Abstract: System integration and testing is very important in any product development. In this paper some steps towards the integration of instrumentation for mini aerial vehicle is discussed. Later Hardware In Loop Simulator for testing of autonomous MAV as a system is discussed. The overall work is a step towards the MAV autonomous flight.

Introduction

In last few years lot of interest has been shown by the researchers in development of mini and micro aerial vehicle as an autonomous vehicle. Initially the design and shapes of the mini and micro aerial vehicles were borrowed from radio control model aircraft. Autonomous flight of a mini aerial vehicle is quite involved and complex. This requires experience in integration of sensor for this class of vehicles and later integration of onboard computer with the sensors and the actuators. Keeping this in view CASDE at IIT Bombay started working towards this in a systematic manner [1-3]. Initially efforts were devoted towards the integration of sensors for inflight measurement [1]. This activity helped in building knowledge and expertise in:

(i) Design, fabrication and operation of remotely piloted aerial vehicles.
(ii) Identification and calibration of sensors for small aerial vehicles.
(iii) Integration of instrumentation payload and other systems onboard aerial platforms.
(iv) Design and development of custom payloads like, data acquisition and flight control hardware.
(v) Low risk approaches in the integration of flight enabling hardware.
The above experience will help in design and implementation of low-level controllers and pave the way for more ambitious controllers integrated with onboard GPS for autonomous flight.

INTEGRATION OF INSTRUMENTATION PAYLOAD ON A REMOTELY PILOTED AERIAL VEHICLE

AIMS:

(i) Identification of sensors and configuration of a generic data acquisition system for implementation on small aerial vehicles.
(ii) Fabrication of a suitable model aircraft to carry the instrumentation payload.
(iii) Flight test the vehicle and acquire flight data under steady state and maneuvering flight.

AERIAL VEHICLE

A model aircraft, KADET Mk II was selected as the instrumented aerial vehicle platform. KADET, a widely used trainer in the aeromodelling community is a proven model aircraft designed by Claude McCullough of the SIG Manufacturing Company USA. The aircraft has a high wing configuration and a wingspan of 1.46 m. The model is particularly suited for the envisaged role due to its roomy fuselage which is capable of accommodating the instrumentation payload. The aircraft has controls for operating throttle, ailerons, elevator and rudder. Rudder is interconnected with nose wheel steering to facilitate ground handling. The wing area is 0.42 m² and the recommended weight of the aircraft is 2.0 to 2.3 kg. A 7.5 cc single cylinder piston engine (OS Max 46 FX) rated at 1.6 BHP was chosen as the powerplant. The weight of the aircraft with instrumentation was estimated at 2.8 kg. The increase in all up weight due to the payload called for certain simple modifications in the design of the aircraft. Final aircraft after the modifications is shown in figure 1. These modifications are
(i) The wing span was increased by 0.25 m in a bid to maintain design wing loading (of 5.5 kg/m$^2$) owing to an increase in all up weight due to the instrumentation payload.

(ii) The fuselage width was increased by approximately 2 cm to facilitate installation and easy access to the instrumentation payload.

(iii) The throttle, elevator and rudder actuation servos were relocated from the fuselage center section to the rear of the fuselage in order to create space for the payload (close to aircraft CG).

![Figure 1. Instrumented aerial vehicle](image)

**INSTRUMENTATION**

The following parameters have been identified for measurement on the aircraft for in-flight recording of flight data.

(i) Angle of Attack ($\alpha$).

(ii) Indicated Airspeed (V).

(iii) Pressure Altitude (h).

(iv) Three axes Acceleration (Nx, Ny and Nz).

(v) Elevator Position ($\delta_e$).

(vi) Pitch Rate (q).

(vii) Flight trajectory using GPS

Selection of the various components of the instrumentation package was based on aspects like small size, low weight, low operating voltage and low power consumption and low cost (preferably). The block diagram schematic of the instrumentation is shown in figure 2.
Angle of Attack (\(\alpha\)): For measurement of angle of attack a wind vane assembly coupled to a precision potentiometer was fabricated and installed on the aircraft wing tip, well outside the influence of the propeller slipstream. Details of the arrangement are elaborated under experimental work.

