

STA

RADAR RECEIVER TRANSMITTER
AUTOPILOT COMPUTER & PROGRAMMER
COMPUTER
STABLE PLATFORM
POWER UNIT (ELECT.)

H₂O BOTTLE (AUX POWER UNIT/100%)
N₂ BOTTLE (H₂O BOTTLE PRESS.)
FIRE EXTINGUISHER BOTTLE
HYDRAULIC RESERVOIR
POWER UNIT

MAIN LANDING GEAR
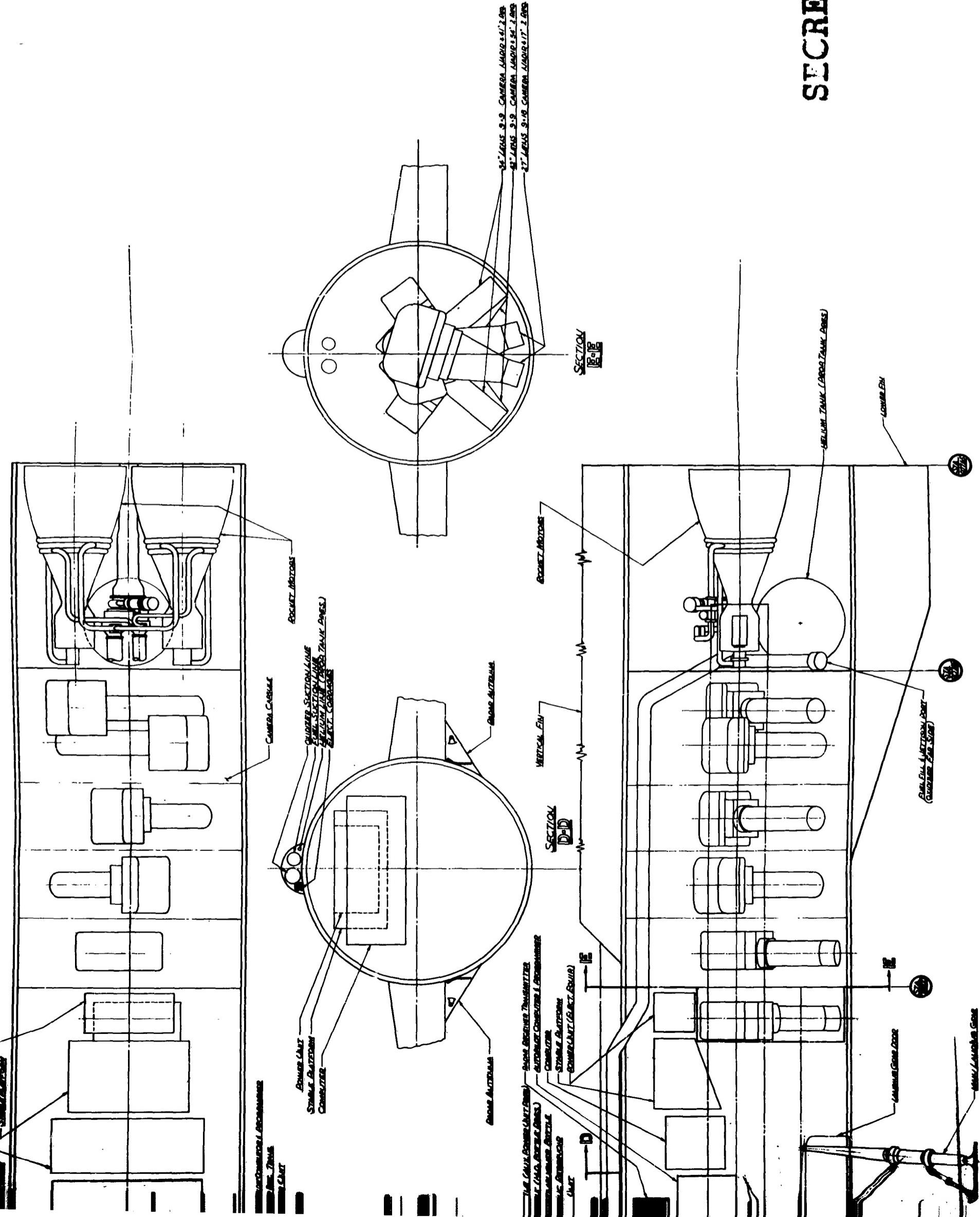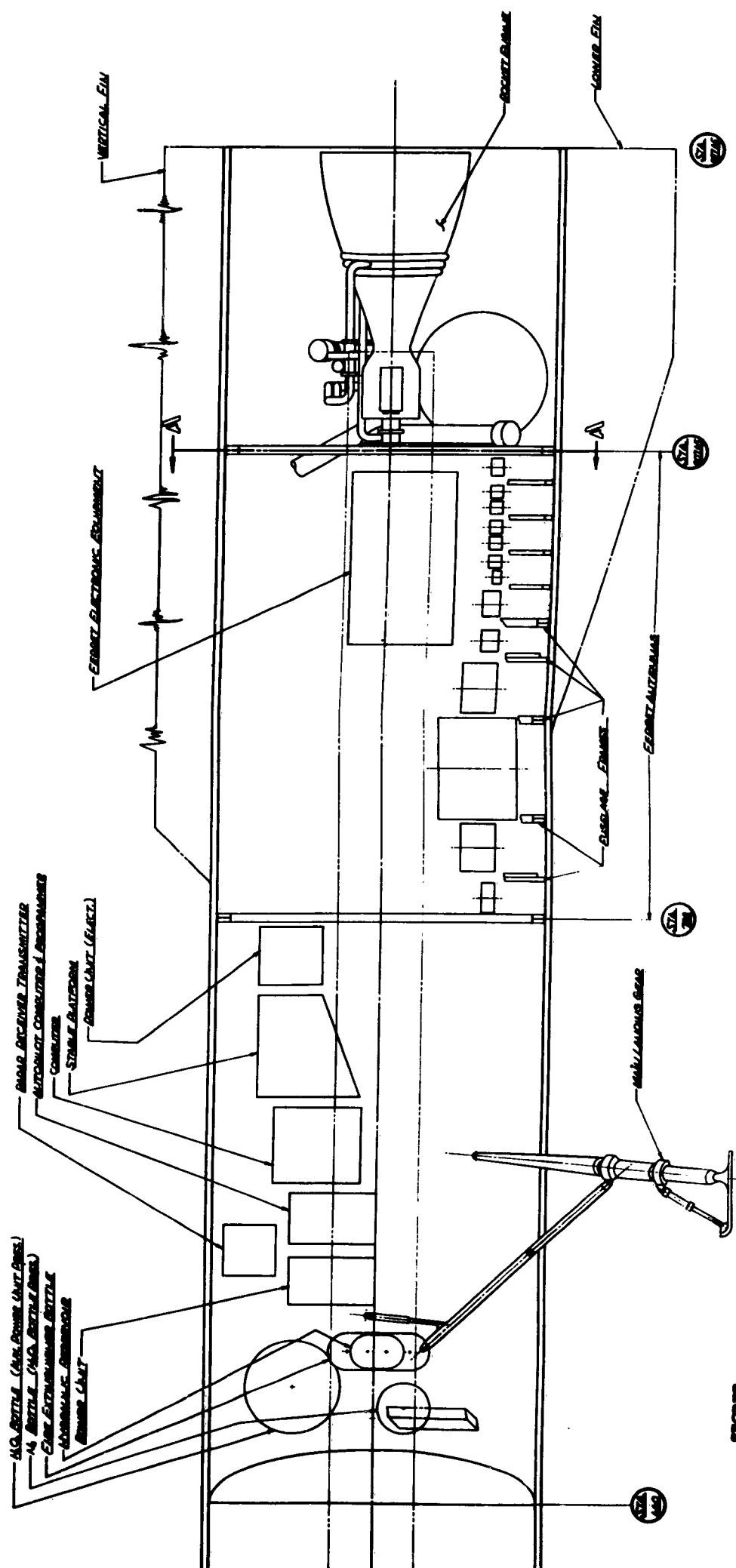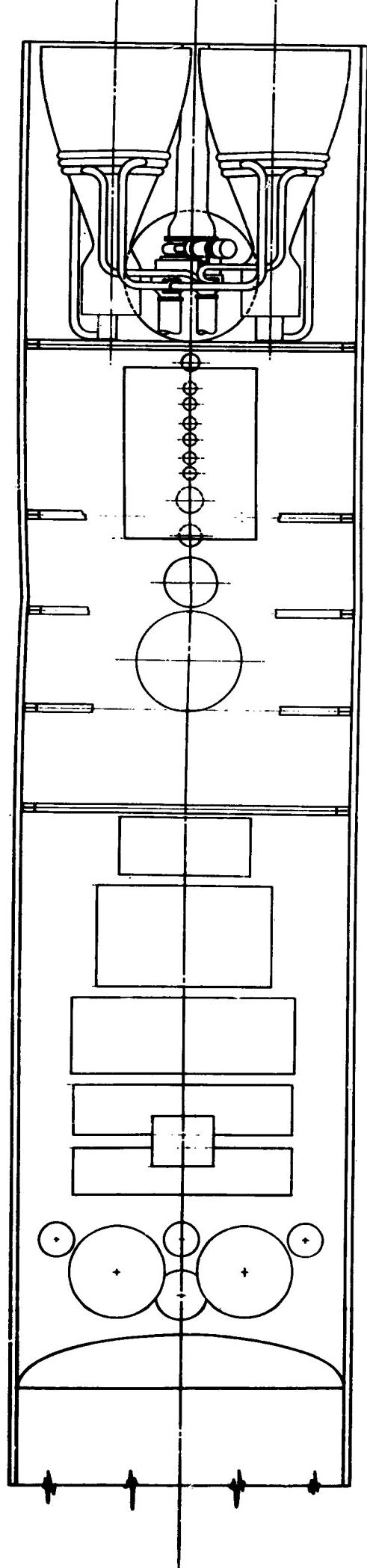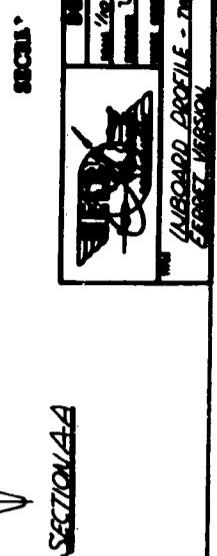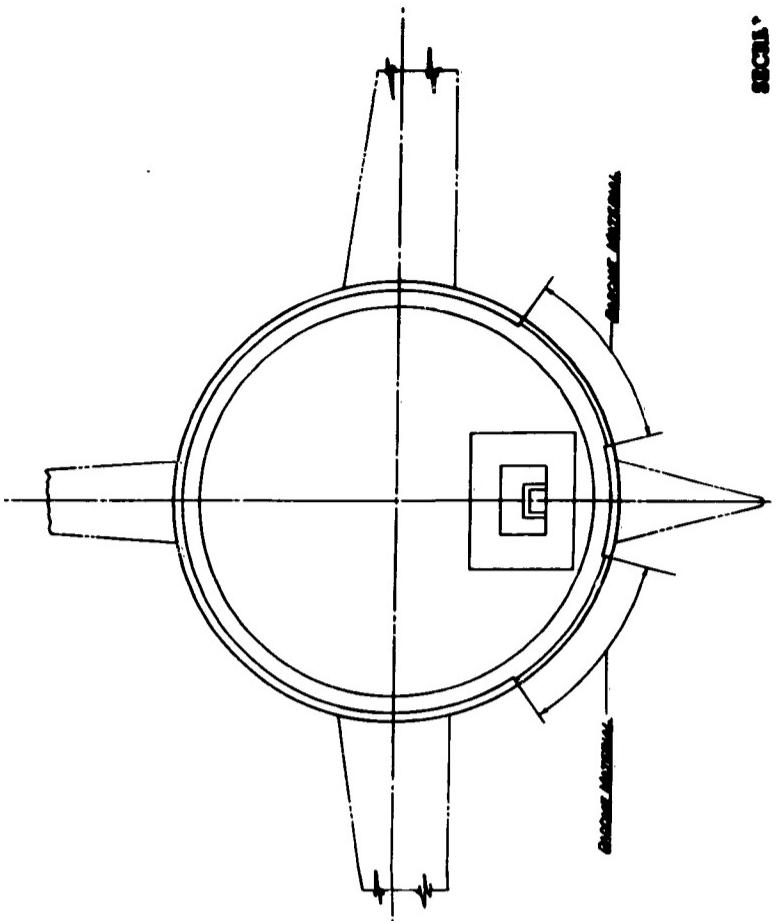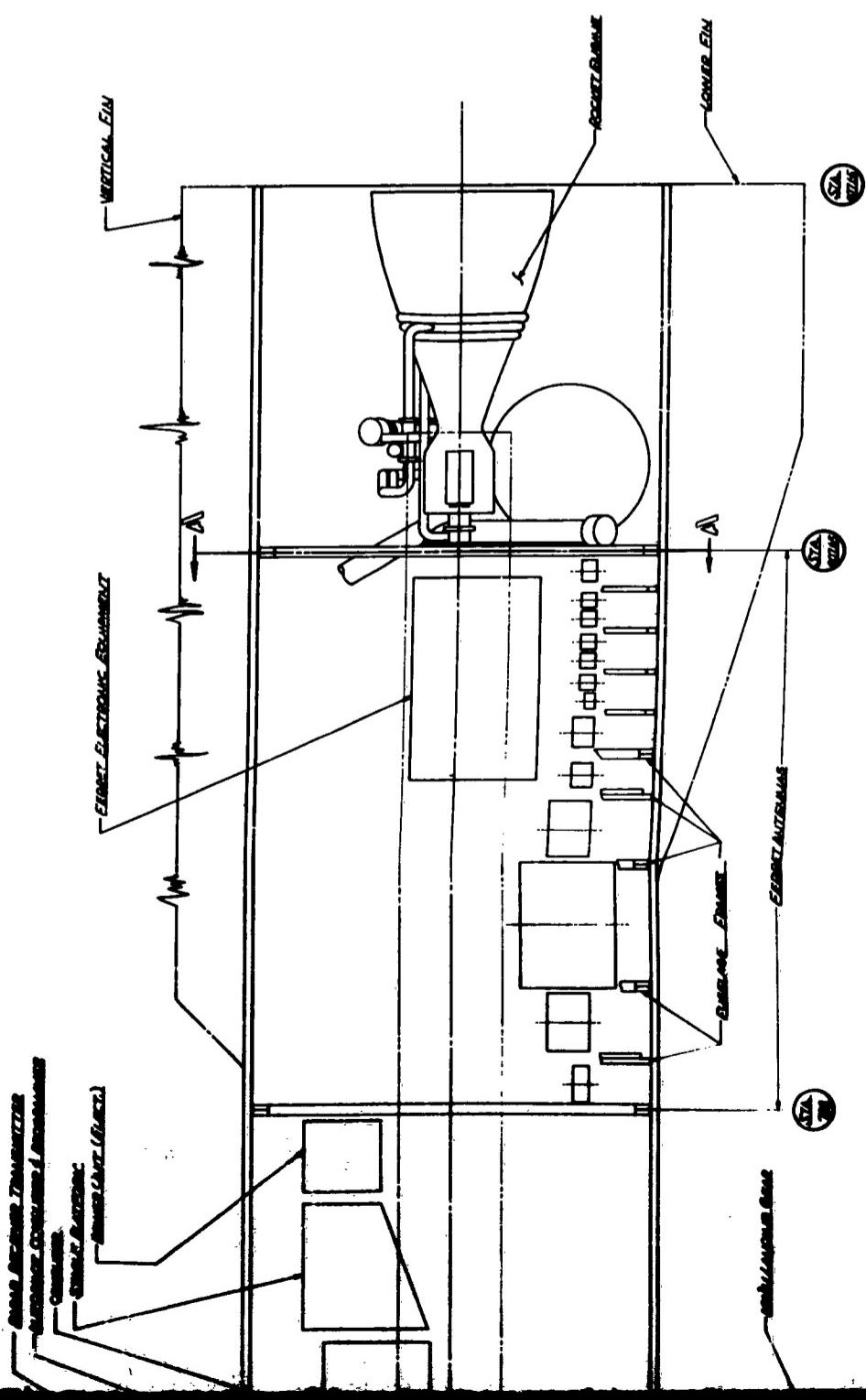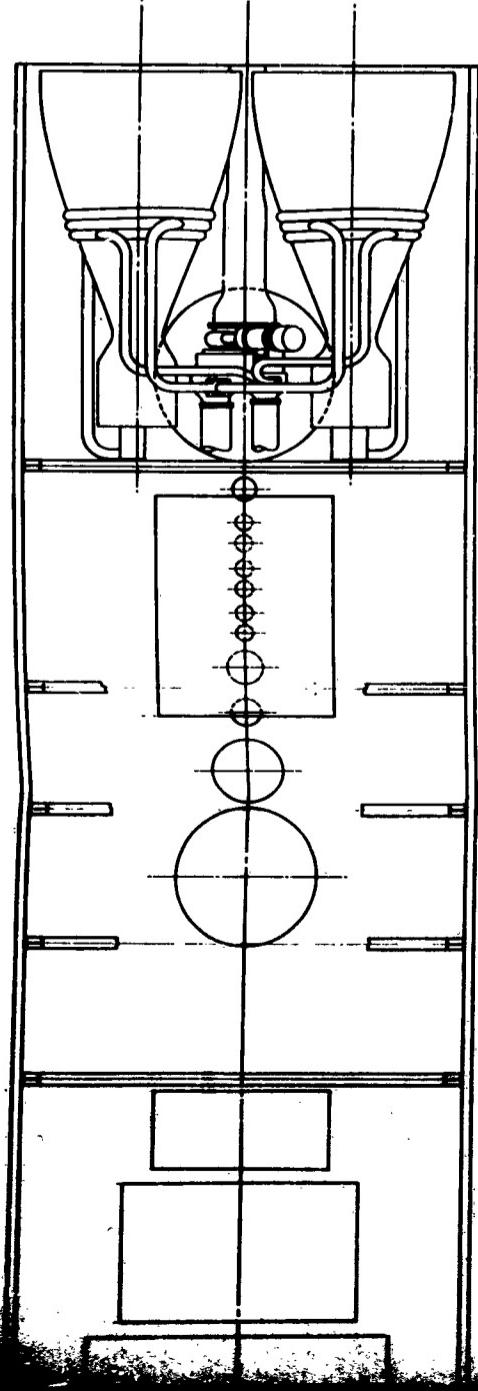
The six camera array shown in Fig. 3a meets the requirements for the search portion of photographic reconnaissance. This camera array and an alternate camera arrangement, which was recently proposed for detailed photographic reconnaissance, are shown and described in Ref. 18. The presently proposed three camera arrangement can be installed without large modifications to the airframe. Investigation of these modifications was not considered important at this time, particularly since the camera design is somewhat flexible, making it quite probable that installation of the two camera arrangements can be made completely compatible.

A description of the ferret reconnaissance equipment is contained in Section VII. It is pointed out that no single installation is capable of acquiring all the desired ferret information. The installation shown is typical; the characteristics of this system and alternate systems are discussed in the above mentioned section.

A preliminary cabin arrangement is shown in Fig. 3a to illustrate the general volume and installation requirements, and special instrumentation is discussed in Section VII. The present fuselage configuration favors a side-by-side seating of the crew, which tends to simplify equipment and instrumentation installations. Standard equipment and instrumentation was used wherever possible, including the ejection seats.

Landing Gear.

The landing gear shown on Fig. 3a is of tricycle type with a nose wheel forward and main skids aft. The basic type of gear is new, but can be shown to be statically stable. It is expected, therefore, that suitable dynamic characteristics can be obtained.

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Airplane _____ Report D143-945-02

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The maximum landing attitude provided is 10°. As shown in Ref. 2, this attitude will provide satisfactory landing characteristics.

Propulsion.

The propulsion system which is shown on Fig. 3a was taken from Section VIII, which contains a detailed description of the entire propulsion system. Ref. 19 contains a description of another propulsion system studied by North American Aviation for this vehicle.

Structure and Heat Protection Method.

The basic structural design and method of heat protection for the fuselage and wing are shown on Figs. 4 and 5, respectively, and are described in some detail in Ref. 17. The vertical surfaces are very similar to the wing, thus separate drawings for these surfaces would contribute little at this time and are not included. On the basis of previous and present studies, an insulated and cooled structure with aluminum as the primary structural material seems best suited.

A water capacity of 945 pounds is provided in the airplane. This capacity provides a margin of at least 15% under normal flight conditions.

An entirely different type of cooling system must be provided for the leading edges, because of the much higher temperatures encountered. A feasible design for a lithium cooling system which provides minimum weight is described in Ref. 17.

3. Booster.

General.

As mentioned earlier the booster configuration is about the most simple design that could be conceived. As shown in Fig. 6. it consists

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CANOPY DOORS

SEAT EJECTION RAILS

INSTRUMENT PANEL

ACCESS DOOR TO EQUIP AREA THRU WHEEL WELL

INSULATED & WATER COOLED SKIN

LANDING GEAR DOORS

SECTION A-A

ALUM ALLOY FRAMES & INNER SKIN

INSULATION

EXPANSION JOINTS

WATER COOLANT PASSAGEWAYS

INCONEL X OR HAYNES METAL

CORRUGATED INNER SKIN WELDED TO OUTER SKIN

SECTION THRU FUSELAGE SHOWING TYPICAL CONSTRUCTION & METHOD OF COOLING STRUCTURE

COCKPIT AREA (PRESSURIZED)

CANOPY

SEAT EJECTION RAILS

NOSE GEAR LAUNCHMENT SECTION

PRESSURE BULKHEAD

LANDING GEAR FITTINGS

FUSELAGE JOINT

H_2O TANK FOR SKIN COOLING

MOLTEN LITHIUM CIRCULATED IN FRAME FOR COOLING

ARC CAST COATED MOLYBDENUM

STA. 847

STA. 576

STA. 846

STA. 0

1

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SECTION D-D

SECTION C-C

SECTION B-B

2

SECTION D-D

VIEW B-B

SECTION D-D

BOOSTER ATTACHING FITTINGS

WING ATTACHING FITTINGS

OIL ATTACHING FITTINGS

LANDING GEAR FITTINGS

STRUCTURAL PANEL BEAM AREA

LANDING GEAR FITTINGS

POWER PLANT SECTION

RETRACTABLE FIN

CAMERA SECTION

MAIN LANDING GEAR FITTINGS

LANDING GEAR & EQUIPMENT SECTION

WATER COOLANT PASSAGEWAYS

PANELS APPROX. 11" LONG.

SECTION C—C
FULL SIZE

0.060"

TORQUE BOX

SHEAR ATTACHMENT
TO FUSELAGE

189.5

73°

FUSELAGE ATTACHING BOLTS

FUSELAGE FRAME

COOLANT PASSAGE

BRAZED JOINT

SECTION D—D
FULL SIZE

RADAR ANTENNA REMOVED
FOR ACCESS TO FUSELAGE
ATTACHING BOLTS

WATER COOLANT PASSAGEWAYS

INCONEL "X" OR MATCHED METAL

ALUM. ALLOY WEBS
& CAPSTRIPS

SECTION A—A
TYPICAL BEAM CONSTRUCTION
¼ SIZE

EXPANSION JOINT

ALUM. ALLOY INNER SKIN

WATER COOLANT DUCTS

DETAIL X
FULL SIZE

₵ ELEVON HINGES

THESE SURFACES MOVE IN THE
SAME DIRECTION ONLY - EITHER
UP OR DOWN.

