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KELLOGG

AIR-TO-AIR

SUPERSONIC PILOTLESS AIRCRAFT

ARMY AIR FORCES PROJECT MX-800

PROGRESS REPORT No. 4

SECRET

FOR

U. S. ARMY AIR FORCES
ENGINEERING DIVISION
AIRCRAFT PROJECTS SECTION
PILOTLESS AIRCRAFT BRANCH
WRIGHT FIELD, DAYTON, OHIO

U-13265

THE M. W. KELLOGG COMPANY

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March 13, 1947

ATI No. 1183

Air-to-Air Supersonic Pilotless Aircraft, Army Air Forces Project MX-800, Progress Report No. 4, SPD No. 66, in accordance with U. S. Contract No. W33-038-ac-14221, is distributed as required by Commanding General, Air Materiel Command, Wright Field, letter TSEON-2/CHB/se, dated 9 December, 1946, and the Joint Mailing List for the Distribution of Guided Missile Technical Information, No 2 February, 1947.

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JOB NO. 6162

FEBRUARY 6, 1947

RESEARCH AND DEVELOPMENT

REPORT - SPD -66

SECRET

AIR-TO-AIR SUPERSONIC PILOTLESS AIRCRAFT

ARMY AIR FORCES PROJECT MX-800

PROGRESS REPORT NO. 4

COPY NO. 10

FOR

U. S. ARMY AIR FORCES, WRIGHT FIELD

U. S. CONTRACT NO. W-33-038-ac-14221

THE M. W. KELLOGG COMPANY

SPECIAL PROJECTS DEPARTMENT

JERSEY CITY, NEW JERSEY

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LIST OF ILLUSTRATIONS

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SECTION I

PURPOSE OF PROJECT

SECTION I

PURPOSE OF PROJECT

Project MX-800 is a nine months' study and research program calling for "investigations in connection with the development of a supersonic air-to-air pilotless aircraft for use as a guided missile for the destruction of high performance hostile aircraft". The study and research are to provide recommendations for the continued development work required for the completion of suitable designe for all necessary components, and will include proposals for the additional engineering studies, development tests, and construction necessary for complete development of this pilotless aircraft.

The missils is to have a tactical range of 6000 yards, a speed of the order of 1500 miles per hour, and is to be used against 750-miles per hour aircraft.

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SECTION II

SUMMARY OF WORK CONDUCTED DURING THE PERIOD DECEMBER 1, 1946 TO FEBRUARY 6, 1947

SECTION II

A. SUMMARY OF WORK CONDUCTED DURING THE PERIOD DECEMBER 1, 1946 TO FEBRUARY 6, 1947.

During the past two months the efforts of Project MX-800 personnel have been devoted to the following activities:

1. An ivestigation of sweptback wings.

2. A study of the effect of the location of the main wing, downwash and tail orientation on performance.

3. A study of rolling moments.

4. Preliminary considerations in the design of the booster.

5. A study of the effect of missiles on the mother aircraft performance.

6. Preliminary mechanical design of a missile with a sustainer motor and a detachable booster.

7. Preliminary mechanical design of a missile, without a sustainer motor and a retained booster.

8. Missile evaluation and tactical analysis.

9. A study of the radio equipment for the two-beam command navigation system.

10. A study of, and design considerations of, the short range seeker.

11. An investigation of the missile equipment for the two-beam command guidance system.

12. A discussion of the operator training equipment necessary for missile operations.

13. A discussion of special test equipment necessary for the MX-800.

14. A study of the radar sets and computers necessary in the launching aircraft.

15. A study and discussion of roll control and stabilization.

16. A comparison of the control of a monowing and a cruciform wing missile.

17. A study of the control system for the two-beam command guidance system.

18. A study of the control system for homing with seeker guidance.

19. An investigation of the effects of aerodynamic parameters an design.

20. A comparison of continuous and discontinuous control systems.

21. A mathematical and electronic analyzer studies of the control problem for two-beam command guidance.

22. An estimate of the errors encountered in a two-beam command system.

23. A preliminary study of the launching problem.

Members of the Engineering Staff visited organizationslisted below:

Organization Visited	Subjects Discussed
Applied Physics Laboratory Silver Springs, Maryland	Boosters and Control Devices
Applied Physics Laboratory Fort Miles, Delaware	Testing grounds, methods and equipment

Personnel of the following Organization visited the M.W. Kellogg Company to discuse the problems noted below:

Organization	Subjects Discussed
Boeing Aircraft Co., Inc.	Guidance Systems
Wright Field (Ordnance Lab.)	Control Equipment

SECTION III

AERODYNAMICS

SECTION III

AERODYNAMICS

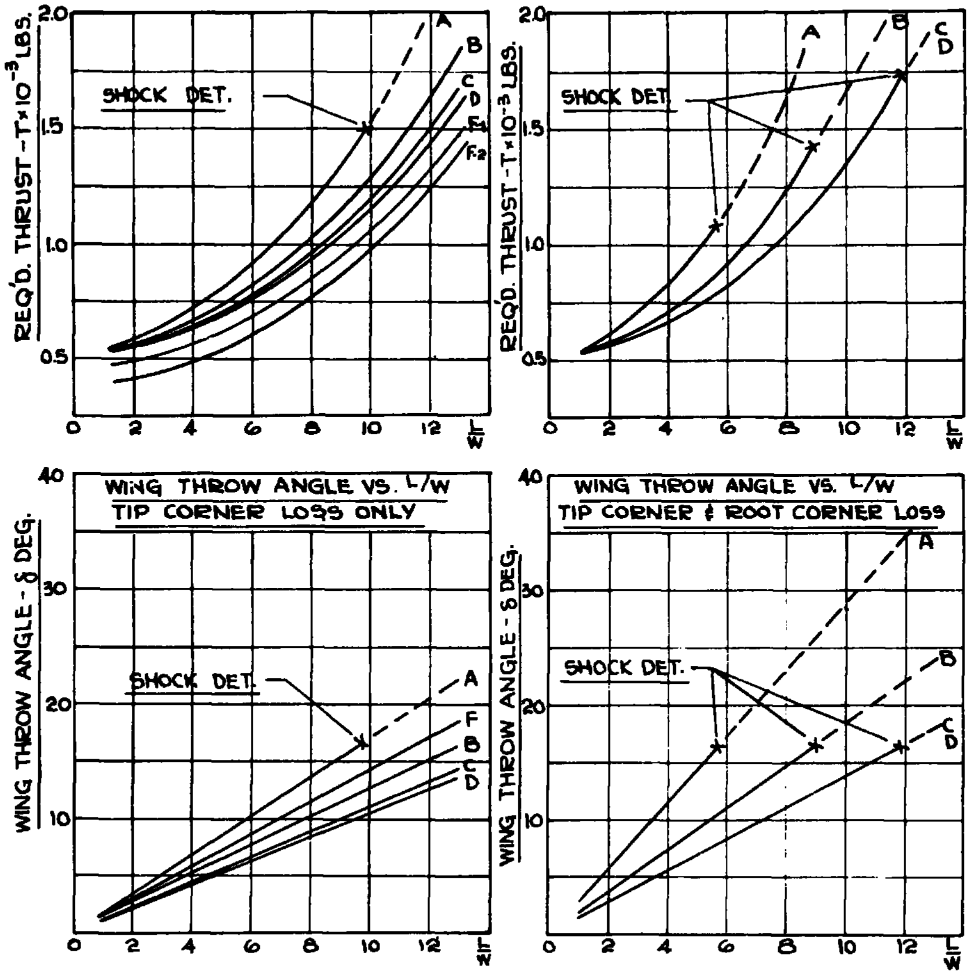
A. SWEPT BACK WINGS:

It is known that relatively high lift-drag ratios can be obtained when the leading edge of the wing lies well underneath the Mach cone. To achieve the latter condition and, at the same time, to avoid an unreasonably long wing compared to the relatively short air to air missile, a speed range lower than the previous range will be considered, namely 1400 to 2400 ft per sec. This speed range and altitude requires that the forward vertex angle of the half wing be not greater than 20 deg. The total drag versus lift-weight ratio for a missile with a delta type wing of 6 sq ft area and a half wing forward vertex angle of 20 deg is shown by curves F-1 and F-2 Figure 1. At a speed of 1700 ft per sec and an altitude of 30,000 ft it is to be noted from the latter curve that the reduction in total drag over the total drag of missiles with wing planforms "C" or "D" (Fig 2) is substantial (11 and 19 per cent respectfully) for lift weight ratio of 9. However, it is also observed that the required angle of attack needed to give a lift-weight ratio of 9 at a speed of 1400 ft per sec at an altitude of 30,000 ft is 17.6 deg compared to 11 deg for that required for the 2.1 aspect ratio rectangular wing, denoted as "C" in Figure 2 . This high angle of attack will certainly prove to be a great disadvantage if the loss of lift occurs at the theoretical shock detachment angle of 9.5 deg corresponding to the flight Mach number. Only if proven that it is the Mach number corresponding to the component of the flow normal to the leading edge which more closely determines the phenomenon will this wing be capable of giving reasonably high accelerations at a reasonable range of speeds. A rough check of the possible "one chance zone" for a missile using a 20 deg delta type planform indicates that it may be smaller than that for a missile having a speed range between 1660 and 2750 ft per sec with a 2.1 aspect ratio rectangular wing. On the other hand the former missile has a launching weight advantage of approximately 50 lb over the higher speed missile.

According to triangular and sweptback wing experiments run by John Stack of NACA (Reference 1) a delta wing experiences erratic behavior of the lift characteristic when the parameter $\sqrt{M-1}\tan w_o$ is near one, where M is the flight Mach number and w_o is one-half the vertex angle of the forward point of the delta. When one examines Stewarts' Theory on delta wings it is expected that this phenomenon is similar to the instability which a straight leading edge wing experiencee when going thru the transonic range. For the 20 deg delta wing under consideration $\sqrt{M-1}\tan w_o$ becomes equal to one at a speed of 2920 ft per sec at 30,000 ft altitude. A maximum boost speed of this missile may have to be limited to about 2400 ft per sec in order to prevent the missile from becoming unstable in the operating range. For unlike the straight leading edge type wing, the delta wing missile does not have the advantage of having the booster thrust available at the time when it could quickly push the missile through its transonic range.

EFFECT OF WING PLANFORM
WITH TIP CORNER LOSS & TIP CORNER-ROOT CORNER LOSS
ARRANGEMENT-B

M = 1.7	W = 350 LBS.	A_t = 3 FT.^{2}
ALT. = 30000 FT.	A_w = 6 FT.^{2}	BODY DIA. = 9"



REQUIRED THRUST & WING THROW ANGLE VS. LIFT WEIGHT RATIO	BY R.B.P. CKD E.S.W. DATE 2-26-47	FIGURE 1

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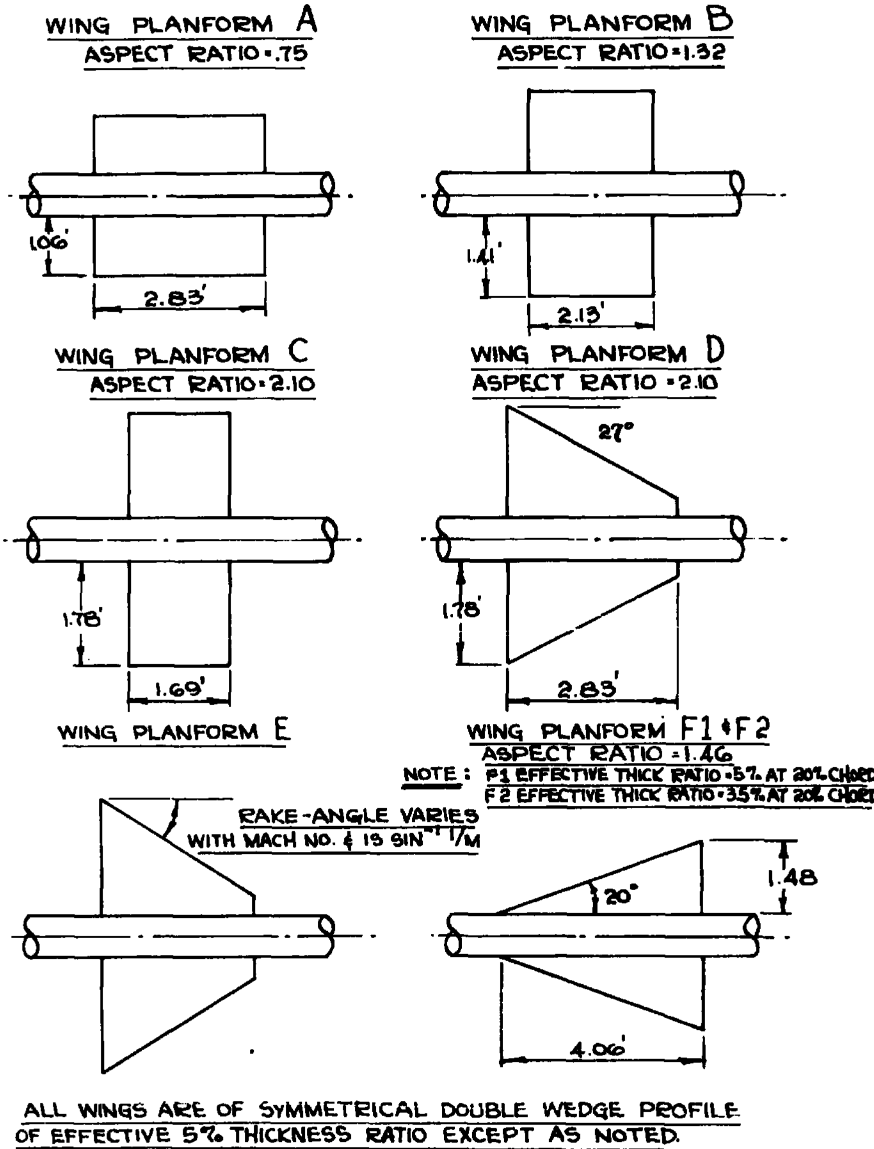


WING PLANFORMS AREA 6 FT.^{2} BODY DIA. 9 IN.	BY R.B.P. CKD E.S.W. DATE 2-26-47	FIGURE 2

Finally a delta wing is indicated to be of questionable value for highly maneuverable air-to-air missilaa by Boeing's wind tunnel experiments. Even when this wing was operating well underneath the Mach cone, tha prasenca of a short body appaars to aliminate its high lift-drag ratio advantage over rectangular wings. Howevar the prasence of a body did not aliminata this advantage for the aweptback wing of constant chord with its leading and trailing edge 21 dag underneath the Mach cone. It appaars that operating thia type wing 21 deg undar a 36 deg Mach cone, corresponding to a Mach number of 1.7, becomes impractical, particularly if the wing is to be made movable.

B. LOCATION OF THE MAIN WINGS:

In Progrese Report No. 3 (Reference No. 2) the effect of the location of the main wing on thatotal drag, attitude of tha fusalage, and the angle of attack of tha wing was shown. By ueing rasults of wind tunnel tests conducted at Aberdeen for tha Boeing GAPA aircraft as a basis, tha affacts of lateral accelerations, interference from the wing, and tail oriantation were computed and are shown in Figure 3.

We may summarize tha reeults as follows:

1. For larga accelarationa, advancing tha wing forward by a small amount results in a decrease in drag, an increase in the angle of attack of the fuaelage, and a decreasa in the angla of attack of tha wing. Thase affects are shown to be small when the accelerations are lass than 5g or when tha wings ara not locatad further than 0.4 ft aft of the cg.

2. Tha effect of intarfarence batween wing and tail ia to mova tha wing location for zaro trim angle aft. This intarfarance has littla affact on the reduction in drag which may ba obtained through moving tha wing forward.

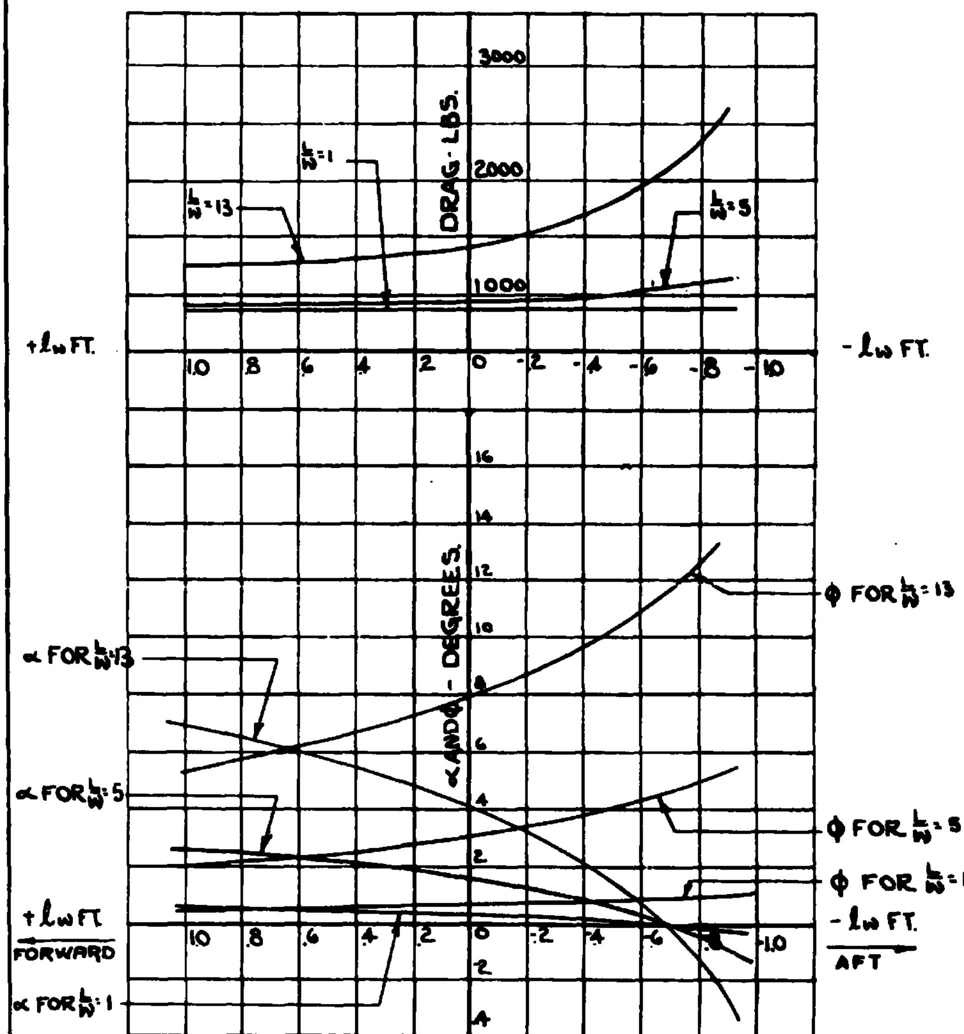
3. By rotating tha tail surfaces 45 deg from the wing surfaces the interference effects from tha wings, ara markedly reduced. Therefora a configuration with a tail rotated 45 dag will be adoptad.