Indicated Airspeed (V): For the measurement of indicated airspeed of the aircraft, a differential pressure transducer model 163PC01D75 of M/s Honeywell Sensing and Control, Canada has been used. The operating range of the device is \(\pm 2.5\) inches of water and corresponds to an
equivalent airspeed of approximately 32 m/s (which is considered adequate to cover the range of operating speeds of the aircraft). The sensor has been coupled to pitot and static probes. Details regarding fabrication and calibration of the pitot and static probes is described under experimental work. 

**Pressure Altitude Sensor (h):** A differential pressure transducer, model 142PC01D, with a range upto 28 inches (0 to 71.12 cm) of water has been used. The range corresponds to an altitude band (at sea level) of approximately 870 m. One of the ports of the transducer has been connected to a sealed tank. An arrangement to equalize the pressure of this tank with respect to the ambient atmospheric pressure on ground before each flight has been incorporated. The other port is open to the atmosphere via the static probe. The differential pressure measured by this system is therefore indicative of the height of the aircraft above ground. This arrangement will have adequate short-term accuracy for typical flight durations of 15 to 20 minutes.

**Tri-axial Accelerometer (Nx, Ny, Nz):** For the measurement of accelerations along all three axes of the aircraft, a tri-axial accelerometer has been used. The transducer, model ADXL105-EM3, is manufactured by M/s Analog Devices, USA. The operating range of the device is ± 4 g along all three axes. The accelerometer was installed close to CG location.

**Elevator Position Sensor (δe):** For the measurement of elevator position, a potentiometric transducer has been employed. The elevator has been coupled to the transducer using a suitable linkage. The sensor installation is shown in Figure 3

![Figure3. Elevator potentiometer installation](image)
**Pitch Rate Sensor (q):** Efforts to identify a miniature rate gyro for measurement of pitch rate did not fructify when the project was started. However, during market survey a piezoelectric gyro based flight stabilization unit was identified. This unit is specially designed for use on radio controlled aircraft as a single axis stabilization system and is not intended for use as an angular rate sensor. However, it was decided to adapt this system for measurement of pitch rate. An electromechanical interface was developed for this purpose and the scheme is elaborated under experimental work. Figures 4 show the gyro sensor installation and interface respectively.

![Gyro Sensor and Interface](image)

**Figure 4. Interface for Gyro Unit**

**Trajectory recording using GPS:** At later stage GPS was also included in the instrumentation and trajectory of the aircraft was recorded. Dual Port RAM was used in interfacing GPS with the on-board data recorder.

**Data Acquisition Unit:** An off-the-shelf miniature data acquisition unit was identified for recording flight data onboard the aircraft. The unit is the Model TFX–11 data logger from Onset Computer Corporation, USA. The device can accept upto 19 channels of analog data and operate at a maximum sampling rate of 3.2 kilo samples/sec. Onboard memory consists of 128 kilo bytes of RAM and 472 kilobytes of flash EEPROM, which can be used for data as well as programs. It is planned to use Motorola 68832 processor (TT8) from Onset Computer Corporation, USA, as a candidate for onboard computer. Flight data recording with all the above parameters has been completed successfully using TT8.