15° UP

15° DOWN

SECTION B—B
TYPICAL RIB CONSTRUCTION - WING & ELEVON
¼ SIZE

1

SECTION B-B
L.E. CONSTRUCTION
FULL SIZE

SECTION D-D
FULL SIZE

PLAN VIEW OF WING & ELEVON
SHOWN WITH CORRUGATED SKIN AND
INSULATION REMOVED.
1/8 SCALE

WATER COOLED ALUM. ALLOY SKIN BETWEEN INSULATION
& PRIMARY STRUCTURE IN THIS AREA ONLY

FUSELAGE LATCHING POINTS

EXPANSION JOINT

LITHIUM TANK

INCONEL X OR HAYNES METL

INSULATION

ELEVON HINGES

INSULATION

WATER COOLANT PASSAGEPOINTS

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Fig. No. 6
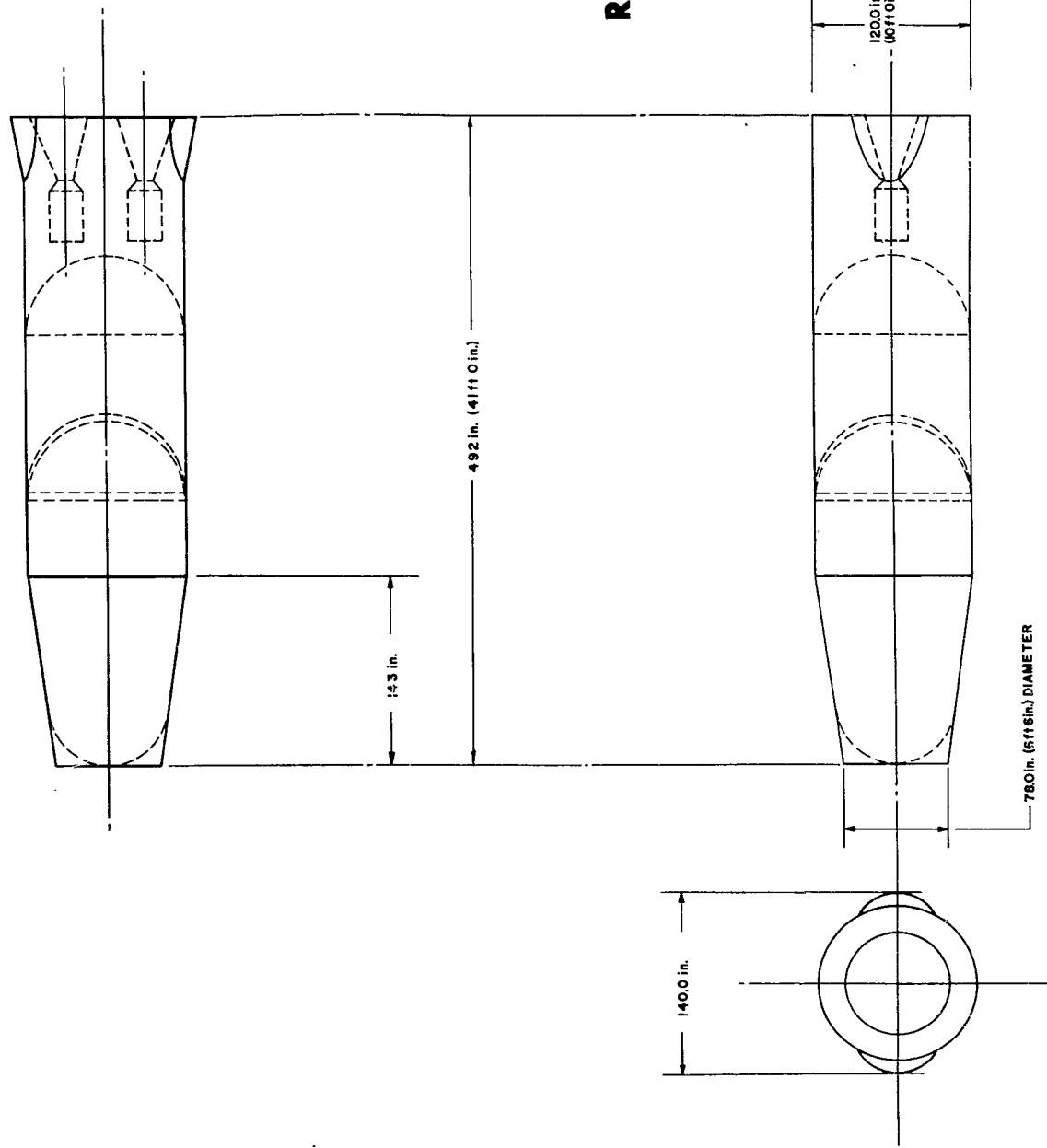
PROJECT MX-2276

U.S. AIR FORCE
RECONNAISSANCE AIRCRAFT
booster

BELL Aircraft CORPORATION

GROSS WEIGHT	142,450 lb
PROPELLANTS (TOTAL)	126,450 lb
JP-4	42,150
LIQUID OXYGEN	84,300
THRUST AT TAKEOFF	300,000 lb

492 in. (41 ft 0 in.)

143 in.

120.0 in. (10 ft 0 in.)

78.0 in. (6 ft 6 in.) DIAMETER

140.0 in.

DESIGN PROPOSAL

SCALE 1/96	DATE 11-3-56
PROJECT APPROVAL	
DESIGNED S.MURPHY	ENGINEERING APPROVAL
WORK ORDER 8744	MODEL
	DWG. NO.
TITLE THREE VIEW RECONNAISSANCE AIRCRAFT BOOSTER	D143-960-008

mainly of propellant tanks. Tank ends are hemispherical, with a nested arrangement being used between the LOX and fuel tanks to keep the length and weight to a minimum. One of the main benefits that is obtained by the use of a cylindrical, integral tank structure is that the stiffening effect of the internal pressure is substantial and consequently provides for much lower over-all structural weights. It is to be noted, however, that the booster has been made sufficiently strong so as to not require internal pressurization in ground handling or storage, or when supporting the full gross weight of the airplane in the launch attitude.

The booster experiences aerodynamic heating in the ascent. As a result it appears that some form of insulation must be provided if the primary structural material is to be aluminum. Even so, a lower total weight is obtained with an insulated aluminum structure as compared to a steel structure which would not require insulation. Several methods of insulating the structure are discussed in Ref. 17, and a comparison is made with an alternative steel structure. It will be seen also in Ref. 17 that a total weight substantially below that initially estimated, and shown in Fig. 6, may be achieved. However, this weight saving has not yet been taken into account in the design of the two stage system for two main reasons. The contingency is desirable in that it will allow for some change in either aerodynamic configuration, propellant loading, or structural design without affecting the take-off weight. In addition, if a parachute recovery system is employed, some or all of the contingency will be used in this system. The latter should include a substantial amount of structure enveloping as much as possible of the rocket motors; the function of this structure being to

absorb the landing shock and thus reduce or eliminate damage to the motors.

Propulsion System.

The propulsion system shown in very brief outline on Fig. 6 is, as mentioned earlier, a two chamber 300,000 pound thrust engine now under development. The nozzle expansion ratio is 8 to 1, which is very satisfactory for the booster under consideration, since its maximum altitude is in the order of 55,000 feet. A larger expansion ratio, say 20 to 1, would provide a higher specific impulse beyond about 30,000 ft., but the resulting over-expansion at lower altitudes would reduce the impulse. The integrated effect is that there is little to choose on the basis of impulse and consequently propellant required, but the motors with the lower expansion ratio are smaller and lighter and thus afford a real advantage.

Additional discussion on the booster propulsion system is contained in Section VIII, and a preliminary performance specification is presented in Ref. 19.

Since relatively little data has been available to Bell Aircraft on the proposed propulsion system, more detailed consideration could not be given at this time to installation details.

Equipment.

The over-all weight shown in Fig. 6 has been based on a very preliminary, but conservative, estimate. Questionable items are the tank pressurization system and the motor actuation system. As pointed out in Section VIII, lack of information of the required pump inlet pressures prevents an accurate calculation of the size and weight of the pressurization

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system. Similarly the size and weight of the motor actuation system can be determined only when the aerodynamic stability and structural flexibility characteristics of the two stage system are known in detail.

It may be noted that no auxiliary power generation problem exists for the booster since the propellant pumping system can provide ample shaft power throughout the entire booster flight, reducing electrical and hydraulic power generation to mainly installation problems.

Shipping Considerations.

The limiting dimensions for shipment of boosters in standard railway freight cars is about 10 feet in diamter and 66 feet in length. Of the two dimensions the diameter is the more important; to some extent greater lengths can be handled with special cars, but an increase in diameter would require special handling and routing techniques as well as the use of special cars. These considerations have been kept in mind and it will be seen that the booster falls within these limits, thus presenting minimum problems in shipping.

Transport Methods.

In support of the investigation of ground handling equipment and operational problems of the weapon system, a brief study was made of the applicable transport methods for the major elements of the vehicle.

Of primary interest in the study was the possibility of air transport for the airplane and booster. It was readily determined that the booster can be carried in several existing transport aircraft such as the C-130 and the C-124. The airplane stage of the vehicle presented a more difficult problem. Only very large aircraft were seen to be suitable and installation

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studies indicated that the B-36 could be used for air ferry, and the C-124 transport may be used for air transport when the lifting surfaces are removed. These aircraft are shown in Fig. 7 and 8 with the MX-2276 airplane stage aboard.

UPPER FIN

WINGS

FUSELAGE

A

A

BY _____ DATE
CHECKED _____ DATE

1
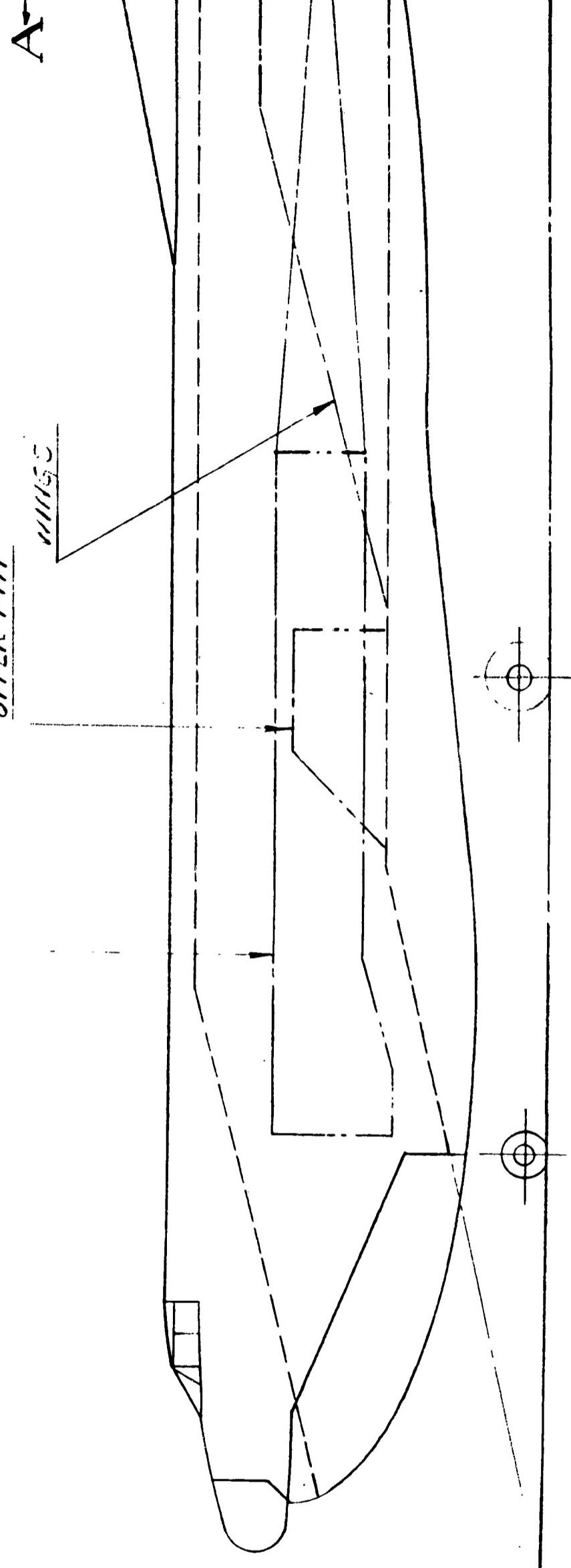
BELL Aircraft CORPORATION

BY ___ DATE ___
CHECKED ___ DATE ___

MODEL
MISSILE
AIRPLANE

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SECTION—A-A

WINGS

A →

A →

V₁
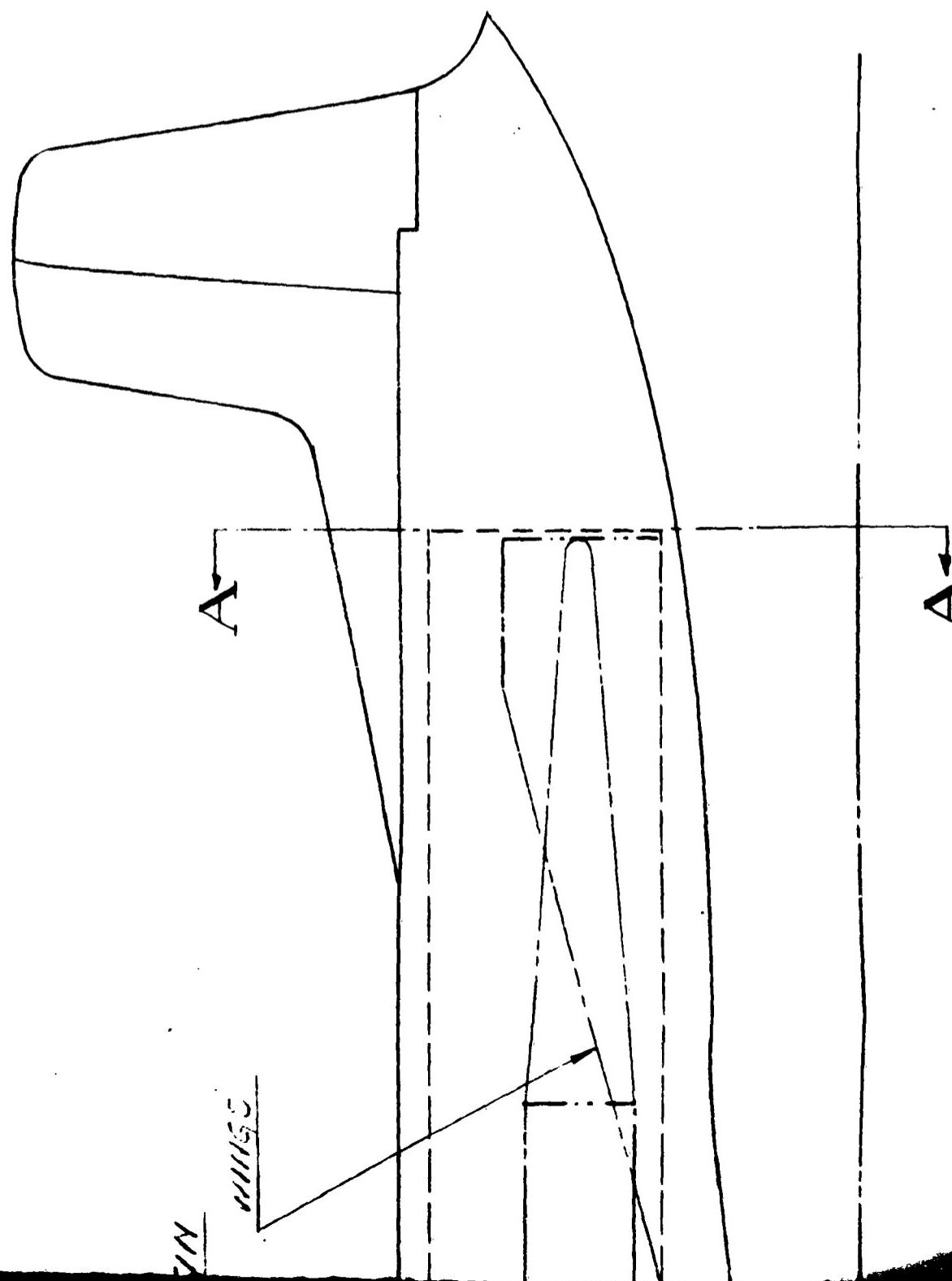
FIG. NO. 7
TRANSPORT OF THE AIRPLANE IN THE C-124

BY

CHECK'D

DA

DA

STA.
783.5

BOMB BAY ENVELOPE

STA.
383

2.06
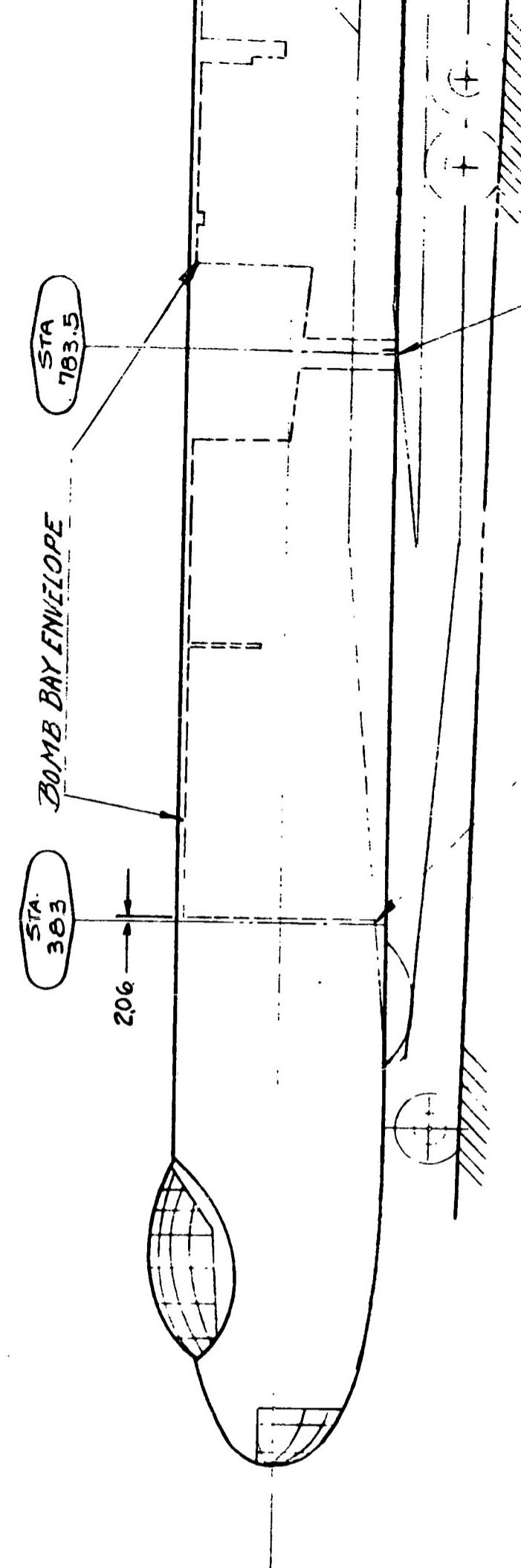
B-36 REVISED BLKHD'S
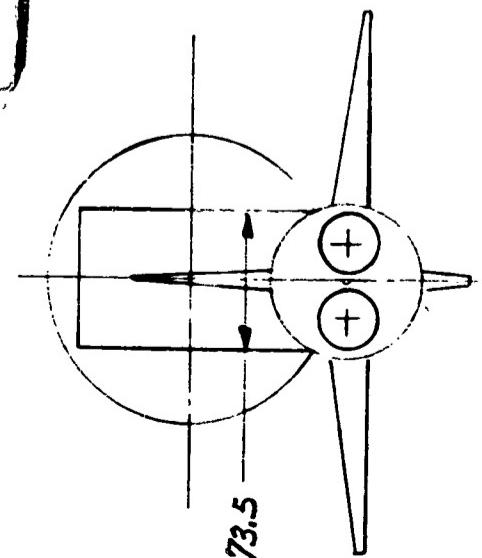
LOOKING FWD @ STA. 1184

73.5

1

BELL *Aircraft* CORPORATION

BY

CHECK'D

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MODEL
MISSILE
AIRPLANE

SECRET

STA 1184

7.06

STA 783.5

BAY ENVELOPE

GROUND LINE 1°-21'-15"

2

B-36 REVISED BLKHD'S
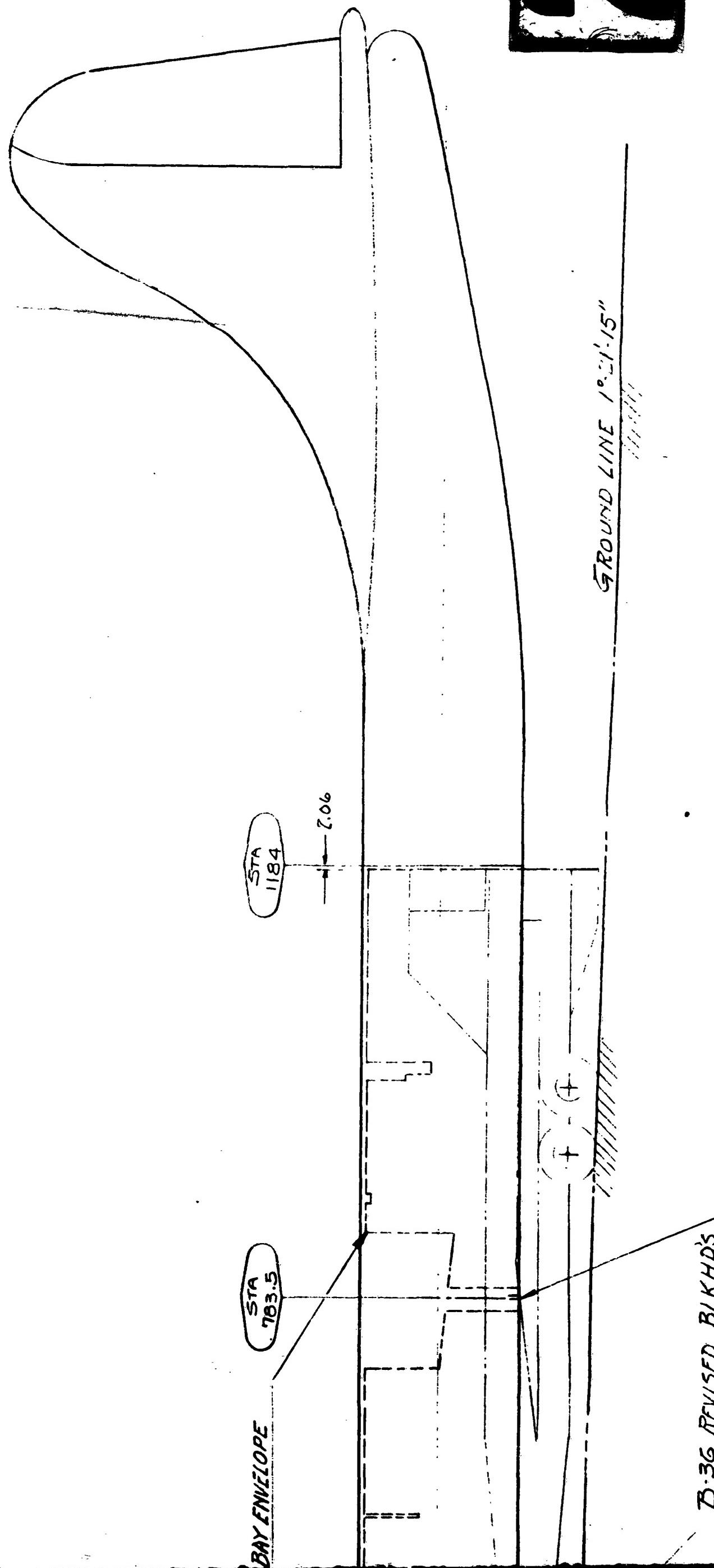
FIG NO. 8

INSTALLATION OF THE AIRPLANE ON THE B-36

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V. CREW EFFECTIVENESS

A. General.

The weapon system has been designed with the intent that its normal mode of operation will be completely automatic. However, a human crew has been included to increase the weapon system reliability and to provide the system with capabilities which cannot be achieved by an automatic system. The normal function of this crew will be to observe the progress of the mission by monitoring the operation of the automatic equipment. In order to perform this function, the proper data must be made available to the crew, and in order to improve reliability, means for implementing any action decided upon must also be provided. Such action will be necessary only if there is a malfunction of some component, subsystem, or major system. In this event, the crew has a choice of substituting either for the part or for the whole system. Through these actions the crew will improve the reliability of the weapon system.