It is to ba noted that for 13 g accalaration the raduction in the total drag raaches a maximum whan the wing is locatad about 0.4 ft forward of the cg; where a reduction in drag of aome 23 per cant is obtainad ovar the zero trim location of the wing. Tha principal advantage of arrangemant B (Reference 3) which ia the identity of the fuselaga axis with tha flight path, is not believed to ba seriously impaired. For this raason tha praaent deaign is based on the center of prassura of tha wing being locatad 0.4 ft forward of tha cg.

C. BOOSTER DESIGN:

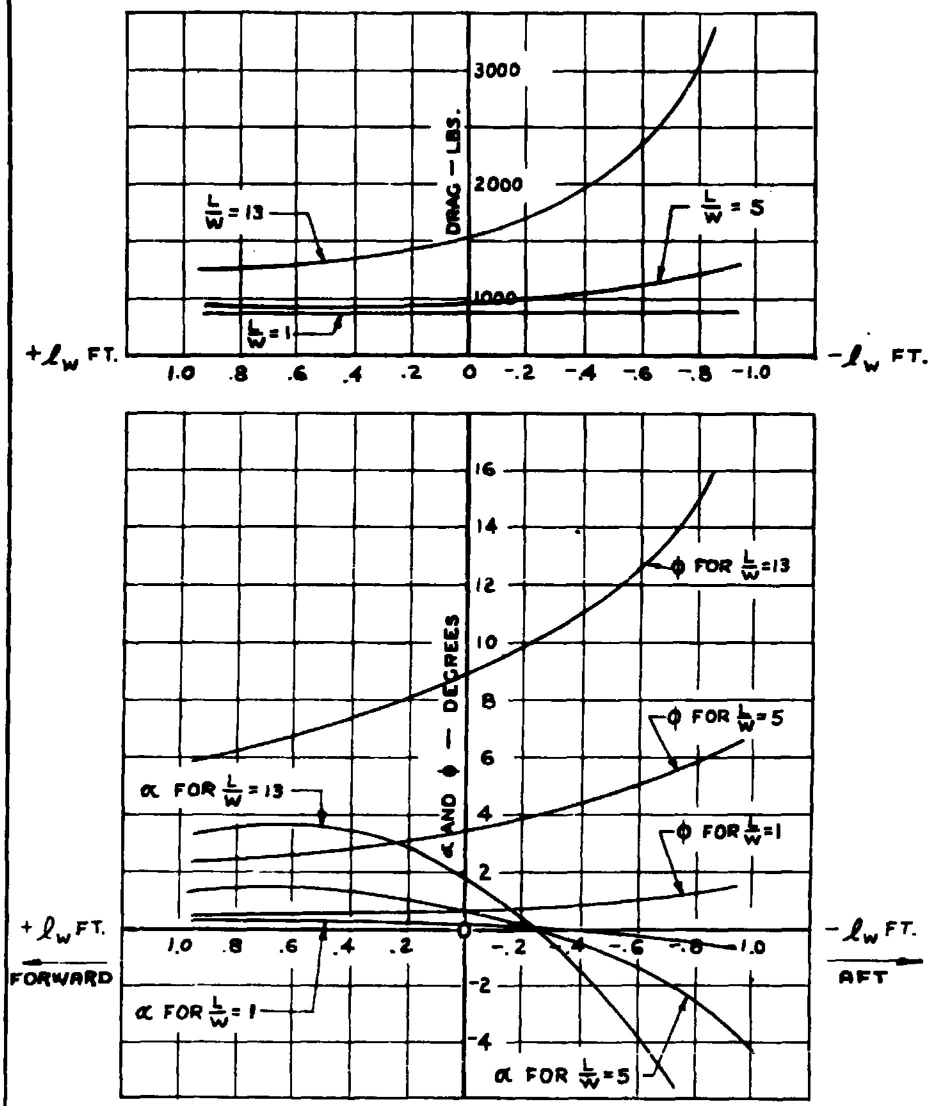
The co-axial type boostar will be tha only type considarad aarodynamically, the parallal type booster baing disregarded bacausa of its more complex jettisoning problem aa well as its apparantly poor aerodynamic faatures.

WEIGHT = 350 LBS. WING AREA = 6 FT.^{2} WING ASPECT RATIO 2:1
FUSELAGE DIA. = 9 IN. TAIL AREA = 3 FT.^{2} TAIL ASPECT RATIO = 2:1
ALTITUDE 30,000 FT. VELOCITY 2700 FT/SEC.



EFFECT OF WING LOCATION ON DRAG, MISSILE ANGLE OF ATTACK α, AND WING ANGLE OF ATTACK φ, FOR TAIL IN HORIZONTAL POSITION AND L/W = 1, 5, AND 13	BY H B CKD A.B. DATE 1-21-47	FIGURE 3

WEIGHT = 350 LBS. WING AREA = 6 FT.^{2} WING ASPECT RATIO = 2.1
FUSELAGE DIA. = 9 IN. TAIL AREA = 3 FT.^{2} TAIL ASPECT RATIO = 2.1
ALTITUDE = 30000 FT. VELOCITY = 2700 FT./SEC.



EFFECT OF WING LOCATION ON DRAG, MISSILE ANGLE OF ATTACK α, AND WING ANGLE OF ATTACK φ, FOR TAIL ROTATED TO 45° POSITION AND L/W = 1, 5, AND 13	BY HB CKD A.B. DATE 1-21-47	FIGURE 3 CONCLUDED

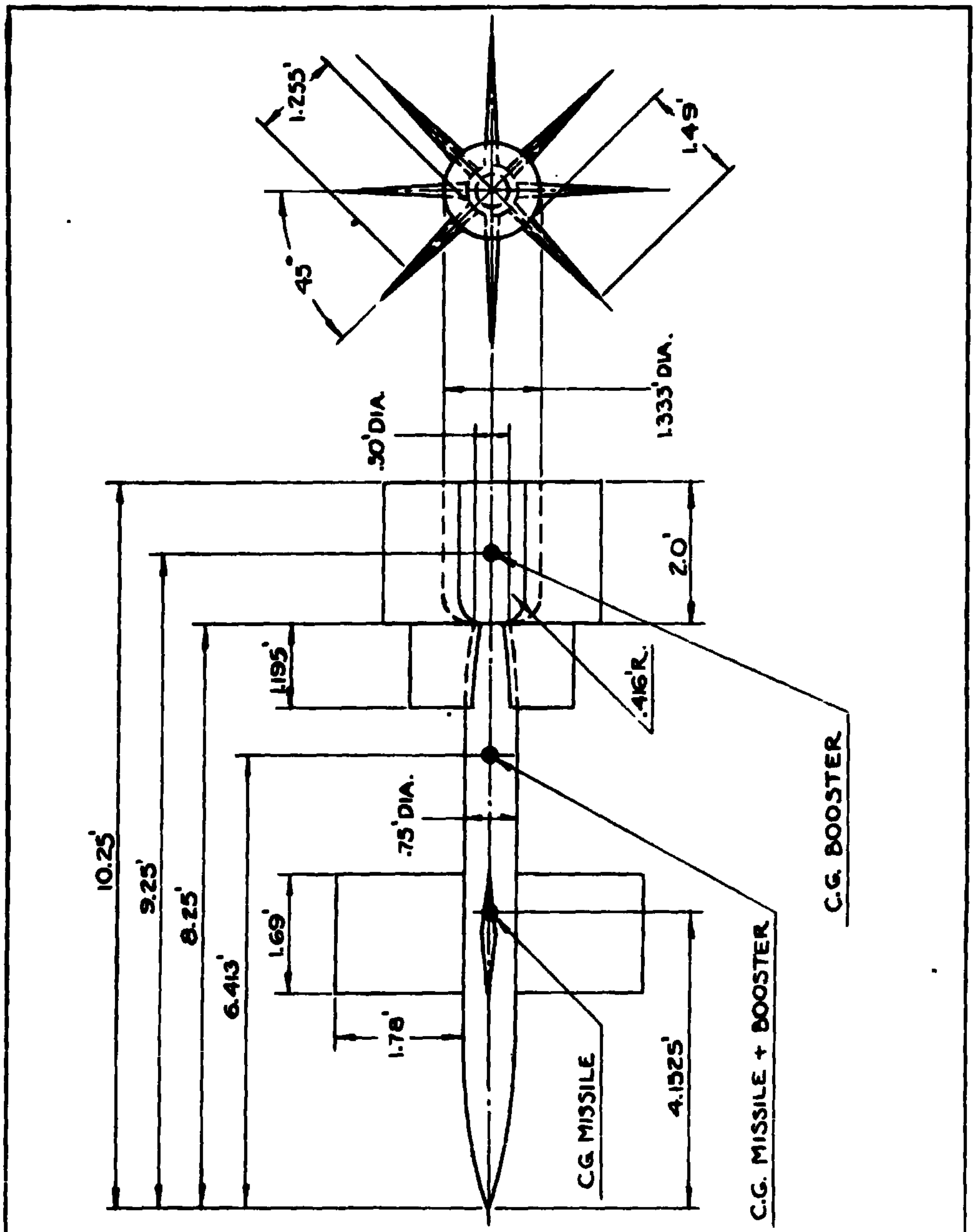
The latter is most important because it is the prime factor in affecting the performance of the mother aircraft. Although the co-axial type at first hand appears to be a less compact design from a launching point of view, a design procedure to minimize this disadvantage has been established.

The problem resolves itself into arranging a given weight booster in the least space by making the booster fins a continuation of the supersonic tail of the missile which minimizes the interference between these two fins as well as the interference from the main wings. In addition the cg of the missile-booster combination is kept as far forward as possible by making the booster of as short a length and as large a diameter as good rocket motor design will permit. In this manner the destabilizing effect of the main wing, is minimized. Furthermore a large enough span of the booster fin is taken so as to minimize the effect of aspect ratio on the lift.

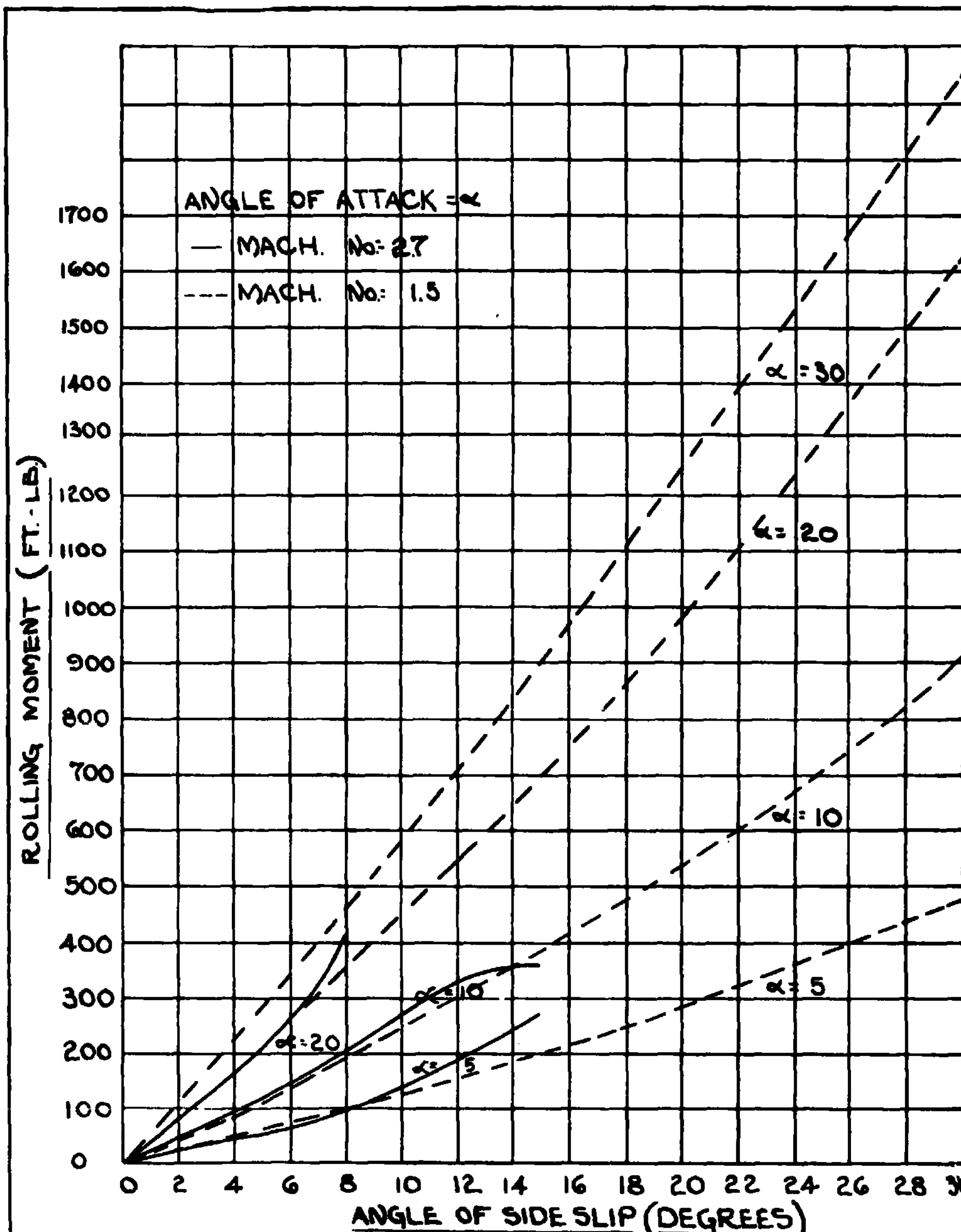
Based upon the above design conditions and having given the relation between booster length and diameter for a given weight of booster the fin area may be determined as the solution of a pair of simultaneous equations. These equations express available fin area as a function of booster length, and required area for neutral stability as a function of booster length. This was done at a speed as high as available data would permit, namely at Mach number of 0.7. The resulting 280 lb booster and fin design for boosting the 350 lb missile in two seconds is shown in Figure 4. It must be stressed that the procedure outlined above is only a first approach to the solution of the problem of designing a tail suitable for the transonic range. The problem of shaping the tail and possibly the wing for stability still remains.

D. ROLLING MOMENTS:

A cruciform missile or any other type which has radial symmetry appears to have no aerodynamic moments which tend to roll the missile continuously about its longitudinal axis. However side slipping a wing causes asymmetrical cone effects on the tips of the wing operating at an angle of attack to produce an unequal distribution of lift across the wing which in turn will produce a rolling moment. A first approximation of the rolling moments of a 2.1 aspect ratio rectangular wing versus angle of sideslip is shown in Figure 5 for several angles of attack and two Mach numbers. It was first thought that a cruciform missile which was designed to trim out at an angle of attack other than zero might have a tendency to roll steadily when both wings were operating at an angle of attack corresponding to a given acceleration. When this missile trims out in such a maneuver, sideslip occurs on both wings such that the asymmetrical cone effects produce rolling moments on each wing which are equal but opposite in direction. The approximate analysis used shows this to be the case no matter what combination of angles of attack of each wing is used since the sideslip is here a linear function of these angles of attack. It is pertinent to mention here that W.F. Hilton of Johns Hopkins Applied Physics group has found a similar cancelling of rolling moments because of the effect of a fuselage on radially symmetrical wing or fin arrangements.



COMPOSITE DESIGN OF 350 LB MISSILE AND 280 LB BOOSTER SCALE 1/2" = 1 FT.	BY WMM CKD A.B. DATE 2-7-47	FIGURE 4



EFFECT OF ANGLE OF ATTACK & MACH NO. ON ROLLING MOMENT VS. ANGLE OF SIDESLIP FOR A RECTANGULAR WING. AREA = 6 FT.^{2} ASPECT RATIO = 2.1	BY R.B.P. CKD A.B. DATE 2-6-47	FIGURE 5

An estimate will now be made of the maximum possible rolling moments which may exist for the missile shown in Figure 6 when flying at 2700 ft per sec at an altitude of 30,000 ft with one wing at a lift-weight ratio of 13 or 7.5 deg angle of attack.

From Figure 5 it is found that a 100 ft per sec gust of wind across the above wing, causing a 2.1 deg sideslip would produce a rolling moment of 30 ft lb. In as much as the missile is trimmed at an angle of 3.5 deg for this case the tail would also produce a rolling moment of 4 ft -lb. A wing dihedral of 2.5 deg resulting from the above load causes an additional moment of 50 ft-lb. Another source of rolling moment could be due to manufacturing tolerance of say 1/4 deg misalignment for each half of the cruciform wing and tail all adding to give a rolling moment of 312 ft-lb. It may further be conceived, for estimating purposes only, that the first mentioned source of rolling moment exists for a very short duration of time only. This would come about by assuming that one set of wings of the cruciform is first being sideslipped by virtue of the finite trim position produced by the other wings and then the first set of wings is suddenly moved to its full angle of attack position (7.5 deg) before the missile has time to trim out. The rolling moment which results from this hypothetical case is 60 ft lbs.

Now assume all of the above conditions exist at the same time to give a possible maximum rolling moment of 456 ft-lbs. For the missile shown in Figure 6 , each half of a pair of the main wings would have to move approximately, 1 degree differentially to stabilize this moment. A set of flippers with a chord of 0.67 ft and half span of 0.7 ft could stabilize this same moment by moving each differentially to an angle of attack of 16 deg. It is to be noted that the wave drag of these flippers alone would increase the total drag of the missile 5 per cent.