**EXPERIMENTAL WORK**

The following experimental work was undertaken.
(i) Fabrication of wind vane assembly for measurement of angle of attack.
(ii) Fabrication of pitot static probes for measurement of indicated airspeed and altitude.
(iii) Fabrication of a stub wing to test the above in a wind tunnel.
(iv) Fabrication of an interface for the gyro system to enable recording of pitch rate.
(v) Design of interface for GPS data acquisition

**Angle of Attack Vane (AOA):** The wind vane assembly was fabricated from an aluminum rod (2 mm dia.) and a balsa wood fin. The extension shaft (4 mm dia.) was notched to receive the vane assembly and was glued in place using cyanoacrylate adhesive. The assembly was statically balanced to reduce measurement errors. Since the aircraft was propeller driven (nose mounted single engine), it was decided to mount the vane on the wing tip well outside the influence of propeller wash. However, in a bid to reduce the effect of wing tip vortices on the vane, the shaft was extended to provide a reasonable separation of approximately 7.6 cm (3 inches) between wing tip and the vane. A ball bearing unit was used for supporting the vane shaft to reduce problems of friction. The vane shaft was connected to potentiometer. A noteworthy feature of the potentiometer is that it has a low value of operating torque, which is of the order of 0.021 Nm. This was an important feature in selecting it since large operating torque is a potential source of error and can be significant at low airspeeds. The schematic drawing of the vane assembly installed on the stub wing is shown in figure 5.

![Plan view of Stub Wing](image)

Figure 5. **Stub wing with AOA vane**
**Pitot and Static Probes:** Standard pitot static probes consist of a coaxial arrangement of two tubes. The inner one measures total pressure and is open to the direction of airflow. The outer tube that measures static pressure has holes tangential to the airflow. This arrangement though compact, presented certain manufacturing difficulties due to the co-axial arrangement of tubes. It was decided to fabricate a simpler arrangement consisting of two separate tubes, one each for total and static pressure measurement. Four diametrically opposite holes were drilled on the surface of the static probe. Stainless steel tubes of 11.5 cm length (exposed length) and 3.2 mm diameter were used for the purpose. The schematic drawing of the static tube is shown in Fig. 6. The pitot pressure probe was simply a length of tube open to the airflow. To minimize errors in measurement of static pressure, it is recommended that the ratio of ‘x’ to ‘d’ (refer Fig. 6) should be in the region of 10 and above [4]. A ratio of ten was implemented during fabrication of the static probe. Ideally, the probe should be as long as possible, to minimize the influence of the probe supports (in this case the aircraft wing) on the measured value of static pressure. However, it was been decided to fix the length at 15 cm (6”) more for convenience rather than any other reason. The pitot and static test probes were initially tested in isolation in the wind tunnel i.e. without the influence of wing. The error in speed measurement with these probes was found to be negligible. The installation of the probes on the stub wing to enable wind tunnel testing is shown in Fig. 7.

![Figure 6. Static Tube Schematic](image)

**Stub Wing:** The wind tunnel test section of, two feet by two feet (61 cm), dictated the size of the stub wing. To ensure that an aspect ratio of at least four is maintained the actual aircraft wing was
scaled down to 73%. Thus wing chord of 29.5 cm was reduced to 21.5 cm on the stub wing. The span was fixed at 42 cm, which together with its image in the tunnel wall yielded an aspect ratio of approximately four. The schematic arrangement of the stub wing in the wind tunnel is shown in Fig. 7. The stub wing was constructed with balsa wood and covered with commercially available polyurethane self-adhesive heat shrink film (*Monokote*). An aluminum tube was embedded spanwise into the stub wing. This enabled the wing angle of attack to be varied from outside the tunnel by rotating the tube. All pressure connections and electrical wires of the pitot static probes and the angle of attack vane were brought out of the wind tunnel through this tube. The arrangement of the stub wing in the wind tunnel is shown in Fig. 8.