The additional capabilities with which the crew provides the weapon system are the abilities of the human. These include the ability to assimilate and interpret data of widely different types and forms and to use judgment in the evaluation of these data. Using this information, the crew can then decide upon a course of action and initiate actions which can vary widely in their nature. A small change in dial reading may be transferred to a heavy torquing action as in the case of the control stick or control pedals. The same crew member is capable of very fine adjustments such as in radar focus or contrast.

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In order that the crew can execute the duties described above at a high level of performance, it is necessary to permit them to concentrate on these duties only. Three major factors which affect their abilities are: work load, environment, and time of concentration. The factor of work load can be controlled by providing enough persons in the crew. The environment should be as comfortable as possible to reduce distractions. Fortunately, the weapon system time of flight is relatively short.

B. Crew Functions.

The performance of the weapon system during the actual operation may be measured by two basic criteria; the accuracy with which the vehicle traverses the desired ground path, and the efficiency with which the potential energy, due to the high velocity, is transformed into range. To monitor the accuracy, the crew must have some indication of the position as it should be, as the automatic system says it is, and as it actually is. To monitor the efficiency with which the energy is utilized, an ideal flight path showing the desired relationship of velocity, and altitude with range must be provided. For comparison with this, the crew must have present velocity, altitude, and range data. Due to the high speeds, the time required for the aircraft to get into an untenable situation from either a range or an accuracy standpoint is very short. The time available for rectifying the situation is similarly short. Therefore, the crew is required to continuously monitor both criteria. By this means, not only will correction time be reduced to a minimum, but it may be possible for the crew to anticipate certain developments.

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C. One- vs. Two-man Crew.

1. General.

As previously stated, the crew is included in the weapon system to provide features of performance otherwise not attainable. This performance can only be attained if the human is provided with good working conditions. These consist, in part, of a reasonable work load, a comfortable environment and a short time of concentration. The work load can be controlled by increasing the number of the crew and dividing the work among the members. Against crew performance must be weighed the increased weight of the weapon system caused by the addition of the crew.

Examination of the system weight indicates that provisions for a crew of one increases the aircraft weight empty by 2%, and a crew of two by 3%. The resulting effect on take-off weight is 3% for one man, and 5% for two men. Thus, the addition of the crew is achieved with a relatively small increase in weight, with the second man increasing the weight appreciably less than the first.

Next, the work load must be considered. The duties outlined under "Crew Function" indicate that the work breaks down generally into two groups, and it is considered that each of these groups requires the full time of one man for maximum efficiency. Since the crew is in the system to provide high performance and two men are required to provide the performance necessary, the advantages to be gained by the addition of the second man exceed the disadvantage of additional weight. In order to illustrate the work load of each crew member, their specific functions and equipment must be considered in some detail.

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2. Pilot.

The first crew member will perform the usual duties of a pilot, combined with those of a flight engineer. This weapon system has no power plant operation during cruise, therefore, the duties of the flight engineer are considerably different from those existing in present day aircraft. The range obtained depends upon the efficiency with which the potential energy of the system is used. This energy exists in the form of aircraft velocity and altitude. Instead of the parameters of SFC, RPM, etc., the flight engineer now is concerned with angle of attack, velocity, altitude, etc. Normally, the automatic equipment will maintain the variables within the proper ratios; however, the pilot must have an independent means of check-ing the operation of this equipment. This will consist of a series of charts showing the proper velocity, altitude, angle-of-attack, lift-drag ratio, etc., as a function of range. From his displays, the pilot will obtain information for comparison with the desired values. (These displays will be positioned by information obtained from the stabilized platform. A secondary set of instruments, using aerodynamic sources, will be provided for use if the stabilized platform malfunctions.)

If the pilot detects any discrepancy in the flight parameters, he must determine the cause of the trouble and take the necessary corrective action, if possible. Whether the malfunction can be corrected or not, the pilot must decide upon the best course of procedure. Several courses of action are available; continue the present path, return to base, fly an alternate mission, fly to an alternate base, abandon the aircraft, etc. The decision involves consideration of the present position and nature of the

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malfunction, plus the distance to the desired region and the amount of turning required to reach this region. The reference charts will indicate the amount of range lost by turns at different velocities. The use of these charts will be the duty of the pilot, the determination of the distance to other sites will be the work of the second crew member.

After the decision has been made, the pilot will fly the airplane along the desired flight path in the most efficient manner possible. He will take advantage of whatever automatic equipment is still available, but will perform whatever function this system cannot handle. This will require a continual checking and rechecking of the progress of the flight.

3. Navigator.

The second crew member's major duties will involve navigation. The automatic system will navigate the airplane to the desired region and on to the designated base. The reconnaissance equipment will also be turned on, function, and be turned off automatically. It will be the second crew member's duty to monitor these functions and also to maintain a continual awareness of the aircraft's position with respect to both enemy territory and to the landing sites. In the event of a guidance error, he must be prepared to give the pilot instructions for rectifying this error. He must also be prepared to provide the pilot with sufficient information for the latter to make a decision if a deviation from the planned program is necessary either due to guidance error, or due to airplane malfunction. Once a decision to deviate has been made, the navigator must be prepared to direct the pilot along the proper route. This navigation must be performed using whatever portions of the automatic equipment are still operating correctly

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and supplementing this with information from secondary equipment such as compass, visual information and/or radar information.

In addition to this work, the navigator will be available to assist in the reconnaissance operation. Certain functions of this equipment may be more readily performed by the navigator. These include such possibilities as light-meter readings and exposure settings for the photographic equipment, and adjustment of gain, focus, or contrast on the radar.

4. Pilot Equipment.

The pilot's main source of information will be a single integrated display showing the main items of interest, velocity, altitudes, angle-of-attack, horizon, attitude, etc. During cruise, the pilot will have no information from the outside, i.e. no transparencies, periscope, or radar presentation. Flight will be entirely from instruments and communication with the navigator. This communication is enhanced by the side-by-side crew location. During the landing, hatches may be removed or dropped providing some side windows, and a periscope will be lowered for the pilot's use.

The integrated display will be operated by the automatic system using information from the stabilized platform. In addition to this information, a secondary source of data is also necessary. Since equipment for determining velocity, altitude, etc., aerodynamically is not in existence for the extremes encountered by this weapon system, a development program for this type of equipment is indicated. The information from this second source will be presented in a separate group of conventional instruments. Since there are a minimum of power plant instruments, mainly a

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group of "on-off" indicators and a switch bank, the total number of instruments should be comparable or smaller than on conventional aircraft.

There are currently several major development programs on integrated displays for the pilot. The basic information from these programs should be applicable to this vehicle even though the vehicles for which these displays are being developed have considerably different purposes. The basic flight variables remain the same although the rates and magnitudes will be different. This material should be presented in such a manner as to indicate divergence of the variables from the desired value and also as to the proper direction to move the controls for correction. These principles should be equally applicable to this weapon system.

The personal equipment of the pilot was described in considerable detail in Ref. 4 . No information has been received since the publication of that material which would indicate that any changes should be considered at this time. Development of the full pressure suit, which is the most important single item discussed in that reference, is continuing. Until such time as this becomes available, the material in that reference continues to apply.

5. Navigator Equipment.

In order that the navigator may perform his duties, he will be supplied with several presentations; a visual display from a periscope, a radar display, and/or a map display. Since the navigational accuracy of the reconnaissance system is within the capabilities of the all-inertial guidance system, the radar and visual displays are included only for information to the navigator. Automatic means for correcting the inertial

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system will not be included. These displays will all be available on a single screen or lens in order to reduce required cockpit space and for the convenience of the operator.

The map, on 70 mm film, will be drawn in two coordinates by the inertial system. It will be presented in such a way that the center of the display will be the aircraft and the centerline of the display will be the direction in which the aircraft is moving. The display will be large enough that the operator will be able to note points of interest which are about to be passed and he can then use the visual or radar presentation to determine whether or not these points are actually flown over. In addition to this map, other more conventional maps will be available in the event the primary map does not function correctly. These maps will include the entire area about the region of flight and contain sufficient information for use in dead reckoning.

The visual presentation will, desirably, cover the region which the reconnaissance system "sees." This is possible with both the photo and radar functions, but not the ferret missions. In order to cover such a large area as necessary, even for this photo and radar, some amount of minification will be required. However, the periscope will contain the equipment necessary to provide magnification, at the expense of reduced field-of-view, for specific areas of interest. Thus the navigator can examine interesting regions which he may encounter. Normally, however, the optical system would function both as a periscope and a viewfinder. The visual presentation will provide a high resolution view of the terrain during good weather flights. In bad weather, the radar information will be available for use.

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The radar presentation, since it is a side-looking radar, will initiate at the center of the other displays. In order to obtain as much information as possible, it may be necessary to move the display forward with respect to the other displays, so that points of interest will be visible longer. A discussion of how this display will be achieved is contained in Section VII.

The operator will be able to change the selected presentation by merely turning a knob or pushing a button.

D. Crew Environment.

The considerations of the crew environment for this vehicle remain the same as for the crew of the third stage of Ref. 4. Although the reconnaissance vehicle operates at lower speeds and altitudes and has a shorter time of flight than the vehicle in Ref. 4 the basic problems remain the same and the proposed means of solving these problems remain unchanged also. Since the environment was discussed rather extensively in that reference, the material will not be repeated here. Additional discussion of the survival problems of both systems are presented in the Supplement to this report.

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VI. EFFECTS OF TARGET AREA AND OPERATING BASE LOCATIONS ON WEAPON SYSTEM PERFORMANCE

A. Summary.

An analysis was conducted to determine the extent of Soviet terri-
tory that can be covered by reconnaissance flights involving the two-stage
MX-2276 weapon system. Major attention was directed toward the launching
and landing sites to be employed, possible flight paths, and the area cover-
age obtainable. Consideration was also given to the number of flights re-
quired, the effects of cloud cover, and the minimum system range necessary
for complete coverage of the USSR.

B. Launching and Landing Sites.

In this analysis, consideration of the reconnaissance capabilities
of the MX-2276 weapon system was restricted to the USSR, and did not include
Red China, North Korea, or the European satellite nations.

Geographically, the USSR is appreciably shorter in a north-south
direction than it is in an east-west direction. Because of this fact,
various airbases have been chosen, in both hemispheres, to allow flights
that are generally north-south in direction. For photographic or visual
observation, it is also desirable to limit the east-west distance traversed
on a flight; otherwise, varying daylight conditions along the way may have
detrimental effects on the data-gathering processes.

The following bases have been selected for consideration: Launch-
ing Sites: Keflavik, Iceland; Thule, Greenland; Fairbranks, Alaska; and
Limeston AFB, Maine. Two additional sites in the US have also been chosen,

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one in northern Minnesota and the other in northern Washington. These two
locations are general in nature, without regard to existing facilities in
their vicinities, and are intended to give an estimate of coverage possible
with stateside bases. Landing Sites: Ankara, Turkey; Tehran, Iran; Dharan
and Al Jazir, Saudi-Arabia; Karachi, Pakistan; Tokyo, Japan; and Okinawa
Island. Some of the above bases may be unfeasible because of local terrain,
difficulties of logistics or defense, or political questions. They are
nevertheless adequate for indicating the coverage that can be expected with
a two-stage system.

C. Flight Paths.

Except for flights which follow great-circle routes, it is assumed
that maneuvers can be performed by the airplane during the glide phase.
When the airplane makes a turn, however, the decrease in altitude and vel-
ocity are greater than would be the case if the same distance along the ground
were traversed on a straight (great-circle) path. Consequently, if Coriolis
effects are neglected, a flight path that involves maneuvers necessitates a
longer-range system than one that does not. Through the employment of aero-
dynamic turning data in the analysis, it has been possible to evaluate the
effects of maneuvers on the range required, and to determine the limits on
the maneuvers that are permissible with the present two-stage system. These
maximum maneuvers, in turn, determine the regions that can be covered by
flights involving a given combination of launching site and landing site.

The turning data employed in this study (Ref. 2) correlate dis-
tance with velocity, such that, when the airplane performs a particular

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maneuver throughout its glide phase, the position (down range and off range) and velocity of the airplane are obtainable from these data for any point on the flight. According to these data, the velocity of the airplane at the initiation of glide is 16,600 ft./sec., and the great-circle range (without maneuvers) to zero velocity is 4600 miles. A nominal value of 300 miles has been assigned to the distance traversed along the ground during ascent and, in general, a flight has been considered as terminated when the velocity dropped to 4000 ft./sec. This occurs about 250 miles before the point of zero velocity is reached. The useable great-circle range of the system is therefore 4650 miles from launch to landing.

Subsequent to the completion of this analysis, certain refinements in the aerodynamic data have resulted in some modifications of the above flight characteristics of the airplane. Notably, the system range has been reduced somewhat. However, by allowing some flights to terminate at zero velocity instead of at 4000 ft./sec., the coverage capabilities of the system remain virtually unchanged. With this variation in terminal velocity wherever necessary, the accuracy of the results of this study is not therefore affected substantially.

The great circle connecting the launching and landing sites is the equator of the navigation coordinate system. By launching at some angle to this great circle, then executing whatever maneuvers are necessary to land the airplane at the desired location, the areas along the ground adjacent to the great circle can be observed. For the sake of simplicity, only three general types of these flight paths are considered (Fig. 9):

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Date _____

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Model _____ Page ___45___

Missile
Airplane _____ Report _D143-945-027_
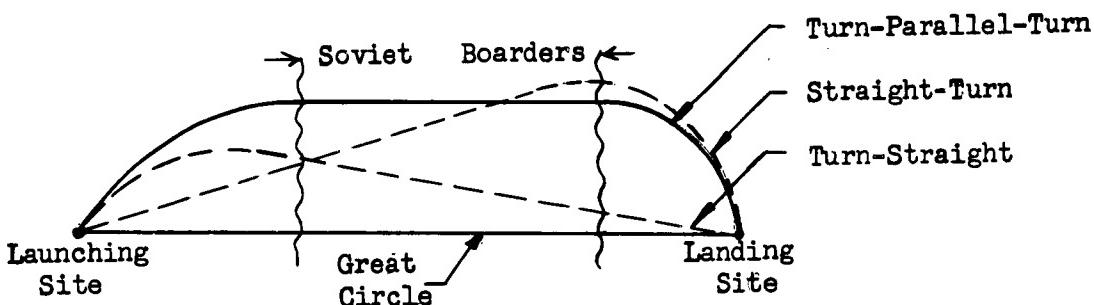
FIG. 9: Types of Flight Paths

1. "Turn-parallel-turn" - when the airplane has attained its maximum altitude, a turn is initiated and continued until the airplane heading is parallel to the great-circle route. This parallel course is maintained until the final portion of the flight, when the airplane begins another turn to bring it to the landing site.

2. "Straight-turn" - after launch, the airplane follows a straight course until the final portion of the flight, when it begins a turn to bring it to the landing site.

3. "Turn-straight" - when the airplane has attained maximum altitude, it makes a turn, then continues on a straight course to the landing site.

A comparison of the coverage obtainable with these three types of flight paths shows that the first (turn-parallel-turn) gives the best general coverage and may find advantageous application in an extensive reconnaissance program. The area to be observed can be divided into strips parallel to the great-circle path. The width of each strip may be determined

from the distance along the ground, normal to the flight paths, that is to be covered on the respective flights. In this manner, any desired degree of overlapping between the strips covered on adjacent flights may be easily obtained.

The tolerances of the navigation system employed may impose restrictions on the maximum allowable maneuvers of the airplane. Two such restrictions have been considered.

1. The airplane velocity component normal to the great circle may not exceed 5000 ft/sec at any time during the flight.

2. Because of the lateral-velocity limitation, the angle between the great circle and the airplane heading at launch may not exceed about 18°.

Either of these restrictions is sufficient to limit the "turn-straight" type of path to small deviations from the great-circle route. The "straight-turn" type of path employs a small initial heading angle but necessitates a sharp (0.5g) turn at the end of the flight, with a consequent excessive normal velocity component. This component may be reduced by the use of a shallower turn, but at the penalty of less coverage, since the airplane must then fly closer to the great-circle path or begin its turn at an earlier point on the flight.

An additional restriction that can be imposed on maneuvers involves the airplane roll angle. Because of structural limitations, the cameras mounted in the airplane are rather inflexible directionally. Consequently, it is desirable to minimize both roll angle and changes in roll angle while photographing enemy territory, and to perform all necessary maneuvers, especially high-bank turns, during those portions of the flight for which

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photographic data is not sought; i.e., while the airplane is over friendly or **neutral** territory. This restriction serves to reduce further the usefulness of the "straight-turn" type of flight path insofar as general coverage is concerned.

D. Area Coverage

The great-circle distances between the stateside launching sites considered and the nearest landing sites are listed in Table I below:

TABLE I - DISTANCE BETWEEN US AND OVERSEAS BASES

Limestone to Ankara	-	4200 nautical miles
Limestone to Tehran	-	4900 nautical miles
Minnesota to Ankara	-	4900 nautical miles
Minnesota to Tokyo	-	5000 nautical miles
Washington to Tokyo	-	4200 nautical miles
Washington to Okinawa	-	5000 nautical miles

From this list, it can be seen that only two pairs of these bases, Limestone to Ankara and Washington to Tokyo, are within range of the present two-stage system. These pairs, at 4200-mile ranges, allow fairly wide maneuvers to be made, but restrict all flights to the extreme eastern and extreme western portions of the USSR, with a total area coverage of about 15% of the Soviet territory. The use of other launching sites within the US will do nothing toward improving this figure. Since, furthermore, flights from overseas launching sites are capable of duplicating this area coverage as well as bringing vastly more territory under observations, the employment of stateside bases has been eliminated from consideration.

The following table lists the great-circle distances between the various overseas bases:

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TABLE II: Distance Between Overseas Bases

Launching Sites

		Thule	Keflavik	Fairbanks
Landing	Ankara	3300	3000	4400
	Tehran	3700	3000	4700
	Dharan	4300	3300	5300
Sites	Al Jazir	5000	4700	5600
	Karachi	4500	4000	5200
	Okinawa	4600	5200	3800
	Tokyo	4000	4700	3000

Distance in Nautical Miles

It can be seen that some of these pairs of bases result in ranges greater than 4650 miles, and are therefore incompatible with the present system. Other pairs require ranges that are at or very near this limit; the usefulness of these pairs is restricted by the fact that only small maneuvers, if any, can be performed along such routes. In the course of investigating the coverage obtainable with various combinations of the above airbases, it is also found that some duplications of coverable areas occur with certain pairs of sites. A comparison of the coverage obtainable in each case has led to the elimination of Ankara, Tehran, and Al Jazir entirely, and to the limitation of Fairbanks to flights over the eastern Siberian regions of the USSR. In this manner, the total number of bases required is minimized without penalizing the reconnaissance capabilities of the system.

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By employing Keflavik, Thule, and Fairbanks as launching sites, and Dharan, Karachi, Tokyo, and Okinawa as landing sites, it is possible to observe about 50% of the land area and about 80% of the key cities of the USSR (see Fig. 10). The region west of 70° E. longitude can be covered almost completely. This area contains the preponderance of Soviet industry, population, and commerce, and would be of prime concern in an operational reconnaissance program. In the eastern Soviet territory, coverage is not nearly so complete. The Pacific coastal regions can be observed fairly well, but the inland coverage is generally poor. To the north of China, vast stretches of land are inaccessible to the two-stage system, nor can the situation be improved appreciably by employing airbases other than those chosen.

From the standpoint of coverage alone, two-way flights (each base can be used for both launching and landing) seem to offer no advantage over one-way flights. Unless the restriction on roll angle variations while over enemy territory is relaxed, the coverage possible with south-to-north flights is actually less than that obtained with north-to-south flights. This is so because, in general, the landing sites indicated above are closer to the Soviet border than the launching sites; and a south-to-north flight path would allow a shorter distance in which to complete maneuvers immediately after launch. To realize the same coverage as with the corresponding north-to-south flight, it would therefore be necessary to complete the post-launch turn over enemy territory in some instances.

Because of the additional aerodynamic heating problems associated with high-g turns, especially through large turning angles, it is desirable to employ the shallowest maneuvers that are compatible with the maximum

Figure 10. Coverage with Two-Stage 4650-Mile System

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possible coverage. This has been done in the above analysis wherever it is feasible. The sharpest turns considered are 0.5g, and shallower turns (0.4 or 0.3g) have been preferred wherever it is possible to do so without penalizing the over-all coverage obtainable.

E. Number of Flights.

The cameras mounted in the airplane have a total look angle of 120° across the direction of flight. This angle subtends a distance of 86 miles along the ground when the airplane is at an altitude of 150,000 ft., and at zero roll angle. Any two consecutive pictures from a given flight duplicately cover some area, which is adequate to insure accurate matching of those two pictures. By properly combining all the pictures from one flight, it is therefore possible to obtain a composite photograph of a continuous strip about 86 miles wide along the ground. Actually, the width of this strip varies somewhat because of changes in altitude along the flight path. But a constant width of 86 miles may be assumed with little error.