Therefore roll stabilizing the missile by moving the main wings differentially is preferred aerodynamically, unless this becomes impractical from the controls point of view.

E. ESTIMATED EFFECT OF MISSILES ON MOTHER AIRCRAFT PERFORMANCE:

An analysis was made to show the effect of two missiles on the performance of the mother airplane. This analysis was made by assuming a set of aerodynamic characteristics typical of a modern fighter airplane and combining these with the thrust values of an actual turbo-jet engine suitable for a fighter design. The following table shows the values assumed for the pertinent airplane and missile characteristics. Included in the analysis are drag compressibility corrections obtained from data in Reference 4. The data are results from high-speed wind tunnel tests conducted on a .3 scale model of the P-47 airplane. Since the airplane analyzed is assumed to be aerodynamically much cleaner than the P-47 airplane, one-half the corrections given in Reference 4 was used in the analysis.

The thrust values used are those for a 011B turbo-jet engine which furnished 3080 lbs of static thrust at SL (5)

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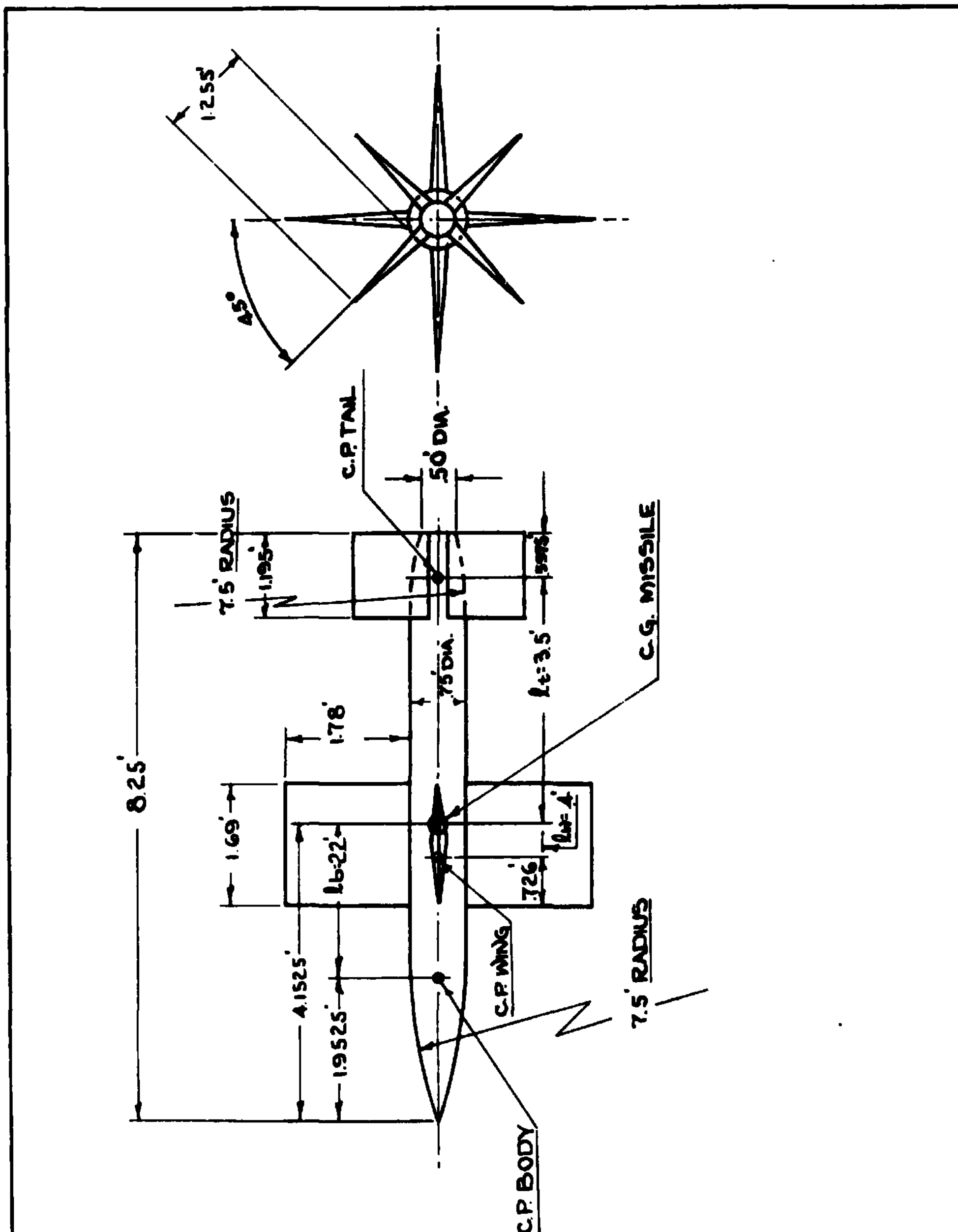
SPECIAL PROJECTS DEPARTMENT

REPORT NO.
SPD 66



COMPOSITE DESIGN OF 350 LB. MISSILE SCALE ½" = 1 FT.	BY R.B.P. CKD A.B. DATE 8-6-47	FIGURE 6

1. **AIRPLANE**

Gross Weight, W, lb	10,000
Wing Area, S, sq ft	250
Wing Aspect Ratio, R	5.0
Efficiency Factor (Oswald), e	.9
Parasite Drag Coefficient, C_{do}	.0180

2. **MISSILE (PER SIDE)**

Weight, W_M, lb	750
Body Frontal Area, A_M, sq ft	.442
Parasite Drag Coefficient, C_{DCM} *	.850
Induced Drag Coefficient, $\frac{dC_{DM}}{d\alpha^2}$ *	.0122

* Based on body frontal area

AIRPLANE AND MISSILE CHARACTERISTICS	BY R.B.P. CKD E.S.W. DATE 2-24-47	TABLE 1

Using these values in conjunction with compressibility corrections obtained from Reference 4, the classical performance was calculated for three configurations:

1. Airplane (no missiles)

2. Airplane plus two missiles (internal stowage)

3. Airplane plus two missiles (external stowage)

The results of these calculations are shown in Figure 7.

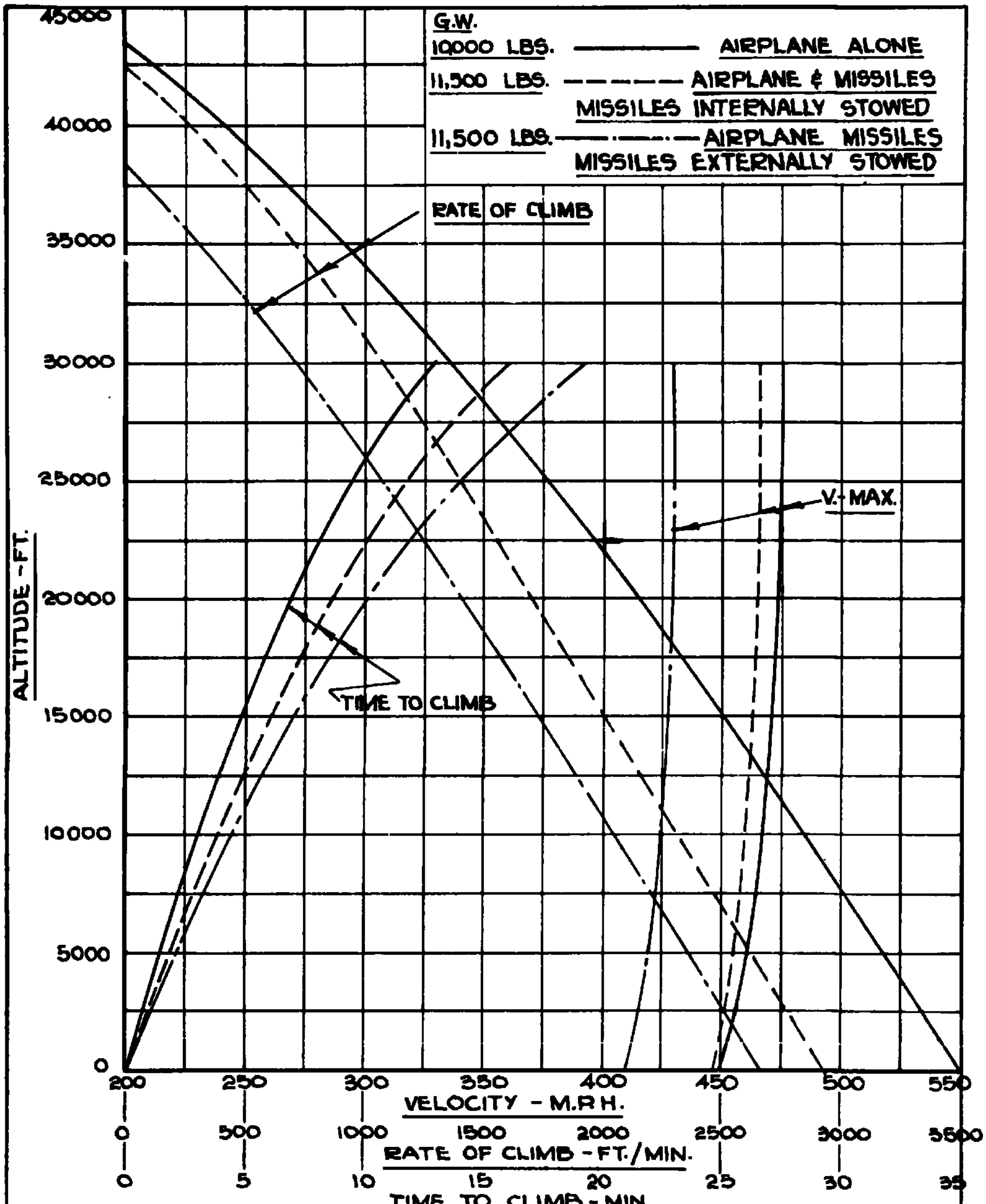
Examination of Figure 7 leads to the following conclusions:

1. Internally stowed missiles yield a reasonably small reduction in mother airplane performance.

2. Externally stowed missiles produce considerable reductions in mother airplane performance, as indicated by Figure 7.

A comparison of the effects of internally and externally stowed missile on the performance of the airplane is given in Table 2.

In conclusion, it should be noted that the values are probably conservative because no account of interference drag between airplane and the external missiles was taken. Experience with underslung bombs and droppable fuel tanks shows that this effect may increase the resultant drag appreciably.



PERFORMANCE OF AIRPLANE WITH TWO 750-LB MISSILES	BY R.B.P. CKD F.S.W. DATE 2-24-47	FIGURE 7

		AIRPLANE PLUS 2 MISSILES INTERNALLY STOWED	AIRPLANE PLUS 2 MISSILES EXTERNALLY STOWED
1.	ΔV max at S.L. - mph	- 5	- 40
2.	ΔV max at 30,000 Ft - mph	- 10	- 45
3.	$\Delta \frac{R}{C}$ max at S.L. Ft per min	- 575	- 800
4.	$\Delta \frac{R}{C}$ at 30,000 Ft. Ft per min	- 275	- 650
5.	$\Delta \frac{T}{C}$ to 3000 Ft. min	+ 2.75	+ 6.25
6.	ΔH_S (service ceiling), Ft	- 1000	- 5300

TYPICAL VALUES FOR EFFECT OF TWO INTERNALLY AND TWO EXTERNALLY STOWED MISSILES ON MOTHER AIRCRAFT PERFORMANCE	BY R.B.P. CKD E.S.W. DATE 2-24-47	TABLE 2

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SECTION IV

STRUCTURES

SECTION IV

STRUCTURES

In the interim period covered by this report the structures group concentrated its efforts on combining all previous mechanical design efforts and the design requirements of the other groups into the mechanical design of two hypothetical missiles. One a 350 pound missile (booster weight not included) with a sustainer motor and a detachable booster and the other a 290 pound missile (weight after boost) without a sustainer motor and with a booster which is retained after the boost period. This work was done for the purpose of illustrating the feasability of physically incorporating the various components into this type of missiles and to bring to light the various problems of a minor nature which may otherwise be overlooked. The results of this work appear in the MX800 Phase I, Final Report (6).

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SECTION V

PROPULSION

SECTION V

PROPULSION

During the period covered by this report the propulsion group worked directly with the tactical analysis group on the missile evaluation problem. This work consisted of the comparison of weights and ranges of missiles having various thrust schedules and motor combinations. The results of this work appear in the MX800 Phase I, Final Report (6).

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SECTION VI

INTELLIGENCE

SECTION VI

INTELLIGENCE

A. RADIO EQUIPMENT FOR THE TWO BEAM COMMAND NAVIGATION SYSTEM:

The two beam command navigation system discussed here is for the purpose of guiding a supersonic pilotless aircraft toward collision with a high performance airborne target. The guidance equipment described operates in conjunction with a target acquistion system and a short range target seeker to guide a missile of either cruciform or monowing configuration along some selected trajectory. The operation of the system as part of the closed cycle missile control system and the selection of trajectories and computation of the command signals is discussed elsewhere (7). The system to be described here may be used with a number of different trajectory choices and provides data transmission channels for a number of differing methods of computing the command signals.

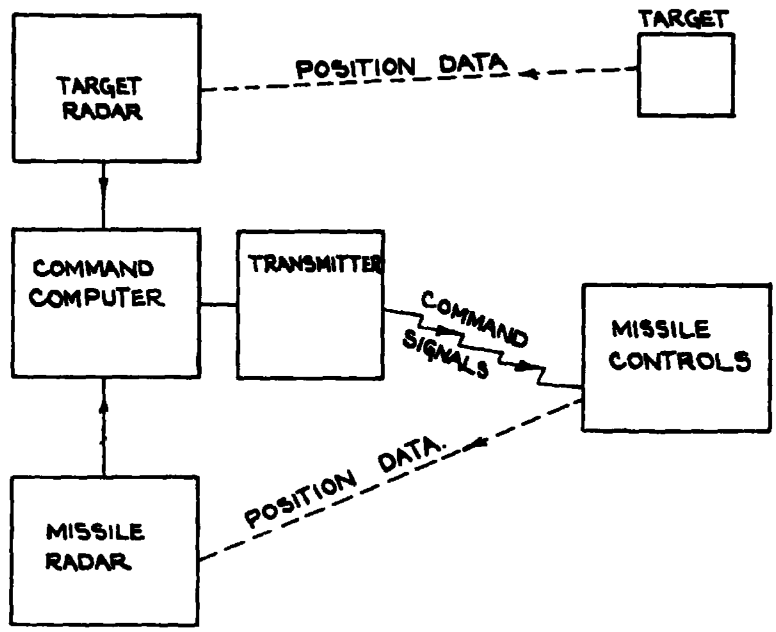
Missiles of the cruciform configuration are equipped with two sets of movable wings or fins which may be turned to give the missile accelerations along two mutually perpendicular axes fixed in the missile. Monowing missiles achieve the desired turns by rolling about the longitudinal axis and turning wings or fins to give an acceleration perpendicular to the longitudinal axis. In either case, two sets of control signals are required in the command system. It is also possible that in the case of a missile powered by a sustainer motor it may prove to be desirable to control the acceleration along the longitudinal axis. This would require a third control channel. As development proceeds, a need for additional control, such as adjusting the sensitivity of a control amplifier as a function of range may be discovered or one or more of the existing needs may be eliminated. Preliminary work has been done on the basis of four proportional channels, or eight on-off channels in the design of the transmission circuit.

A factor which limits the number of channels which may be carried on one transmission system is the maximum permissible time delay in transmission of a command. Since the command transmission link is a part of a closed cycle control system, the time delay must be held to a minimum. If, for example in the pulsed command system, we use 1,000 pulses per second, and a train of 10 pulses is required for a complete set of command signals, then the complete set is sent 100 times per second, and the maximum delay between insertion of the command in the transmitter and its reception at the receiver is 1/100 of a second plus the time of transmission through the circuits and space between the receiver and transmitter (50 microseconds at 10 miles). If two sets of pulses bearing the same command are required for security, the overall transmission time is a little more than 1/50 second under the given assumptions. If the number of channels is increased, increasing the number of pulses per set, then the overall transmission time is also increased.

Figure 8 is a block diagram of the major elements of the two beam command navigation system. In this system, the target is observed continuously by a tracking type of radar set mounted in the launching aircraft, the missile is observed continuously by another tracking set, and the command signals required to make the missile follow a given trajectory are computed at the launching aircraft and transmitted by radio to the missile. In practice, each of the blocks shown is fairly complex because of the complicated operations carried out, and because of the three primary coordinate systems used. Since the missile and the launching aircraft are both moving with respect to a fixed frame of reference, and the air through which the flight is carried out is a fixed or rather slowly moving medium, we need coordinate systems fixed in the missile, the launching aircraft, and the air (or space). The coordinate systems listed are all free to rotate with respect to each other, and we must therefore carry out transformation from one to another in order to relate the data observed in the aircraft coordinate to a command signal which should be given in missile coordinates. This problem is discussed from the standpoint of the computer in another report (7), but it also touches on the design of the command radio equipment, since it shows that one or more channels for information from the missile to the launching aircraft may be required. Following the terminology of a captured German report on the subject (8), we may have the command signals referred to any one of the three sets of coordinates and evaluated either at the launching aircraft or at the missile. Figure 9 shows block diagrams for the following cases: 9a, missile referred missile evaluation; 9b, missile referred aircraft evaluation with the assumption that the missile is roll position stabilized and flies along its longitudinal axis; 9c, missile referred aircraft evaluation with roll position transmitted from missile to aircraft and with the assumption that the missile flies along its axis; 9d, missile referred aircraft evaluation with the missile roll position stabilized and with the missile attitude transmitted to the launching aircraft. The last system listed will require two data channels from the missile to the launching aircraft to provide coordinate transformation data.

As we may see from Figure 7, a command which is sent to the missile influences the direction of flight of the missile and thus its space position at any instant. The change in space position is observed by the missile tracking radar set, and the position data fed into the computer is one of the things used in computing further commands to the missile. This means that the missile control system contains a feedback path. The operation of the entire system is thus that of a closed cycle control system or servomechanism. The existence of the feedback imposes certain conditions on the operation of some of the components used. These are discussed in a report on the overall control problem. It must be remembered, however that changes in any of the parts of the closed loop may have an adverse effect on the stability or accuracy of the loop.