![Schematic Arrangement of Stub Wing in Wind Tunnel](image1)

**Figure 7. Schematic Arrangement of Stub Wing in Wind Tunnel**

![Stub Wing in Wind Tunnel](image2)

**Figure 8. Stub Wing in Wind Tunnel**
**Wind Tunnel Tests:** A platform was constructed on the aluminum tube embedded in the stub wing and extending out of the wind tunnel wall. The platform was rigged to have zero inclination with respect to the wing chord. Thus, angle of attack of the stub wing was measurable at all times by placing an inclinometer on this platform. The wing incidence was varied between – 6° to 16° during the tests. All pressures were measured in mm of water using a micromanometer. The airspeed measured by the pitot static test probes was plotted against the tunnel speed for different inclinations of the wing (wind tunnel speed was varied between 5 m/s to 30 m/s). The plot is shown in Fig. 9

![Figure 9. Measured Velocity versus Tunnel Velocity for Different AOA](image.png)

It can be seen that the slope of these plots increases with increase in angle of attack. Ideally, they should have coalesced, however they are close and linear. The percentage error in measured airspeed was then plotted versus angle of attack and is shown in Fig. 10. It is evident from the plot that the errors are within +6% and –3%. Therefore, the probe arrangement was considered acceptable for measurement of airspeed on the actual aircraft.
Gyro Interface: To enable recording of angular rate using the available gyro system an electromechanical interface was developed. The gyro system consists of a piezoelectric sensor and an amplifier unit. The amplifier is capable of driving a servo, which in turn can move a control surface in the appropriate direction to annul the effect of disturbances. The system can be rigged to act as a roll, pitch or yaw damping system. In the present application however, a precision potentiometer was coupled to the output shaft of a servo, which in turn was connected to the gyro amplifier. The arrangement together with the sensor unit was then placed on an angular rate calibration turntable. The servo movement, which is proportional to the angular rate, was then picked off by the potentiometer and its output was calibrated for different angular rates. The block schematic of the system is shown in Fig. 11.
GPS Interface: GPS outputs data at every one second. Latitude, longitude, UTC, speed over ground, heading etc are the parameters available in NMEA format. The GPS output is serial data at 9600 bps. The length of GPS sentence is around 54 bytes. It also means the transmission time is around 54 msec. Datalogger can be directly connected to GPS using serial port. During the GPS data receiving the datalogger will be busy and no other task can be performed. To overcome this limitation, GPS was connected to another microcontroller. This microcontroller is working in parallel with the datalogger and receives the GPS data at 9600 bps. After receiving the data it stores the data in DPRAM using parallel port. DPRAM data is accessed by the datalogger using parallel port. The transfer time for data using ports is in microseconds and save lot of time for datalogger to perform other tasks. The schematic is shown in figure 12.

FLIGHT TESTS

Lack of a suitable flying site at Mumbai necessitated flying the aircraft at an outstation venue. The vehicle was thus flown in the end of September 2000 at the Gliding Center at Hadapasar, Pune. Initial flights were planned for familiarization, checking control
harmony and trimming the aircraft. During these flights, the instrumentation payload was removed as a precaution. The flying setup at the field is shown in figure 13.

Fig 13. Field setup for the Instrumented Aerial Vehicle

Thereafter instrumented flights were attempted. On examination of the flight data it was immediately apparent that the accelerometer data was too noisy to be usable. Quality of the other recorded parameters was found to be satisfactory. Data for one level flight and one loop are presented in figures 14 and 15. From figure 15 the loop diameter can be determined (approx. 26 m).
Figure 14. Level flight data.
Solving accelerometer noise problem

Accelerometer data acquired during flight tests pointed to the need for filtering. It was felt that the principal source of noise in the data was engine vibration. To confirm the same, ground runs were conducted in which the accelerometer data was sampled at close to 400 samples per second. The data was acquired at different throttle settings and engine rpm was measured using a tachometer. The spectral content of the data was then determined by FFT techniques. The plots revealed that the principal source of noise was the engine indeed. Thereafter first order filters with a cutoff frequency of 5 Hz were implemented on the three accelerometer channels (the cutoff frequency of 5 Hz was chosen as it is unlikely that any of vehicle dynamics will exceed this frequency). Subsequent data acquisition during ground run revealed significant data.
smoothening on all accelerometer channels. Fig. 16 shows time history of Nz data for unfiltered and filtered channels. Fig. 17 shows the spectral plot of the data. It is readily apparent from the figures the extent to which filtering has helped in smoothening the data.