If each reconnaissance flight were successful, and weather conditions permitted clear photographing of the ground at all times, a minimum of about 22 flights would yield complete coverage of the observable areas in the western portion of the USSR. With regard to points of launching and landing, these flights may be scheduled in various ways. The following table shows two such schedules, (a) and (b), of the many that are possible:

TABLE II - SAMPLE FLIGHT SCHEDULES

	Dharan		Karachi	
	(a)	(b)	(a)	(b)
Keflavik	6	7	1	7
Thule	7	1	8	8

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Thule	7	1	8	8

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Schedule (a) totals 22 flights; schedule (b), 23. These involve only the "turn-parallel-turn" type of flight path, with all maneuvers being completed outside the USSR, and do not exceed the limits imposed on initial heading angle (18°) and normal velocity component (5000 ft./sec.).

The above figures assume that the data gathered on each flight will not be duplicated on any other flight involving the same combination of air-bases; i.e., that there is no overlapping of the composite strips covered by adjacent flights. In an operational program overlap will be desirable to match and correlate the data obtained on the various photographic missions. Without overlap, it is also possible for a slight error in the position or attitude of the airplane over Soviet territory to result in a lack of coverage of some small areas. The use of overlapping techniques can therefore obviate this discontinuity in coverage.

The following example serves to illustrate the manner in which the concept of overlap is interpreted and applied in this analysis:

The first flight from Thule to Karachi involves the greatest deviation westward from the great-circle route between these bases. The second flight lies closer to the great-circle, and is so programmed that the area along the eastern side of the strip covered on the first flight is duplicated by the area along the western side of the strip covered on the second flight. The third flight is closer yet to the great-circle, and a similar duplication of covered area occurs with the second and third flights. The area of the strip covered on the second flight is therefore overlapped along both sides: along the western side by the first flight, and along the eastern side by the third flight.

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The overlap factor is the ratio of the width of the area, along one side of a covered strip, that is duplicated by an adjacent flight, to the total width (86 miles) of the covered strip. Thus, an overlap of 25% (see Fig. 10) means that, on the first and last flights, 25% of the area

Overlapped
Areas

| 1st Flight | | 3rd Flight |

| 2nd Flights |

Overlapped
Areas

Direction
of
Flights
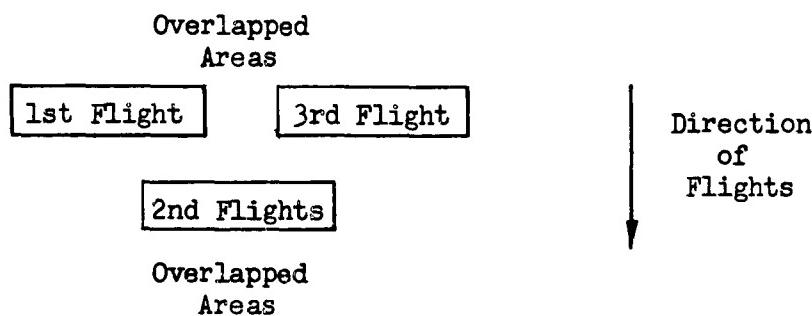
FIG. 11: Illustration of 25% Overlap

covered on each of these flights is also covered on adjacent flights; on any of the other flights, 50% of the total area covered on that flight is duplicated by adjacent flights, 25% along each side.

The net effect of overlapping is an increase in the total number of flights required for each pair of bases:

$$N' = \frac{N-r}{1-r}$$

where

N' = number of flights required when overlapping occurs,

N = number of flights required without overlapping,

r = overlap factor, defined above.

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N is determined by dividing the over-all width of the area to be covered with a particular pair of bases by the width of the strip (86 miles) observable on each flight. The figure of 10% overlap (r = 0.1) has been suggested for a general reconnaissance program. On this basis, about 9 flights from Thule to Karachi would be required to cover the same area that could be covered by the 8 flights previously considered without overlapping.

If radar mapping of Soviet territory is to be considered without photographic coverage on the same flights, a minimum of about 19 flights is required over the western portion of the USSR. This assumes that each flight is capable of providing continuous coverage of a strip 100 miles wide along the ground, and that no overlapping occurs. It also assumes that any weather conditions encountered on these flights have no adverse effects on the functioning of the radar equipment. The flight paths are of the "turn-parallel-turn" type and, except for the strip width, the treatment and results of overlapping are the same as for photographic coverage.

F. Cloud Conditions.

If it is assumed that a cloud cover of more than 2/10 over any portion of the area to be photographed on a given flight results in an unsuccessful reconnaissance mission, then the number of flights required for complete coverage of the observable western Soviet territory is in the order of 200. This value has been arrived at through a consideration of probability estimates* of cloud cover over the USSR, and makes no allowance for overlapping of data from adjacent flights.

* These probability estimates are presented in BAC Report D161-945-002 (March 1954)

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In general, undesirable cloud conditions are more prevalent during the winter than during the summer. In an operational program, the above figure of 200 could be reduced somewhat by the proper scheduling of flights. In this manner, seasonal variations in cloudiness could be utilized advantageously to provide the highest possible probability of success along a given path. Also, any knowledge of the weather conditions over a given area on a particular day could be employed. For example, flights over an area known to be cloudy could be rerouted to areas known to have clear skies. The use of such techniques could bring about a further reduction in the total number of flights required.

The effects of overlapping data from adjacent flights are the same as indicated previously.

G. Range for Total Coverage of USSR.

With the two-stage system, coverage of the western USSR is rather complete. This, however, is not true for the eastern portions of the country where, except for the coastal regions which can be covered fairly well by flights to Tokyo and Okinawa, only a small part of the total area can be observed.

A study has been conducted to determine the minimum system range required to obtain complete coverage of the USSR. Fig.12 shows the coverage obtainable with a 5500-mile system. It has been found that the portion of Soviet territory to the north of China that is beyond the range of the 5500-mile system can be covered by a 6000-mile system. Therefore, a range of between 5500 and 6000 miles is necessary to open the entire USSR to aerial

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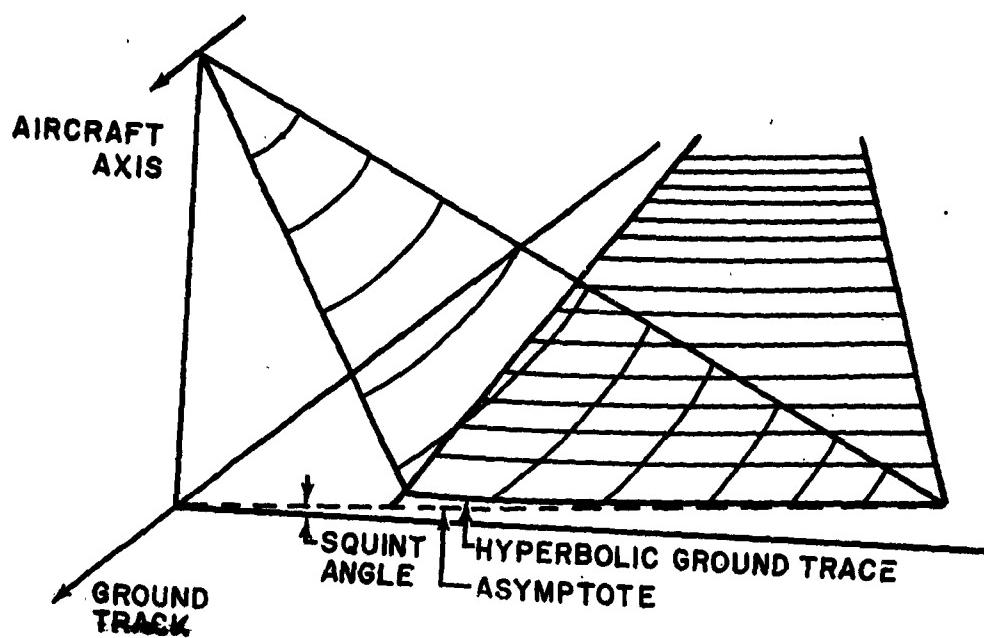
Figure 13a. Ground Intersection of Radiation Beam with Airplane in Level Flight
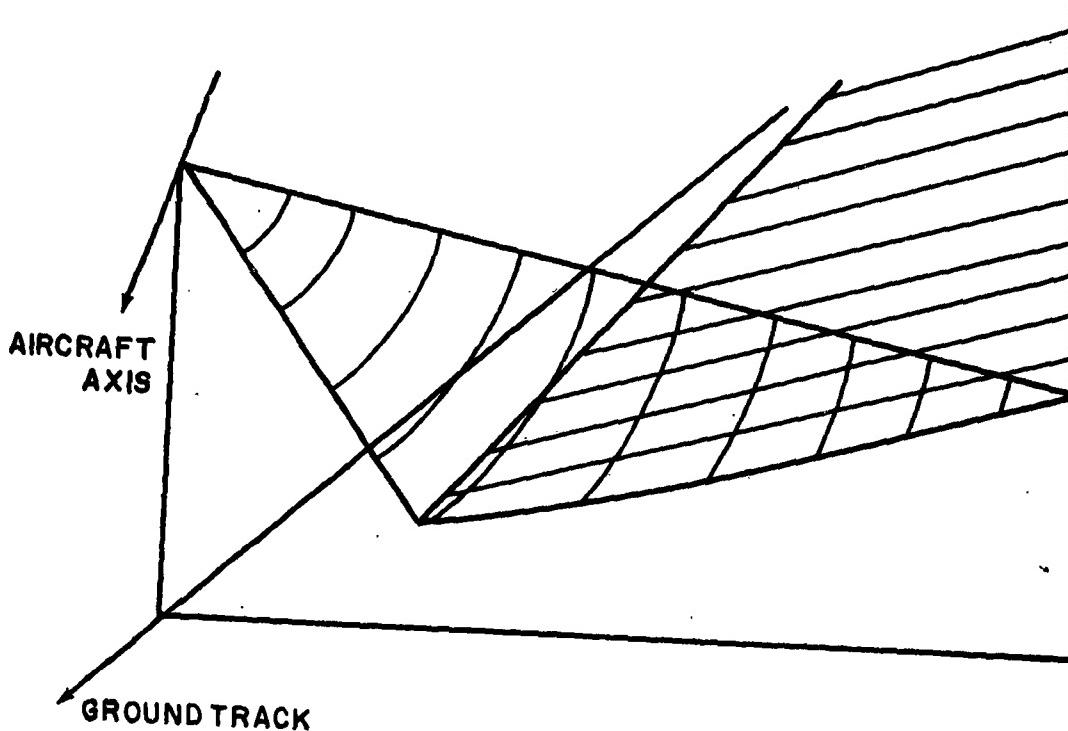
Figure 13b. Ground Intersection with Yaw, Pitch and Roll

Date _____

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Airplane _____ Report __D143-945-027

reconnaissance with the airbases considered. These figures cannot be

bracketed more closely by the methods of analysis employed, nor has it been

deemed advisable to conduct a longer and more complex study at the present

time for the purpose of arriving at a more precise value of this minimum

system range.

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VII. RECONNAISSANCE SYSTEMS

A. Radar Reconnaissance.

1. General.

Radar reconnaissance should provide useful information for present and future navigation, guidance and bombardment systems. Successful pursuit of this objective indicates two functions:

a) The radar must detect previously unknown points of interest. These may be actual targets, or guidance and navigational checkpoints.

b) It should locate these and other points as accurately as possible with respect to some known coordinate system or point.

Because of the high antenna gain and excellent resolution available from a side-looking radar, this type is preferred for the reconnaissance vehicle. The radar chosen in the previous one-year study period (Ref. 5) is well suited to reconnaissance purposes. The bombardment requirements of high resolution and accurate presentation satisfy requirements (a) and (b), respectively. Other radar types were considered but were not deemed as acceptable for an aircraft with the MX-2276 speed and altitude perform- ance. For a complete study of possible radar configurations the reader is referred to Ref. 5.

One feature of the sidelooking radar should be investigated further; it illuminates a given target from only one aspect angle. In this way it differs from a mechanically rotated aperture which illuminates every target from a variety of angles as the aircraft approaches and recedes. The degree of deterioration due to this has not been established, and in fact, may be

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negligible. It appears unlikely that target detection will be a sufficiently sensitive function of aspect angle as to allow certain targets to go un-noticed by the side-looking radar. Shadowing will be different than for a PPI; this might require reduction of the side-looking data into a form more acceptable to navigators using a PPI presentation over the area. However, it is quite likely that a navigation or bombing system will use a number of strong checkpoints rather than terrain detail. The side-looking radar sup-plies accurate location of these points combined with a degree of resolution unattainable by a rotating antenna.

System design problems which were not covered in the previous study are discussed in the following paragraphs; additional information on previous studies is included.

2. Data Stabilization.

If the radar map is to be an accurate representation of the terrain scanned, provisions must be made to compensate for aircraft pitch, roll and sideslip angle. It is readily visualized that a plane of radiation perpendi-cular to the aircraft's longitudinal axis will intersect a plane earth in a straight line. Yaw or pitch of the aircraft will change this intersection line. If the radiation fan is displaced from perpendicular to the aircraft axis by a small amount, roll in addition to yaw and pitch will move the intersection. Movement of this trace has two effects; it illuminates a different section of the ground than what would be covered if the aircraft were not maneuvering, and it causes the radiation intensity at the ground to be greater or less than the proper amount for uniform illumination. The

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first effect will give a distorted map if the display is not properly compensated; the second will result in a loss of radar coverage for some areas with over-bright returns from others.

The radiation pattern of a linear slot array as proposed for this vehicle is a section of a conic surface squinted a few degrees from perpendicular. The actual trace on the ground is a hyperbola asymptotic to a line displaced from the perpendicular to the aircraft axis by an amount equal to the squint angle. This is illustrated for level flight in Fig. 5a.

Three methods of stabilization are available to maintain a true ground presentation of the radar picture.

1. The antenna can be space stabilized so that the radiation pattern covers a constant range on the ground and is at all times perpendicular to both the ground and the aircraft's ground velocity vector.

2. The display can be distorted in such a manner as to automatically compensate for all changes in aircraft attitude. This is known as data stabilization.

3. A combination of the above two methods can be used.

In present day bombing radars using mechanically scanned apertures of about 60 inches, a combination of space and data stabilization is used. (Ref. 6 and 7). The antenna is pitch stabilized about the line of sight to insure uniform radiation at the ground. The PPI presentation is north stabilized to compensate for aircraft yaw. Correction for roll about the line of sight is computed for the particular target being tracked. This type of data stabilization allows correct measurement of range and azimuth for the one target, but leaves the rest of the PPI presentation distorted.

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A reconnaissance radar must have undistorted mapping of all ground

points, as should a bombing radar which obtains position information by

map-matching techniques. The method suggested herein for MX-2276 is a

complete data stabilization system. The reasons for this are twofold:

1) The side-looking antenna is very large and cumbersome. An-

tenna stabilization would require a considerable increase in weight and

space; the difficulty of holding dimensional tolerances would be increased.

2) Attitude changes in the airplane are sufficiently small that

radiation intensity levels at the ground will not be too adversely affected.

For bombing radar systems ± 4 db is the largest uniform radial variation in

signal strength that may be tolerated as a result of incorrect tilting of

the fan beam. (Ref. 8). This restriction makes it necessary to stabilize

the beam within a tolerance of $0.23\ \theta_o$, where θ_o is the beam depression

angle. Under normal flight conditions the airframe is stabilized within

these tolerances.

Fig. 13a shows the antenna radiation pattern for level flight.

Fig. 13b shows the pattern when the aircraft has yaw, pitch and roll devia-

tions from the ground velocity vector. Comparison of Fig. 13a with Fig.

13b illustrates the difference between beam intercepts under level flight

and under combined yaw, pitch and roll. The displacement between the two is

the correction to be computed in the data stabilization computer.

An analog solution to the problem is suggested. The beam intercept

is computed in a rectangular coordinate system where the ground velocity

vector is the ordinate and a line orthogonal to this vector is the abscissa.

The choice of coordinates is a natural one, since scanning motion is supplied
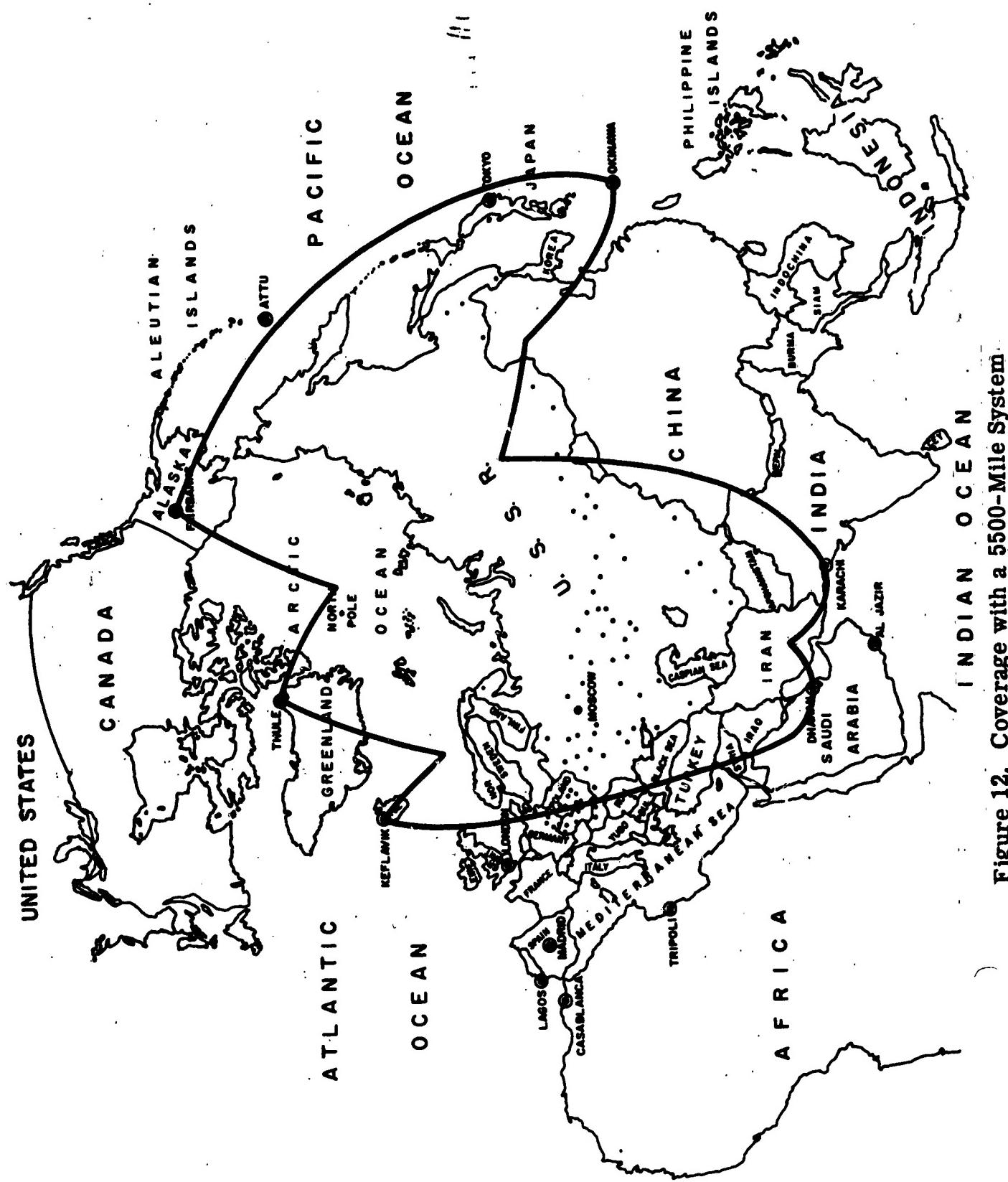
Figure 12. Coverage with a 5500-Mile System

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by the aircraft ground velocity. Hence, the viewing and storage media are
driven porportional to this velocity. The result is a trace on the cathode-
ray tube analogous to the ground intersection of the radiation pattern. The
CRT sweep is hyperbolic in both time and space; the former is to display
true ground range, the latter is because the radiation pattern is a section of
a cone's surface.

A hyperbola is the sum of a linear term plus an infinite series of
exponential terms. The sweep generator and data stbilization computer ap-
proximate it by a linear term plus a finite number of exponential terms. The
hyperbolic ground intercept trace is modified by the data stabilization
computer in accordance with aircraft attitude information from the navigation
system.

Derivation of the data stabilization equations, and possible instru-
mentation thereof are shown in Appendix A.

3. Accuracy Considerations.

The ability of radar to accurately locate a target in respect to the
aircraft or in respect to another point on the ground, is of prime importance
in both a bombardment and reconnaissance vehicle. During the bomb guidance
phase of the MX-2276 bombardment mission, the locations of radar targets in
the aircraft's inertial coordinate system are used to compute corrections
to the bomb's flight path. Similarly, a reconnaissance system must determine
the relative locations of targets and checkpoints to an accuracy suitable for
navigation and guidance systems. These position measurements will greatly in-
fluence the CEP of the final mission, whether it be bombardment or naviga-
tional.

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Table IV shows the CEP of a position fix from the aircraft. Para-
. ster values are attainable within the time period of this program.

Assumptions made are as follows:

1) Vehicle altitude is 150,000 feet.

2) The aimpoint density is uniform in the ground strip 10 to 50
miles from the aircraft ground track. The tabulated errors are obtained by
integrating individual errors over this range.

3) Matching accuracy, either placing cross hairs over an aimpoint
or matching a point on the radar map to a point on a prepared map, is a
critical function of the operator and his environment. It is felt that the
assumed 220 feet CEP is a conservative estimate, since it represents the sum
of target size and spot size.

4) Random errors due to flexure of the antenna, beam shift caused
by magnetron frequency changes, and inaccuracies in the circuits generating
the hyperbolic trace have been combined under squint error. At the present
time little is known of the first two, but efforts will be made to keep them
small. Total CEP for all three is estimated at 200 feet.

5) Aircraft attitude information, accurate to 0.05 degree, is
available for data stabilization.

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TABLE IV

RADAR FIX ERRORS

(Assumed Target of 100 feet Square)

A. Parallel to Ground Velocity

Spot Size (2000 spots per 40 miles)	120 ft.
Altitude Computer	224 ft.
Pulse Width (1/2 of total = $\frac{c\gamma}{4}$ sec.θ)	149 ft.
Hyperbolic Sweep (0.1% of maximum range)	304 ft.
Target Size	100 ft.
Data Stabilization	300 ft.
Matching Accuracy	200 ft.
Over-all	565 ft.

B. Perpendicular to Ground Velocity

Spot Size	120 ft.
Target Size	100 ft.
Beamwidth (1/3 of total to half power points)	222 ft.
Radom Fluctuations in Film Drive	200 ft.
Data Stabilization	240 ft.
Matching Accuracy	200 ft.
Squint	200 ft.
Over-all	530 ft.

Total CEP	775 ft.

Map matching techniques where accurate radar maps (either real or synthetic) of the target area are available can reduce this CEP. If the matching errors at the n different points are uncorrelated, the mean value of the error is equal to $\frac{1}{\sqrt{n}}$ times the mean error of all the points.

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4. Altitude Measurement.