The equipment for a typical two beam command system includes a target tracking radar set, a missile tracking radar set, a command computer and auxilliary equipment installed in the launching aircraft; and a command receiver, demodulator, beacon, and auxilliary data transmitter installed in each missile to be controlled.



MAJOR ELEMENTS OF THE TWO-BEAM COMMAND NAVIGATIONAL GUIDANCE SYSTEM	BY G.L.H. CKD E.S.W. DATE 2-1-47	FIGURE 8

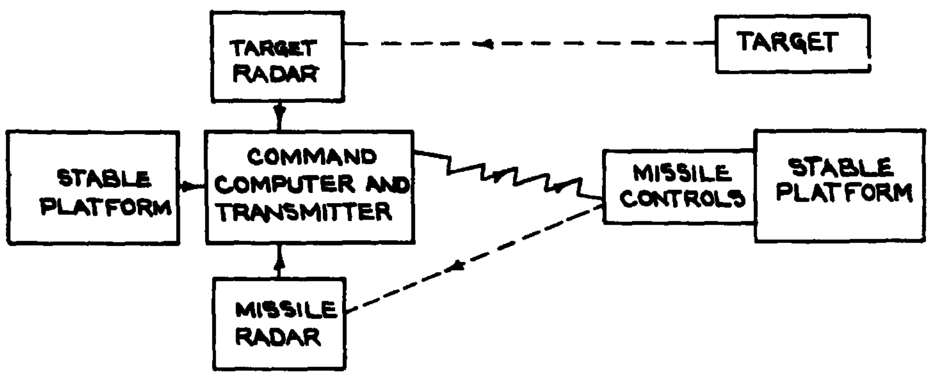


Figure 9a. Missile referred missile evaluation. Commands are computed with reference to space axes and transformed at the missile to missile axes

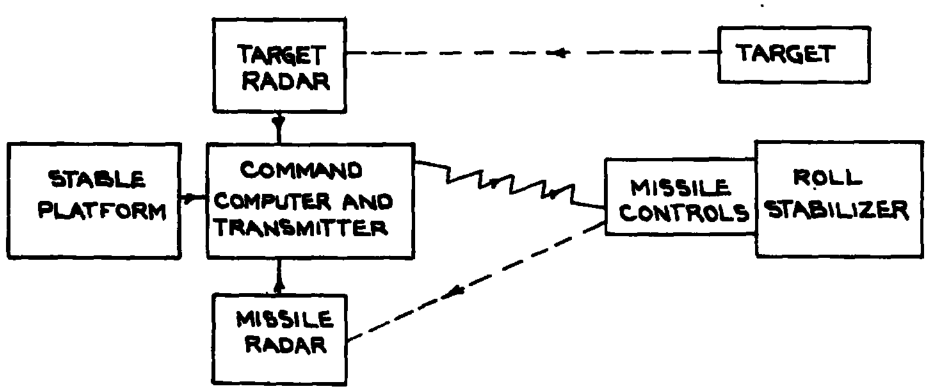


Figure 9b. Missile referred aircraft evaluation. Commands are computed in approximate missile axes by knowledge that the missile is roll stabilized and assumption that it flies along its longitudinal axis.

POSSIBLE REFERRAL AND EVALUATION SCHEMES	BY G.L.H. CKD E.S.W. DATE 2-1-47	FIGURE 9a 9b

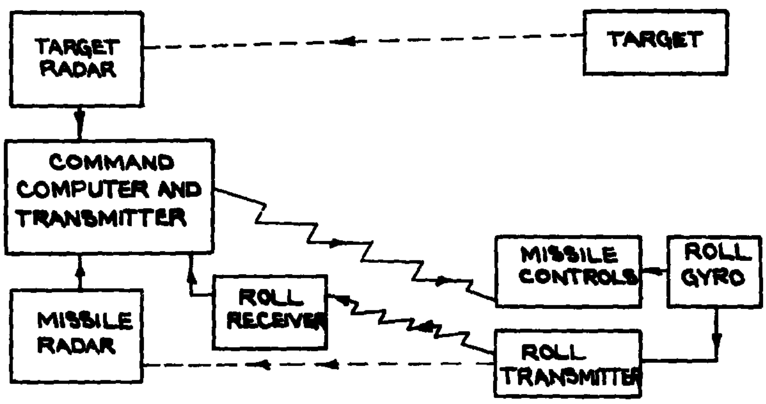


Figure 9c. Missile referred aircraft evaluation. Commands are computed with respect to missile axes. Missile roll position is transmitted to computer and it is assumed that missile flies along its longitudinal axis.

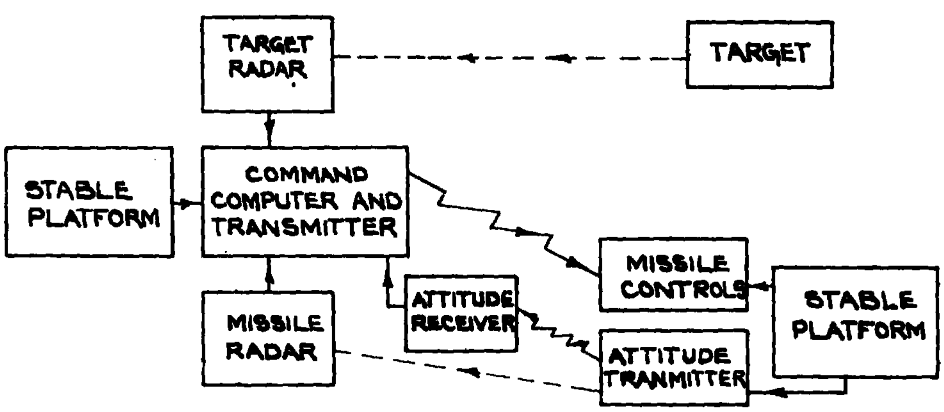


Figure 9d. Missile referred aircraft evaluation. Commands are computed in missile axes. The missile is roll stabilized and the attitudes of the missile in space axes is transmitted to the command computer.

POSSIBLE REFERRAL AND EVALUATION SCHEMES	BY G.L.H. CKD E.S.W. DATE 2-1-47	FIGURE 9c 9d

The target tracking radar set may be one of the type now normally used for gunfire control from aircraft. It must track very accurately in angle at low angular rates. Range accuracy should also be good, but it is of secondary importance to angular accuracy. The antenna system is probably of the moving axis type, using either a conical scan or the newer monopulse technique for accurate direction finding. The servos which drive the antenna may contain extra smoothing circuits to improve the accuracy in slow tracking rates at the expense of rapid angular tracking. The radar antenna may also have data resolvers directly attached to transform the radar tracking data from the coordinates of the launching aircraft to the coordinates of the missile or of free space, depending on the type of computation which may be employed.

Tracking of the missile with a simple radar set may be possible to the desired ranges, but the dependability of the system can probably be improved considerably by the use of a radar set which tracks a beacon in the missile. The missile tracking radar may then be a fire control radar set similar to that used to track the target, except that the missile radar receiver may be tuned to a beacon frequency different from the frequency of its own transmitter, and the antenna system may be modified slightly in view of the higher received power available from the beacon and the possible desire to stabilize the polarization of the transmitted energy from the missile tracker. The stabilization requirement will arise only if it is not possible to supply sufficient power to the missile to trigger the beacon from all parts of the conical scan. In case the monopulse type of tracker becomes available in time for use, the stabilization will have been accomplished. It will probably prove economical to use the missile tracker transmission to carry commands to the missile and the response of the missile beacon to bring necessary guidance data back from the missile to the command computer.

It is possible to describe the separate components of the complete system quite fully in terms of present techniques. A complete discussion of the equipment for data transmission from aircraft to missile and from missile to aircraft, using the pulsed transmissions of the missile tracker and beacon and the receivers of the missile and the tracking radar may be found in reference (9).

B. <u>SHORT RANGE SEEKER:</u>

1. <u>Introduction.</u> In an earlier report (10) the fundamental aspects of the seeker problem were developed and a general procedure for selecting the optimum seeker type was outlined. In the interim period covered by this report, a detailed study of the trajectory problem was conducted, tentative performance specification for a seeker were presented and the development of a novel fixed-axis relative-amplitude seeker was proposed.

2. <u>Relation to Navigation System.</u> From a study of the errors encountered in the two beam command system (17) it is quite clear that barring a substantial (about 5:1) increase in radar set accuracy, a relaxation of the present 30 foot miss distance requirement, or a restriction of launching aircraft to detonation point range to 10,000 feet or less, a seeker of some sort will definitely be required in the MX-800 Missile. The 5:1 improvement in

radar set accuracy is a possible but not a probable outcome of refinement of the monopulse tracking technique. The 30 foot miss distance now specified can be shown to be very nearly optimum for a non-atomic warhead. If the miss distance is increased by some factor $K > 1$, the present 90 lb warhead weight must be multiplied by a factor roughly equal to K^3 in order to maintain the same kill probability. Tactical analysis for MX-800 have shown that the major portions of the lethal zone must be at distances greater than 20,000 feet from the launching point for effective missile design. The launching aircraft travels at less than half the missile speed and the launching aircraft detonation point range will therefore, exceed 10,000 feet for detonations more than about 20,000 feet from the launching point. There appears to be little likelihood that use of a seeker in MX-800 can be avoided.

3. Trajectories. In making a detailed study of seeker design features (17), including weight and space requirements, it was necessary to establish tentative specifications covering the required seeker performance. An analysis of the seeker trajectory problem and relative considerations was made and the approximate translational behavior characteristics which the MX-800 must have during the seeking phase are tabulated below:

a. Seeking Initiation Program. Seeking should be initiated at a constant missile-target distance for all approach angles. This procedure is nearly optimum from a trajectory view point and is simplest from an equipment point of view.

b. Seeking Initiation Range. The optimum missile-target range for initiation of seeking varies with the navigation and seeking systems details and appears to be between 4000 to 2000 feet for the conditions assumed. The optimum range is the range at which the seeker's ability to recognize trajectory errors first becomes equal to that of the navigation system.

c. Missile Course. The trajectory analysis indicates that an approximated zero rate of change of bearing angle course $\dot{\psi}_{MT} = 0$ is best. The ability of MX-800 to make a kill is severely limited during the seeking phase by the poor resolution of any practicable seeker and by the large minimum turning radius of the missile. Full control must be applied quickly whenever the trajectory error becomes great enough to be observed by the seeker if the missile is to be effective.

d. Seeker Angular Resolution. The trajectory analysis indicates that, for the assumptions stated, an angular resolution width of between 1/2 deg and 1 deg will provide sufficient seeking accuracy.

e. Smoothing Time. For the conditions assumed in the trajectory studies the optimum data smoothing time for the control of translational motion in the seeking phase is about 0.5second. Because of servo and control system limitations, discussed in a parallel report on the MX-800 control system, it may be advisable to make the seeker smoothing time as short as practicable and effect the necessary smoothing in the missile control system.

f. Response Time. The trajectory analysis indicate that the translational response time of the missile to guidance data during the seeking phase should be of the order of 0.1 second or less. Control system and angular stability considerations may require that the response time of the seeker itself be considerably shorter that 0.1 second.

4. Seeker Circuit Considerations: In considering the specific seeker design it was necessary to choose between various possible methods which offer nearly equal advantages. In order that the most promising systems may be chosen for future study the various possibilities were investigated (17) and some of the results are presented here.

On the basis of an advantage in the relative sizes and weights of the pulse radar seeker and the unavailability of the necessary components for the CW radar seeker, it was decided that further investigation of specific seeker designs will be devoted principally to pulsed seekers. This does not discontinue all considerations of CW radar seekers for many of the basic principals of pulsed radar seekers apply with slight modifications of technique to CW seekers as well.

At the present time it is not clear whether it will be more desirable to illuminate the target by a radar transmitter carried in the missile or by energy from the parent plane. The principal faults in the parent plane illuminated system are that the varying illumination of the target resulting from conical scanning of the transmitter antenna bean of the launching aircraft will be difficult to cope with and that difficulty may be encountered at the seeker discriminating between the energy arriving directly from the launching aircraft and that reflected from the target, especially at close range when the times of arrival nearly coincide. These difficulties are not insurmountable and the parent plane illumination system may prove to be advantageous by reason of weight and space savings in the missile.

The choice between fixed and movable axis antenna systems for the seeker involves comparison of the relative sizes and weights and the relative accuracies of the two systems. A movable axis antenna seeker is probably somewhat larger and heavier, but fixed axis seekers compose a relatively new field whose potentialities are not yet fully known. In general, the fixed axis seeker is probably less accurate than a movable axis seeker employing relative amplitude null-seeking for bearing determination.

In addition to the problems just discussed, there arise questions as to the manner of obtaining bearing data from the signals received at the antenna, especially in the case of the fixed axis antenna seeker, the form which the seeker output signals should take, the coordinate system in which the output data should be presented, etc.

As previously indicated, a final decision as to the type of seeker to be employed in MX-800 must wait further information and investigation. However, more detailed discussion of some of the various possibilities, including block diagrams of representative circuits are given in reference (17).

The receiver shown by the solid lines of the block diagram of Figure 10 represents a typical receiver for the simplest form of pulsed seeker employing either target illumination by the missile or by the launching aircraft. It consists of a scanned antenna, which may be of either the fixed or movable axis type, and a conventional superheterodyne receiver for the reception of pulsed signals. Automatic frequency control of the local oscillator is employed and automatic gain control is applied to the six stage intermediate frequency amplifier. One or two stages of video frequency amplification supply video signals to the data reduction system.

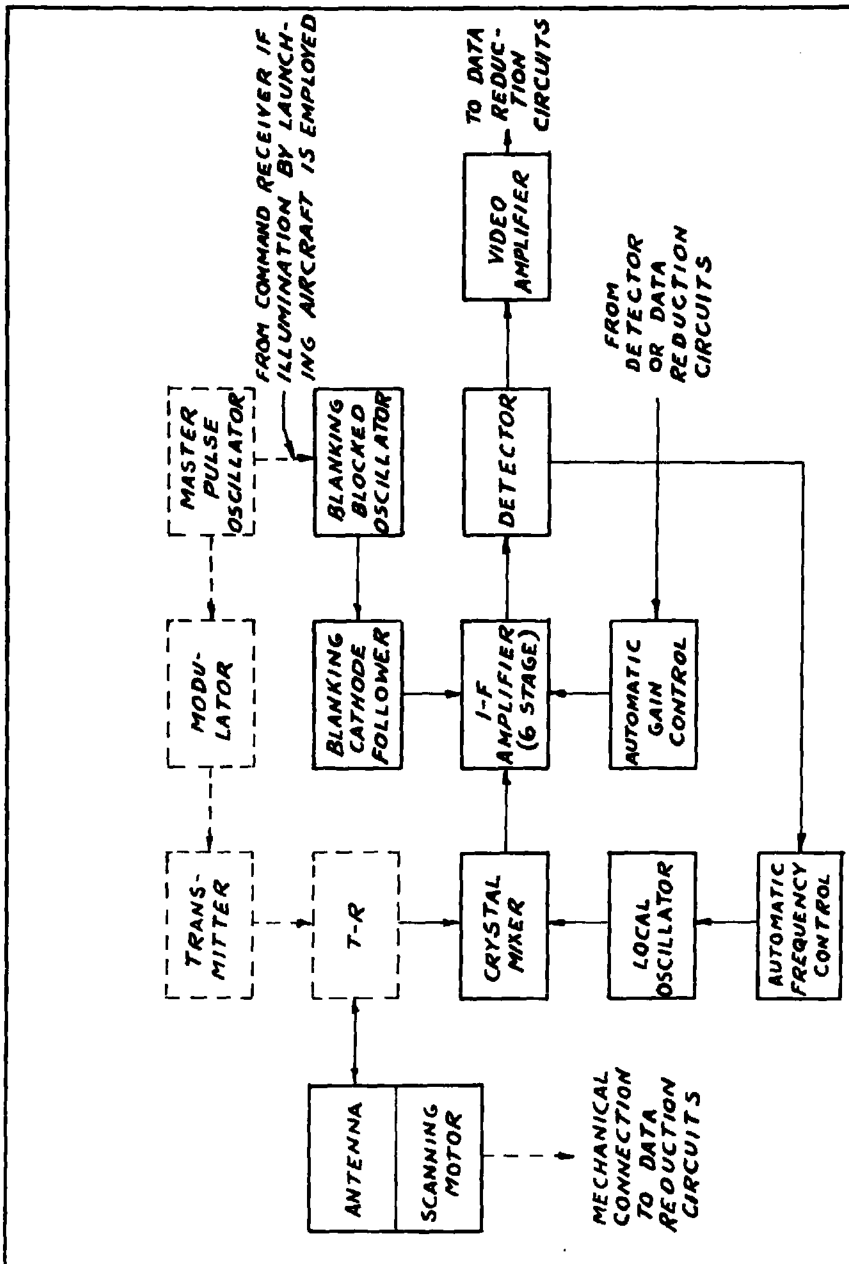
A blocked oscillator and a cathode follower are provided to supply short duration blanking signals to the intermediate frequency amplifier. The blocked oscillator receives triggering voltage from the transmitter master oscillator in the case of missile illumination of the target or from the command receiver in the case of launching aircraft illumination of the target.

The required sensitivity and general excellence of the automatic gain control circuits is to some extent a function for the data reduction method employed. A number of systems for seeker data reduction have been studied and a complete discussion of these systems may be found in reference (17). Figures 11 and 12 are block diagrams typical of these systems. Figure 11 is a block diagram of a seeker data reduction system, which is particularly suited for use with a movable axis antenna system and Figure 12 is a block diagram of a seeker which might be employed with a fixed axis antenna.

5. Heat Seekers. Use of a heat rather than a radio seeker offers a number of advantages, particularly for short ranges where atmosphere attenuation does not limit performance seriously. Perhaps the most important advantage in the case of MX-800 is that the wave-length of the radiation involved is shorter with the result that the seeker antenna or reflection directivity is limited by the ratio of antenna dimension to detector dimension rather than by the ratio of antenna dimension to wave-length.