Figure 16. **Time history of unfiltered and filtered Nz data**
Figure 17. **Spectral plots of unfiltered and filtered Nz data.**

**Trajectory of flight using GPS:** Four or more satellite should be visible to GPS antenna for accurate position data. The antenna was mounted just ahead of the wing tip for better satellite visibility. 12 channel GPS was used for this experiment. From the recorded data it was observed that GPS signal was never lost during the flight test. Aircraft performed various maneuvers climb, turn, level flight and almost in all direction. Trajectory of the aircraft is shown in figure 18. Altitude measured using GPS during the flight is shown in figure 19. Large error in the altitude measurement can be observed. The flight was conducted at sea level for ~320 secs. Maximum altitude above ground level during the flight is ~ 240 m.
Figure 18. Trajectory of the aircraft recorded using GPS
CURRENT STATUS AND FUTURE PLANS

Regular flights are conducted to test the sensor integration and issues related to these are resolved. Presently flight test are conducted at Virar, where large open space is available for such activity. Control and guidance strategy for altitude hold, speed hold and heading hold are under development and in near future it will be implemented on the MAV. Present instrumentation weight is ~500 gm. After complete testing of the individual system, reduction in size and weight of the instrumentation package will be attempted.
Hardware In Loops Simulator for MAV

Modeling and simulation technologies have helped in system development and successfully shortened design and development cycle of complex systems. Full software simulation of a system requires a high fidelity model of the complete system. In case a subsystem cannot be adequately characterized by mathematical models, it is safe to embed it, as it is, into the simulation e.g. actuator dynamics is often difficult to model and embedded microcontroller with its resident code is hard to take into account. Use of Hardware-In-the-Loop Simulation (HILS), where actual hardware is embedded into the simulation started in aircraft and space applications and is slowly percolating to other industries. Onboard-controller-in-the-loop simulation (OILS) is also a popular strategy.

The basic principle of HILS is that some subsystems are physically embedded within a real-time simulation model. In HILS the embedded system is fooled into thinking that it is operating with real-world inputs and outputs, in real-time. A computer software with real-time simulation capabilities and a computer with necessary communication abilities (A/D, D/A converters for communications with analog signals and digital ports for communication with digital signals) is necessary to perform hardware-in-the-loop simulation.

HILS for Mini Air Vehicle (MAV)

An automated MAV will have the following components:

- Air vehicle
- Sensors
- Controller
- Control actuators

Air vehicle is the plant, sensors are what provide feedback to the controller and actuators are what move control surfaces (elevator, ailerons, rudder) and throttle to control pitch, roll, yaw and thrust of the air vehicle respectively.
In the Hardware-In-the-Loop Simulation (HILS), the air vehicle is modelled to simulate its flight dynamics. This plant model is used to test the actual controller before putting it on flight. Similarly, various environmental conditions can be simulated in computer and controller can be tested under these conditions. Sensors are used to sense various states of the aircraft which are required by the control strategy. Various sensors which are used by existing control strategies are vertical gyros for sensing aircraft orientation along the three axes, rate gyros for sensing rotation rates along the three axes, pressure sensors for height and velocity measurement, Global Positioning System (GPS), magnetic heading sensor and accelerometer. These sensors have to be modelled to emulate the real ones. Sensor models must output data to the controller in the same format as the actual sensors on real aircraft. The system in simulation will look as shown in Figure 20.

![Simulated and actual components of HILS](image)

**Figure 20. Simulated and actual components of HILS**

There is a need for HILS real time simulation platform for:

- Designing and verifying the control and navigation strategies for the MAV.
Deciding on the controller sampling time step.

Gauging the effect of quantization error while taking sensor inputs.

Seeing the effect of actuator dynamics, which is difficult to model.