An accurate indication of altitude is required for the inertial guidance navigation system and for image motion compensation in the photographic installation. Three methods are feasible; the choice among them depends upon the type of mission being flown.

1) The side-looking radar antenna will have a vertical lobe from which altitude may be determined when the radar is operating. Bell Aircraft Corporation has had considerable experience in the design of such a system for the Rascal missile.

2) Under conditions where it is required to keep radio frequency emanation to a minimum, a reduced output of the radar may be directed downward. The return time of the pulse leading edge is an indication of altitude.

3) A conventional radar altitmeter designed to operate at MX-2276 altitudes may be installed in the aircraft.

The requirements for the first method are covered in the preliminary performance specifications.

During periods when minimum radiation is required, a portion of the magnetron output may be switched from the side-looking antennas to a horn pointed toward the earth; the remainder would be dissipated in a dummy load. The leading edge of the return energy is tracked as in normal operation and the altitude is computed. This is not a completely non-emanating system; however, a small area is illuminated for a very short time and the risk of detection is minimized. Additions to the proposed radar consist of a waveguide switch and a Ku-band horn. The circuitry of the altitude computation is common to both methods.

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On missions where radar reconnaissance will not be used, it may be feasible to remove the total radar system for space and weight considerations. Installation of a radar altimeter is then required for an accurate indication of altitude. A pulsed method has been chosen over an FM CW system for two reasons:

1) Pulsed systems have a relatively small percentage error and are therefore more accurate for high altitude operation.

2) There is no antenna isolation problem in a pulsed altimeter. An FM CW system is subject to noise caused by changes in the leakage path length. These may arise from antenna vibration or from scattering at the ionized boundary layer. Tentative specifications for the radar altimeter are given in Table V.

TABLE V

TENTATIVE SPECIFICATIONS FOR RADAR ALTIMETER

Frequency	4200 mc
Peak Power Output	500 watts
Pulse Width	0.4 microsecond
Fixed Error	±50 feet
Percentage Error	±0.25%
Weight	70 pounds
Volume	2 cubic feet
Power Consumption	400 watts

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5. **Natural Ionization Effects.**

The bombardment version of MX-2276 flies in the lower regions of the E layer of the ionosphere. It has been shown that the bending and absorption effects of the slightly ionized (equivalent electron density of 1.24×10^{10} electrons per cubic meter) D layer below the aircraft are negligible. An investigation of two other natural ionization processes shows that their effects will not be detrimental to radar operation. These are discussed below.

There is considerable literature on the scattering of radio energy from the ionization in meteor trails. (Ref. 11, 12, and 13 contain additional bibliographies.) The magnitude of backscatter obtained from high density trails can be calculated from equation 2 of Ref. 11.

$$\frac{Pr}{Pt} = \left[\frac{GrGt}{32\pi^4} \left(\frac{\lambda}{R} \right)^3 \right] \left(\frac{\pi}{4 \text{ exp}} \right)^{1/2} \left(\frac{\mu_o e^2}{4m} \right)^{1/2} q^{1/2}$$

where Pr   = maximum peak echo power at the input of a receiver matched to the receiving antenna

Pt   = peak transmitter power

Gr, Gt   = receiving and transmitting antenna power gains

λ = wavelength

R   = range from common transmitter receiving site to the meteor trail

μ_o = permeability of free space

e   = electronic charge

m   = electronic mass

q   = total number of electrons per meter of length along the meteor trail.

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The duration of the trail is given by

$$T = \frac{\lambda^2}{4\pi^3 D} \left(\frac{\mu_0 e^2}{4m}\right)^2 q$$

where D = the diffusion coefficient $\simeq 3m^2$/second.

Calculations show that trail densities of 10^{14} electrons per meter will give detectable echoes at the proposed radar frequency. Their durations are short and the effect will be similar to noise spikes. Densities greater than 10^{16} electrons/meter might saturate the receiver for short periods of time. The incidence of high density trails is very low; however, they are most common in the lower E region. From Ref. 12;

$$N = \frac{160 \times 10^8}{q}$$

where N = number of centers of meteor trails of line density q or greater per square kilometer of meteor layer per second. This number should probably be increased by a factor of 100 to account for meteors whose centers are outside the antenna beam, but whose trails are within the beam. Nevertheless, the narrow antenna beamwidth of the search radar illuminates such a small area of the meteor layer that the echo rate will not interfere with normal radar operation.

Of particular interest to operations over the north polar regions is the aurora borealis (Ref.14). In these regions ions are produced in the atmosphere by corpuscular particles emitted from the sun. This is in addition to the normal ionization effects of ultraviolet light. The effects of these corpuscles extend down into the E region. They produce very intense ionization in addition to the visible radiation known as the Northern

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7. Preliminary Performance Specifications.

Tentative data for the proposed side-looking radar are given in Table VI. Preliminary performance specifications for each unit are included in subsequent paragraphs.

TABLE VI

TENTATIVE SPECIFICATIONS
PROPOSED SIDE-LOOKING RADAR
(Fixed Antennas - Simultaneous Lobing)

		Near Range	Far Range
Frequency	16 kmc/s		
Peak Power	120 kw (each side)		
Pulse Repetition Frequency	1200 pps		
Pulse Width	0.4 microsecond		
Receiver Bandwidth	3.0 mc/s		
Antenna Length	22 feet		
Antenna Width	8 inches		
Ground Coverage	10 to 50 miles each side		
Pulse Packet Size:		Near Range	Far Range
Azimuth		500 ft	1000 ft
Range		535 ft	225 ft
Indicator Resolution	2000 spots for 50 miles		
Weight Estimate: Antennas	320 pounds		
Radar	250 pounds		
Size Estimates: Antennas	22 ft x 1 ft x 8 in. each		
Radar	6 cubic feet		
Power Requirements	2500 watts		

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a. Power Supply.

The power supply unit will furnish the unregulated and regulated low voltage power for the radar system. Percentage regulation will be determined by allowable sweep and data stabilization errors. It will occupy approximately 0.8 cubic foot and weigh about 40 pounds.

b. Junction Box.

The junction box will interconnect the radar system with the inertial guidance system and the primary power sources. Filters in the a-c and d-c power lines will be included if necessary for interference protection. It will occupy about 0.2 cubic foot and weigh approximately 9 pounds.

c. Receiver-Transmitter.

This unit will consist of three parts:

1. The transmitter will generate high power radio frequency pulses from the modulator high voltage pulses.

2. The transmission circuits and mixer will divide the power between the two antennas, and will mix the sum and difference signals from each antenna with the local oscillator output to obtain a 60 mc intermediate frequency.

3. The receiver will amplify and detect the sum and difference signals before transmission to the video amplifier.

Preliminary design data:

Weight	50 pounds
Volume	1.3 cubic feet
Magnetron power output	250 kw peak
Receiver sensitivity	-87 dbm minimum

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Bandwidth	3 mc
Frequency	16.0 kmc ± 200 mc

d. Synchronizer.

This unit will consist of four parts:

1. The video amplifier will amplify the video output of the receiver-transmitter before transmission to the display unit.

2. The altitude computer will track the altitude return to obtain a voltage proportional to aircraft altitude above terrain.

3. The sweep generator will generate the horizontal and vertical deflection currents for the display indicator.

4. The data stabilization computer will accept attitude information from the inertial guidance system. It will use this to compensate the sweep generator output for aircraft maneuvers.

Preliminary design data:

Weight	40 pounds
Volume	1 cubic foot
Sweep accuracy	0.1% of maximum range
Altitude accuracy	±200 feet
Data stabilization computer accuracy	±200 feet in direction of aircraft ground velocity
	±200 feet perpendicular to aircraft ground velocity

e. Display Unit.

This unit will present to the pilot a radar map of the terrain illuminated by the side-looking radar. If deemed necessary, provisions are to be made to obtain position measurements by either a cross hair or a map-

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matching scheme. Means for permanent data storage in the form of a continuous radar strip map or overlapping frames will be included.

Preliminary design data:

Weight	60 pounds
Volume	1.2 cubic feet
Range resolution	2000 spots for 50 miles ground range
Film requirement	7 in. x 25 ft. for 5000 miles

f. Modulator.

This unit will generate the high voltage pulse used to modulate the magnetron, and will supply trigger pulses to the synchronizer. Provision will be made to synchronize it from an external source.

Preliminary design data:

Weight	70 pounds
Volume	1.2 cubic feet
Pulse repetition rate	1200 pulses per second
Pulse width	0.4 microsecond
Pulse rise time	) Determined by final magnetron
	)
Pulse amplitude	) design

g. Antenna.

The antenna will shape the radiated energy to provide uniform ground coverage between 10 and 50 miles from an altitude of 150,000 feet. There will be a vertical lobe for altitude determination. Two units are required.

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Preliminary design data:

Length - line of flight	22 feet
Width - lateral	8 inches
Depth - horizontal	1 foot
Frequency	16.0 kmc
Bandwidth	400 mc
Weight (total)	250 pounds
Gain	40 db
Sidelobe level	18 db below main lobe
Squint angle	2 degrees, maximum
Horizontal beamwidth	$0.2°$ at 1/2 power points
Vertical coverage	csc^2 within 1 db from $26.5°$ to $68°$ below the horizontal

h. Radome.

The radome provides mechanical protection for the antenna while not deteriorating aircraft performance. Small radome losses and minimum beam bending are to be design objectives.

Preliminary design data:

Weight	70 pounds
Temperature	$1700°$ F., maximum

Fused silica is a material capable of withstanding the very high temperature. Pertinent electrical and mechanical characteristics are tabulated below: (Ref. 9)

Specific gravity	2.2
Coefficient of expansion	5.6×10^7 per degree C
Loss tangent	0.0001

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Dielectric constant	3.76
Softening point	2885°F
Half wavelength at Ku band	0.190 inch
Strain point	1724°F

i. Operator's Control Panel

This unit will make available to the operator those switches and controls necessary for operation and in-flight adjustment of the radar equipment.

Suggested controls are:

Main power switch

Receiver gain

Video gain

Cathode-ray tube bias

Variable autotransformer to apply magentron voltage

Receiver tuning switch to select AFC or manual tuning

Manual tuning control

Meter and meter selector switch to show:

1. Magnetron current

2. Receiver crystal current

3. AFC crystal current

4. Primary supply voltage - ac

5. Primary supply voltage - dc

6. Pertinent regulated d-c voltages

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j. Environmental Control.

Necessar., pressurization and cooling equipment will be provided for proper operation of the radar set. Liquid and gaseous coolants will be investigated. The use of pressurizing agents, such as freon and sulfur hexafluoride, which have a much higher breakdown potential than air, are to be considered.

B. Ferret Reconnaissance.

1. General.

The objective of airborne ferret reconnaissance is to monitor all usable frequencies whose propagation characteristics make their detection by ground based receivers unlikely. Under normal conditions the lower limit is determined by the maximum usable frequency of the ionosphere; the upper limit is set by the highest frequency expected to be in use at the operational time of the equipment. Present knowledge places these limits at approximately 30 mc and 40 kmc, respectively. Information desirable from ferret reconnaissance is:

> Frequency of the transmission
>
> Location of the transmitter
>
> Type of modulation
>
> Modulation characteristics
>
> Polarization
>
> Antenna pattern and scan rate

These parameters are not necessarily of equal importance; their relative values are determined by the type of mission.

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Ferreting missions can be divided into two types. The first is to determine the radar order of battle. The objective is to make a map of a given area showing all radar installations with as much information as possible about the individual radars. The emphasis, however, is on detection and location. The second is to obtain intelligence of an enemy's technical capabilities. The accent here is on new and different transmissions; the use of new frequencies or of unusual types of modulation must be determined. This classification of objectives has led to two rather broad categories of ferret equipment.

1) High order ferret systems obtain a maximum amount of information about a relatively small number of transmissions. Accuracies of frequencies, locations and modulation types are very high; however, the number of intercepts analyzed is low.

2) Low order ferret reconnaissance will record a large number of intercepts with rather poor accuracy.

The former obtains a greater amount of technical information; the latter has a high probability of intercept. Low order ferreting implies a wide open system which seeks out every transmitter. This is necessary for early reconnaissance to determine what frequency bands are being used and with what types of modulation. Immediate countermeasures programs would depend upon this information. High order ferreting requires the extraction of finer detail from the intercept transmission. This would indicate the enemy's present technical capabilities and aid in forecasting his future abilities. In addition, specialized countermeasures programs are more effective if accurate modulation characteristics are known.

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Obviously, there are other equally valid ways of classifying ferret systems. One method is to distinguish between CW and pulse ferret systems. Another is to divide the frequency spectrum into certain bands and classify the equipment as to the band covered; a third is to distinguish between automatic and manned installations. Combinations of the above are possible; equipment for pulse S-band ferreting or for CW UHF band analysis is certainly logical.

At the present time it is difficult to visualize what degree of specialization will be required of ferret operations by 1965. Certainly the burden on the ferret system is inversely proportional to the available information on the radio frequency transmissions of any potential enemy. It is readily seen that as the number of ferret missions increases, the degree of specialization of a given mission may very well do likewise. Any system operational in the time period of the MX-2276 should contain a very high degree of flexibility. It should be capable of the low order initial reconnaissance while maintaining the ability to accurately analyze any type of transmission anywhere in the 30 mc to 40 kmc band.

It is unreasonable to expect that highly accurate measurements of all parameters can be made on every intercept during a given flight. To do so would require prohibitive weights and volumes using present techniques; and it is not readily visualized that normal advances in the state of the art will make such an all-purpose system feasible. However, a system containing basic elements common to all ferret requirements plus additional units chosen for the particular mission is feasible. Such a building block approach is recommended for the MX-2276.

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2. Recommended Ferret System.

Two general types of ferret installation are recommended for the MX-2276 reconnaissance vehicle: a system with high probability of intercept will obtain the radar order of battle while a system capable of detailed analysis will furnish technical intelligence. Common to both installations will be digital data recording equipment. Tables VII and VIII show the preliminary specifications for both systems.

It is not anticipated that a reconnaissance mission will be limited specifically to the use of one or the other of these types. Development of a complete ferret system should follow a building block concept so that the reconnaissance emphasis can be rapidly shifted as the need arises. The different building blocks, or subsystems, can be somewhat independent; the requirement for similarity is that all outputs be compatible with the common recording unit. Of course, the total capacity of any recording device is fixed, but in this way it can be used for information which is of maximum value at the time. Probability of intercept, frequency coverage, the number of parameters analyzed, and accuracy of analysis can be emphasized or de-emphasized as the situation requires, with the total amount of information obtained remaining approximately constant.

It may be that detailed modulation analysis can be most economically performed using a manned rather than a completely automatic system. An operator has the ability to recognize and discard transmissions previously analyzed. He can concentrate his attention upon new and unusual intercepts. Provisions are therefore made for inclusion of a presentation unit whereby an operator may select and record significant signals.

Figure 14 is a tentative block diagram of an over-all system.

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TABLE VII

PRELIMINARY SPECIFICATIONS
PROPOSED FERRET RECONNAISSANCE SYSTEM
TO OBTAIN RADAR ORDER OF BATTLE

Frequency Coverage	30 mc to 40 kmc
Frequency Accuracy	±10%
Directional Coverage	150 degrees on either side of the aircraft
Directional Accuracy	±1 degree
Modulation	Pulse only
Pulse Analysis	
Pulse Width – Range	0.1 to 10 microseconds
– Accuracy	±0.1 microsecond
Pulse Rate – Range	50 to 25,000
– Accuracy	±3 percent

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TABLE VIII

TENTATIVE SPECIFICATIONS
PROPOSED FERRET RECONNAISSANCE SYSTEM
TO OBTAIN TECHNICAL INTELLIGENCE

Frequency Coverage	30 mc to 40 kmc
Frequency Accuracy	±1.0 percent
Directional Coverage	150 degrees on either side of the aircraft
Directional Accuracy	Unimportant
Polarization	Determine the type
Modulation	Determine whether pulse or CW

Modulation Characteristics

A.	Pulse Width	± 0.1 microsecond for pulse width greater than 0.2 microsecond. ±0.02 microsecond for pulse width less than 0.2 microsecond
B.	Interval between pulses	±1.0 percent
C.	Frequency Modulated CW	Determine frequency deviation and modulating wave form.
D.	Amplitude Modulated CW	Determine modulating waveform
	Antenna Pattern and Scan Rate	Determine duration of illumination within ±10% and relative amplitude of signal within 10% of peak value.

3. Boundary Layer Ionization.

The effects of certain types of natural ionization have been discussed under radar reconnaissance. Ionization produced at the shock front and diffused along the vehicle will have a similar effect on electromagnetic propagation. Depending upon the electron density and the frequency under consideration, the effect can vary anywhere from total reflection to insignificant refraction of the radio energy.

Calculations have been made of the electron density at the shock front (Ref. 2), but little is known of the concentration in the boundary layer. Experimental methods are recommended to determine the rate of recombination and diffusion behind the shock front. In lieu of this knowledge a discussion of boundary layer ionization must be limited to qualitative effects.

The refractive index, η , of an ionized medium is given by (Ref. 2)

$$\eta = \left[1 - \frac{Ne^2}{\varepsilon_0 \, m\omega^2} \right]^{1/2}$$

where N = number of electrons per cubic meter

e = the electronic charge

m = the electronic mass

ε_0 = dielectric constant of free space

ω = angular frequency of the incident wave.

when N is great enough so that the refractive index is equal to zero, an electromagnetic wave cannot be propagated; all frequencies less than ω will be reflected. Solution of the above equation under this condition gives

$$f_N^2 = 80.5 \, N$$

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where f_N = the highest frequency totally reflected by the ionized medium. A graph of f_N vs. N is shown in Fig. 15. Frequencies below f_N will not pass through a boundary layer of electron density greater than N.
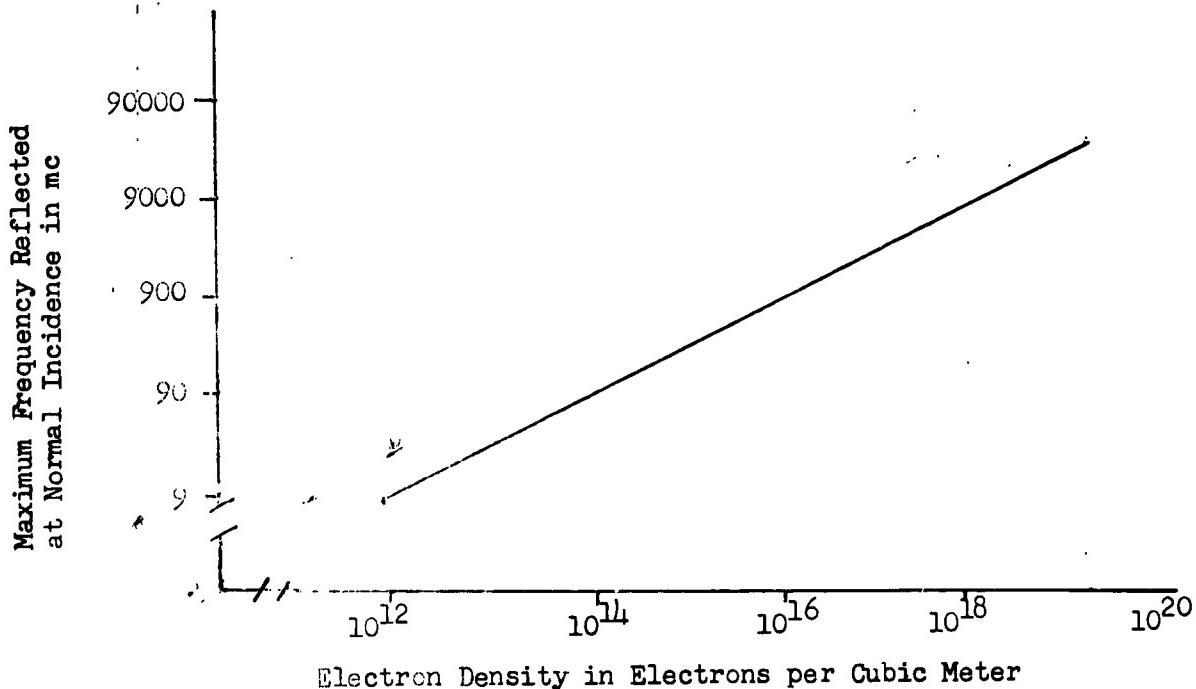
FIGURE 15: Critical Frequency as a Function of Electron Density

Fig. 15 is plotted for a wave at normal incidence to the ionized layer. For energy at oblique incidence the maximum reflected freqiency is:

$$f_{ob} = f_N \sec i$$

where i = angle of incidence measured from the normal. Thus at oblique incidence, some frequencies above f_N are totally reflected. In this discussion of ferret reconnaissance we will assume the incoming energy is perpendicular to the boundary layer so that f_N represents the lowest usable frequency for a given electron density. Of course, this may not be true,

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in which case detailed information of the boundary layer and of the antenna configuration is needed. The highest frequency which is totally reflected may therefore determine the lowest limit at which ferret reconnaissance may be accomplished. However, the lower limit of 30 mc as chosen previously will be considered valid until further data on the boundary layer ionization are obtained.

The ionized layer in addition to reflecting a certain band of frequencies will refract all higher frequencies. The amount of refraction is a function of the index of refraction which has been shown to depend upon the electron density and the frequency used. If we let r be the angle which the direction of propagation makes with the normal to the ionized layer after refraction and i the angle which the direction of propagation makes with the normal on entering the boundary layer, Snell's law may be used to determine the amount of refraction:

$$\sin i = \eta \sin r$$

where the refractive index of the space outside the ionized layer is unity.

Refraction will cause errors in both radar and ferret data. Because of the high frequency of the radar (16.5 kmc), little beam bending is expected. The lower ferret frequencies may be quite adversely affected; however, considerably more refraction can be tolerated for ferret intercepts than for the radar beam.

If experiments show that the electron concentration will significantly affect the radar operation, means should be devised to divert the ionized layer away from the radar antennas. Both aerodynamic and electronic means can be investigated,

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4. Radome.

The radar radome problem is discussed in Ref. 5. Similar stringent requirements as to heat resistance exist for the ferret radome. The relatively wide beam ferret antennas and relatively high-power received signals allow the transmission characteristics for ferreting to be considerably less severe, however. While low loss tangent and minimum beam bending are always desirable, they can be compromised to a certain extent for heat resistance, light weight and good structural characteristics.

The suggested ferret installation for MX-2276 requires a section of the fuselage to be transparent to the radio frequency portion of the spectrum. Since ferreting antennas are quite broadband, the use of a quarter wavelength or half wavelength radome is unlikely. A single radome which is electrically thin to all wavelengths in the 30 mc to 40 kmc band would be ideal; however, the extreme thinness of such a radome would be structurally unsatisfactory. A suggested solution is to use a radome thin to the longer wavelengths and to furnish small windows for the high frequency antennas. Experimental investigation is recommended to determine optimum window thickness from both a structural and an electrical point of view. Both very thin windows, and windows of half wavelength thickness at the center of the frequency band should be investigated.