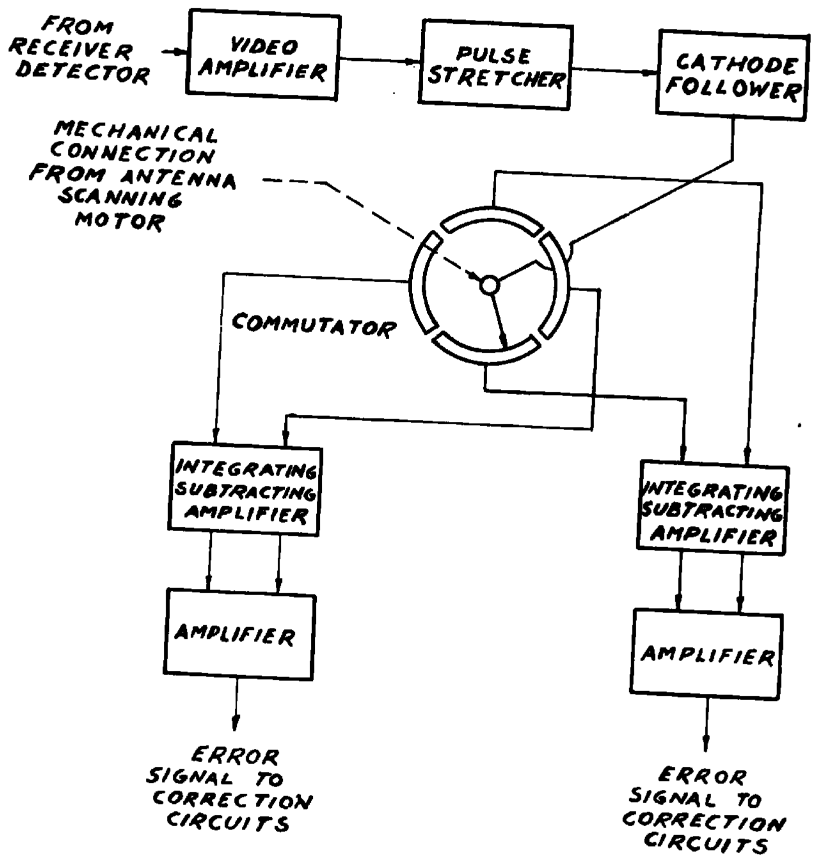
Disadvantages of the infra-red or heat seeker are that the present non-existence of suitable sources preclude the use of a reflection type seeking system, and make the seeker dependent upon radiation originating at the target. Since there are many other possible sources of heat radiation, notable the sun, inability to select and track the desired target may detract seriously from the reliability of the heat seeker.

Data reduction or evaluation circuits for the heat seeker approximate closely those required for the corresponding radio seeker. The basic classifications of seeker systems, as described elsewhere, hold equally for the heat and radio seeker. Given a sufficiently small detector of adequate sensitivity, the antenna or reflector problem is simpler for the heat seeker than for the radio seeker. Elaboration of the above general comments must await a detailed study of the heat seeker.

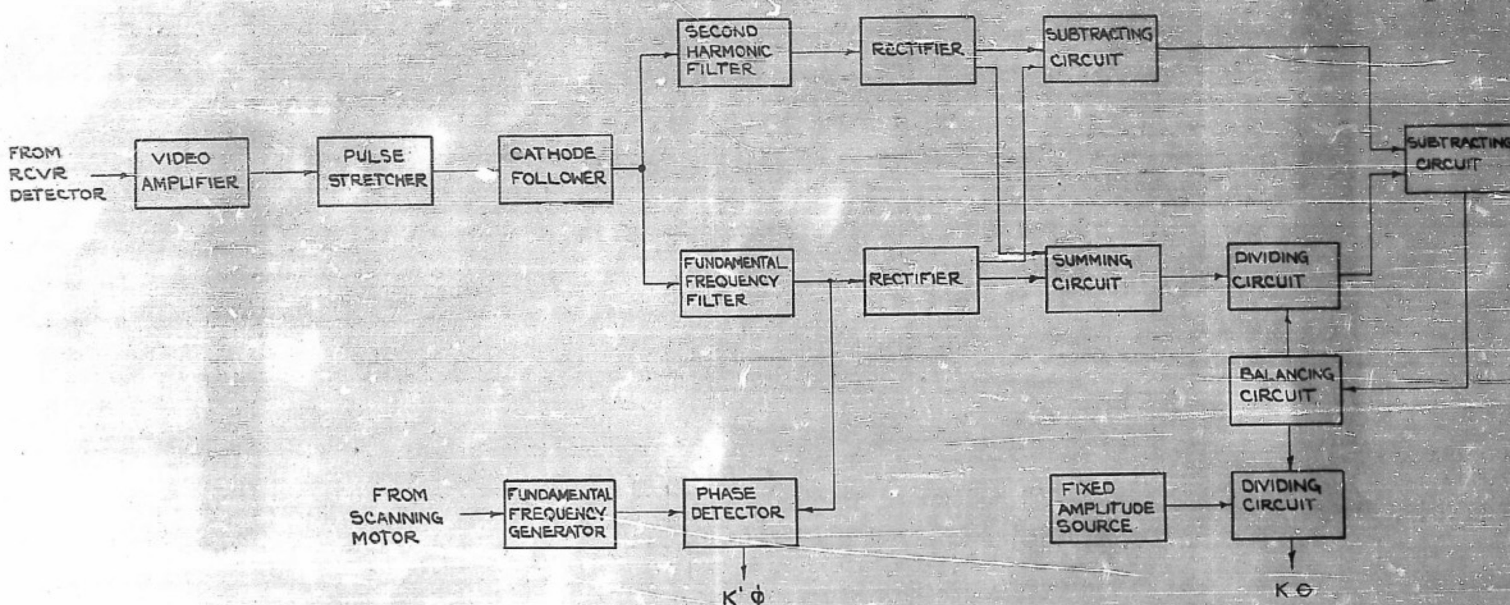


NOTE: BLOCKS SHOWN BY DASHED LINES ARE REQUIRED ONLY FOR TARGET ILLUMINATION BY MISSILE.

BASIC SEEKER RECEIVER AND OPTIONAL PULSE TRANSMITTER	BY G.L.H. CKD E.S.W. DATE 2-1-47	FIGURE 10



DATA REDUCTION CIRCUITS FOR SEEKER - MOVABLE AXIS ANTENNA	BY G.L.H. CKD E.S.W. DATE 2-1-47	FIGURE 11



FIXED AXIS SEEKER - MODIFIED RATES OUTPUT BY HARMONIC COMPARISON	BY G.L.H. CKD E.S.W. DATE 2-1-47	FIGURE 12

6. Seeker Antennas. As a result of the trajectory studies which have been made, a tentative angular resolution requirement of 1/2 to 1 deg have been estabilished for the seeker. Under favorable conditions this angular accuracy requirement can be met with a flexible axis seeker of conventional design incorporating a paraboloidal antenna of 6 inch diameter. It is believed that this requirement can also be met with the less conventional seeker designs proposed in this report and in the references. The flexible axis seeker may be more subject to angular errors caused by the presence of a dielectric nose cap, which according to current aerodynamic data must be in the form of a cone with half angle of 12.5 degrees or less.

Except for the preliminary surveys of the seeker antenna problem reported on in references (10) and (11), antenna developement for MX-800 has awaited specification of seeker accuracy requirements and classification of missile dimensions and nose shape. An active antenna research program is now being initiated on the basis of available information.

C. EQUIPMENT FOR THE TWO-BEAM COMMAND GUIDANCE SYSTEM:

The overall plan for guidance of the MX-800 missile envisions the use of radar for target detection and tracking before launching. Pre-launching maneuvers will be directed by a computer so that the missile may be launched in the most favorable direction and at the optimum range from the target. After, launching, control of the missile will be by a two-beam command radar system, in which one radar beam tracks the target and another the missile, and the control signals directing the missile are determined in the launching aircraft through the operation of a command computer. The commands will be sent to the missile through the missile tracking beam, and a beacon located in the missile will respond to the tracker and transmit missile data to the tracker for computing purposes. At the end of the trajectory, control will be transferred to a target seeker, probably of the radar type, which will make last minute corrections in the flight path of the missile, and as the missile approaches the target, it will be detonated by a radio type of fuse.

The radar sets and computers which must be mounted in the launching aircraft and the missile in order to accomplish the above functions have been studied and are described and discussed in references (18) and (19).

D. SPECIAL TEST EQUIPMENT AND OPERATOR TRAINING EQUIPMENT:

While it is still to early in the development of the MX-800 to design the test and training equipment in detail, a preliminary study of the situation was conducted in order to help in estimating the magnitude of the needs and in the laying out of a program of development to go along with the missile development. The results of this study may be found in reference (18).

SECTION VII

CONTROLS

SECTION VII

CONTROLS

A. ROLL CONTROL AND STABILIZATION:

In preceding reports and discussions one important basic assumption was made that the missile is roll stabilized or roll controlled. Roll as here interpreted means pure roll of the missile about its longitudinal axis in contradistinction to an apparent roll which might result from a maneuver. Roll stabilization thus infers that, ideally, in the case of a cruciform missile that the axis of one pair of control surfaces remains at all times horizontal. Roll control as applied to the monowing type missile implies that the missile rotates about its longitudinal axis in such a manner as to maintain the main wing in the correct plane of orientation to give the desired trajectory.

In a spinning cruciform missile the missile is caused to spin about its longitudinal axis and each of the four control surfaces changes from rudder-action to elevator-action every quarter of a revolution. A vertical reference is necessary to transform guidance control signals to wing angle signals as the missile rolls. The amount of instrumentation necessary in a spinning missile is not essentially reduced in amount from that necessary in a roll stabilized type and the method of intelligence data transmission is more complex in nature. Because of the changing of action of each wing, there is a high power loss in the main wing control servo for a highly maneuverable missile. For example, a control surface may be required to change from one extreme position to the other extreme position in a time representing one-quarter of the period of rotation of the missile. Thus, in general, the control surface would oscillate back and forth and absorb relatively large amounts of power which would not directly contribute to the maneuver of the missile.

In a cruciform type missile utilizing the two-beam command system of guidance, roll stabilization is not absolutely necessary but it is highly desirable since it prevents the "flip-flop" of the wings described above, and it makes the data transmission and interpretation system simpler. The former is true since one particular control surface acts either as a rudder or an elevator and retains its single action at all times. The latter is true since guidance data sent to a control surface, to effect a maneuver of the missile, if computed on the basis that the control surface concerned has a definite normal orientation at all times, requires less modification than if these data must be modified to take into account motion of the control surface as the missile rolls.

The use of a target seeker does not require a vertical reference, but lack of stabilization imposes serious conditions on the seeker. Since a target seeker will not be used alone but only in conjunction with a two-beam command phase, it will be necessary to have a vertical reference in the missile.

In the stabilized missile a gyroscope or similar device is used to establish a reference vertical. The roll stabilization system has the functions of detecting deviations of the vertical axis of the missile from this reference vertical and correcting these deviations in such a manner as to restore the missile to normal attitude in roll in the shortest practicable time. The roll stabilization system also has the important function of establishing the vertical reference as quickly as possible after the missile is launched from the parent plane so that lateral and vertical maneuvers can begin early in the flight.

For one missile under consideration there will be a 2 second boost from a speed of 500 ft per sec to a speed of 2600 ft per sec and for the remainder of the trajectory it will coast. It seems, in view of uncertainties in the prediction of wing loads in the subsonic and transonic regions of missile speed, that roll control might be ineffective or unsatisfactory prior to the time that the missile reaches a speed corresponding to a Mach number of one. In this case, roll stabilization would have to be established in the time between missile velocities of 1100 ft per sec and 2700 ft per sec. Assuming that a lineer variation of speed exists during boost, the time to establish roll control would be 1.45 seconds. This appears entirely feasible and will allow lateral control at the instant the boost is ended.

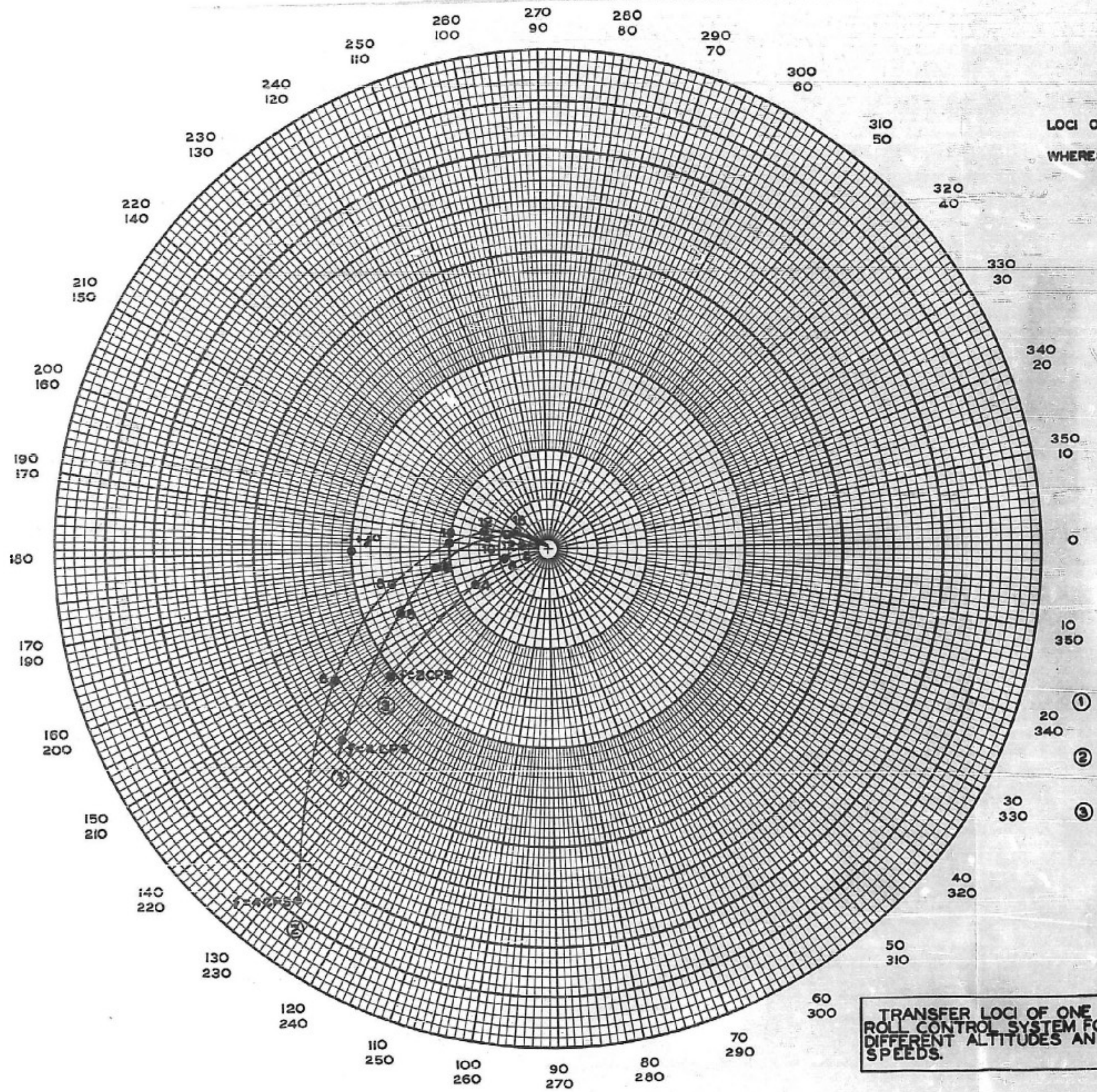
Some of the problems in obtaining good roll stabilization are:

1. The natural presence of wing misaligment torques which vary with speed and altitude. The presence of such torques indicates the desirability of a torque compensated roll control which will develop a steady control torque without requiring an error in stabilization. It is proposed to include an amplifier stage sensitive to the integral of its imput as well as to its imput to accomplish this.

2. The possible appearance of oscillating torques caused by simultaneous slip of the missile in yaw and pitch planes. These oscillating torques will be worse if the pitch and yaw controls are poorly damped. It is therefore advisable to make the roll control system have a high resonant frequency (7) and a lead network is recommended for inclusion in the roll control to aid in accomplishing this, as well as to improve the degree of stability of roll control.

3. The aerodynamic coefficients change with speed and altitude and alter the dynamics of the missile. In Figure 13 are sketched several transfer loci of a roll control system showing the change in its performance as a function of speed and altitude. The worst cases between 15,000 ft and 50,000 ft are shown.

4. If a free vertical gyro is used, the roll control system sensitivity must be compensated as a secant function of the angle of climb or dive and becomes ineffective when these maneuvers are vertical. Since this is a possibility according to the tactical analysis it is recommended that an



LOCI OF $\frac{\phi_i}{\phi_o}(j\omega)$

WHERE: ϕ_i IS "VERTICAL" REFERENCE

ϕ_o IS MISSILE ROLL ORIENTATION

ω IS FREQUENCY OF SINUSOIDALLY VARYING INDEPENDENT VARIABLE ϕ_i

① SYSTEM AT 30,000 FT. AND 2700 FT./SEC.

② SYSTEM AT 15,000 FT. AND 2700 FT./SEC.

③ SYSTEM AT 50,000 FT. AND 1400 FT/SEC.

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TRANSFER LOCI OF ONE ROLL CONTROL SYSTEM FOR DIFFERENT ALTITUDES AND SPEEDS.	BY I.P.G. CKD I.P.G. DATE 1-6-47	FIGURE 13

Form KL 13

integrating rate gyroscope be used for the basic reference.

5. There is inadequate viscous damping from roll rate torques to provide good stability of the roll control without servo compensating networks.

A detailed mathematical development of one roll control system is decribed in Reference (7)

B. COMPARISON OF MONOWING AND CRUCIFORM CONFIGURATIONS:

Most of the studies made as a part of the MX-800 project have been based on a cruciform configuration of the airframe. Several months before the end of the study phase of the contract the mechanical design group and the aerodynamicists proposed that a monowing configuration might have an advantage as carried by a parent plane. The possibility of using the monowing configuration was then referred to the guidance and control groups.

The majority of the work on the monowing configuration thus far has been a library research to study the efforts made by the Germans in this direction. Two technical papers were found (References (8) and (12) which proposed to compare the monowing with the cruciform from the point of view of control. In introductory comparison of the aerodynamic properties of the two configurations References (13) and (14) do not exactly agree.