The controller going onboard the MAV needs to be tested thoroughly to check for flaws in control strategy and other implementation details. For that sufficient number of test cases have to be generated to make sure that every possibility is explored and there are no errors. Putting the controller without doing all these tests may lead to crashes resulting in huge losses in terms of time and money. All this can be achieved using the HILS under development at CASDE, IIT Bombay.

**HILS Configuration**

CASDE is presently engaged in the development of an autonomous MAV. A HILS facility to support the development effort was planned. Since the MAV activity is in its initial stages, it was decided to develop HILS without relying on specific inputs from MAV team, i.e. actual plant data, exact control strategy etc. But overall MAV requirements are to be kept in mind. So, components of HILS simulator are chosen as follows.

- Use a well documented aircraft Flight Dynamic Model (FDM). As a complete mathematical model of the aircraft to be automated is required, aircraft data should be available i.e. aerodynamic coefficients, stability derivatives, engine model and aircraft physical parameters etc.
- Use a known and verified control strategy. This will remove uncertainty with respect to control strategy while proving HILS.
- Use a simple navigation strategy.
- Design and develop the complete HILS facility.
Choices Made

The following specific choices have been made for HILS components so that complete system integration can be achieved and it is flexible to reconfigure the system for new onboard computers and actuators.

Flight Dynamic Model (FDM)

The flight dynamics simulation (FDS) has been done for DeHavilland DHC-2 `Beaver' aircraft, The data of this aircraft is available from a matlab add-in module Flight Dynamics and Control toolbox} (FDC) written by M.O. Rauw [5]. This aircraft is well documented in the Ph.D. thesis of M.O.Rauw as well as the reference manuals for Flight Dynamics and Control Toolbox. This helped in verification of flight dynamics simulation.

Navigation Strategy

Waypoint navigation, where the controller is supplied with a set of waypoints it has to pass through has been chosen as the navigation strategy. It has to fly at a given altitude and velocity between any two waypoints. Navigation is based on Air Data Dead Reckoning (ADDR) using IAS and heading [6]. For offline simulation GPS data was incorporated to correct the trajectory obtained from ADDR.

Guidance Strategy

Two guidance strategies have been implemented. In first, aircraft tries to follow the straight line joining the waypoints. In second, aircraft follows the line of sight to the target waypoint.

Straight Line guidance: In this strategy guidance is achieved by cross-track error (also Y-error). Y-error is the perpendicular distance between the aircraft position and the line
joining the two way points as shown in figure 21. The reference heading given to aircraft tries to minimize the Y-error by giving an extra heading command proportional to Y-error. Figure 21 also shows how reference heading is derived in case of straight line guidance.

**Figure 21. Straight line guidance**

**Line of sight guidance:** At any instant aircraft tries to follow the line joining its position at that instant and the waypoint it is heading towards. This guidance strategy is shown in figure 22. Reference heading in this case is heading of line joining aircraft and next point.
Control Strategy

Control strategy of ‘Nishant’ UAV developed by Aeronautical Development Establishment (ADE), Bangalore is being used for this purpose. This is available in open literature [6]. Longitudinal control loop is shown in figure 23. Directional control loop is shown in figure 24.
Figure 23. **Longitudinal Control Loop**

Figure 24. **Directional Control Loop**
**Project Scheme**

Schematic of HILS is shown in figure 25. It contains Simulator, onboard computer and the actuators in the loop. The figure 25 also shows the connectivity between these.

![Schematic of HILS Facility](image)

Figure 25. **Schematic of HILS**

**HILS component**

While setting up HILS, choices for various components of the system have to be made. In this system few of the components are generic in nature and few are dependent on the system under development. Some of these are listed below.
Application independent components

- An operating system which is able to meet the time constraints for flight dynamics simulation.

The flight dynamics has been implemented using a time frame of 1 msec. A real time operating system, RT Linux has been chosen to accomplish this. Maximum scheduling delay is lesser than 20 µsec on Pentium 120. Flight dynamics model is scheduled as a single thread in real time. All file read write operations are done using RealTime FIFOs.