5. Preliminary Performance Specifications.

Preliminary performance specifications of the ferret reconnaissance system follow.

a. Data Recording Unit.

This unit will record on magnetic tape the binary digits from the coordination unit. It will have a minimum total capacity of 2.8×10^7 b.

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bits and will be capable of storing 40 binary digits from each trunk line during the readout time. Provision will be made for simultaneous recording on ten channels. The unit will occupy approximately one cubic foot and weigh about 30 pounds.

b. Coordination Unit.

The coordination unit will accept and store coded information from the analysis equipment. It will separate the intercepts by means of time simultaneity, and record the parameters to an accuracy determined by ground adjustment of the unit. The information will be transmitted to the recorder over one or more of ten available trunk lines. The forty binary digits available for each reading may be used to represent one or more of the recorded parameters. Approximate weight and volume will be 100 pounds and 2 cubic feet, respectively.

c. Polarization Antennas and Analyzer.

This equipment will consist of vertically and horizontally polarized omnidirectional antennas of sufficient number to cover the required frequency band. An amplitude comparator will determine the relative magnitudes of each polarized component. If they are within 6 db of each other, the polarization will be considered mixed. Type of polarization will be transmitted in digital form to the coordination unit.

d. Directional Antennas and Analyzer.

This will consist of the proper antennas to obtain the direction of the radiation. A phase comparison method is recommended for high accuracy; where this is not required, directional antennas such as horn arrays or Luneberg lenses may be used at high frequencies and rotating rhombic antennas

at the lower frequencies. The information will be transmitted in digital
form to the coordination unit. The directional analyzer will weigh approx-
imately 70 pounds and occupy about 1.5 cubic feet.

    e. Frequency Antennas and Analyzer.

    Use will be made of a sufficient number of omnidirectional
circularly polarized antennas to cover the frequency range. The output of
each antenna will feed a bank of filters to give the desired frequency
resolution. Frequency information from the filters will be digitalized
before transmission to the coordination unit. Analysis equipment weight and
volume will be approximately 145 pounds and 3 cubic feet.

    f. Pulse Analyzer.

    This unit will determine pulse width and the interval between
pulses. It may be a low sensitivity device since pulse peak power is
usually quite high. Being of low sensitivity it will discriminate against
CW signals. The pulse parameters are transmitted in digital form to the
coordination unit. The unit will weigh about 40 pounds and occupy 1 cubic
foot.

    g. CW Analyzer.

    This will be a high sensitivity device. Automatic analysis of
CW signals will be investigated; if this is not feasible the signal will be
stored for detailed analysis on the ground. The use of an operator will be
considered to aid in detecting and analyzing CW transmissions.

    h. Scan Rate Analyzer.

    This unit will measure and store the amplitude distribution of a
pulse signal as a function of time. It will measure the elapsed time between

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periods of illumination by the same radar. The amplitude distribution will be sampled and the samples coded into binary form for transmission to the coordination unit simultaneously with the interval between illuminations.

C. Ground Support Equipment.

The general electronic test equipment required for this installation is listed in Table IX. Special radar test equipment is shown in Table X.

The ferret reconnaissance system will require two types of specialized ground equipment in addition to conventional measuring instruments. These are discussed below.

A data reduction unit must convert the digital information on magnetic tape into a form suitable for strategic and tactical use. A punched card system appears the most feasible. Efforts will be made to make the MX-2276 ferret recording unit compatible with other ferret installations so that a single type of ground computer can process all ferret data.

A special ferret test set must be developed. It should be capable of radiating from test antennas various pulse and CW signals in any portion of the 30 mc to 40 kmc band. Modulated noise will be investigated as the wide band signal source. Definite intercept problems will be fed into the ferret installation so that over-all system performance can be ascertained.

D. Personnel Training

Skilled electronics technicians will be required to maintain the radar and ferret installations. The display unit should be serviced by men with photographic experience.

Pre-Flight Tests.

It is expected that a one-hour pre-flight test period exclusive of warm-up time will be needed by both the radar and the ferret reconnaissance equipments. Conventional radar measurements such as frequency, power output and receiver sensitivity will be made. A special problem will be devised to test the accuracy of the presentation unit. The ferret installation will be tested by a number of dummy intercepts generated by special test equipment. Processing of these intercepts will determine the over-all system sensitivity and accuracy.

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TABLE IX

GENERAL ELECTRONIC TEST EQUIPMENT

Electronic Tube test set   Similar to I-177

Voltmeter - Ohmmeter - Ammeter   Similar to IS-189

   Measure d-c voltages from 0-5,000 volts in 6 ranges

   Measure a-c voltages from 0-1,000 volts in 5 ranges

   Measure d-c current from 0-10 amperes in 3 ranges

   Measure resistance from 0-10 megohms in 3 ranges

Voltmeter - AC Vacuum tube   Similar to ME-6A/U

   Frequency Range   40 cps to 100 kcs

   Voltage Range   0.001 to 100 volts in decade

                                           steps

   Input impedance   500,000 ohms

   Accuracy   2% of full scale reading

Crystal test set   Similar to TS-268 C/U

Insulation tester   Commercial megger

Voltmeter - DC Vacuum tube   Commercial Voltohmyst

Ground Power Source

   Supplies necessary a-c and d-c power to operate electronic

    equipment during ground tests

Cooling Unit

   Supplies coolant for radar and ferret installations during ground

    operations

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TABLE X

SPECIAL TEST EQUIPMENT FOR RADAR SET

Echo Box   Ku band

Pulse Voltage Divider   Similar to TS-89/AP

   Provides a 100:1 divider for pulses of 2,000 to 20,000 volts

   Provides a 10:1 divider for pulses of 200 to 2,000 volts

Modulator Dummy Load   Similar to TS-90/AP

   Provides a matched load for the modulator

Radar Range Simulator   Similar to Sperry T-101015

   Provides variable delayed radio frequency pulses for range calibration

Ku-band Power and Frequency Meter   Similar to TS-729/UPM-29

Ku-band Signal Generator   Similar to TS-729/UPM-29

Ku-band Spectrum Analyzer   Similar to AN/UPM-28

Ku-band RF Dummy Load   Similar to Sperry 401

Crystal Current Meter   Similar to I-139A

Aircraft Attitude Simulator   New equipment

   Generates signals to simulate aircraft altitude, roll, pitch and

   yaw for calibration of sweep and data stabilization circuits.

C.   Photographic Reconnaissance.

The photographic subsystem is the subject of a separate report (Ref. 18). Two camera installations are provided, one for search missions and one for detail information missions. Fig. 3a shows the six camera array recommended for the search function. This array consists of two 27 inch, two 34 inch, and two 42 inch focal length cameras. The array for detail information consists of three 72 inch focal length cameras. Since this system is discussed in detail in Ref. 18, no further discussion is necessary here.

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VIII PROPULSION

A. General.

Both stages of the proposed vehicle will be powered by liquid propellant rocket engines. The airplane or manned stage will require 60,000 pounds of thrust and the first stage will require 300,000 pounds.

The selection of the power plants has been based on the policy of using existing units if the cost or the performance of the over-all weapon system is not seriously penalized. The engine requirement for the booster can take advantage of the double 150,000 pound thrust LOX-JP-4 unit now being developed for another Air Force Project. A preliminary performance specification for this engine system is contained in NAA proposal. (Ref.19). The propulsion unit for the airplane will require a new engine using higher energy propellants.

The performance of the LOX-JP-4 propellants is considered the most efficient of presently used propellant combinations. The use of liquefied fluorine as an oxidizer makes possible a significant increase in the performance of rocket engines. The decrease in weight, size and cost for high performance rocket vehicles using fluorine in the oxidizer is sufficient to justify the development of fluorine or fluorine-oxygen mixtures as a practical rocket propellant. Therefore the cost of learning to handle this oxidizer and the complications of the logistics problem should not be charged entirely to this project.

One very interesting possibility in the use of Fluorine-LOX oxidizers requires further exploration. Mixtures of liquid fluorine and oxygen when

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used with JP-4 give best performance when mixed at the ratio of 70% F_2 and 30% O_2. (Fig. 16 & 17). It happens that there is a chemical compound, oxygen bifluoride (OF_2), which contains these elements in this weight ratio. It has good rocket characteristics in that it will yield slightly better impulse than the mixed liquids and its bulk density is excellent. More important however, is the fact that its boiling point at atmospheric pressure is $-227°F$ and its freezing point is $-371°F$. Thus it can safely and easily be cooled below its boiling point with liquid nitrogen ($-320°F$) considerably reducing the handling and storage problems in comparison to the mixed liquids. (In addition, there are indications that it is compatible with Teflon which can then be used for gaskets and seals.) This chemical has been produced in laboratory quantities and it should be possible to produce commercial quantities if required. This compound will be the subject of further extensive evaluation.

The fuel used for all stages of the present proposal can be JP-4. It is the least expensive and most plentiful fuel in the present Air Force supply program. There are other fuels such as ammonia and unsymmetrical dimethyl hydrazine (UDMH) which offer slight gains in performance over JP-4. However, the gain may not be worth the increased complication, development cost, and logistic problems.

The use of liquefied hydrogen offers further performance growth with LOX in the booster and with fluorine in airplane. The H_2 and O_2 combination has poor density characteristics which require large propellant tanks but this may not be so critical as it is generally supposed. F_2 and H_2 can be used at the very high oxidizer to fuel ratios of 9 to 1 or even 16 to 1

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Specific Impulse of Various Fuels with Fluorine Oxidizers for a Chamber Pressure of 300 psia and Isentropic Expansion to 14.7 psia with Equilibrium Composition Maintained.

FIGURE 16
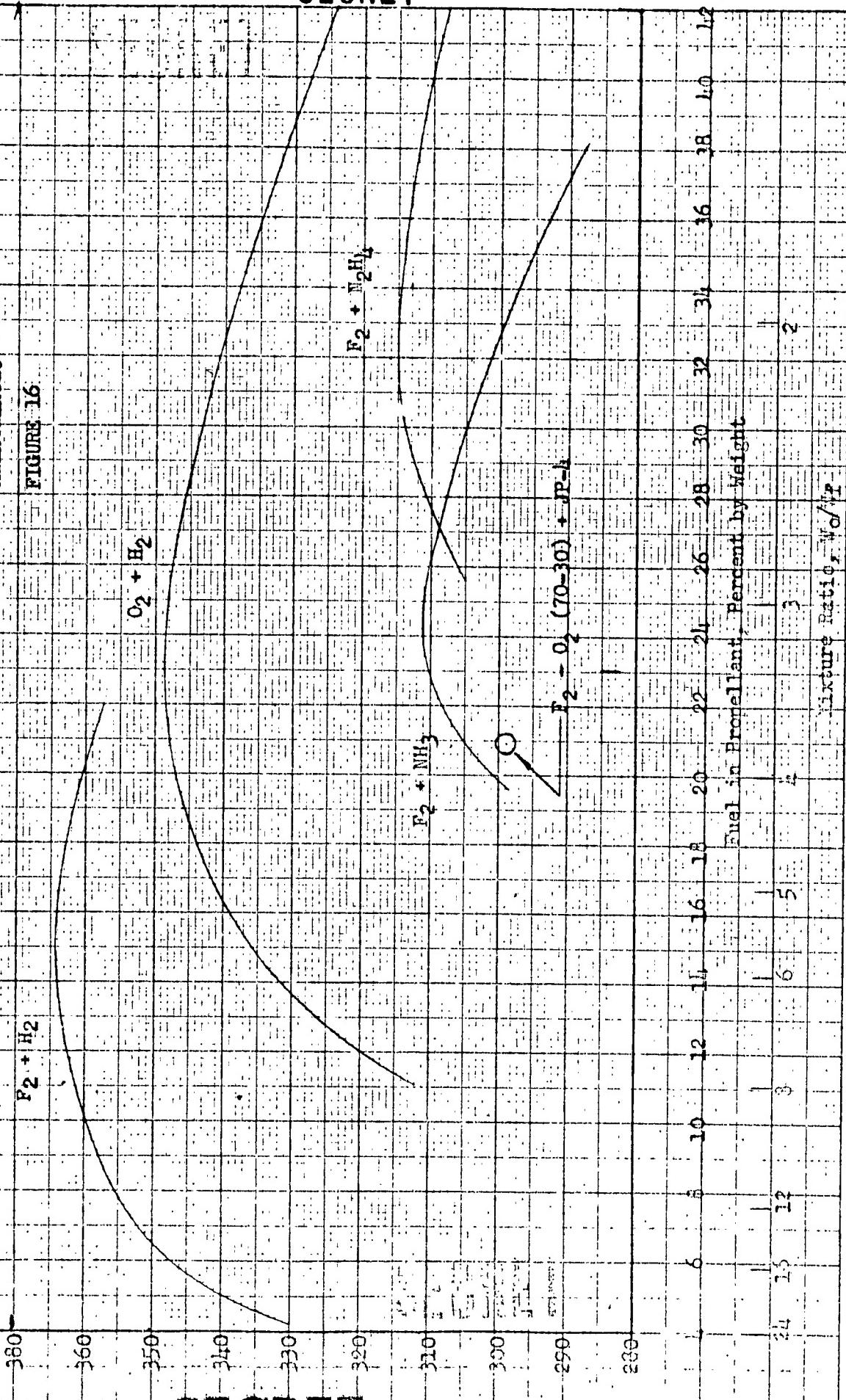
$F_2 + H_2$

$O_2 + H_2$

$F_2 + N_2H_4$

$F_2 + NH_3$

$F_2 - O_2 \ (70-30) + JP-4$

Fuel in Propellant, Percent by Weight

Mixture Ratio, w_o/w_f

380

360

350

340

330

320

310

300

290

280

Combustion Temperature of Various Fuels
with Fluorine Oxidizers at a Chamber
Pressure of 300 psia.
FIGURE 17

$F_2 + N_2H_4$

$F_2 + NH_3$

$F_2 = O_2 (70-30) + JP-4$

$O_2 + H_2$

$F_2 + H_2$

Fuel in propellant, percent by weight

FIGURE 2
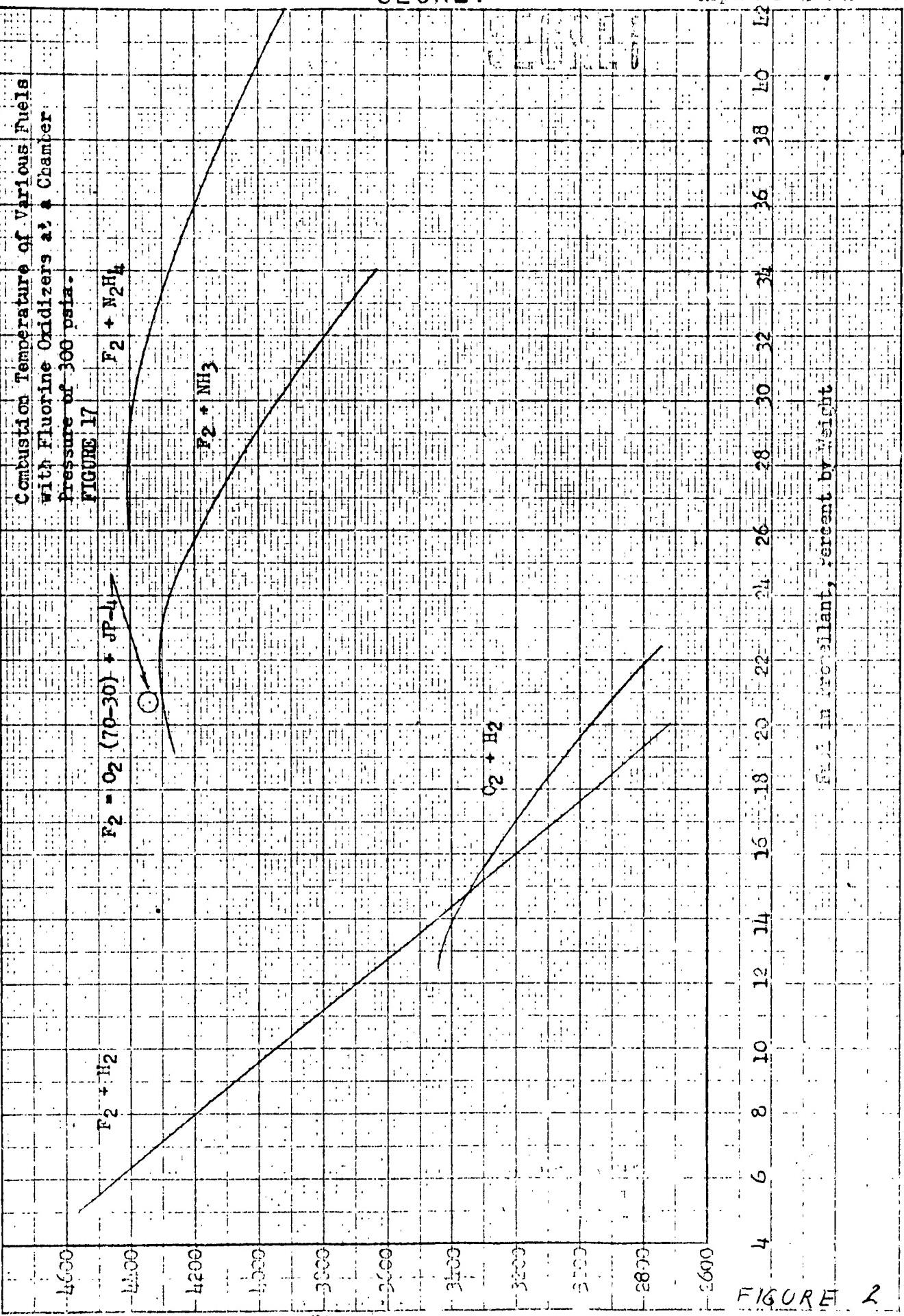
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to get reasonable density. Hydrogen, of course, is one of the most diffi-
cult liquefied gases to handle, but interest in its use is increasing rapidly
because of its very high heat energy per pound. Methods and equipment for
handling and storage are expected to be available in a comparatively few
years. Therefore, further consideration should be given to the use of
fluorine and hydrogen as a later development for this program.

One of the most significant differences between the consideration
of rocket engines for this project and the large rockets that have been
developed up to this time is the presence of the crew in the vehicle. A
propulsion system for a missile must be dependable, but does not require
the safety systems demanded for manned rocket aircraft.

B. Airplane Propulsion System.

The rocket engines for the airplane stage of this weapon system
were studied by both the North American Aviation Corporation and the Bell
Aircraft Corporation Rocket Department. No detailed evaluation of these
proposals will be attempted in this report since the selection of power
plants may be influenced by Air Force development contracts for the general
state of the art advancement of rocket engines. Proposals and studies from
both organizations are included as a part of this study.

It is expected that an over-all specific impulse of 334 seconds will
be obtained at 55,000 feet altitude and 346 seconds at 155,000 feet as
shown in Figure No. 18 . The weight of the engine will be approximately
710 pounds.

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FIGURE 18

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VARIATION OF PERFORMANCE WITH ALTITUDE

$A_e/A_t = 25.0$ $P_c = 500$ PSIA
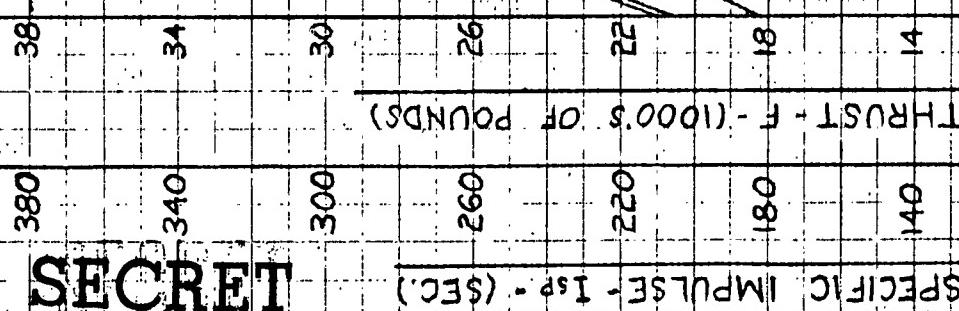
ACTUAL THRUST CHAMBER SPECIFIC IMPULSE

ACTUAL ENGINE SPECIFIC IMPULSE

ACTUAL THRUST PER THRUST CHAMBER

THRUST - F - (1000's OF POUNDS)

SPECIFIC IMPULSE - I_{sp} - (SEC)

ALTITUDE - (1000's OF FT)

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1. Tanks.

The propellant tanks are integral with the airframe structure. This
takes advantage of the excess strength of a cylindrical pressure vessel that
is not required for internal stresses and takes advantage of the stiffness
of the pressurized tanks during the critical phases of powered operation.
The tanks are approximately 75 inches in diameter with elliptical heads.
The fuel tank volume is 187.4 cubic feet. Oxidizer volume is 364.6 cubic
feet including 3% expansion space. The total tank structural weight is 1413
pounds. Tanks with elliptical ends of a 2 to 1 ratio of the axes are about
70 pounds lighter than tanks having 3 to 1 elliptical heads. However, they
are 7 inches longer and the weight of the heat protection for this extra
length more than offsets the lighter tank structure. Pilot safety in tank
design is assured by adequate margins of safety and relief valves to prevent
over-pressurization. In addition, the tanks are designed for all flight
and landing loads without depending on the tank pressure.

In the event that the engines do not start or an early malfunction
shutdown occurs the remaining propellants in the tanks must be jettisoned
to reduce the weight to the design loading condition. These jettison valves
can be operated by a completely independent system from a cockpit control.
Expulsion can be accomplished by the normal tank pressurization system or by
an emergency pressure source.

The same jettison valve can be used on the ground prior to takeoff
to quickly unload the oxidizer in the event of a cancelled flight or in case
of accident such as a propellant leak. For this purpose, the oxidizer load-
ing vehicle could remain connected to the jettison valve until the tank is

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pressurized prior to takeoff. If the oxidizer is jettisoned, the ground
vehicle can safely receive the oxidizer to prevent contamination of the
area.

For added safety to the pilots in the combat zone and during landing,
the oxidizer tank and system will be completely emptied and purged immediately
after power plant shutdown. The combination of natural evaporation of the
small amount of remaining oxidizer and a helium purge will effectively elim-
inate any toxic fumes. Extra capacity is included in the helium storage
system for this purpose.

2. Pressurization System.

There are at least four different types of systems that can be used
for propellant tank pressurization.

The system now being developed by this contractor for MX-1964 is
perhaps the simplest and lightest. It generates high pressure gases from
the combustion of a solid propellant. The gas is filtered and cooled before
introduction into the propellant tanks. Compatibility with RFNA and JP-4
has been satisfactory but it is very doubtful that safe conditions could be
maintained when pressurizing a fluorine tank. In addition, the safe opera-
tion of the engine will be enhanced by purging before and after operation.
An inert gas would be easier to handle for this purpose.

A liquefied gas offers the advantage of compact storage and rela-
tively light weight. Liquid nitrogen evaporation systems have been studied
and tested by this contractor for other projects and the equipment design
requirements have been established. However, the system is somewhat compli-
cated and the nitrogen gas introduced into the F_2 O_2 propellant tank will
tend to be recondensed to some extent by the cold oxidizer.