	Per Unit Weight		Per Unit Drag		
	Ref 8	Ref 12	Ref 8	Ref 12	MX-800
Monowing	0.8	0.9	0.87	0.65	0.85
Cruciform	1.0	1.0	1.0	1.0	1.0

But both indicate a trend which has been checked by aerodynamicists for the MX-800 project (calculations based on a 350 lb 9 inch diameter missile)

The control signals for a roll stabilized cruciform missile are quite simple to conceive since one can reason without considering a guidance interaction between pitch and yaw axes as a first order effect. In the case of a monowing configuration the situation is somewhat more difficult. Obviously, the aircraft must be rolled so that the main wing axis will be at right angles to the required lift. Considering incremental quantities, and an initially correct missile flight, an error vector can be thought of in terms of components along the main wing axis and perpendicular to it and to the longitudinal axis. Alternately the error vector can be assigned a magnitude and an angle with respect to a line perpendicular to the plane of the main wing axis and the longitudinal axis. The former would be called rectilinear missile evaluation and the latter polar coordinate missile evaluation. In the latter case the roll control may act to keep the angle zero, but then there is an indefinite control when the error integrating rate gyroscope be used for the basic reference.

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The use of a monowing configuration for a highly maneuverable missile has been questioned on the basis of the rapidity with which it can act to correct deviations from the desired trajectory. This question is based on the procedure of the monowing of rolling before acquiring lift in the proper direction whereas in a cruciform the lift is obtained directly by setting the perpendicular wings at appropriate angles of attack. A detailed analysis has not been made of this as a part of the MX-800 project because of lack of time, and because also it can be done more easily with an analyzer than by mathematics because of the complication introduced by the guidance interaction among axes in the monowing case.

In Reference (14) a comparison was made in the calculated performance of a beam guiding system of a low speed (200 m per sec) F25 missile and an equivalent cruciform.

It was concluded that there exists no essential difference between control half periods with rudder and aileron control of the monowing. However, the cruciform missile was characterized by a faster response having a half period 40 per cent less than that of the monowing. This would seem to favor the cruciform from the point of view of speed of response.

In concluding this discussion of the monowing two factors should be pointed out:

1. That guided missiles of the glide bomb type such as the "Bat" have been built with monowing configurations and successfully flown (Reference (15).

2. That supersonic flight should offer the possibility of improving tremendously the speed of response in roll, but that the success of monowing control depends on good precision in roll control which has not yet been demonstrated for this class of missile.

C. CONTROL SYSTEM FOR TWO-BEAM COMMAND GUIDANCE:

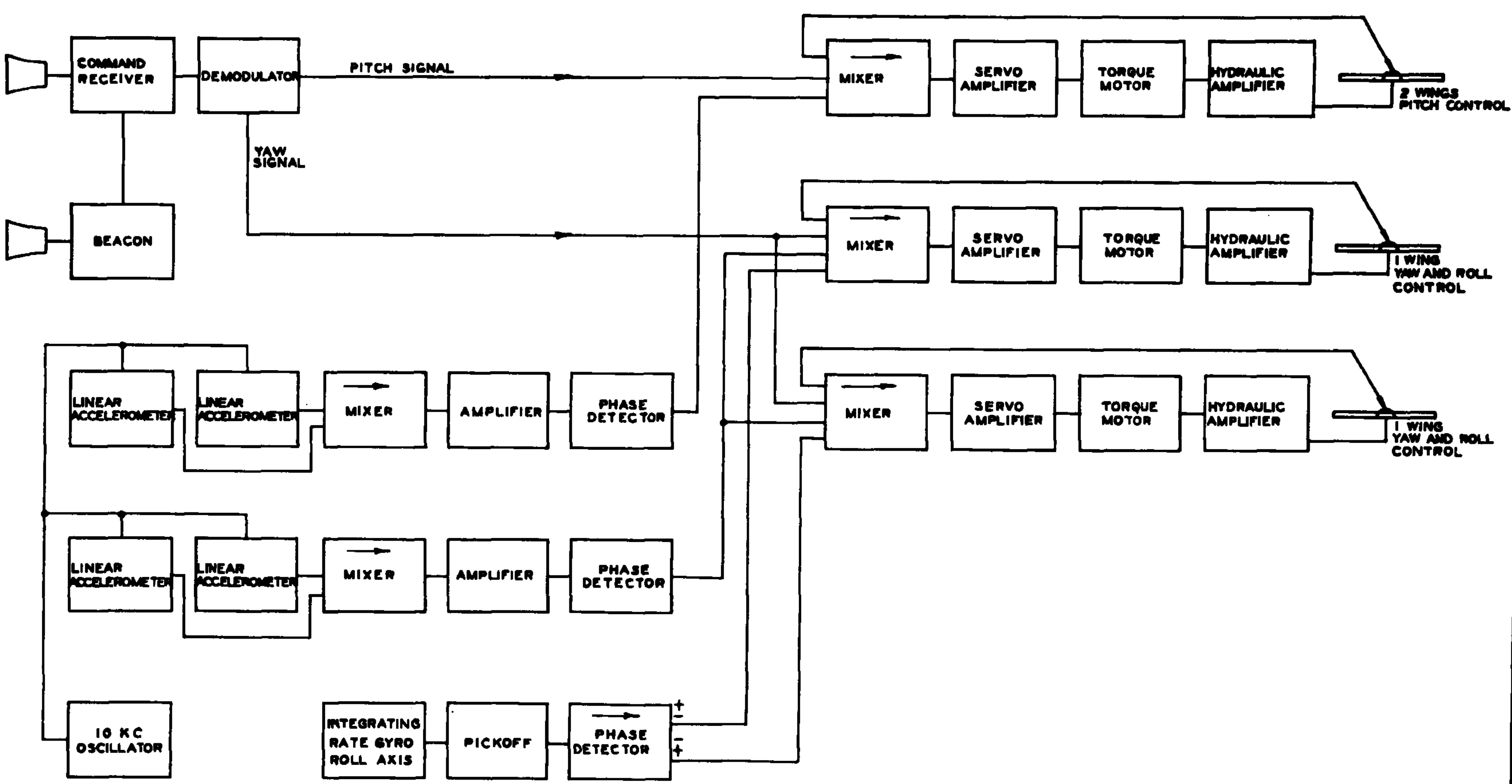
It has been recommended, as a result of guidance studies, that at least the initial stages of flight of an air-to-air supersonic missile should be commanded from the parent plant by a two-beam command system. In this type of flight there are two control loops which merit investigation. The first of these considers target motion as an independent variable and missile motion as the controlled variable. The controlled variable is adjusted so that there is no rotation in space of the line joining the missile and the target. The second control loop is a subsidiary to the first and deals with the ability of the missile to carry out the commands received by it and produce a stable flight. As a part of this second problem it is recommended that a roll stabilized cruciform missile be developed.

The mathematical requirements of the command computer to be carried in the parent plane is discussed in Reference (7) . The inter-connection of the components of a proposed control system for the missile is shown in Figure 14 . Control signals are received from the command receiver and after demodulation are sent to the pitch and yaw channels. Wing motion control of a cruciform airframe has been assumed with differential control of the motion wings for roll stabilization. Here, the roll control is shown as operating differentially on the same wings which receive the command yaw signals. Absolute motion transducers are used in the pitch and yaw controls systems in order to improve stability of the missile flight. The analysis of the two-beam command system implies that the desired feedback at absolute motion can be accomplished by the use of two linear accelerometers placed on either side of the center of gravity. This need for improved stability stems from the little, or no, inherent stability of a highly maneuverabile air-to-air missile.

A theoretical analysis has been made of the system described in Figure 14 and some supporting data obtained from an electronic control analyzer. From these studies some estimate can be made of the dynamic requirements of components of the control system, and also of an optimum position for the location of the main wings of the aircraft. These estimates are, of course, dependent upon the estimated aerodynamic coefficient for the air-frame. In Table 3 are given the equations of motion of the aircraft in the traverse plane, the equation used for motion of the aircraft about its roll axis, and the values of the coefficients used for the present studies. These numerical values are purely estimates made for the purpose of allowing a demonstration of the analytical technique which would be used to systhesize a control system were these the known coefficients.

In Table 4 are itemized the estimated requirements of the dynamic properties of system components. As the actual equipment is designed and developed more complete analyses will be performed continuously which will alter the above recommendations in a manner particularly suited to the aircraft.

The control system that is recommended is not of the discontinuous (bang-bang) type. The elements are expected to act in a linear manner up to saturation. Justification for this is less positive in the case where no seeker is anticipated than in the case where a seeker will be used. This is so because of the direct effect of oscillations of the missiles longitudinal axis on the performance of a seeker. If further development indicates that the airframe can be made to have a reasonable amount of inherent stability, and if no seeker is required then it is possible that a more simple control system of the discontinuous type could be devised. For this reason it is recommended that studies be continued on discontinuous type control systems for a two-beam command guidance system. More data pertinent to discontinuous control will be obtained from a wind tunnel and flight tests on the performance of the airframe.



TWO BEAM COMMAND CONTROL SYSTEM FOR MX 800

BY T. G. M.
CKD J. R.
DATE 1-20-47

FIGURE 14

Lateral Moment Equation $N_{\gamma}\gamma + l F_{\delta_{MW}} \delta_{MW} + N_{\dot{\psi}_M} \dot{\psi}_M = I_Z \ddot{\psi}_M$

Lateral Force Equation $F_{\gamma}\gamma + F_{\delta_{MW}} \delta_{MW} = \mu(\dot{\psi}_M - \dot{\gamma})$

Roll Moment Equation $L_C + L_D - L_{\dot{\phi}}\dot{\phi} = I_X \ddot{\phi}$

TERM	DEFINITION	NUMERICAL VALUES		Units
γ	Angle of Side-Slip			
δ_{MW}	Angle Between Wing Chord and Longitudinal Axis			
ψ_M	Angle from Space Ref. to Long. Axis in Traverse Plane			
ϕ	Angle of Roll			
m	Ratio of Weight to Gravity, W/g, of Missile	11	11	lb ft sec^2
u	Velocity Along Longitudinal Axis	2700	2700	ft/sec
I_Z	Lateral Moment of Inertia of Missile	54	54	lb ft sec^2
I_X	Roll Moment of Inertia of Missile	2	2	lb ft sec^2
F	Side Force on Missile			
F_{γ}	$\partial F/\partial\gamma$	36100	36100	lb
$F_{\delta_{MW}}$	$\partial F/\partial\delta_{MW}$	22700	22700	lb
N	Yawing Moment			
N_{γ}	$\partial N/\partial\gamma$	-15800	-34000	lb ft
$N_{\dot{\psi}_M}$	$\partial N/\partial\dot{\psi}_M$	-63.6	-58.2	sec lb ft
L	Roll Moment			
$L_{\dot{\phi}}$	$\partial L/\partial\dot{\phi}$ Damping Coefficient	19.9	19.9	ft lb/rad/sec
L_C	Control Torque in Roll			
L_D	Disturbance Torque in Roll			
l	Wing Location Foreward of Center of Gravity	0.8	0.0	ft

Notes:
1. Aspect Ratio of Wings Assumed to be 0.75
2. Area of Wings Assumed to be 6 sq. ft. per plane
3. Area of Tail Assumed to be 3 sq. ft. per plane
4. 9" Diameter Missile, 350 lb
5. Altitude for Numerical Values is 30,000 ft.
6. All Angles are in Radians
7. Assumed Cosines of Small Angles to be Unity

SIMPLIFIED EQUATIONS OF MOTION FOR AIRFRAME USED IN MX800 PHASE I CONTROL STUDIES	BY B.A. CKD H.T.M. DATE JAN. 15, 1947	TABLE 3

ROLL CONTROL

Allowable time lag of wing controls	0.02 sec
Load network time constant (Attenuation ratio of 10)	0.16 sec
Integral network time constant (Amplifier characteristic $K(\epsilon + \frac{1}{\tau_1}\int \epsilon \, dT)$	$\tau_1 = 0.18$ sec
Resonant frequency of missile in roll (30,000 ft alt 2700 ft per sec)	6 cycles/sec
Magnification of sinusoidal variation of apparent vertical at resonance	2

PITCH AND YAW CONTROL FOR TWO-BEAM COMMAND GUIDANCE

Approximating wing controls, which use position feedback, by a simple quadratic resonant system	
a. Undamped natural frequency of wing control	50 rad/sec
b. Damping ratio of wing controls	0.5
Percentage longitudinal axis angular acceleration feedback	10
Percentage longitudinal axis angular velocity feedback	0
Percentage of wing load available for guidance	90

PITCH AND YAW CONTROL FOR HOMING WITH SEEKER GUIDANCE

Approximating wing controls, which use position feedback, by a simple quadratic resonant system	
a. Undamped natural frequency of wing control	50 rad/sec
b. Damping ratio of wing control	0.5
Characteristics of amplifier preceding command controls	$K_4 = 18.6$
Amplifier characteristic $\delta_c = K_4\delta_4 + \int \frac{K_4}{\tau_1} \delta_4 \, dt$	$\tau_1 = 0.05$
Resonant frequency of Antenna servos	12 rad/sec

DYNAMIC CHARACTERISTICS FOR CONTROL SYSTEMS AND COMPONENTS ESTIMATED AS SATISFACTORY FOR ONE SET OF AERODYNAMIC COEFFICIENTS (TABLE 1 ℓ=0)	BY HTM CKD HTM DATE Feb. 1, 1947	TABLE

D. CONTROL SYSTEM FOR HOMING WITH SEEKER GUIDANCE:

Should a seeker be required to improve the accuracy in the final stages of flight then the control system should operate satisfactory either with a two-beam command system or with a seeker. It is proposed that in a system using seeker guidance the controls regulate the angular velocity of the line of sight to zero. A block diagram for the recommended control system is shown in Figure 15. It will be noted that the same essential elements are used in this system as for control from two-beam command guidance. The seeker shown schematically uses a movable dish antenna with a servo drive for automatic tracking. A tachometer is used to measure the angular rate of the antenna relative to the airframe. In some seeker systems under consideration this angular rate would be measured electronically and the seeker antenna would be fixed. In order to obtain the desired trajectory with seeker guidance (space angular velocity of line of sight = 0) a signal indicating the angular velocity of the longitudinal axis is fed back regeneratively and added to the tachometer to obtain the space angular velocity of the line of sight. A theoretical analysis was made on this type of control system (7) and the following was concluded.

1. The system is likely to be unstable with wings as far forward of the center of gravity as 0.8 feet. Calculations show absolute instability for the case of an oncoming target.

2. The system should be satisfactory with wings approximately at the center of gravity if downwash is negligible. The calculations for the case of an oncoming target also indicate satisfactory performance.

3. The system can be made stable over a satisfactory range (missile to target) change.

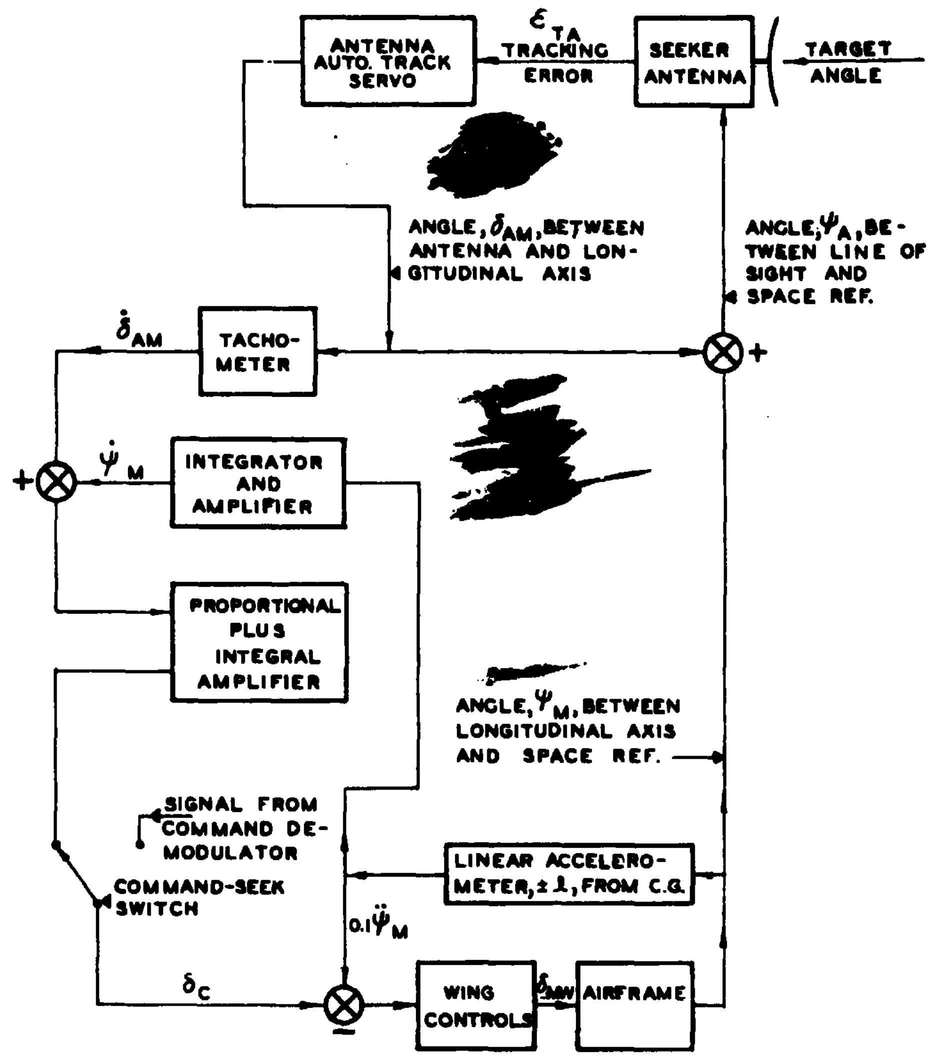
4. The system speed of response is limited primarily by the dynamics of the seeker.

5. The same components used in the two-beam command system can be used after a switching operation in the control system for seeker guidance. This is indicated in Figure 15 .

6. The dynamics of the seeker must be matched to those of the airframe by the appropriate parameters of a proportional plus integral amplifier preceding the command loop.