- Data acquisition cards for A/D, D/A and digital I/O required for communication between flight dynamics simulation computer, micro-controller and actuators.

The I/O cards the interface with the external world for sensor output and actuator feedback. The I/O card chosen are PCI-DDA08/12 and PCI-DAS1002 from Measurement Computing. These cards are supported by Comedi drivers available in Linux. Number of channels and conversion time were also the parameters governing the choice of cards. All the interface cards were verified under RT Linux before incorporating into HILS.

Application dependent components

- Micro-controller, which will be going onboard the aircraft.

Micro-controller acquired for on-board computation is Motorola 68332. This unit is available from Onset Computers USA. The choice was made on from the literature survey [7] and size and weight of the onboard computer. Some of the important features are given below.

<table>
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<tr>
<th>Parameter</th>
<th>Value</th>
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</thead>
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<td>Size (cm)</td>
<td>5.08 x 7.72 x 1.27</td>
</tr>
<tr>
<td>Weight (gm)</td>
<td>30</td>
</tr>
<tr>
<td>Processor</td>
<td>68332</td>
</tr>
<tr>
<td>Clock</td>
<td>160 kHz to 16 MHz</td>
</tr>
<tr>
<td>Data Capacity</td>
<td>1 MB</td>
</tr>
</tbody>
</table>
Flash EEPROM 256 KB
A/D converter 12 bit
Analog Channels 8
Peak Current 200 mA
TPU 14
Voltage input 7-15 V

❖ *Actuators, for the vehicle*

Actuators used are FUTABA FP-S148 radio controller servos. Position of these actuators is controlled using Pulse Width Modulation signal. TPU available on the onboard computer are directly compatible for actuator motion.

**Hardware In Loop Simulation**

After implementing and verifying all HILS components and with actuators calibrated, the final HILS setup was assembled and the simulation were run for various test cases. Two simulation results are presented in this paper. It was found that as there was considerable noise in the system, gains as used in offline simulations have to be considerably reduced for getting a smooth actuator movement. There was jitter in the actuator movement and when the steady state is reached, actuators start vibrating around the steady state value with amplitude depending on the noise level and gains. Gains were modified to reduce this with little loss in controller performance.

**Six point simulation in vertical plane**

This simulation study was conducted to understand and implement altitude hold and speed hold algorithms. Waypoints were set in one vertical plane, the aircraft will climb or descend depending upon the situation and maintain the altitude. As shown in figure 26, the aircraft flies in N-H plane. Trajectories from HILS and offline simulation for the same set of gains are shown. It is seen that apart from the height at which they settle,
these trajectories are very much alike. The difference in height at steady state is due to error in height sensed by the micro-controller.

![Six point simulation in vertical plane](image)

**Figure 26. Autonomous flight simulation in vertical N-H plane**

In Figure 27, throttle movement in HILS for trajectory shown in Figure 26 is compared with the movement in off-line simulation. The two curves match except there is a deviation towards the end of the curve (t = 200 sec - 300 sec) though trend of movement is the same. Also, there is a jitter in HILS throttle movement due to noise present in the system.
Simulations in horizontal plane

The second example is considered in which the aircraft flies in a horizontal plane and the waypoints are four corners of a square. Figures 28 and 29 show trajectories for aircraft flying through the four way points following the straight line guidance (SLG) strategy. The trajectories from off-line simulations are also plotted for comparison. It can be seen that the trajectories match very well.
Figure 28. **Autonomous flight simulation in horizontal plane**

Corresponding aileron movement for the above horizontal plane autonomous flight is shown in figure. Match between offline simulation and HILS is very good.
Simulation in 3-D space

Good match obtained from vertical plane simulation and horizontal plane simulation encouraged for full 3D simulation. Figure 30 shows a 10 point simulation in 3-D space. The aircraft tends to lose height at the start of the turn but later recovers height. Loss in height occurs because while turning, the component of lift acting in vertical direction reduces considerably due to large roll angle. When the aircraft loses height, the altitude control loop acts to recover height. Trajectories have been plotted for HILS and off-line simulations and they match well.
Results & Discussion:

DAC card was giving high frequency (>1KHz) noise and this was suppressed using hardware filter (15 Hz) between the onboard computer and simulation computer. The output was within 5% of the desired value.