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The storage at high pressure of a gas at ambient temperature is the simplest of all systems since no heat exchanger or insulation is required. Systems of this kind have been successfully used by this contractor and others for all kinds of propellants. A gas such as helium can be used since it is completely inert and will not be condensed by the cold oxidizer. However, the weight of this type of system for the size and with the critical performance of this project is prohibitive. Some complication can be justified for a significant weight saving.

A chilled helium storage system is concluded to be the best compromise of compatibility, complication, and weight. The system consists of a chilled high pressure spherical storage tank, a start valve, two heat exchangers in the turbine pump exhaust duct, a first stage regulator, two second stage regulators, and a purging system. It was determined that if the gas mean temperature were raised to 500°F, the final average gas temperature in the oxidizer tank would be approximately -100°F and approximately 160°F in the fuel tank. By using these values, the required amount of helium was determined to provide a 60 psia suppression head pressure to the oxidizer tank and a 39 psia suppression head to the fuel tank. The tank volumes used in determining these gas requirements were 364.6 ft^3 for the oxidizer and 187.4 ft^3 for the fuel tank.

A sufficient amount of helium has also been included to allow for purging the two thrust chambers and the gas generator prior to start and at shutdown.

The size of the required helium storage sphere is 6.5 ft^3. The helium is loaded in the sphere at a maximum temperature and pressure of -250°F and

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4000 psia. A safety valve set for 5000 psia will prevent over-pressurization. The sphere is cooled by an oxidizer jacket in order to maintain low gas temperature. The pressure in the sphere is allowed to decay only to 200 psia during operation to provide a sufficient drop across the second stage regulator at shutdown to assure accuracy in controlling tank pressures.

In order to prevent cold gases from entering the first stage regulator during operation, a first stage heat exchanger is required to pre-heat the helium gas to approximately -100°F. This is done because of the difficulty encountered in properly seating the regulator valve with extremely cold gases. The size of this heat exchanger is small since it operates on the high pressure side of the regulator. The second stage heat exchanger will operate on the low pressure side of the first stage regulator and will increase the temperature of the helium gas to 500°F. The total amount of heat to be transferred to the gas in both heat exchangers is 112 BTU/sec. Of this amount, 22 BTU/sec will be provided in the first stage and 90 BTU/sec will be provided in the second stage. Both heat exchangers will be of the one-pass, cross flow type.

The helium storage sphere will be made from cold drawn age hardened "K" monel material and will be machined to final configuration to minimize the weight. The local areas in the vicinity of the welds will have an increased wall thickness to obtain a lower stress level in the welds.

The heat exchangers will be made from 300 series stainless steel. The tubing will be 3/16 O.D. x 0.022 wall and will be "Nicrobrazed" to the header plates. This type of construction has been used successfully on several heat exchangers made at this facility. "Nicrobrazing" allows for a

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more compact and easily fabricated unit in that the tubes are joined to the
header plates in one operation. This type of design will result in a light-
weight, efficient unit. The total empty weight of the pressurization system
is 405 pounds.

3. Mixture Ratio Control System.

Mixture ratio control of propellants is certain to benefit any
rocket-powered vehicle utilizing a bi-propellant combination. Thus, whether
the system has a fixed burning time or runs until all propellants are ex-
hausted, the speed and ultimate range at burnout will be considerably greater
if some control is maintained on matching the consumption rate of each prop-
ellant to minimize the total weight of propellants remaining at burnout.
This is accomplished to some degree by the normal control tolerance of the
engine, to an accuracy of about $\pm3\%$. Assuming extreme mixture ratios, it is
obvious that an "outage" of 6% of either propellants can exist. Thus, the
burning time of the engine is cut down and the unburned propellant results in
dead weight to be carried. This effectively reduces the mass ratio of the
system.

It is the purpose of the mixture ratio control to monitor the rate
of consumption of propellants and effect the necessary adjustments to the
system flow constants so that the propellants are exhausted simultaneously.
This may generally result in a slightly altered firing time; however, it will
preserve the required total impulse. A sample calculation using a vertical
trajectory and neglecting aerodynamic drag will illustrate the magnitude of
the effect of propellant mismatch. If a hypothetical vehicle with no mismatch
could attain a velocity at burnout of 4860 ft/sec. it would coast to an alti-

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tude of 367,000 ft. or 69.5 miles. This same vehicle with 3% of the prop-
ellants remaining at burnout would attain a velocity of only 4280 ft/sec and
a maximum altitude of 284,000 ft. or 54 miles.

Several methods of mixture ratio control are possible, among which
are:

1) Tank level sensors.

    a. Float type - a float secured to an endless belt or in a
       guided tube.

    b. Hot wire indicator type - since the wire loses heat more
       readily in the liquid than above the liquid, the resistance
       of the wire is a function of the liquid level.

    c. Sonic pipe type - sonic resonant frequency changes as the
       liquid level in a pipe changes.

    d. Differential pressure type - measurement of propellant mass
       to effect mixture ratio control.

2) Volumetric Flow Rate - Real flow rates of each propellant by
    installed flowmeters.

3) Ultrasonic Mass Flowmeter - Acoustic measurement of fluid flow.

All methods utilize the balanced bridge principle in conjunction
with an amplifier circuit to control a motor-driven valve in the propellant
feed system.

In view of the constant positive acceleration of the expected flight
plan, it would appear that the tank level sensor type would be most appro-
priate for the proposed application. Of the systems mentioned above, the
differential pressure type has been tested in two flights of the Viking

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program. Although no attempt was made to control flow, sufficient information was attained to prove the feasibility of the application.

It is known that other weapon system projects are faced with this problem and that systems are being designed to reduce the mismatch to a tolerable amount. Unfortunately, this contractor has not been able to obtain sufficient information to discuss the latest development. However, it is known that the state of the art is being advanced and the results of the effort will be available for this application.

4. Auxiliary Power Units.

For the proposed flight plan, auxiliary power cannot be obtained from the rocket engine turbine or from a ram air turbine for the duration of the flight. Similarly, a system that uses the energy available from aerodynamic heating cannot be used for the take-off or the final phases of the flight. To reduce complications and to improve reliability, it is very desirable to have a self contained auxiliary power unit that does not require switching and is not dependent on other parts of the weapon system.

For these reasons this study was confined to an examination of a monopropellant turbine drive or reciprocating engine drive self-contained and independent system. The total power required was established at 25 horsepower for a duration of 72 minutes.

The load has been divided equally between two units so that one unit could continue to supply sufficient hydraulic and electrical power to enable the pilot to make a safe landing in case of a unit malfunction.

The turbine drive unit is simplest and has the lightest dry weight but the weight of the propellant consumed is greater. This type of unit is now being developed and used on several Air Force projects and no general

development difficulty is expected. However, the development of a specific size for this man-carrying application will represent a considerable expenditure of time and effort.

The reciprocating engine powered unit is an outgrowth of conventional gasoline and air powered units which have been used for years for ground and moderate altitude airborne applications. However, for the extreme altitudes encountered here the combustion must be independent of an ambient air supply. The use of a monopropellant or bi-propellant in a reciprocating engine has been successfully demonstrated by several organizations and the basic development problems are reasonably well understood. It is expected that the development of a reciprocating engine for this specific project would be possible in the time available if an early start is made.

The reciprocating engine can use the propellants more efficiently than a turbine so that its original weight disadvantage is overcome for long durations. For the 72 minutes established for this study, the total weight of the turbine system is 10 pounds heavier as shown on Fig. 19. This difference is not significant for the presently required duration. However, in view of the fact that the growth factors of this weapons system would extend the range and the duration of auxiliary power plant operation it seems advisable to start a detailed investigation of both types of units. Both units have been shown on the propulsion system layout, Dwg. No. 02-960-129 (Fig. 20) to show the comparative size of the units and their propellant tanks.

FIGURE 19

WEIGHT COMPARISON OF PROPOSED AUXILIARY
POWER UNITS @ AN AVG. 12.5 B.H.P.

WEIGHT (LBS)

TOTAL WEIGHT OF TURBINE UNIT

TOTAL WEIGHT OF RECIP UNIT

DRY WEIGHT OF RECIPROCATING UNIT

DRY WEIGHT OF TURBINE UNIT

DURATION   (MINUTES)

OXIDIZER SUCTION LINE

FUEL SUCTI

MONOPROPELLANT TANK FOR
RECIPROCATING ENGINE UNIT

HELIUM LINE
PRESSURIZAT

ALTERNATOR

HYDRAULIC PUMP

RECIPROCATING ENGINE DRIVEN
AUXILIARY POWER UNIT

SECTION A-A

MIXTURE RATIO CONTROLLER

FUEL TANK PRESSURE REGULATOR

OXIDIZER TANK PRESSURE REGULATOR

FUEL TANK

OXIDIZER TANK

OXIDIZER PUMP INLET

FUEL PUMP INLET

ENGINE SEQUENCE CONTROL BOX

FUEL FILL & JETTISON PORT

OXIDIZER FILL & JETTISON PORT

OXIDIZER JACKETED HELIUM TANK

SECTION B B

02-860-124 ROCKET ENGINE ASSY

B

B

ENGINE SEQUENCE CONTROL BOX

B

FUEL FILL & JETTISON PORT

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5. Propulsion System Tolerances.

A rigorous analysis of the exact effect of the accumulated tolerances of the propulsion system has not been attempted during this study because of the extreme details of design that must be considered. Several of the primary causes of variations have been reduced by appropriate provisions as indicated below but it is recognized that the others will require further considerations. The following items contribute to variation in the thrust and efficiency of the propulsion system.

1. Basic engine control tolerances.

2. Propellant purity and fuel mixture variations.

3. Temperature variations of the propellants.

4. Evaporation or boil-off.

It is intended that the oxidizer will be subcooled by the ground service unit and that this temperature will be maintained by recirculation of cooled oxidizer, until just before takeoff. This will eliminate the escape of dangerous fumes and the loss due to boil off. In addition it will reduce the tank pressure required to attain the required pump suppression pressure. The variation of the JP-4 fuel temperature will not be directly controlled but by proper design and insulation from the oxidizer the variations due to extreme ambient conditions will be minimized.

5. Tank Expulsion Efficiency.

The absence of compartmentation to control center gravity shift, the expected stability of the flight path during engine operation and the high axial acceleration combined with good detail design should reduce the unusable propellants to a comparatively small figure. Baffles will also be used to eliminate vortex formation and to reduce splashing effects during expulsion.

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6. Propellant Mismatch.

A mixture ratio controller is required to reduce this loss to the minimum.

7. Variations in start and shutdown transients.

8. Seal drains, leaks, etc.

9. Weight variations.

If all the tolerances go to their extremes simultaneously, the total effect would be appreciable and would adversely effect either the performance or the weight of the weapon system. The expected accumulation is serious enough to be considered a problem area and should require detailed study.

Auxiliary thrust is available during the flight of the vehicle.

The sources of this thrust include:

1. Turbine exhaust (during boost).

2. Auxiliary power unit exhaust.

3. Cooling water (steam) ejection.

Because of the high altitude of operation, reasonably good thrust efficiency can be obtained from these sources.

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TABLE XI

AIRPLANE OR SECOND STAGE

PROPULSION SYSTEM WEIGHTS

Item	Number Required	Weight
Engine Mount	1	55
Tank Pressure Relief Valve	2	10
Tank Pressure Jettison Valve	2	2
Check Valve	2	2
Second Stage Regulator	2	10
First Stage Regulator	1	10
First Stage Relief Valve	1	3
Prop Lines, Supports, and Tank Fittings	1	128
Helium Start Valve	1	3
Helium Start Pilot Valve	1	2
Helium Source Pressure Relief Valve	1	5
Helium Fill Valve	1	3
Helium Jettison Valve	1	2
Propellant Fill and Jettison Valves	2	18
Turbine Pump Exhaust Duct	1	30
Heat Exchanger	1	15
Fuel and Oxidizer Tanks	1	1413
Helium Tank and Mount	1	350
Engine Assembly	1	710
Auxiliary Power Unit Install.	2	229
Total Dry Weight		3000

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C. Booster Propulsion System.

The propulsion system for the first stage of this weapon system has not required detailed consideration during this study program since it is not considered a problem area. The motor has been selected as the 300,000 lb. thrust assembly that has been developed for another project. Adequate information about this motor has not been available but its specific impulse is expected to be 250 seconds and its weight about 3400 pounds, including a safety system.

The increased development required for a piloted application is indicated by the different specification requirements for unmanned and manned use. Proof of endurance and reliability for aircraft rocket engines requires at least 35 full duration firings compared to 9 firings for missile engines. Proof of reliability and safety for the aircraft rocket engine is based on the performance of at least 100 starting and stopping tests including proof of safety for all conditions of single malfunction occurrences during the entire duration of rocket engine operation as compared to 10 tests for a remotely launched missile rocket engine.

The propellant tanks for Stage I will comprise a large part of the basic structure. They will be cylindrical with a hemispherical head of approximately 120 inches diameter and 392 inches over-all length. The tank pressure is determined by the design requirements of the turbine pump. Recent information on these requirements has not been available so a 35 psi pressure has been used for structural analysis purposes. The stiffening effects of internal pressure have been considered in determining the structural conditions when the engines are operating. However, to insure the safety of the

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pilot in the event of a loss of pressure just before takeoff the tank struct-
ure is made strong enough to take the static weight of the fully loaded
vehicle without the benefit of pressurization.

A mixture ratio control system will be required for this stage as
well as for the final stage to reduce the amount of propellant mismatch. It
is expected that the system under development can be used with very little
change.

The pressurization system for this stage will probably be chilled
helium since the design conditions are very similar to those for the airplane
system. Here again, lack of knowledge of the pressures required and other
details prevents an accurate calculation of the size and weight of the pres-
surization system.

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IX. GROUND HANDLING AND OPERATIONAL PROCEDURES

A. General.

The functional units of Ground Support Equipment have been considered as an integrated part of the Weapon System. They have been briefly studied in order to establish type, size and amount of equipment presumably needed, and their relation to the other elements of this Weapon System. In general the operations are similar for the reconnaissance or bombing missions with the exception of warhead handling.

The following general characteristics were considered:

1. Choice and/or design of equipment to be independent of geographical location and climate.

2. Design of single functional units with a high degree of reliability, especially for equipment that is used under adverse environmental conditions.

3. Low level of maintenance required.

4. Elimination of extensive concrete construction at the launch site.

5. Self powered units independent of local power supply.

6. 100% mobility.

7. Air, Road and Railroad transportability.

8. Low shipping weight of parts. Easy handling and transportability.

9. Use of GFP or commercial equipment as far as possible.

The operational ground handling phases of the Weapon System have been assumed as follows:

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The elements of the system are shipped from manufacturers' plants or depot to the AF Base by suitable transportation means such as air ferry, air transport, truck trailer, railroad and steamship or a combination of these. At the base the stages of the vehicle and reconnaissance equipment are kept in covered storage in ready alert status by periodic tests and checkouts.

The launch site at or near the AF Base needs no special preparation. Only a trench is dug for ducting away the gases of the rocket power plant during take-off. Over this trench the Assembly and Erection Equipment will be put together using a crane. A timber foundation provides bearing for this equipment which will be leveled by jack pads. A road from the hangar or storage building to the launch site will be necessary for transporting of the various units.

After transportation to the launch site the airplane and booster are lifted by a crane to the erection boom of the Assembly and Erection Equipment and are assembled in a horizontal position. The assembled vehicle is then erected to the vertical position and put over the gas duct pit. During the final checkout, adjustment and propellant servicing phases, the vehicle is secured to the erection boom which is lowered shortly before launching.

The three kinds of propellants are supplied and serviced as follows:

The JP-4 is available at the AF Base. Servicing trailers will be used.

Fluorine will be transported in the servicing trailer from the manu-facturer to the AF Base. Storage for a period of time in this trailer is possible.

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The LOX will be produced at the AF Base. Storage tanks of sufficient capacity will be provided. Trailers will be used for servicing the vehicle.

The JP-4 and LOX are pumped from the trailers to the vehicle tanks. For the fluorine servicing, gas pressure will be used. The servicing lines are mounted on the erection boom where platforms are also provided for work at the various stations.

B. Ground Handling Sequence and Equipment.

Table XII shows a preliminary sequence for ground handling activities from the manufacturers' plant to take-off, and the kind, number and type of necessary equipment for the various ground handling phases of the MX-2276 two-stage weapon system as a reconnaissance airplane. It is assumed that three complete vehicles (airplane and booster) are in ready alert storage at the base and that a mission is scheduled every 3 days.

If an organization plan shows that the same equipment can be used for several phases, the number of the listed items may be reduced. The numbers are based on the functional requirement and do not include spare units. In the columns listing "Type of Equipment" some information is given for the presumable state of the art at the time when this equipment will be needed.

The information in Table XII is not definitive. A close definition can only be established during advanced studies and/or R & D work.

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TABLE XII - GROUND HANDLING SEQUENCE AND EQUIPMENT

Phase	Airplane	Type of Equip.	Booster	Type of Equip.	Reconnaissance Equipment	Type of Equip.	Remarks
1) Preparation for shipping, packaging, etc. at Mfg. plant	Dolly for airplane skid. Number: One for every airplane	Special	Skid with casters for transportation and storage. Number: One for every booster. Crane: (Mobile) Useful Load 10t Height of Hook: 30' Range: 25' Number: One	Special / GFP	Container Hoist or crane suitable for loading operation Useful Load: It Number: One	Special / GFP	
2) Loading	Tractor, B-36, Jacks Slings and Hoist device on B-36 suitable for loading the airplane on Ferry aircraft	GFP / Special	Crane like above Tractor suitable for towing skid loaded with booster Loading and unloading equipment for Air Freight. (One set)	GFP / Special	Hoist or crane like above Normal loading and unloading equipment (One set)	GFP	
3) Shipping	Air Ferry: B-36 Air Transport: C-124	GFP	Road: Flat bed Trailer Air Freight: C-130 Railroad: Flat bed car	Special / GFP / Commerc.	Trailer: Useful Load 4t Cargo Airplane Railroad car	GFP / GFP / Commerc.	
4) Unloading & Storage at the Base	Sling, hoist device on B-36 Jacks & Tractor (See 2) Dolly for airplane skid (See 1)	Special / GFP / Special	Crane,Tractor (See 2) Loading & unloading equip. for air transport(See 2) Flat Bed Trailer (See 3)	Commer/GFP / Special / Special	Hoist or Crane (See 1 & 2) Normal loading or unloading equip. (See 2) Cradle with casters	GFP / GFP / Special	

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TABLE XII - Cont'd.

Phase	Airplane	Type of Equip.	Booster	Type of Equip.	Reconnaissance Equipment	Type of Equip.	Remarks
5) Storage	Building (Hangar)				In separate part of the building (Hangar)		
Inspection Mainten-ance & ready alert storage	Visual inspection Power Units for: Hydraulic Pressure Electrical	GFP		GFP	Power Units for: Hydraulic Pressure Electrical	GFP	
	Hydraulic test equipment	Special or GFP		Special or GFP	Hydraulic test Equipment	Special	
	Pressure test equipment	Special or GFP		Special or GFP	Pressure test Equipment	Special	
	One Unit of each item I.G. System C/O Equipment	Special		Special	Servo System C/O Equipment	Special	
	Radar System C/O Equipment	Special		Special	Radar Receiver C/O Equipment	Special	
	Reconnaissance System C/O Equipment (Radar and Photographic) Crew-Compartment C/O Equipment Rocket Engine Analyzer (Pressure & Electric)	Special		Special			
	Tools (3 sets)	Commer. & Spec.		Commer. & Special	Tools (3 sets)	Commer. & Special	
6) Flight Prepara-tions (one mis-sion every 3 days) 6-1, In Storage Building or Hangar			**Assembly - Payload Section** Stand, Hoist and Slings (One set) Tools	Special Comm. & Special		Special	
			Cradle - skid with casters	Special Special			
	Loading in airplane Load.equip.(1.set) C/O Equipment for assembled aircraft	Special Special					
	Tools (One set)	Comm. & Special			Tools (one set)	Comm. & Special	

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TABLE XII - Cont'd.

Phase	Airplane	Type of Equip.	Booster	Type of Equip.	Reconnaissance Equipment	Type of Equip.	Remarks
6-2) Transfer from Storage to launch site	Dolly for Airplane skid (See 1) / Tractor (See 2)	Special / GFP	Skid (See 1) / Tractor (See 2)	Special / GFP			
6-3) Assembly Alignment, Erection and fix vertical position	A pit is provided on launch site for gas - duct / Assembly and erection equipment (portable) (One unit) / Crane, mobile, Useful load 10t; Height of hook 30', Range of boom 25 ft (See 1) (One unit) / Slings (one set) / Tools (two sets)	Special	Special	Special / Commer. & Spec. / Special / Special			
6-4) Final C/O & Adjustments of assembled airplane-booster in vertical position	Final C/O Equipment, Go-No-Go (One Unit) / Power Supply (One Unit) / Connection Lines (One Set)		Final C/O Equipment, Go-No-Go (One unit) / Power Supply (One unit) / Connection Lines (One Set)				
6-5) Propellant Servicing	Trailer JP-4 (Minimum 2000 gal.) (One Unit) / Trailer Fluorine, Liquid (2500 gal) (One Unit)	Commer. or GFP	Trailer JP-4 (6000 gal) (One Unit) / Trailer, LOX (4000 gal) (One Unit) / Trailer, LOX (4000 gal) (Two Units)	GFP / GFP / Commer. or GFP / Commer. or GFP			Normal JP-4 Storage at Base / Storage at Base in Trailers

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TABLE XII – Cont'd.

Phase	Airplane	Type of Equip.	Booster	Type of Equip.	Reconnaissance Equipment	Type of Equip.	Remarks	Type of Equipmt.
6-6) Power Supply for Ignition & Take-off			1 LOX Storage Tank 66 tons at base LOX Production Units at Base 24t/24 Hr)	Special			Open Syst. for LOX	
			That is: Mobile unit at 6 tons/24 hours (Four Units) or: Mobile unit at 10 tons/24 hrs(3 units)	Commer. or GFP				
			Connection lines for JP-4 Fluorine & LOX (One set)	Com.Mod.				
			Return lines for closed system for JP-4 and Fluorine (One set for every one)	Commer. Modified				
			Pumps for JP-4 & LOX Service attached to the trailer	Commer.				
			Pressure Gas Generator for Fluorine Service (One unit)					
			Gas Pressure (One Unit)	GFP				
			Hydraulic Power if necessary (One unit)	GFP				
			Electrical Power (One Unit)	GFP				
			Take-Off					

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C. Brief Description of Equipment

Some of the major items of equipment are described below. Their use in the operational sequence can be established in Table XII.

Phase 1) - <u>Dolly for Bomber Skid.</u>

This is a mountable device with casters. It should be designed for towing on good roads over short distances.

- <u>Skid for Booster.</u>

This equipment must support and cradle the booster over its whole length. It will stay attached to the booster during the transport, handling and checkout phases. Mountable casters are provided for mobility by tow on good roads over short distances.