E. EFFECTS OF AERODYNAMIC PARAMETERS ON DESIGN:

The dynamics of a missile control system must match those of the guidance system with those of the aircraft to produce a satisfactory flight. Consequently where it is feasible the requirements of the control system should be taken into account in determining the dynamics of the airframe. Where the dynamics of the airframe are determined by other considerations a thorough knowledge of its behavior is a prerequisite of the intelligent



CONTROL SYSTEM FOR HOMING WITH SEEKER GUIDANCE	BY H.T.M. CKD HTM DATE 1-25-47	FIGURE 15

design of the flight control system. Several phases of the aerodynamic behavior of the missile have been investigated as a part of control studies.

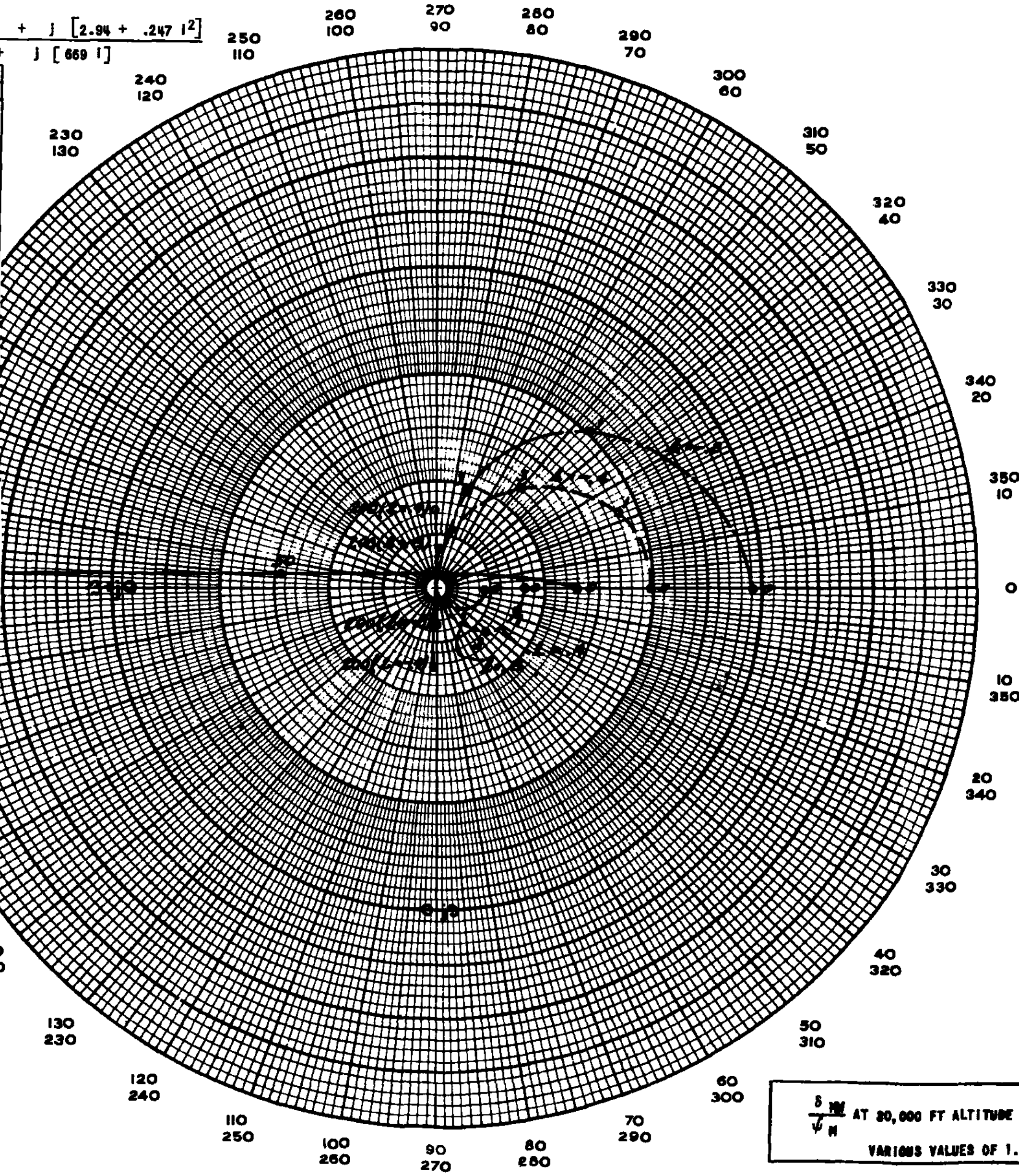
Certain characteristics of the airframe are chosen with only secondary consideration of the control system such as the maximum lift available (which has been taken from trajectory considerations) and the general size and shape of the missile (which has been taken from overall project considerations). Other aspects of the airframe more directly affect the controls work such as, the arrangement of the control wings on the airframe and the location of the main wings relative to the center of gravity.

The location of the main wings to be used for control is important in determining the dynamic response of the missile to control signals. This is illustrated in Figures 16 and 17, for an altitude of 30,000 feet as the dynamic response of the longitudinal axis of the missile to sinusoidal changes in main wing position relative to longitudinal axis and also the response of the missile flight path angular velocity, due to sinusoidal changes main wing angle relative to the longitudinal axis for five different wing locations (7). These calculations are based on the aerodynamic coefficients described in Table 3 and downwash has been neglected. At the present time it is not practical to make a recommendation for the best wing location, although it appears that the two most desirable locations are 0.4 feet forward of the center of gravity and at the center of gravity. Actually most of the calculations for control systems were based on the wing location 0.8 feet forward of the center of gravity. Calculations were also made for 15000 feet and 50000 feet altitude.

The relative importance of effects of the oscillating airfoil at supersonic speeds was investigated as it affects the control theory of the airframe (16). This study indicated that the equations of motion based on stationary flow should be adequate in the frequency range important for automatic control of the missile.

TABLE OF • $\frac{\delta_{MW}}{\dot{\psi}_M} = \frac{[1002 - 669 l - \omega^2] + j\,[2.94 + .247\, l^2]}{[904\, l + 772] + j\,[669\, l]}$

ALTITUDE 30,000 FT			
ω	L	AMPLITUDE	PHASE
0	.6	.461	0
	.4	.823	0
	0	1.90	0
	-.4	1.948	0
	-.8	2.90	0
5	.8	.155	67.2
	.4	.443	55.1
	0	1.283	.85
	-.4	.889	64.7
	-.6	.554	79.4
10	.6	.0672	74.5
	.4	.225	68.9
	0	1.171	1.9
	-.4	.426	77.9
	-.6	.269	95.5
19.4	.8	.0130	59.5
21.6	.6	.0067	4.0
24	.8	.01025	60.2
	.4	.0270	58.7
27.1	.4	.0111	6.7
28.5	.4	.0147	48.9
	0	.271	23.7
31.7	0	.121	90.9
	-.4	.033	105.2
35.3	0	.342	158.6
	-.4	.0115	182.4
	-.8	.0169	108.6
39.2	-.4	.0282	242.5
	-.6	.0059	175.5
43.1	-.8	.0154	245.9
50	.8	.091	69.1
	.4	.133	99.0
	0	1.447	174.4
	-.4	.092	260.7
	-.9	.0366	259.9
100	.9	.179	99.9
	.4	.347	90
	0	11.65	178.1
	-.4	.326	266.7
	-.9	.159	267.3
200	.9	.971	96.6
	.4	.737	90.1
	0	50.4	179.1
	-.4	.727	91.6
	-.9	.361	91.2



DEFINITIONS

δ_{MW} = Angle in radians between the missile axis and the wing chord.

$\dot{\psi}_M = \frac{d\psi_M}{dt}$

ψ_M = Azimuth of longitudinal axis of missile.

t = Time

l = Distance in feet, forward of the missile center of gravity, to wing shaft centerline.

ω = Radian frequency of oscillation of the wing chord, thus $\delta_{MW} = \bar{\delta}_{MW} \sin \omega t$, where $\bar{\delta}_{MW}$ is the amplitude of oscillation

j = $\sqrt{-1}$

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$\frac{\delta_{MW}}{\dot{\psi}_M}$ AT 30,000 FT ALTITUDE FOR VARIOUS VALUES OF l.	BY BJA CKD BJA DATE 1-22-47	FIGURE 16

Form EL 10

DEFINITIONS

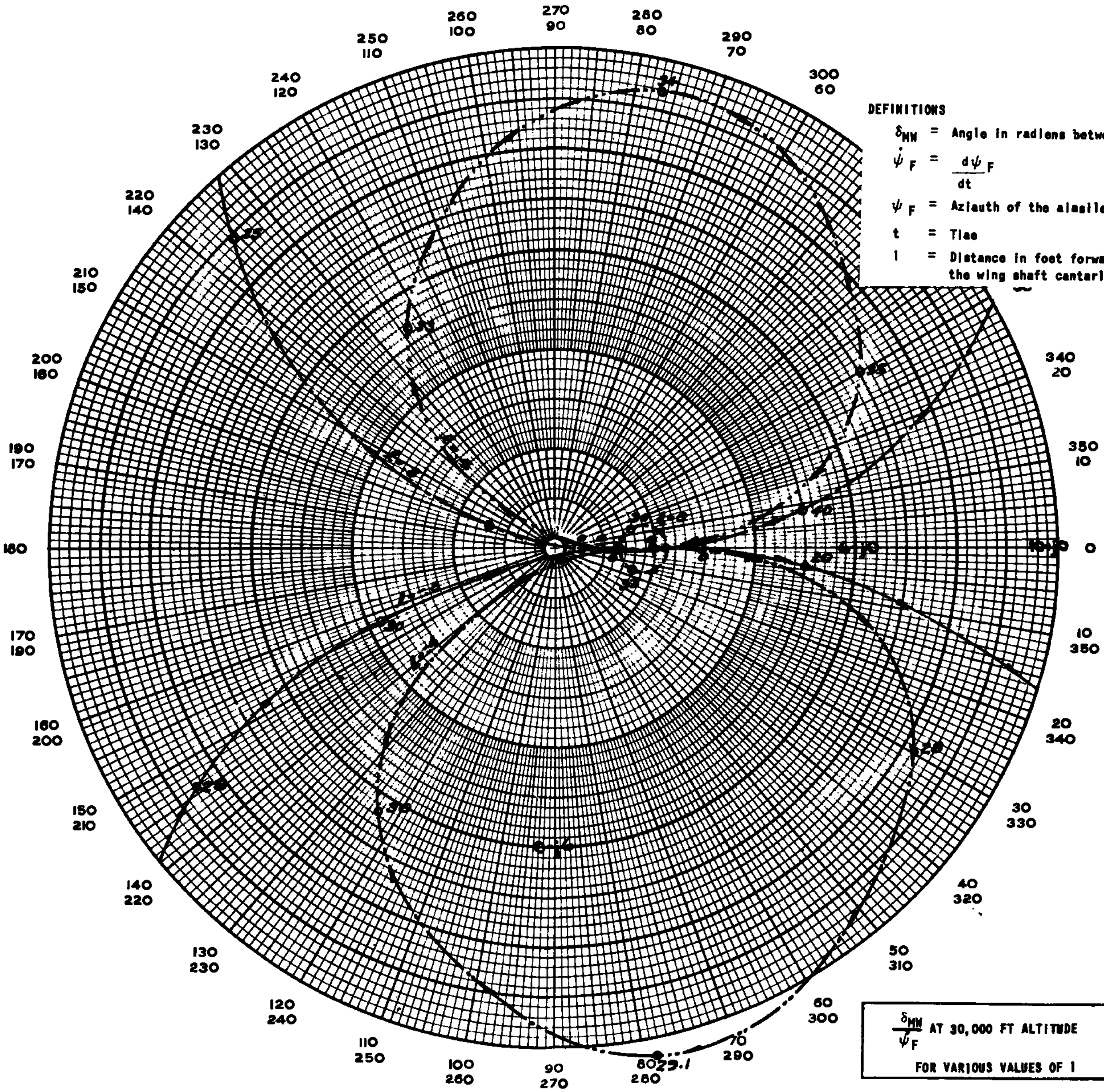
δ_{MW} = Angle in radians between the missile axis and the wing chord.

$\dot{\psi}_F = \frac{d\psi_F}{dt}$

ψ_F = Azimuth of the missile flight velocity, assuming level flight.

t = Time

l = Distance in feet forward of the missile center of gravity to the wing shaft centerline.



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$\frac{\delta_{MW}}{\dot{\psi}_F}$ AT 30,000 FT ALTITUDE FOR VARIOUS VALUES OF l	BY BJA CKD. BJA DATE 1-22-47	FIGURE 17

SECTION VIII

TRAJECTORIES

SECTION VIII

TRAJECTORIES

A. ERRORS IN TWO-BEAM COMMAND:

An estimate of errors encountered in a two-beam command system was made. The method of guidance considered is one based on the value of $\dot{\beta}$ ($= d\beta/dt$) (see Figure 18 a). $\dot{\beta}$ is computed at the control station from the measured values of R_T, R_M, $\dot{R}_T$, $\dot{R}_M$, β_T, β_M, and γ ($= \beta_T - \beta_M$). The complete mathematical consideration is given in Appendix A.

From the considerations in Appendix A we have:

a. From the figures given in Appendix A it appears that the worst errors are likely to be those due to β_M and β_T. It is true that as R gets smaller these errors increase approximately as R^{-1}, whereas the error due to γ increases as R^{-2}, and so this last error may be the decisive one when the missile gets close to the target. But in a given example (Appendix A) this would not occur until R was less than a half second away from the target, and in this short time interval no appreciable guidance could be done anyway.

b. The possible effect of an error in $\dot{\beta}$ can be very roughly estimated by considering a missile moving in a straight line so as to miss a stationary target by an amount h. (Figure 18 b). We have

$$\dot{\beta} = \frac{vh}{R^2} .$$

For $\dot{\beta} = .04$, R = 3500 ft, v = 2000 ft per sec, this gives h = 200 ft. Assuming that the errors here are comparable to the ones obtained in Appendix A this would be an upper bound to the miss that might be obtained. Actually, the average miss would be much smaller than this, for the overall accuracy of the missile will depend on the values of $d\beta$ over a considerable period of time, and as these can be expected to vary in a more or less random manner the average accuracy will be much greater than the crude estimate above. The statistical problem involved in estimating the accuracy has not yet been solved.

c. A suggestion has been made that the error in $\dot{\beta}$ might be decreased by installing in the missile a simple device to give an accurate measurement of R, and possibly $\dot{R}$. That this is not the case follows from the fact that most of the error in $\dot{\beta}$ is due to errors in β_M and β_T, and these quantities do not enter in the expressions for R or $\dot{R}$.

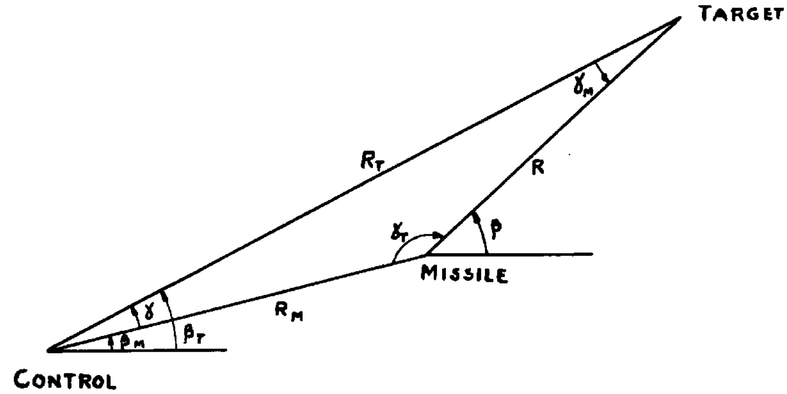

Figure 18a

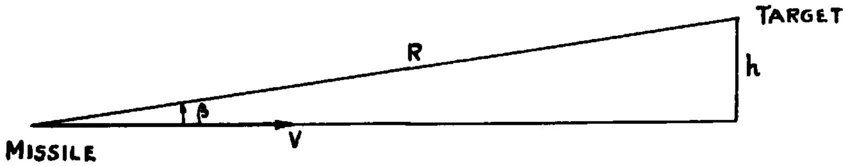

Figure 18b

TWO-BEAM COMMAND GUIDANCE DIAGRAM	BY J.B.R. CKD E.S.W. DATE 1-11-47	FIGURE 18

SECTION IX

LAUNCHING

SECTION IX

LAUNCHING

A. GENERAL:

During the boost period (transonic range) present indications show that there can be little or no guidance on the part of the missile operator over the missile. It is necessary to assume, therefore, that only simplified control may be attained by automatic means (i.e. gyroscopes) causing the missile to traverse a predetermined course. Such a course might have to be set by the operator before the time of launching according to the conditions fixed by the particular launching. This requires provisions in the booster design to accomodate such adjustment.

There is some indication that subsonic control of the missile can be attained by use of sufficiently large lifting surfaces to support the missile from launching speed to the beginning of the transonic speed range, at which time these surfaces can be jettisoned. If this is possible, a configuration may be arrived at which allows maneuvering at subsonic flight speeds, thus permitting launching to be restricted to one direction only with respect to the carrier aircraft. Guidance to the proper attacking position is achieved during this period. Such a system might be composed of a missile and a three stage booster as follows:

1. A jettisonable subsonic propulsion unit with subsonic lifting and control surfaces.

2. A jettisonable booster for operation in the transonic range so as to bring the missile up to supersonic speed.

3. Missile with sustainer rocket (internal).

It is important to note that the complexity of the foregoing arrangement may be reduced if it is possible to use the missile lifting surfaces (supersonic wings) during subsonic flight. The booster can be designed to exert a low propulsive effort during the time required to turn and then a high propulsive effort which is required to attain supersonic velocities. Another point to be considered is that the subsonic surfaces may be retained until the end of the total boost period, thus simplifying the mechanical design of the booster. This would give a missile configuration of a tandem monoplane. The practicality of this proposal depends on the effects of such an arrangement on the total drag during the boost period and also the effects on static stability of the missile.

The problem of launching may be clarified by subdividing it into two general cases; launching for offensive uses and launching for defensive uses. These two cases determine the missile arrangement required and also the provisions necessary in the carrier airplane to provide for launching.