Off line simulation: Offline simulation was done using 'C' code and the results were compared with the data available in the literature. After verifying the flight dynamics, NGC algorithm was added for autonomous mission simulation. Stability augmentation system for yaw motion is used.
NGC algorithm: The Navigation strategy is waypoint navigation. In this navigation scheme aircraft passes through various points in 3D space (waypoints). This is achieved by implementing two control functions. i) Altitude and speed hold ii) Heading and track hold. Navigation is based on Air Data Dead Reckoning (ADDR) using Indicated Air Speed (IAS) and heading. The navigation error is corrected by GPS data which is available every second. For guidance two strategies are used i) Straight Line Guidance (SLG): In this strategy aircraft tries to follow the straight line joining two waypoints. Guidance is achieved by reducing cross track error as shown in Figure 21. ii) Line of Sight Guidance (LOSG): In this strategy, at any instant the aircraft tries to follow the line joining its position at that instant and the waypoint it is headed towards as shown in Figure 22. Reference heading i.e. heading of line joining aircraft and next waypoint is the guidance criterion. Choice of guidance strategy depends on the mission specification. In case of SLG aircraft will follow a corridor along the line joining two waypoints. Whereas in case of LOSG it is only guaranteed that aircraft will pass the waypoint. In the present case when the aircraft reaches within 100 m radius of target waypoint, the waypoint is assumed to be reached and it starts heading towards the next waypoint.

Onboard software: Onboard software was developed in the 'C' using the Tattletale development environment. Present micro-controller is a 32 bit architecture and without floating point unit, each trigonometric calculation takes about 2 msec. This was reduced by using lookup table to about 0.6 msec.

Real-time simulation: Real-time simulation was done in asynchronous mode. This means that there is no communication between FDS and on-board computer to synchronise events. Both are running at their own real time clock and it is assumed that both are working with wall clock with different offset. Flight dynamics time step is 1 msec and controller time step is 15 msec. The mismatch in the sensor output and on-board computer sensing can be maximum of 1 msec. Advantage of asynchronous mode simulation is that the on-board computer can be used after testing without any modification.
Offline simulations were done to compute the first guess of the gains for the control strategy. Later these gains were modified in HILS. Simulations were carried out for SLG and LOSG under different wind conditions using only ADDR and with GPS assisted corrections. It was also noted during off-line simulations that GPS assisted navigation gives path closer to the intended path especially in presence of winds.

Using HILS setup complex missions consisting of 10 way-points as shown in Figure 30 were carried out. The trajectories from HILS compare well with those of off-line simulations but for small deviations. In Off-line simulations actuator dynamics and interface issues (noise, filter delays etc.) are not included and these are difficult to model. The difference in trajectories obtained using off-line and HILS can be attributed to these. Study was also carried out to see the effect of time step of the on-board controller and good path following was observed over a range of 10 msec to 250 msec.

In the present work balance between cost and the efforts required for development of the system were considered. Decision on selection of DAC/ADC cards was based on the support available in RTLinux to reduce the cost of drivers. Use of higher language in embedded system is common and 'C' development environment for this became handy in the present case.

GPS simulator is ready and it will be incorporated in HILS for a realistic mission. Radio controlled equipment will be used for take-off and landing of autonomous aircraft. Transmitter and receiver will be part of the HILS. It is expected that with all these components a very realistic autonomous mission will be flown in which radio transmitter-receiver, on-board computer and actuators will be actual hardware in the simulation loop.

Work on development of altitude hold, speed hold and heading hold is under progress and this will be thoroughly tested on HILS before flight test.

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