Phase 5) and 6-1) - The listed special test, checkout and associated equipment will be designed as a console type equipment and/or as bench sets for in-door use. They will not be mobile. The configuration of this equipment can only be established at a time when the systems in the aircraft are designed. If possible, this item should be of the Go-No-Go type.

Phase 6-3) - The Assembly and Erection Equipment is used in connection with a crane to assemble airplane and booster horizontally at the launch site, to put the assembled aircraft in the vertical position and to secure the erected aircraft in this position during preflight preparations. The unit consists of a basic frame, the erection boom, the crane tower and cable winch machinery. The propellant servicing lines are mounted on the erection boom as well as platforms at various stations for the ground crew. These will be accessible by a ladder. The boom is erected and lowered by cables and a spring assembly takes care of smooth operation in either dir-

ection. This equipment is shipped in parts and assembled at the launch site using a crane. A concrete foundation is not necessary since there are no external forces. The weight of the unit, including payload is supported on a timber foundation. Jacks are provided to level this equipment. In vertical position the aircraft stands on a grill which is supported and fixed on the basic frame. This unit is not removed during take-off. Principal dimensions are:

Length	130 ft. approx.
Width	35 ft. approx.
Height (Tower)	44 ft. approx.
Weight	50 tons

Phase 6-5) The final checkout equipment of Go-No-Go type will be designed for use at the launch site. It must be mobile and adequately designed for adverse environmental conditions. It will be used for final checkout of the erected aircraft.

D. Time Requirements for Flight Preparation.

Table XIII lists the various activities during flight preparations at the launch site, and the approximate time involved.

The amount of time may be cut down when experience shows that some steps can be accomplished concurrently. It is assumed that the Assembly and Erection Equipment is already installed over the pit of the launch site.

TABLE XIII. PRELAUNCH OPERATIONS

Operations	Hours
Transport of Booster and Airplane to Launch Site	.25
Place booster on Assembly and Erection Equipment	.25
Place airplane on Assembly and Erection Equipment	.25
Assemble Airplane and Booster	.50
Put Grill on Assembly and Erection Equipment and Close Restraining Ring	.25
Check	.25
Total	1.75

Remark: During this time all other checkout and servicing equipment will be moved to the launch site

Operations	Hours
Erect the Vehicle	
Lock Erected Boom and Unlock Grill from Boom	
Unfold and Secure Work Platforms	.75
Make Connections for final checkout	
Level and Secure Vehicle in Position	
Final Checkout and Adjustments	1.00
Disconnect Final Checkout Lines	.25
Connect Propellant Servicing Lines	
Fuel Servicing	.50
Cool LOX-Fluorine Tanks = 20 minutes	
Fill Tanks with LOX and Fluorine, with Trailers working in Parallel = 60 minutes	1.50
Disconnect Lines = 10 minutes	

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TABLE XIII - Cont'd.

Operations	Hours
Connect Start Power and Ignition Lines and Open the Restraining Rings	.25
Lower the Erection Boom	.25
Total	4.50
Elapsed Time to Take-Off	6.25 hours

X. DEVELOPMENT PROGRAM

A. General Description.

The program for the design, development, and test of this reconnaissance aircraft weapon system is shown in Fig. 21. First flight would occur 6 years after the start of the program. However, experimental flight testing without the booster and not all of the prototype subsystems could be started a year earlier using the B-36 as a carrier for air launch operation. The technique of air launch (used successfully in the past with a number of airplane systems) would permit an evaluation of handling characteristics for the unpowered landing condition. This type of testing could be extended to include engine operation tests and airplane evaluation to a speed equivalent to approximately $M = 11$. Ultimately initial crew training and reconnaissance equipment capability tests could be performed economically in this manner because booster hardware would not be required and therefore not expended.

To complete the flight test program as shown, 10 airplanes would be required. Approximately 30 flights with the airplane plus the booster have been estimated for the performance evaluation testing.

B. General Work Statement - 1st Year

The first year of the program provides the time to obtain the experimental tests which will prove the practicability of certain design approaches. It also provides a period for clarifying the military requirements with respect to range and special equipment requirements. Generally this work will include the following three important items:

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DEVELOPMENT PROGRAM
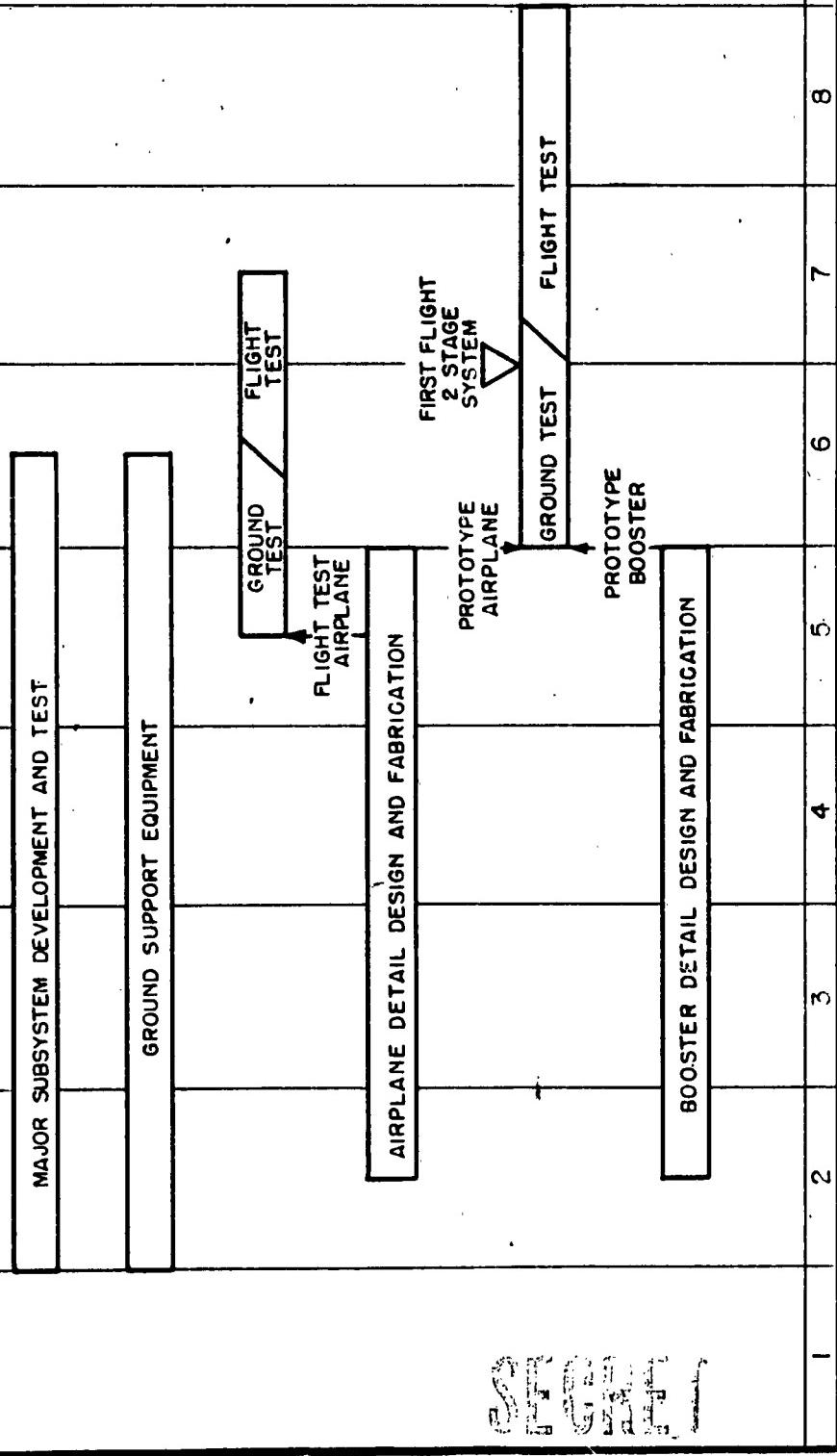
	1	2	3	4	5	6	7	8	9

PREL. DESIGN

PHASE I

MAJOR SUBSYSTEM DEVELOPMENT AND TEST

GROUND SUPPORT EQUIPMENT

GROUND TEST | FLIGHT TEST

FLIGHT TEST AIRPLANE

AIRPLANE DETAIL DESIGN AND FABRICATION

PROTOTYPE AIRPLANE

PROTOTYPE BOOSTER

FIRST FLIGHT 2 STAGE SYSTEM

GROUND TEST | FLIGHT TEST

BOOSTER DETAIL DESIGN AND FABRICATION

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1. **Preliminary Airframe Design and Layout.**

Airplane configuration optimization, booster arrangement, and structural layout of the major part of the airplane would be completed. Equipment requirements would be established and preliminary arrangement in the airframe would be accomplished. Major subsystem requirements would be determined.

2. **Weapon System Analysis.**

This work would include the logistic and ground support aspects of the weapon system, crew work load and effectiveness, and emergency investigations, and the effect of operating bases and target coverage on the ground to-go range requirement.

3. **Experimental Programs.**

Aerodynamic and structural tests to prove the practicability of the weapon system would be initiated. Specific aims of the test program are to verify aerodynamic heating values, control characteristics, and structural cooling and insulation methods (double-wall). Work concerning human factors would also be started using simulators to determine crew capabilities and instrumentation and display requirements.

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XI. CONCLUSIONS

1. The MX-2276 concept of a boost-glide path to achieve a long range flight can be adapted to satisfy the Development Requirements of System 118P.

2. The weapon system is a two-stage vehicle consisting of an airpane with a tandem booster. The booster can be powered by a large thrust rocket engine currently under test for another AF project. The airplane could use the fluorine motor system under development by NAA. However, the fluorine program would have to be accelerated.

3. The airplane configuration will possess suitable flight character- istics for all speed regimes encountered.

4. The crew effectiveness study shows the value of the two man crew which is recommended.

5. Alternate installations of photographic and ferret reconnais- sance equipment are possible. A high resolution side looking radar can be carried on all missions.

6. The search mission camera installation will permit the detection of 25 foot objects and the target detail mission equipment will permit the detection of 9 foot objects.

7. The use of overseas bases for both take-off and recovery sites will permit this weapon system to cover 80% of the key cities of the USSR. A range of 5500 to 6000 nautical miles would provide 100% coverage of the USSR.

8. Subsystem development and airframe design and fabrication can proceed on an accelerated basis permitting the first flight of the weapon system to occur six years after the start of the program. Experimental flight could be initiated one year earlier.

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XIII. APPENDIX

Appendix A. Data Stabilization for Side-Looking Radar

1. Definitions.

q = antenna squint angle; squint forward is positive.

α = total sideslip angle; heading left of ground track is positive

p = pitch angle; nose up is positive

r = roll angle; right wing up is positive

V_g = aircraft ground velocity vector

The total sideslip angle, α, as measured in the horizontal plane. It is the angular difference between the aircraft ground velocity and heading.

Pitch is defined as an angle from the horizontal. It is measured in the vertical plane containing V_g.

Roll is about the longitudinal axis of the aircraft projected onto the vertical plane containing V_g.

The sweep on the cathod-ray tube is a function of x, y and t where x is perpendicular to V_g and y is parallel to V_g.

The trace on the cathode-ray tube is a function of x and y.

2. Derivation of Formulae.

Where α, p and r are zero, the trace is a hyperbola due to the squint angle of the antenna. This is shown in Fig. A-1 for the right side antenna.

3. Instrumentation.

With an antenna radiation pattern that intersects the ground in a straight line, ground range is a hyperbolic function of time; $Rg^2 = Rs^2 - H^2$, where Rg and Rs are the ground and slant ranges respectively, and H is the altitude. The side-looking radiation is a section of a cone's surface and the hyperbolic sweep is not exactly correct. With small squint angles, however, the error is negligible. Therefore a sweep hyperbolic in time is generated. This sweep is made to trace a hyperbola on the cathode-ray tube; the shape of the hyperbola is determined by the altitude and the antenna squint angle. Departure from a straight line, D, is given by: (Ref. 6)

$$D = zq \tan \frac{\theta}{2}$$

where z = altitude in feet

q = squint angle in degrees

θ = beam depression angle in degrees

This departure is approximated by an exponential. The actual trace is then a linear term plus the exponential. The cathode-ray tube sweep is therefore hyperbolic in both time and space.

The sweep circuits generate the parametric voltages $x_3(t)$ and $y_3(t)$ which under no maneuver conditions are correct for driving the horizontal and vertical deflection circuits. During maneuvers the data stabilization computer modifies these in accordance with the aircraft attitude to obtain $x(t)$ and $y(t)$. These new sweep functions are obtained by substituting x and y for x_3 and y_3 according to the equations derived above.

A block diagram of a possible instrumentation is given in Fig. A-5. Computation is done for one antenna only.

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**Ground
Track**

Hyperbolic
Trace

Asymptote

q
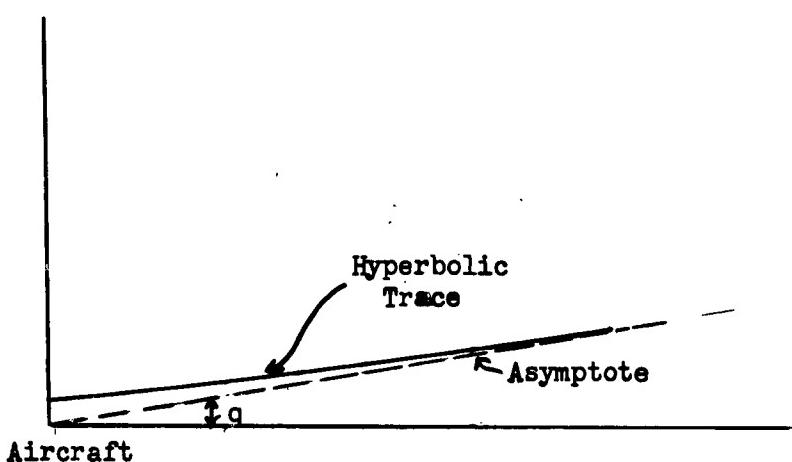
Aircraft

FIGURE A-1 - Radiation Pattern Ground Intercept Under
Conditions of Zero Sideslip, Pitch and Roll

Assume that the hyperbolic trace is known. It is determined by the aircraft

altitude and the radiation pattern of the antenna.

Let (x_3, y_3) be a generic point on this trace. The problem is to

determine the new coordinates of (x_3, y_3) under conditions of sideslip,

pitch and roll. These are determined by assuming successive maneuvers in

the above order.

Sideslip:

Let x_2 and y_2 be the coordinates of the point under conditions of

sideslip only. The values of x_2 and y_2 are readily seen from Fig. A-2.

**Aircraft
Heading** V_g

α

y_2 y_3

x_3

x_2

$$x_2 = x_3 \cos \alpha - y_3 \sin \alpha$$
$$y_2 = x_3 \sin \alpha + y_3 \cos \alpha$$
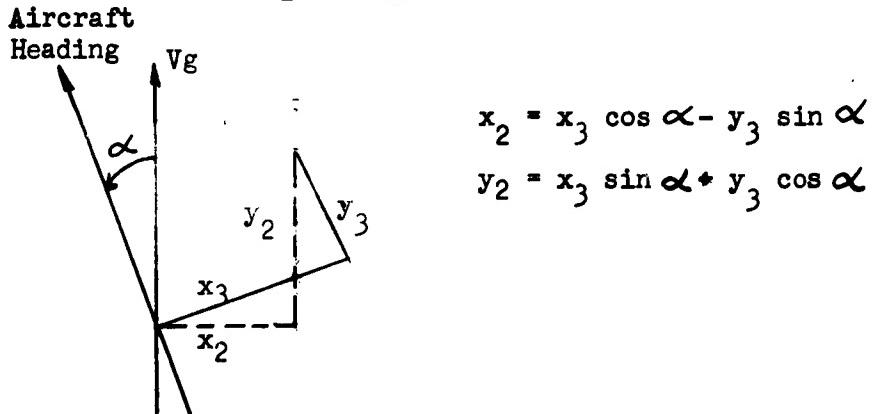
FIG. A-2 - Coordinate Translation with Sideslip

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Pitch:

Let x_1 and y_1 be the coordinates of the point under conditions of sideslip and pitch. The aircraft altitude above terrain is z_2. This is shown in Fig. A-3.
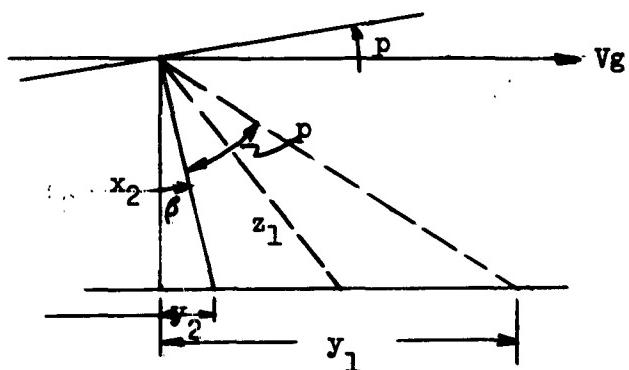

FIG. A-3 - Coordinate Translation with Pitch

Let $\beta = \tan^{-1} \dfrac{y_2}{z_2}$

$$z_1 = \frac{z_2}{\cos p} \simeq z_2 \quad \text{with negligible error}$$

$$y_1 = z_2 \tan(\beta + p) = z_2 \left[\frac{\tan\beta + \tan p}{1 - \tan\beta \tan p} \right]$$

$$y_1 = z_2 \frac{\dfrac{y_2}{z_2} + \dfrac{\sin p}{\cos p}}{1 - \dfrac{y_2}{z_2} \dfrac{\sin p}{\cos p}} = z_2 \left[\frac{y_2 \cos p + z_2 \sin p}{z_2 \cos p - y_2 \sin p} \right]$$

$$y_1 = \frac{y_2 \cos p + z_2 \sin p}{\cos p - \dfrac{y_2}{z_2} \sin p}$$

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Airplane _____ Report D143-945-027
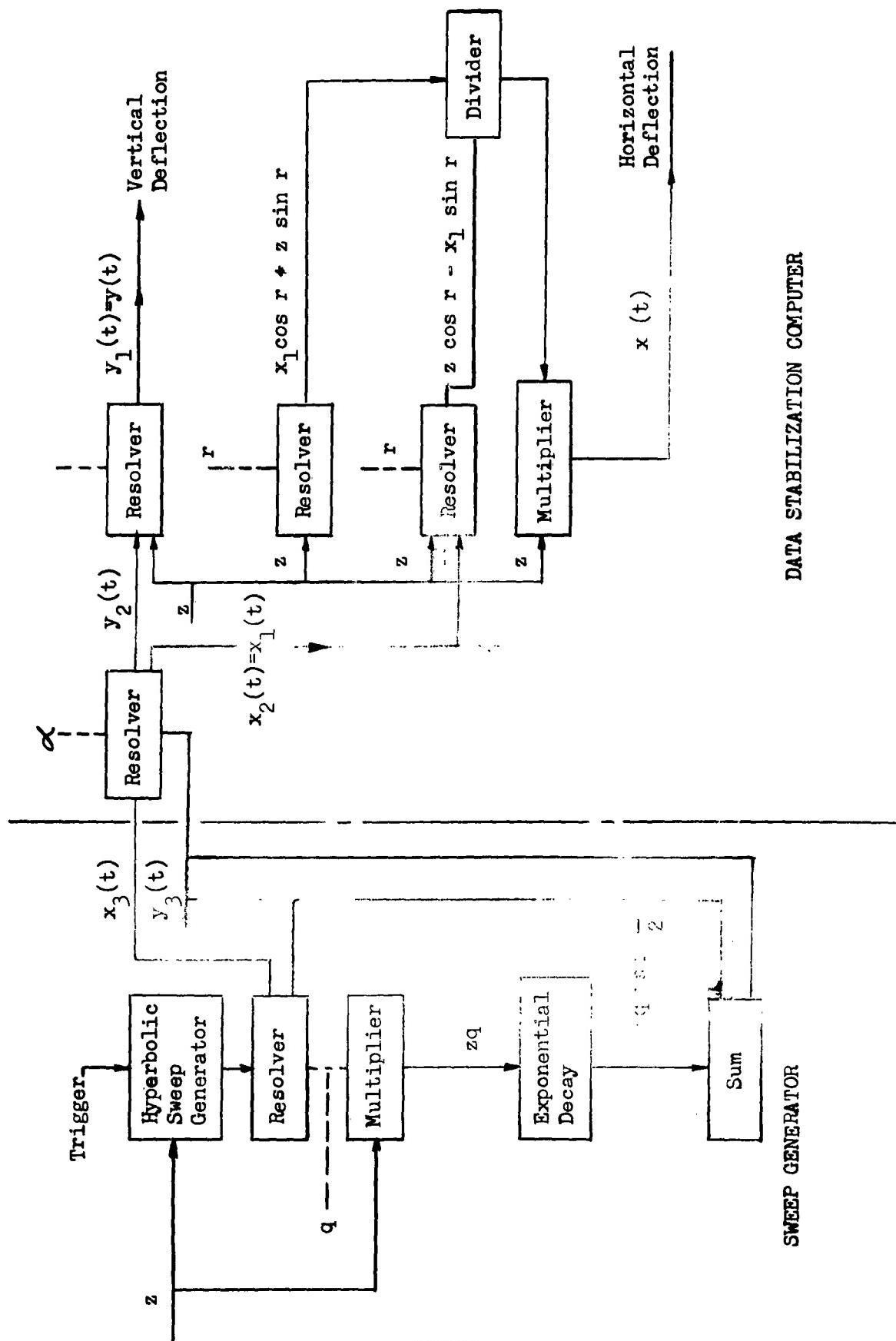
FIG. A-5 - Data Stabilization Instrumentation

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DEPARTMENT OF THE AIR FORCE
HEADQUARTERS 88TH AIR BASE WING (AFMC)
WRIGHT-PATTERSON AIR FORCE BASE OHIO

MEMORANDUM FOR: DTIC – OQ   23 October 2006
                            Attn: Larry Downing
                            8725 John J. Kingman Rd.
                            Ft. Belvoir VA 22060

FROM: 88 CG/SCCM (FOIA Office)
           Bldg 1455
           3810 Communications Blvd
           WPAFB OH 45433

SUBJECT: Freedom of Information Act (FOIA) Cases, WPAFB FOIA Control # 06-443AB and
06-444AB

1. On 17 May 2006, we received a FOIA request for document:

AD125726, "MX-2276 Reconnaissance Aircraft Weapon System. System Design," Bell
Aircraft Corporation, 1 Dec 1955, 157 pages.

AD318479, "Dyna-Soar. Cockpit Display Report," Bell Aerospace, March 1959

2. The documents have been reviewed by the Aeronautical Systems Center (ASC/XR), and
    it has been determined that the distribution statement should be statement A (publicly
    releasable).
3. I am the point of contact and can be reached at (937) 522-3092 or DSN 672-3092.

ABBY L BOGGS
Freedom of Information Act Analyst
Management Services Branch
Base Information Management Division

Attachment
AFMC Form 559
AFMC Form 556
FOIA Request