The problem of defensive launching, although not directly related to the MX 800 requirements, is considered here to show that this missile can easily be adapted to defensive use by bomber type aircraft. These considerations also apply to the offensive missile should it be necessary to fire it in other than the line of flight direction.

B. OFFENSIVE USES:

It is assumed that for offensive uses, the carrier airplane locates the target and is able to maneuver so as to aim the missile approximately at the target. It is also assumed that the release attitude of the carrier is such as to cause the missile and carrier to separate.

For external stowage, the present "zero length" launchers may be used on the condition that sufficient clearance between the missile and the wing of the carrier can be maintained. If this is not possible, displacing gear may be necessary to allow sufficient ground clearance during take-off while allowing for missile-wing clearance during launching.

For internal stowage such displacing gear is also necessary since gravity release may not provide sufficient accuracy and control.

The design of such mechanisms is feasible and have been used previously, notably in release of bombs from dive bombers.

For offensive uses, present launching methods and equipment may be used as is or may be adapted to accomodate the MX 800 missile. This use, therefore, presents no serious problems and only influences the missile design to the extent that clearances between missile and ground or missile and airplane must be provided for.

C. DEFENSIVE LAUNCHING:

In defensive launching, it is necessary to assume that the missile is carried in a long range, medium speed type of airplane (i.e. bomber at speeds equal to 300 mph, approximately). For this type of use it is further assumed that the carrier airplane remains on a given course during launching and that the target plane may be at any azimuth position with respect to the carrier airplane. Several possible methods of launching are noted below:

1. The missile is aimed in the direction of the line of flight of the carrier and is guided along the correct flight path required to engage the target.

a. The missile negotiates the turn at subsonic velocities (up to 700 mph).

b. The missile negotiates the turn at transonic velocities (during the boost period).

2. The missile is aimed opposite to the direction of the line of flight of the carrier and is guided along the correct flight path required to engage the target.

3. The missile is aimed directly at the target (in azimuth), regardless of approach angle.

4. The missile is aimed in some optimum direction and then its flight path is directed to engage the target.

It should be noted that in all cases except (3), above, it is necessary for the missile to negotiate a turn of comparitively small radius. For practical considerations, this should be done during the low speed portion of the flight. In order that these turns may be accomplished during this period it is necessary either to provide low speed control surfaces, as described previously, or to provide some means for turning and stabilizing such as vanes in the exhaust of the booster rockets. The latter is under consideration in connection with a Navy research program.

D. SUMMARY:

The effect of the offensive launching method is not a controlling factor in the design of the MX 800. The effects of the exhaust gases on the wing structure of the carrier have been found to be negligible by the U.S. AAF in tests conducted recently using "Tiny Tim" rockets. These rockets have a starting thrust of about 30000 lb which is 100 per cent greater than that contemplated for the MX 800 missile. The missile design for this type of use will, therefore, be limited by other considerations, such as range and thrust schedule desired.

From an examination of the various methods of defensive launching considered the following conclusions can be drawn:

Method (1)-a. The missile is required to negotiate the initial turn toward the target at velocities below 700 mph. The time required to make the maximum turn of 180 degrees is approximately 6 seconds.

Also it is to be noted that the use of subsonic surfaces materially complicates the missile both as to stowage and design of the booster required. For this type of arrangement, it may be possible to store the missiles externally under the carrier airplane's wings, thus overcoming some of the space requirements incurred by internal stowage.

Method (1)-b. If the problem of stability during subsonic flight and the problem of dynamic stability of the missile at launching can be solved, this method presents a better launching program, since the initial maneuvering time can be materially reduced from 5 seconds minimum for method (1) to approximately 0.1 second for method (1)-b.

This method considerably simplifies the design of the booster arrangement, eliminating the subsonic wings and possibly the need for any fins on the booster at all. The price for this simplification is the need for rocket blast vanes and the necessary mechanism for their control. It should be noted, at this point, that the loads on these vanes are high and that they must be resisted when the vane material is at an elevated temperature.

Method (2). This method includes all the problems of methods (1)-a and -b in addition to the problem of accelerating the missile through zero velocity to its normal flight velocity. This method of launching, however, serves the purpose of imposing the most severe conditions for the booster design; thus indicating the upper limit on the size of the booster. Rough calculations show that a booster capable of exerting approximately 24000 lb of thrust for 2 seconds is maximum required when a 2 second boost period is used for a missile plus booster weighing 600 lb.

Method (3). This method imposes no new conditions on the design of the missile and booster but does indicate the size and weight of the required launching gear. Preliminary strength considerations indicate that a suitable launching mechanism should weight in the order of 200 lb.

E. CONCLUSIONS:

From the discussion in (B), Offensive Uses, considerations of offensive launching are not the controlling factor on the design of the missile, and will in general effect only the design of the missile supports and launching gear.

The largest number of problems relating to the design of the missile were encountered in (C), Defensive Launching.

The follwing conclusions may be drawn from the discussion in (D), Summary:

1. Method (1)-b or method (3) presents the most promising launching procedure for defensive uses.

2. A booster exerting a maximum of 24000 lb for 2 seconds is the upper limit of the required booster size.

3. In method (1)-b or method (3) the need for fins on the booster may be eliminated by use of vanes in the exhaust of the booster rocket.

4. The structural weight of a launching mechanism need not exceed 200 lb.

SECTION X

WARHEAD INSTALLATIONS

SECTION X

WARHEAD INSTALLATIONS

The design of the warhead has not been considered in any detail as yet. Up to the present, a 90 pound warhead has been assumed. Warhead shapes were assumed in representing the MX-800 pictorially. This was done simply to produce a reasonable picture and does not represent a warhead of proper design. The warhead problem will not be studied in detail until the missile design is further crystalized.

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SECTION XI

BIBLIOGRAPHY

SECTION XI

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APPENDIX "A"

ERRORS IN TWO-BEAM COMMAND

APPENDIX "A"

ERRORS IN TWO-BEAM COMMAND

The method of guidance considered is one based on the value of $\dot{\beta}$ ($= d\beta/dt$). (See FigureA1.) $\dot{\beta}$ is computed at the control station from the measured values of R_T, R_M, $\dot{R}_T$, $\dot{R}_M$, $\dot{\beta}_T$, $\dot{\beta}_M$, and γ ($= \beta_T - \beta_M$).

We wish to estimate the error in the computed value of $\dot{\beta}$ due to observed errors in the measured quantities.

The Formulas.

We have

$$R^2\dot{\beta} = R_T^2\dot{\beta}_T + R_M^2\dot{\beta}_M + (R_T\dot{R}_M - R_M\dot{R}_T)\sin\gamma - R_TR_M(\dot{\beta}_T + \dot{\beta}_M)\cos\gamma,$$

where

$$R^2 = R_T^2 + R_M^2 - 2R_TR_M\cos\gamma.$$

This gives

$$R^2\frac{\partial\dot{\beta}}{\partial\dot{R}_M} = R_T\sin\gamma, \qquad R^2\frac{\partial\dot{\beta}}{\partial\dot{R}_T} = -R_M\sin\gamma,$$

$$R^2\frac{\partial\dot{\beta}}{\partial\dot{\beta}_M} = R_M^2 - R_MR_T\cos\gamma, \quad R^2\frac{\partial\dot{\beta}}{\partial\dot{\beta}_T} = R_T^2 - R_MR_T\cos\gamma,$$

$$R^4\frac{\partial\dot{\beta}}{\partial R_M} = \left[(R_T^2 + R_M^2)\cos\gamma - 2R_MR_T\right]R_T\dot{\gamma} - (R_T^2 - R_M^2)\dot{R}_T\sin\gamma$$

$$- 2(R_M - R_T\cos\gamma)R_T\dot{R}_M\sin\gamma,$$

$$R^4\frac{\partial\dot{\beta}}{\partial R_T} = -\left[(R_T^2 + R_M^2)\cos\gamma - 2R_MR_T\right]R_M\dot{\gamma} - (R_T^2 - R_M^2)\dot{R}_M\sin\gamma$$

$$+ 2(R_T - R_M\cos\gamma)R_M\dot{R}_T\sin\gamma,$$

$$R^4\frac{\partial\dot{\beta}}{\partial\gamma} = (R_T\dot{R}_M - R_M\dot{R}_T)\left[(R_T^2 + R_M^2)\cos\gamma - 2R_MR_T\right]$$

$$- (R_T^2 - R_M^2)R_MR_T\sin\gamma\dot{\gamma}.$$

By introducing the angles γ_T and γ_M (FigureA1), these can be reduced to:

$R^2 \dfrac{\partial \dot{\beta}}{\partial \dot{R}_M} = R_T \sin\gamma\,,$ $\qquad$ $R^2 \dfrac{\partial \dot{\beta}}{\partial \dot{R}_T} = -R_M \sin\gamma\,,$

$R^2 \dfrac{\partial \dot{\beta}}{\partial \beta_M} = RR_M \cos\gamma_T\,,$ $\qquad$ $R^2 \dfrac{\partial \dot{\beta}}{\partial \beta_T} = RR_T \cos\gamma_M\,,$

$$R^3 \frac{\partial \dot{\beta}}{\partial R_M} = (-R_T\cos\gamma_T - R_M\cos\gamma_M)\, R_T\,\dot{\gamma}$$
$$+ (-2R_T\dot{R}_M\cos\gamma_T - (R_T\cos\gamma_M - R_M\cos\gamma_T)\dot{R}_T)\sin\gamma\,.$$

$$R^3 \frac{\partial \dot{\beta}}{\partial R_T} = -(-R_T\cos\gamma_T - R_M\cos\gamma_M)\, R_M\,\dot{\gamma}$$
$$+ (2R_M\dot{R}_T\cos\gamma_M - (R_T\cos\gamma_M - R_M\cos\gamma_T)\dot{R}_M)\sin\gamma$$

$$R^4 \frac{\partial \dot{\beta}}{\partial \gamma} = R^2(R_T\dot{R}_M - R_M\dot{R}_T)\,(\cos\gamma - 2\sin\gamma_M\sin\gamma_T)$$
$$- R_M R_T (R_T - R_M)\,(R_T + R_M)\,\sin\gamma\,\dot{\gamma}$$

If γ is small, then γ_M is also small and γ_T is near 180 degrees. So we can put approximately

$\cos\gamma = \cos\gamma_M = 1$

$\cos\gamma_T = -1,$

$R_T - R_M = R.$

Then we get approximatsly

$R^2 \dfrac{\partial \dot{\beta}}{\partial \dot{R}_M} = R_T \sin\gamma\,,$ $\qquad$ $R^2 \dfrac{\partial \dot{\beta}}{\partial \dot{R}_T} = -R_M \sin\gamma$

$R^2 \dfrac{\partial \dot{\beta}}{\partial \beta_M} = -RR_M\,,$ $\qquad$ $R^2 \dfrac{\partial \dot{\beta}}{\partial \beta_T} = RR_T\,,$

$$R^2 \frac{\partial \dot{\beta}}{\partial R_M} = R_T\,\dot{\gamma} + \frac{2R_T\dot{R}_M - (R_T + R_M)\dot{R}_T}{R}\sin\gamma\,.$$

$$R^2 \frac{\partial \dot{\beta}}{\partial R_T} = -R_M\,\dot{\gamma} + \frac{2\dot{R}_M R_T - (R_T + R_M)\,\dot{R}_M}{R}\sin\gamma\,.$$

$$R^2 \frac{\partial \dot{\beta}}{\partial \gamma} = R_T\dot{R}_M - R_M\dot{R}_T - \frac{R_M R_T\,(R_T + R_M)}{R}\sin\gamma\,\dot{\gamma}\,.$$

In most cases the last term of the last equation is negligible.

<u>An Example</u>.

We take the following values as probable errors of the measured quantities:

$$dR_M = dR_T = 30 \text{ ft},$$

$$d\gamma = .004 \text{ rad},$$

$$d\dot{R}_M = d\dot{R}_T = 120 \text{ ft per sec},$$

$$d\dot{\beta}_M = d\dot{\beta}_T = .01 \text{ rad per sec}.$$

Figure 2 shows a sample trajectory. Regardless of whether the control ship pursues a straight course or turns we heve roughly

$$R = 3500 \text{ ft} \qquad R_T = 12300 \text{ ft},$$

$$R_M = 8800 \text{ ft} \qquad \dot{R}_T = -700 \text{ ft per sec},$$

$$\dot{R}_M = 1600 \text{ ft per sec}$$

$$\gamma = -1.5 \text{ deg}, \qquad \dot{\gamma} = .015 \text{ rad per sec}$$

$$\sin\gamma = -.025, \; R_M \sin\gamma = -200 \text{ ft}, \; R_T \sin\gamma = -300 \text{ ft}.$$

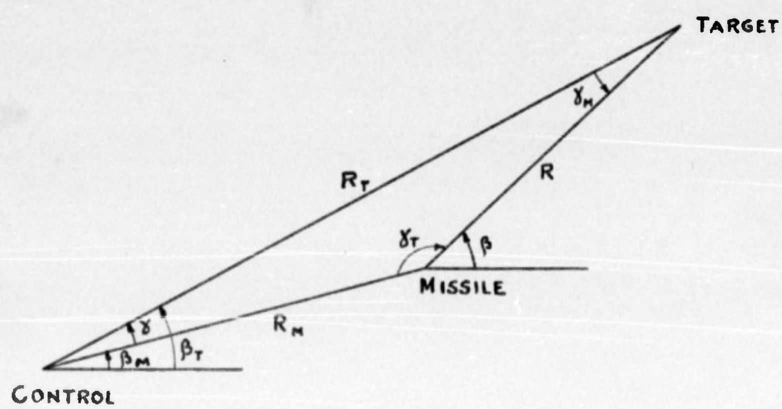
These give, roughly,

$$\frac{\partial \dot{\beta}}{\partial R_M} d\dot{R}_M = -.003 \qquad \frac{\partial \dot{\beta}}{\partial \dot{R}_T} d\dot{R}_T = .002$$

$$\frac{\partial \dot{\beta}}{\partial \beta_M} d\dot{\beta}_M = -,025 \qquad \frac{\partial \dot{\beta}}{\partial \dot{\beta}_T} d\dot{\beta}_T = .035$$

$$\frac{\partial \dot{\beta}}{\partial R_T} dR_M = .000 \qquad \frac{\partial \dot{\beta}}{\partial R_T} dR_T = .000$$

$$\frac{\partial \dot{\beta}}{\partial \gamma} d\gamma = .008$$



TWO-BEAM COMMAND GUIDANCE DIAGRAM	BY J.B.R. CKD E.S.W. DATE 1-11-47	FIGURE A-1

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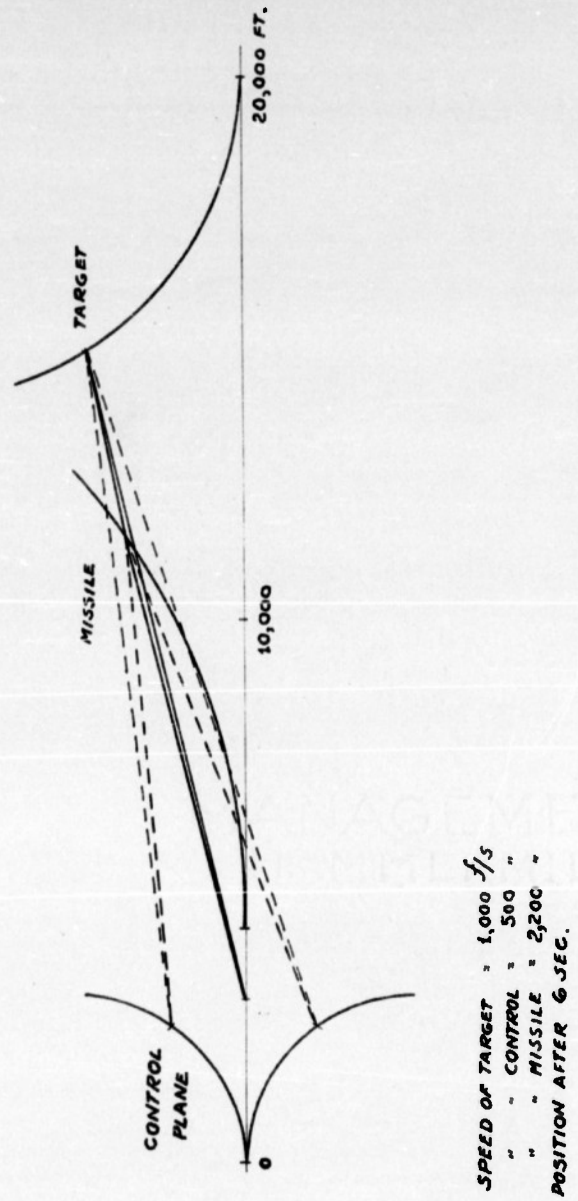
PAGE A-5



SAMPLE TWO BEAM COMMAND TRAJECTORY

BY J.B.R.	FIGURE
CKD E.S.W.	A-2
DATE 1-11-47	

REEL

C 35

FRAME

1183

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ABSTRACT:

This report summarizes the work conducted by the Project MX-800 during the period December, 1948, through February 6, 1947. The Project MX-800 is engaged in the development of a supersonic, air-to-air guided missile to be used against hostile aircraft. The missile is to have a range of 6000 yards and a speed of 1500 mph, and is to be capable of intercepting aircraft flying at 750 mph. The use of radar for target detection and tracking is planned for guidance of the MX-800. The size and weight of the warhead were not yet determined.

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DIVISION: Guided Missiles (1)
SECTION: Design and Description (12)

SUBJECT HEADINGS:
Missiles, Guided - Specifications (63825); Missiles, Guided - Production - Design (63250); Mx 800 (83250)

ATI SHEET NO.: S-1-12-75

DEPARTMENT OF THE AIR FORCE
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MEMORANDUM FOR DTIC 18 July 2017

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The controlling DoD office (AFRL/RQ) reviewer, Reid Melville, for the subject report has affirmed that the report for ADC800676, Title Unknown, Distribution Statement E, is to be declassified.

Additionally, the subject report was reviewed by the 88ABW Public Affairs Office (PAO), and the PAO reviewer, Jeannie Masters (jeannie.masters@us.af.mil), approved the report for public release (Case Number 88ABW-2017-3296, cleared 13 Jul 2017).

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